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Final
Report

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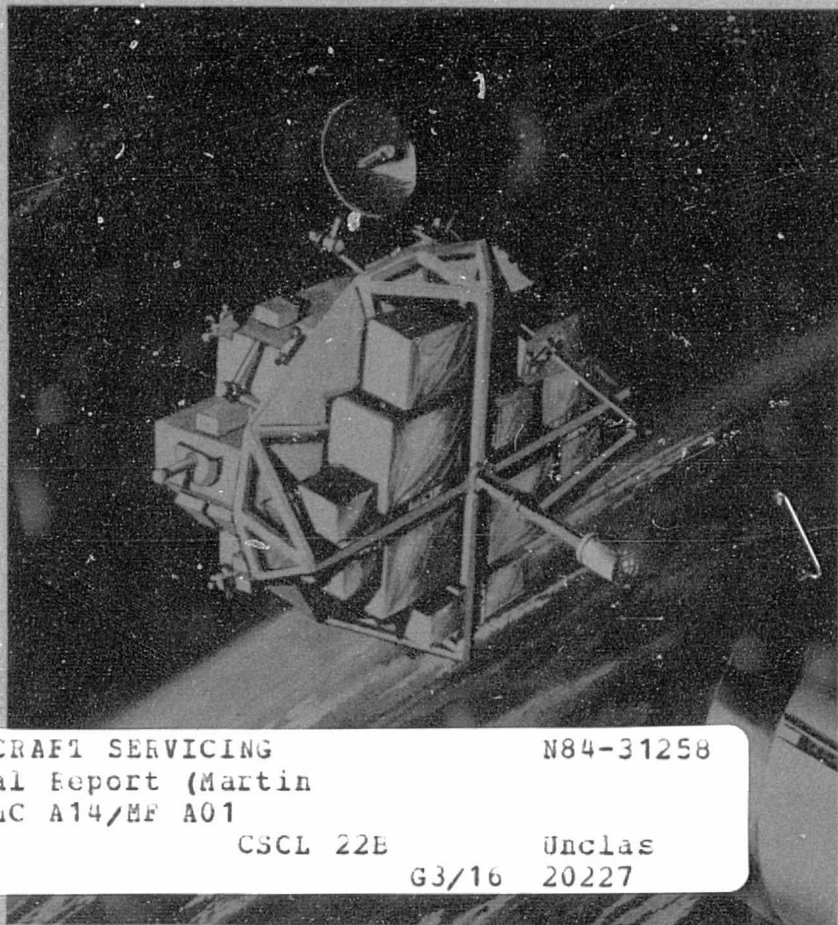
Spacecraft Servicing Demonstration Plan



MARTIN MARIETTA

MCR-84-1866
CONTRACT NAS8-35496 FINAL REPORT JULY 1984

SPACECRAFT SERVICING DEMONSTRATION PLAN

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FOREWORD

This study was performed under Contract NAS8-35496 for the George C. Marshall Space Flight Center of the National Aeronautics and Space Administration under the direction of James R. Turner, the Contracting Officer's representative.

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1.0 SUMMARY

A preliminary spacecraft servicing demonstration plan has been prepared that leads to a fully verified operational on-orbit servicing system based on the module exchange, refueling, and resupply technologies by mid 1992. The resulting system can be applied at the space station, in low earth orbit with an Orbital Maneuvering Vehicle (OMV), or be carried with an OMV to geosynchronous orbit by an Orbital Transfer Vehicle (OTV). The three recommended overlapping phases are:

1) Ground demonstrations using the MSFC Information and Electronics laboratory;
2) Cargo-bay demonstrations in the orbiter using the Remote Manipulator System to dock a spacecraft mockup to the servicer and the Multimission Modular Spacecraft flight support system to support the servicer and stowage rack. Two cargo-bay servicing demonstration flights are recommended, one for module exchange and the other for a refueling demonstration;
3) Free-flight verification using the OMV as the carrier vehicle and a rented spacecraft bus to carry the serviceable spacecraft mockup.

The plan emphasizes the exchange of Multimission Modular Spacecraft (MMS) modules as the MMS is a significant ongoing program involving space-repairable satellites.

Three servicer mechanism configurations are included in the plan:

1) The Engineering Test Unit currently in use at MSFC would be used for ground demonstrations, procedures development, and training;
2) A protoflight quality unit would be used for the two demonstration flights in the orbiter cargo bay;
3) Two fully operational units that have been qualified and documented would be used in the free-flight verification activity. However, deletion of the second unit would save $3.0M.

The plan balances costs and risks by overlapping study phases, utilizing existing equipment for the ground demonstrations, maximizing use of existing MMS equipment, taking advantage of the ongoing NASA-JSC

orbital refueling program, and rental of a spacecraft bus rather than building a new unit for a one-time use in the free-flight verifications. The preliminary funding estimate is $1.5M for the ground demonstrations, $20M for the cargo-bay demonstrations, $35M for the free-flight verifications, and a total of $56.5M in 1984 dollars.

The plan must be significant and long-term to encourage users and spacecraft designers to include on-orbit servicing in the form of module exchange in their plans.

1.1 INTRODUCTION

Many studies and demonstrations during the past decade have clearly proven the overwhelming cost effectiveness benefits of an unmanned on-orbit satellite servicing capability. The ability to change out failed or worn-out satellite modules and to replenish fuels and other expendable commodities offers satellite programs a greatly reduced operating cost when compared with replacement of an entire satellite. Development activities that will eventually lead to routine orbital servicing operations were initiated in the early 1970s. Several alternative servicing systems including satellite modules and component design approaches were defined and evaluated during this period. With the shuttle vehicle now operational, the capability exists to deliver and retrieve an operational servicer system. It is thus appropriate to initiate planning that will lead directly to the operational servicing capability.

Various alternatives for satellite maintenance have been identified, conceptualized, and evaluated--unmanned orbital servicing systems, manned extravehicular activities, highly reliable expendable designs, and retrieval for ground refurbishment and return to orbit. The first Integrated Orbital Servicing System (IOSS) study completed in September 1975 along with a parallel study, Integrated Orbital Servicing and Payloads Study, conducted by COMSAT Laboratories of the Communications Satellite Corporation, jointly concluded:

1) On-orbit servicing is the most cost-effective satellite maintenance approach;

2) Development of a single on-orbit servicer maintenance system is compatible with many spacecraft programs;
3) Spacecraft can be designed to be serviceable with acceptable design, weight, volume, and cost effects;
4) The evolving Space Transportation System (STS) is designed to support on-orbit maintenance;
5) Users need guarantees that servicing will be available and assurances that it will be cost effective.

As satellite designs continue to evolve and the space station era approaches, it becomes apparent that there is room for virtually all the alternatives of satellite maintenance at one point or other in the future. However, to minimize servicer system development costs, the IOSS follow-on study, completed in June 1978, recommended that a single servicer system having the capability to accommodate both low earth and geosynchronous orbit applications should be evolved. This requirement has been satisfied effectively by the servicer mechanization (Fig. 1-1) conceputalized during the IOSS studies. The single design is compatible with maintenance of most spacecraft of the Space Transportation System era. Adapters may be used to accommodate support structure differences across the applications. An effective interface between the spacecraft and the servicer was defined and breadboarded. The interface mechanism provides a logical and cost effective method of incorporating orbital replaceable units (ORU) for module exchange in all spacecraft.

The value of demonstrations in furthering on-orbit servicing development was recognized in the decision to build a 1-g version of the Integrated Orbital Servicing System of Figure 1-1. The result is the Engineering Test Unit (ETU) of the IOSS shown in the photograph of Figure 1-2. This unit was built and delivered to MSFC in 1978. It has been used for over 250 demonstrations during the intervening six years. The ETU has shorter segment lengths that the IOSS as it was designed initially for axial module exchange only. The later addition of a sixth degree of freedom extended the ETU's capability to radial module removal, albeit at a radius less than that of the orbiter cargo bay.

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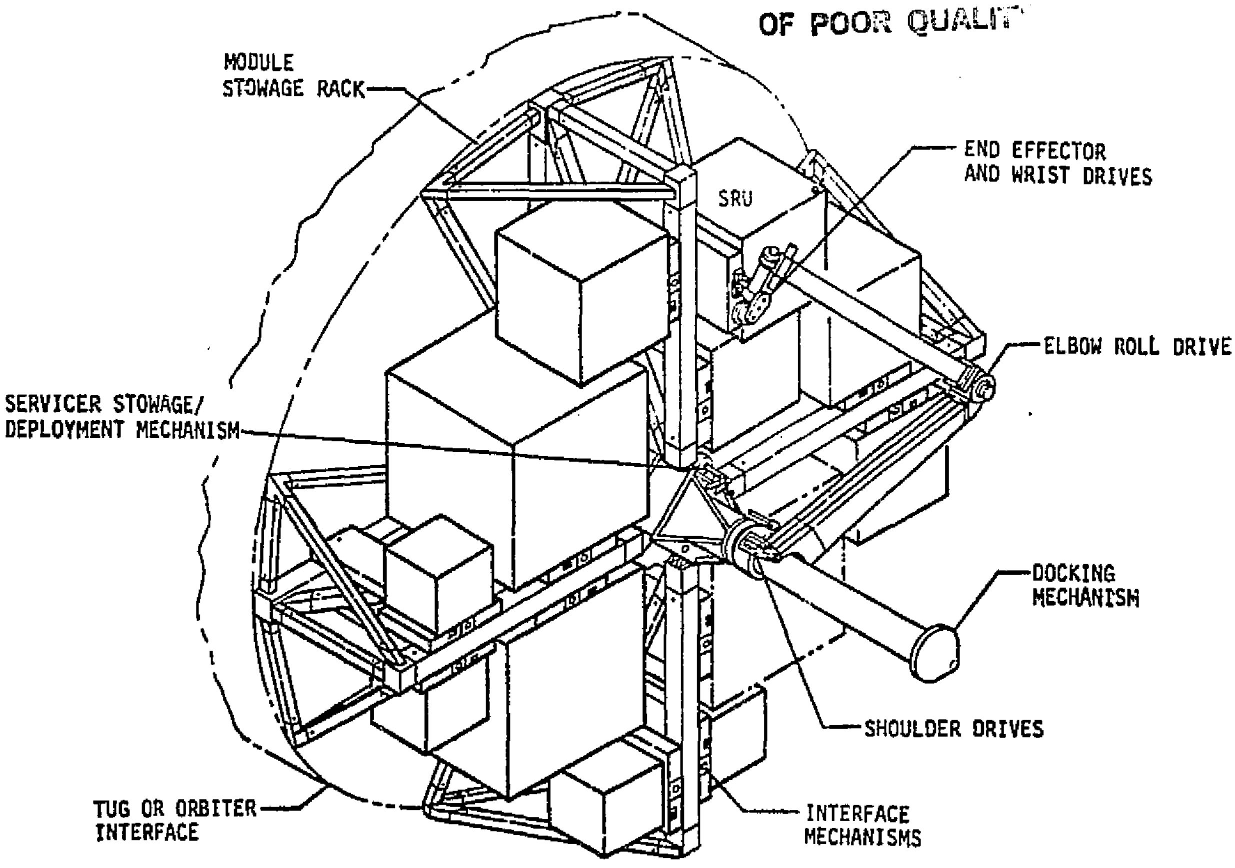

Figure 1-1 IOSS On-Orbit Servicer Configuration

To date, satellite systems in general have not been designed and built with the capability of changeout of subsystem or component modules. One satellite that is currently in use and has an extravehicular activity (EVA) module exchange capability is the Goddard Space Flight Center's Multimission Modular Spacecraft. This spacecraft is in operation in several programs and is projected for continued use throughout the remainder of this century. The Marshall Space Flight Center's Space Telescope (ST) was designed for EVA module changeout and is expected to fly soon. The MMS modules are more accessible for remote module exchange.

Considerable interest in spacecraft maintenance was expressed by both the Department of Defense and the commercial sector, however, the general tenor of their support was that a demonstration of orbital maintenance must be conducted prior to any commitment on their part. A flight demonstration of the all-up maintenance capability is also a NASA requirement prior to wholesale commitment to the concept.

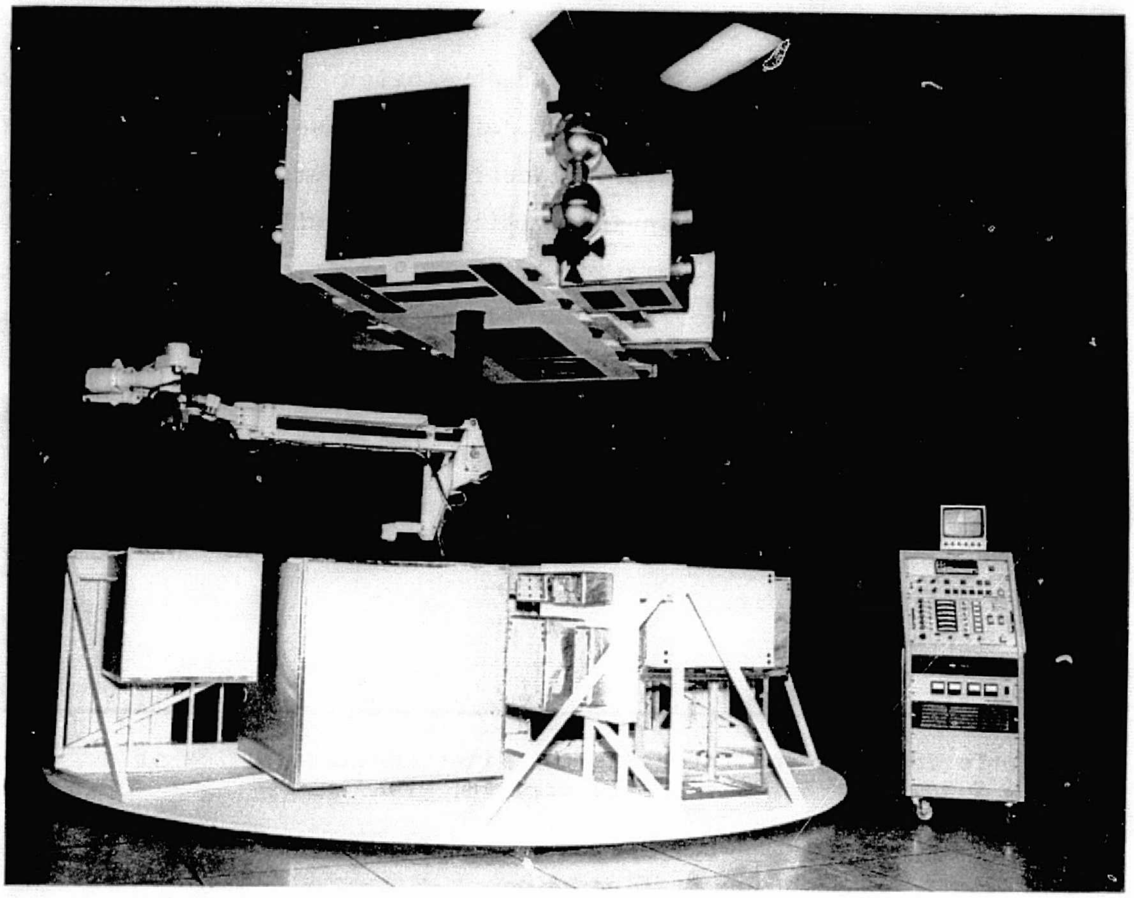

Figure 1-2 Engineering Test Unit

However, a reduced capability test that exercises the basic concept and exchanger capability can and should be demonstrated prior to the time that a full capability will exist. With this background material in hand, and with renewed interest by the space flight community, it was appropriate to perform a study that would define a path to culminate in demonstration of the servicing capability. The objective of this study was to provide a single unified development program for use by both servicing implementors and users to guide their future development and operational plans for this important technology.

1.2 STUDY OBJECTIVES

The objectives of this spacecraft servicing demonstration plan study are to identify all major elements and characteristics of an on-orbit servicing development program and to integrate them into a coherent set of demonstrations. The goal of the development program is a fully verified operational on-orbit servicing system based on the module exchange, refueling, and resupply technologies. The existence of the plan, combined with NASA's support, will increase user acceptance of on-orbit servicing. The study objectives are summarized in Table 1-1. A ground demonstration plan is envisioned that will provide confidence in the development and operation of the on-orbit system. The servicing ground demonstrations cover a wide range of satellite module sizes and include the ability to service propellant systems. They also include a servicing mechanism configuration that is representative of an eventual flight unit. Emphasis was placed on the exchange of MMS modules. The ground demonstrations will screen the elements and characteristics of the development program to identify activities to be demonstrated in the orbiter cargo bay.

Table 1-1 - Study Objectives

To identify and integrate the major characteristics of a Spacecraft Servicing Demonstration Plan
Major plan elements
Ground demonstrations
Orbiter cargo-bay demonstrations
Free-flight verification

The orbiter cargo-bay demonstrations utilize a protoflight version of the servicer mechanism to reduce project costs. Two flights are planned. The first is to demonstrate the exchange of a variety of modules, operation of the communications system, adequacy of control using three different control system approaches, and accuracy of spacecraft to stowage rack alignment when the Remote Manipulator System end effector is used as a docking mechanism. The second flight will involve a demonstration of refueling (propellant replenishment) and resupply (other fluid replenishment). It was decided, as a result of the study, to add a free-flyer (Orbital Maneuvering Vehicle) demonstration to the plan as a way of verifying the capabilities of an operational servicer.

The objective of the ground demonstration task was to produce a conceptual design for a ground demonstration system that is capable of exchanging various spacecraft-sized subsystem modules and performing connections of umbilicals and propellant-line connection devices. The conceptual design selected was based on the requirements for flight servicing as well as the availability of existing hardware. The study emphasized evaluations of the types and configurations of satellite subsystem and major assembly modules, the latching mechanisms, the end effector configurations for the servicer unit, the types of refueling and electrical connections that must be made, and the configurations that are available for each and it recommended that the existing ETU be used as the ground demonstration servicing unit.

The ETU is representative and provides a high-fidelity simulation of the flight servicing system. The ground demonstration program is broader in scope than the flight program. The ground demonstration program can address many possible operating conditions to assure that demonstrations in significant servicer techniques are performed. The flight program can then verify certain elements of the ground program to provide confidence across the full spectrum of operations.

Before specific ground demonstration activities could be identified and defined it was necessary to perform some supporting analyses and to select a servicer configuration for the ground demonstrations. The objective of the supporting analyses was to better define what should be done in the ground demonstrations. The objective of the first supporting analysis was to identify a recommended servicing configuration for the MMS. The selected configuration is to dock the servicer to the MMS triangular module support structure using a docking probe adapter. This puts the servicer mechanism where the individual MMS modules can be exchanged in an axial direction with respect to the servicer mechanism.

The objective of the second ground demonstration supporting study was to select arrangements for the demonstration of refueling, resupply and electrical connection. The recommended refueling interconnection approach involves a set of fluid disconnects mounted on a translation device and connected to supply tanks in the stowage rack by constrained flexible hoses. The servicer mechanism picks up the translation device and moves it to a connection point on the spacecraft. The translation device is attached to the spacecraft and then it mates the fluid quick disconnects and any electrical connections that are included. The servicer arm can then free itself from the translation device and perform other servicing tasks while propellants and the commodities are being transferred.

The objective of the third ground demonstration support task was to select representative modules and servicing tasks other than the MMS module exchange. The recommended modules include (1) 24 in. cube with side mounting interface mechanism, (2) AXAF focal plane interconnect module, (3) individual component level modules, (4) thermal covers, and (5) representative geostationary satellite modules.

The objective of the fourth ground demonstration support task was to recommend an end effector configuration for the servicer mechanism. The IOSS end effector, complemented by a series of tools and adapters, was recommended. A primary adapter is a modified version of the MMS module servicing tool for use with MMS modules.

The objective of the servicer configuration analyses was to select a configuration from six candidates for the ground demonstrations. The primary candidates were the Engineering Test Unit and the Proto-Flight Manipulator Arm (PFMA). The PFMA is a general purpose six degree-of-freedom manipulator that has been in use at MSFC for six years and is capable of performing a wide variety of tasks. The ETU was selected because the PFMA would require extensive rework before it could be used and the ETU had been designed to do the module exchange task.

The final objective of the ground demonstration analyses was to identify representative activities and to prepare schedule and cost estimates. The recommended activities are:

1) Control system upgrading;

2) MMS module exchange;
3) Representative module exchange;
4) Refueling demonstration;
5) Automatic target recognition;
6) Flight demonstration simulation;
7) Flight training;
8) Available for problem solving.

The last three items are in support of the flight demonstrations.

The objective of the flight demonstration analyses was to identify and define the major elements of an on-orbit servicing demonstration in the orbiter cargo bay. The objective of the cargo-bay demonstration is to help convince satellite designers that on-orbit servicing in the form of module exchange can be done and that the major elements of the system can be designed, built, and operated. Additionally schedule and cost estimates were to be prepared. It was recommended that two cargo-bay flights be conducted. The first to demonstrate module exchange and the second to demonstrate refueling. A special protoflight version of the on-orbit servicer mechanism is to be designed, built, and used for the two demonstration flights. The above objective was expanded to include a free-flight verification of on-orbit servicing using an OMV as the servicer carrier vehicle and a rented spacecraft bus for support of the serviceable spacecraft mockup. The cost estimates were based on two OMV-compatible units of a fully qualified and documented servicer system being designed and built for use in the free-flight verification and for subsequent operational use.

The objective of the servicing development plan activity is to integrate the results of the ground and flight demonstration activities into an orderly development plan leading to a fully verified operational on-orbit servicing system based on the module exchange, refueling, and resupply technologies. The word refueling is used to denote the replenishment of any or all fluids involved in the spacecraft propulsion or attitude control systems, while the word resupply is generally used to denote the replenishment of all other fluids including cryogenics used for instruments. The resulting plan overlaps the ground, cargo-bay, and free-flight phases to lead to a free-flight verification in mid 1992.

This study was performed to provide implementors and users with a single development approach that will culminate in orbital servicing operations. The study is necessary at this time because only by providing a planned development program will both development and user support be focused on the servicing issue. Current planning for the Orbital Maneuvering Vehicle is such that servicer development must be started soon if a servicing capability is to exist shortly after the OMV reaches an operational status. Verification of a servicing capability with the OMV will result in a well-proven system being available for use with the space station. Many prior and current studies have addressed individual elements of servicing. Many tools and support hardware elements have been defined that will aid a future servicing program. These efforts, however, have not culminated in a general move on the part of the user community to incorporate serviceability into their spacecraft designs. It is only through the implementation of a development program that produces a demonstrated on-orbit servicing capability that the benefits of this program will be realized in future spacecraft operations. The preliminary development program plan described in this report was prepared to satisfy this need.

1.3 RELATIONSHIP TO OTHER NASA EFFORTS

Prior and ongoing NASA activities, as well as future plans, in the area of satellite servicing are discussed in relation to the objectives and approach of this spacecraft servicing demonstration plan study.

Servicing development activities were initiated in the early 1970s and continue through the present time. Studies and development work have been performed by NASA, other government agencies, and contractors. Early study results concluded that on-orbit servicing was a more cost effective approach than ground refurbishment of satellites.

Recommendations included that spacecraft be designed for servicing and that module exchange was the most cost-effective method of servicing. During the Integrated Orbital Servicing System study an Engineering Test Unit was designed and built and has been in use at MSFC since 1978 for ground demonstrations of remote satellite servicing and other development activities. A wealth of experimental data has been accumulated during this servicer demonstration and development program and constitutes the basis for the next step in the development of on-orbit satellite servicing capability.

As the Space Transportation System is operational, satellites in low earth orbit are accessible for on-orbit maintenance and repair. Many NASA efforts are now directed towards definition of the requirements, interfaces and programmatic aspects of the three main approaches to satellite servicing: (1) manned, using extravehicular activities, (2) remote servicing, using a simple specialized mechanism for module exchange, refueling, and resupply and controlled in manual and automated modes, and (3) remote servicing operations using telepresence technology and artificial intelligence.

The EVA satellite servicing culminated recently with a successful demonstration during the Solar Maximum Repair Mission when equipment modules were exchanged on a Multimission Modular Spacecraft utilizing the orbiter Remote Manipulator System (RMS), the Manned Maneuvering Unit (MMU) and a module servicing tool (MST). Another candidate for a similar repair mission is a failed Landsat D communications satellite. Many tools and auxiliary devices have been developed for use by the shuttle or space station EVA crews to perform various servicing tasks. The accummulated EVA experience emphasizes the need for simple, easy maintenance and repair tasks, ample clearances to accommodate the rather bulky EVA suit, and provision for handrails and foot restraint brackets. Due to EVA time and space limitations and the high cost and risk involved, baselining EVA for maintenance, repair and refueling/resupply of spacecraft needs to be determined by the user on an individual basis. Because of man's direct involvement in the operations, the safety aspects are particularly important and difficult to resolve. However, EVA remains the main back-up system for repair in contingency situations at the orbiter and space station, due to its superior flexibility and ability to perform unscheduled and unplanned repair operations.

An Orbital Maneuvering Vehicle is being developed by NASA-MSFC, with the participation of other NASA centers, to supplement the STS for satellite delivery, retrieval and on-orbit servicing. It will utilize the orbiter for launch and will have applications in both low earth orbits (LEO) and geostationary earth orbit (GEO), when transported to GEO by an Orbital Transfer Vehicle (OTV) or other orbit transfer stage. Early availability of the OMV as a reusable vehicle will obviate the necessity of including integral propulsion in many new space initiatives for satellite deployment or retrieval. The OMV will have a man-in-the-loop control capability from a ground control station (GCS). Rendezvous and docking capability and an OMV compatible servicer kit will be developed in subsequent phases to add satellite retrieval and on-orbit servicing capabilities. The servicer will be controlled from the GCS of the OMV.

The servicer system will be composed of a servicer mechanism, a docking probe and a stowage rack for spare modules of equipment and refueling/resupply units. The servicer will be capable of manual and automated modes of control. The OMV will provide attitude control, power, thermal protection, servicer control electronics, two way communications links for RF and video, rendezvous and docking, and structural support. The OMV and the servicer will utilize the present state of the art technology in order to become operational in the 1990-1992 time period. They will provide a much needed satellite deployment, retrieval and servicing capability to supplement and enhance the STS operations.

Specific servicing aspects are being defined by NASA in connection with space station operations. Maintenance and repair missions are being evaluated for the space station. For the proximity operations an RMS will be used, with manual control from a special servicing platform.

For LEO satellite deployment and retrieval, the OMV will be used. On-orbit satellite servicing at LEO will be performed using an OMV and a servicer from the space station. Similar operations at GEO will use an OTV from the space station to deploy and retrieve the OMV and the servicer. The control of the servicer can be from the space station or from the ground. Operating the OMV/servicer or OTV/OMV/servicer from the space station will provide better availability of servicing and will reduce the launch cost.

Many studies during the past decade proved the cost benefits of on-orbit refueling. The areas of fluid management requiring new technology have been identified. Cargo-bay experiments are now planned by NASA-JSC to demonstrate fluid transfer in 0-g and to test new quick disconnects and sensors. For these first experiments, EVA operations are planned. Safety aspects are of prime concern. Standardization of the refueling interface is an important issue affecting the economics and ultimately the success of the satellite refueling activities. An interface standardization project is being pursued by NASA-JSC. The objective is to develop a standard propellant servicing interface for all satellites. A committee will be formed consisting of appropriate NASA elements, the DoD and those industrial firms active in the design and fabrication of satellite propulsion stages. This committee will define the fluid interconnects, mechanical attachment hardware, isolation philosophy, data format requirements, and instrumentation and control interfaces consistent with safety requirements and minimization of crew time lines. The program objectives are to develop and certify a standardized disconnect design for on-orbit resupply of earth storable, gaseous and cryogenic fluids and to provide earth storable fluid disconnect flight hardware for the Gamma Ray Observatory by March 1986.

This Spacecraft Servicing Demonstration Plan makes use of the experience accummulated during the IOSS demonstrations and expands its scope to encompass demonstrations of Multimission Modular Spacecraft servicing, other module and component exchange, and refueling demonstrations utilizing the present state of the art technology. The plan provides ground demonstrations, cargo-bay experimental demonstrations, and free flight verification of an operational, OMV-compatible, servicer system. Timing of various planned activities is such that it takes advantage of the results of the NASA-JSC refueling development effort and matches the milestones of the OMV development program schedule.

Most of the routine servicing tasks can be accomplished by a remotely controlled servicer built with existing technology and performing module exchange and refueling/resupply of fluids. These tasks involve handling heavy and sometimes bulky modules, long refueling/resupply operations, handling of hazardous fluids, all of which are performed safer, faster and at less cost than by EVA. A smaller number of the servicing tasks, comprised of exchange of equipment at the component level such as batteries, access doors, and thermal covers and some of the unplanned repair tasks, such as deployment of a stuck antenna, can also be performed by this relatively simple servicer mechanism when it is provided with special adapters. A few servicing tasks and some of the unplanned repair tasks, however, can presently be performed on-orbit only by EVA in the proximity of the orbiter. The cost and the risks involved in using EVA and the associated operational constraints justify the present NASA efforts to develop a new generation of automatic servicing systems.

A simple, proven servicer mechanism, with a standardized end effector interface and supplemented by specialized adapters and interface mechanisms, like the IOSS, can be built today with the present technology. It will provide the much needed satellite servicing capability now and the ability to test and develop the elements of future generation servicers.

1.4 STUDY APPROACH

Figure 1-3 shows the major tasks of this study and their interrelationship. Task 3, the flight demonstration plan, is the key element in that it can provide the basis for the satellite servicing capability that will exist in the future. This capability will be utilized by future spacecraft designers in establishing servicing concepts for their space vehicles. This task was performed in parallel with Task 1 in order to influence the selection of the type of servicer and the hardware concept to be utilized for the ground demonstrations. Conversely, the hardware availability output from Task 1 affected the selection and requirements for hardware considered in Task 3. These two tasks had a strong synergistic effect and were performed in a manner to produce a maximum amount of commonality in hardware to be used. The objective of the ground demonstrations was to reduce risk and verify the approach for the flight demonstrations.

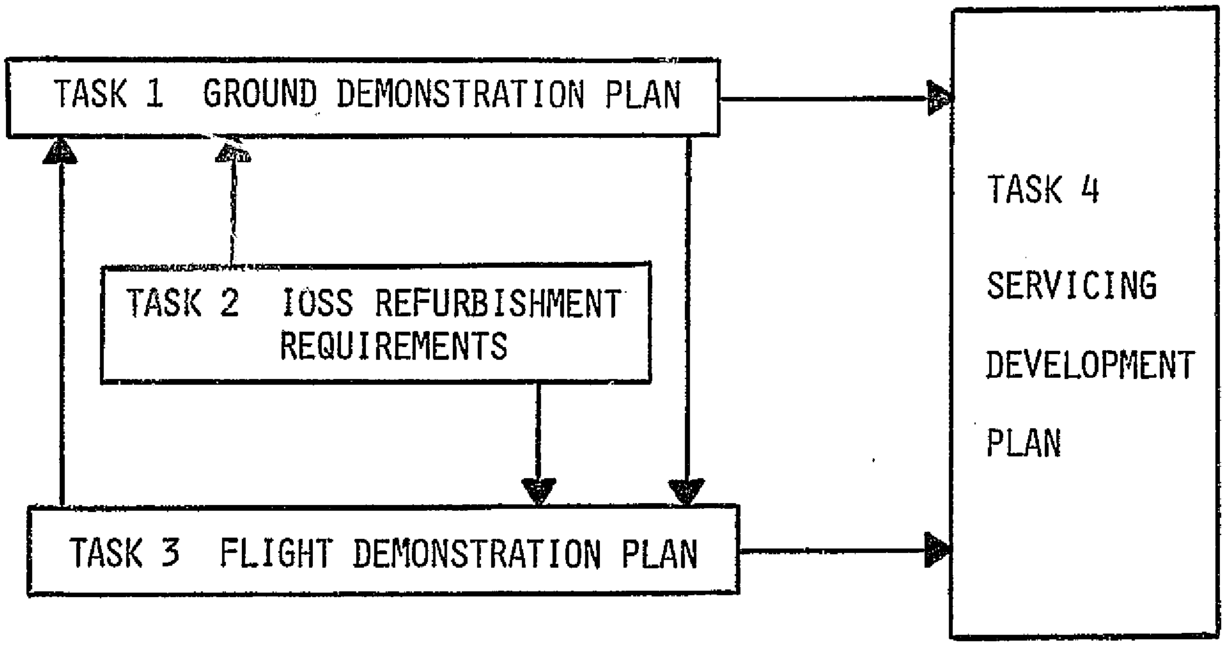

Figure 1-3 Task Logic Flow Diagram

The current status of the Engineering Test Unit of the IOSS was reviewed under Task 2 to establish the ETU's capability to be utilized in the ground demonstration planning. With the major elements of both the ground and flight demonstration plan established, a development plan with overall cost estimates was prepared in Task 4 of this study. A more detailed description of the approach to each of the four tasks of this study is presented next.

1.4.1 Task 1 - Ground Demonstration Plan

Analyses were performed to determine the type and the size of the modules to be used, as well as the refueling and resupply hardware, the type of end effector and the special adapters to be incorporated in the ground demonstration system. An evaluation of existing designs of servicer mechanisms was performed to select the type of servicer to be used. The Integrated Orbital Servicing System, the Proto-Flight Manipulator Arm, the Remote Orbital Servicing System (ROSS), the STS Remote Manipulator System and other servicer mechanisms and systems were analyzed and traded off against the requirements for the ground demonstrations. This analysis was performed in parallel with the servicer system selection for the flight demonstrations of Task 3 so that commonality of design, hardware and procurement could be achieved. The results of the inspection of the ETU of the IOSS, performed under Task 2, were considered in the final selection and recommendation of the type of servicer to be used for ground demonstrations of satellite servicing.

A modified IOSS servicer was recommended, capable of demonstrating MMS type module changeout and refueling in addition to the existing cube modules with a side attachment interface mechanism. The flexibility of the servicing system was enhanced by using special adapters and modular refueling units. Other servicing task demonstrations were proposed, such as thermal cover removal and changeout of component level modules, communications satellite module, OMV module and AXAF module, to further demonstrate the flexibility of the system. Cost estimates were given for each of these demonstrations.

A ground demonstration plan was developed including a recommended schedule for the design and development of the servicing hardware for the ground demonstrations.

1.4.2 Task 2 - Engineering Test Unit Refurbishment Requirements

The Engineering Test Unit of the IOSS was inspected at the Marshall Space Flight Center, Information and Electronics Laboratory. The purpose of the inspection was to determine the refurbishment requirements for using the ETU in the ground demonstration program.

A review of the ETU failure history was performed and the ETU's condition was assessed and compared with the needs and requirements of the demonstration program. Electromechanically, the system is in very good condition. Problem areas in the software and controls were identified as well as the need for further analysis and design effort.

1.4.3 Task 3 - Flight Demonstration Plan

The approach to this task was to review prior work on the subject to identify elements of the operational system, requirements, constraints, and alternative concepts. The desirable characteristics of a cargo-bay experiment were then identified and the rationale was stated. This was followed by an identification of candidate flight demonstration activities. These are discussed only in a general way as the specifics are expected to evolve as new spacecraft are designed and new functional equipment, such as for refueling, becomes available. Several arrangements of equipment in the orbiter cargo bay are then described, evaluated, and a recommended arrangement is selected. This is followed by a discussion of a free-flight demonstration and a summary of the flight demonstration plan. Schedules were developed for both cargo-bay and free-flight demonstrations from an OMV development schedule provided by MSFC. The key points from the OMV schedule were an OMV authority to proceed for Phases C and D on January 1, 1986, an end of supporting development for a servicer kit in July of 1988 and a first flight on January 31, 1990. Detailed schedules were prepared for the development of a servicer system and for the serviced spacecraft. Cost estimates both for cargo-bay and free-flight servicer demonstrations were prepared.

1.4.4 Task 4 - Servicing Development Plan

The results of the ground demonstration and flight demonstration studies were integrated into an orderly development plan leading to a fully verified operational on-orbit servicing system based on the module exchange and refueling/resupply technologies. The key servicing development plan issue was the need to balance the number and complexity of development activities against available funds. The proposed approach was to lay out a program with most of the desired features, that overlaps the 0-g, 1-g, and operational servicer demonstrations, and attempts to get an early operational capability. This approach minimized costs by taking advantage of parallel activities such as the JSC refueling program, and advocated renting a spacecraft bus rather than buying a new one. The program was also scoped large enough to become a recognized part of NASA's long-range plans. A schedule of the overall servicing development plan, based on the OMV development schedule was prepared and a funding estimate by development phase and by year was given.

The promise of a clear plan by NASA to develop and use module exchange for many years will encourage the user, or spacecraft designer, to incorporate module exchange in his plans.

1.5 SIGNIFICANT CONCLUSIONS AND RECOMMENDATIONS

The significant conclusions and recommendations from this Spacecraft Servicing Demonstration Plan study are presented below. Many secondary conclusions and recommendations are given in Sections 3 through 6. The conclusions and recommendations are presented in order from the bottom up except that those conclusions spanning the study are given first.

1.5.1 On-Orbit Servicing Development

The following conclusions and recommendations apply to the overall on-orbit servicing development:

1) The recommended plan leads to the free-flight verification of an operational servicer suitable for use with the OMV and the space station;

2) The plan has three phases:
   - Ground demonstrations,
   - Cargo-bay demonstrations,
   - Free-flight verification;
3) The free-flight verification can be completed by mid 1992 (Fig. 1-4);
4) The total estimated cost is 56.5 million 1984 dollars;
5) The plan includes three servicer mechanism configurations:
   - The Engineering Test Unit currently in use at MSFC would be used for ground demonstrations, procedures development, and training,
   - A protoflight quality unit would be used for the two demonstration flights in the orbiter cargo bay,
   - Two fully operational units that have been qualified and documented for use in the free-flight verification activity;
6) A user's council should be formed to direct the implementation of an on-orbit servicing capability.

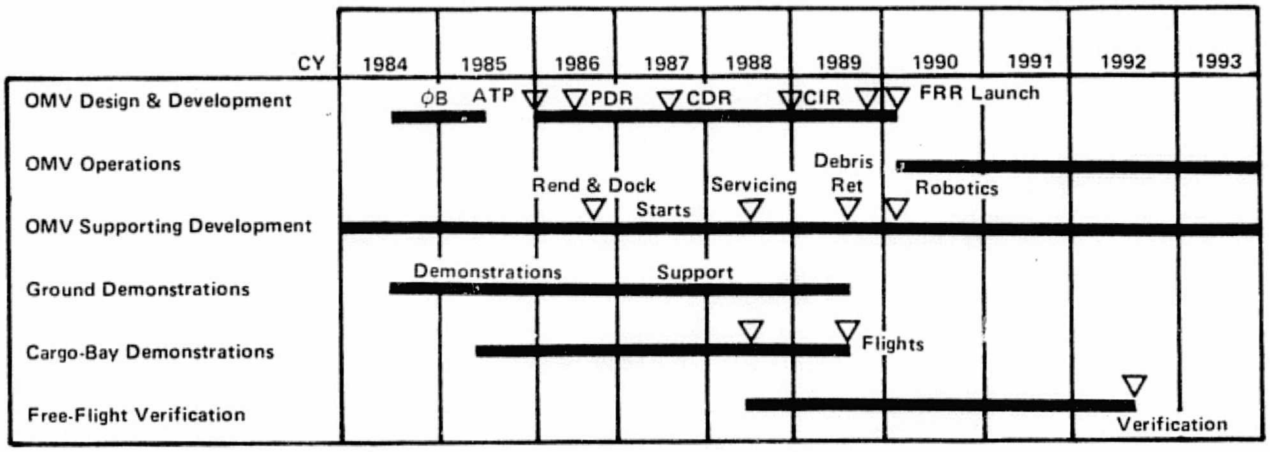

Figure 1-4 On-Orbit Servicing Development Schedule

1.5.2 Multimission Modular Spacecraft

The following conclusions and recommendations apply to the involvement of MMS equipment in the demonstration plan:

1) Primary emphasis would be on demonstrating the exchange of MMS modules (Fig. 1-5);
2) Lightweight MMS module mockups with standard MMS attachment fixtures and connectors should be used for ground demonstration;

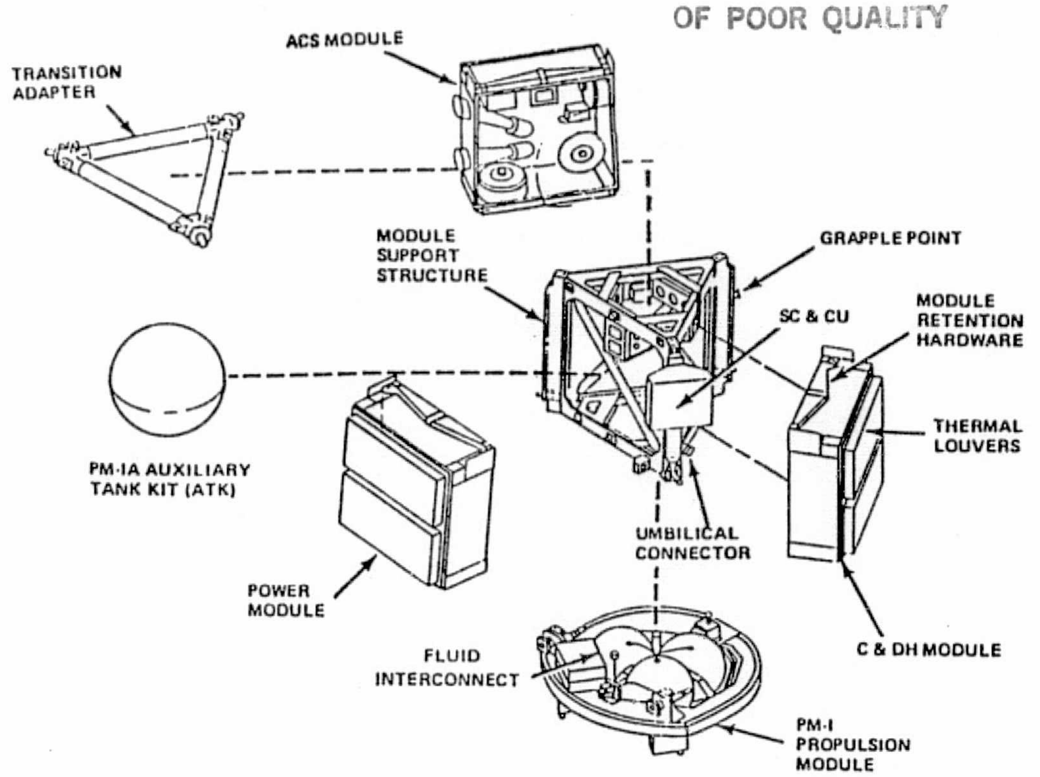

Figure 1-5 Multimission Modular Spacecraft Mechanical System

3) On-orbit servicing of MMS modules should be effected by use of lateral docking with a straight docking probe adapter, tool adapter and modified stowage rack (Fig. 1-6);

4) The MMS flight support system should be used to support the stowage rack and servicer during the cargo-bay demonstrations.

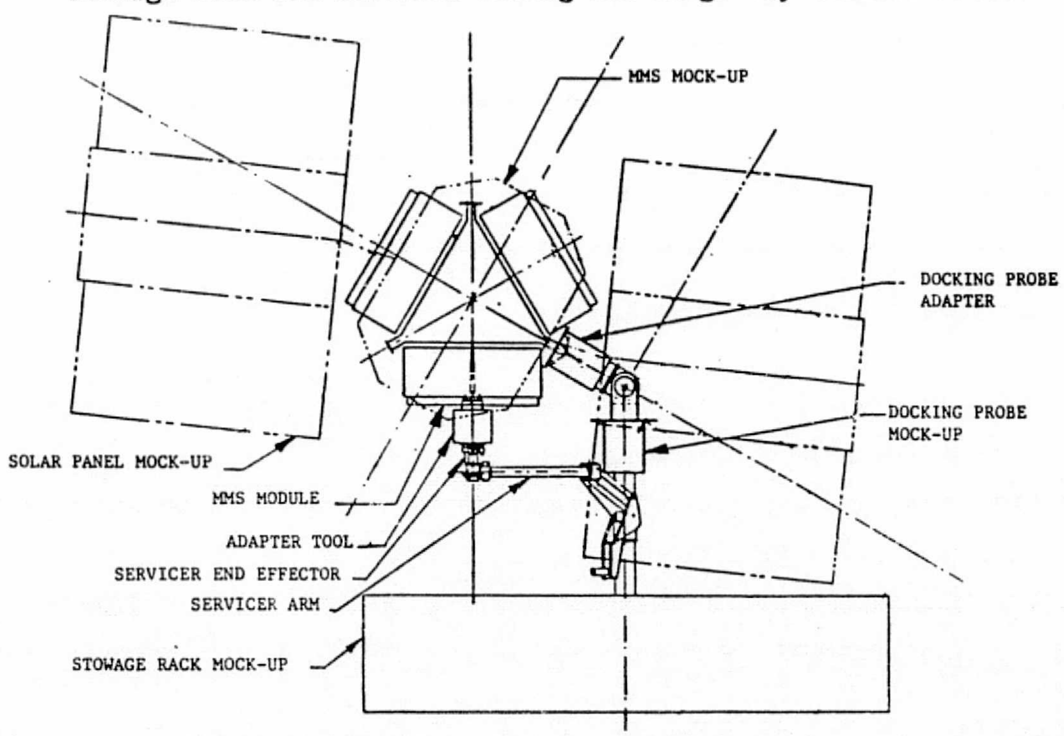

Figure 1-6 MMS Module Exchange 1-g Configuration

1.5.3 Refueling Demonstrations

The following conclusions and recommendations were made with respect to refueling demonstrations:

1) Refueling should be demonstrated;
2) Refueling and resupply modular units should be mounted on the stowage rack and connecting hoses should be positioned and connected by the servicer arm (Fig. 1-7);
3) The refueling demonstration equipment should be based on the NASA-JSC standardization effort;
4) Development work is necessary for in-line coupling and mate/demate mechanisms.

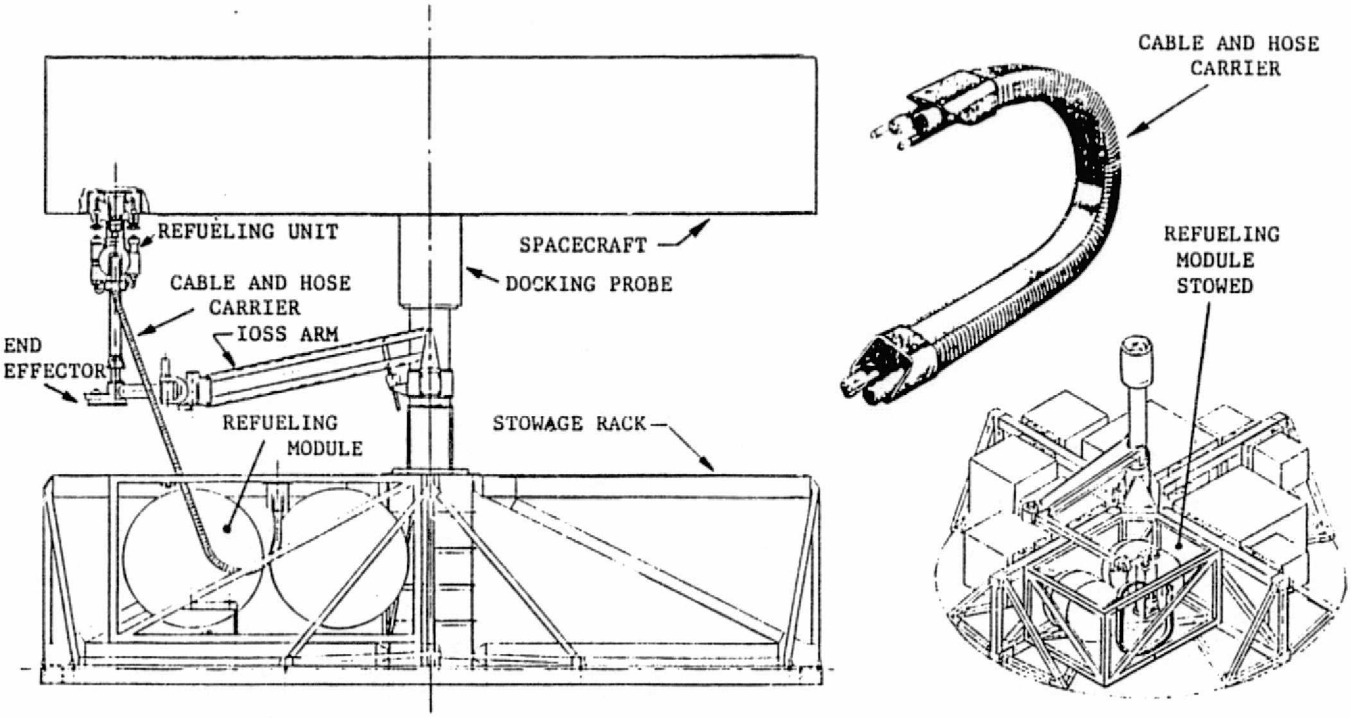

Figure 1-7 Refueling and Resupply Module on Stowage Rack

1.5.4 Representative Satellite Modules

The following conclusions and recommendations were made with regard to selection of representative or generic module exchange:

1) A variety of modules other than MMS modules should be involved in the demonstrations;
2) Thermal cover removal/replace mechanisms and sensors for fastener and attach interface status need to be developed;

3) Changeout of modules representative of the OMV, AXAF, and communications satellites should be included in the program;
4) Axial, near-radial, and off-axis module removal directions for spacecraft modules should be included;
5) Changeout of modules on the stowage rack need be in the axial direction only;
6) A variety of interface mechanisms are possible and could be useful.

1.5.5 End Effector Selection

The following conclusions and recommendations were developed as part of the end effector configuration selection process:

1) The IOSS end effector is recommended for the ground and flight servicing demonstrations (Fig. 1-8);
2) The IOSS end effector meets the end effector requirements and when complemented by a series of adapters can perform the servicing tasks considered;
3) An electrical disconnect should be added to the ETU end effector.

1.5.6 Servicer Mechanism Selection

The following conclusions and recommendations were developed as part of the servicer mechanism selection process:

1) The Engineering Test Unit should be used for ground demonstrations (see Fig. 1-2);
2) The servicer mechanism selection was based on high fidelity, accuracy, versatility, reliability, cost, and risk avoidance;
3) The ETU servicer mechanism is compact and performs module exchange and other tasks efficiently. It was designed to conduct 1-g module exchange demonstrations and it has an effective counterbalance system;
4) The Proto-Flight Manipulator Arm is not as desirable as the ETU because it requires important development work in order to integrate it in a servicer ground demonstration system.

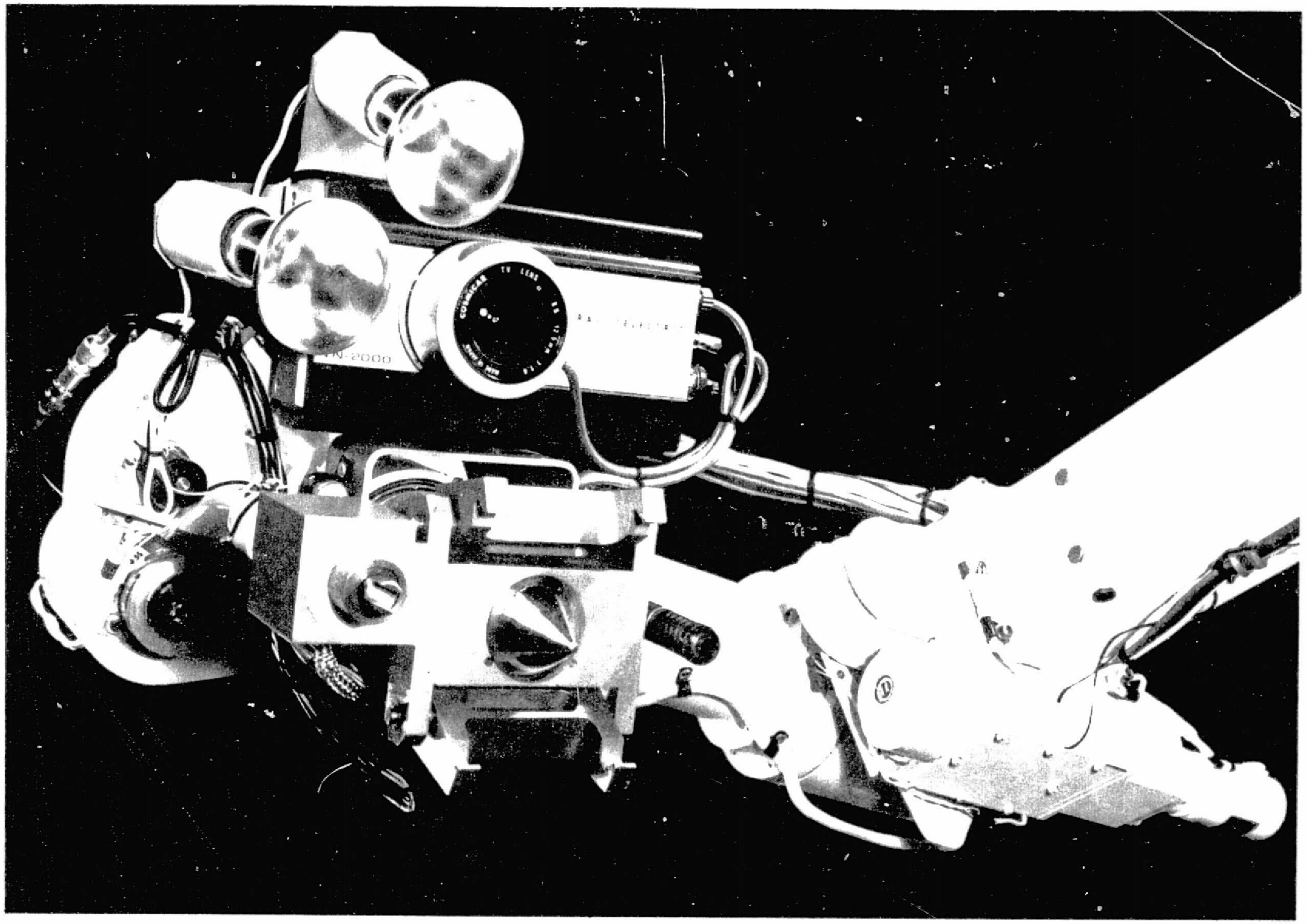

Figure 1-8 ETU End Effector and Wrist Drives

1.5.7 Engineering Test Unit Condition

The following conclusions and recommendations were developed as part of the review and evaluation of the condition of the Engineering Test Unit at MSFC:

1) The Engineering Test Unit is in very good electromechanical condition and no dismantling was necessary;
2) Recent ETU accuracy test data is similar to that taken when the unit was built;
3) Software modifications are needed for smoother operations and to obtain complete module trajectories.

1.5.8 Ground Demonstrations

The following conclusions and recommendations were developed during the ground demonstration analyses:

1) The control system software of the MSFC servicing demonstration facility should be upgraded;
2) MMS module exchange should be the first ground demonstration activity;
3) The exchange of other generic modules--AXAF or communications satellite--should be coordinated with the respective project offices and then demonstrated;
4) Refueling and resupply hardware should be developed and the processes demonstrated;
5) An automatic target recognition and error correction system should be developed and demonstrated;
6) The MSFC servicing demonstration facility should be made available for support of flight operations in terms of simulations, procedures development, training, and problem solving. The facility should also be made available as a laboratory development tool;
7) The first five ground demonstration activities can be accomplished by mid 1986 (Fig. 1-9).

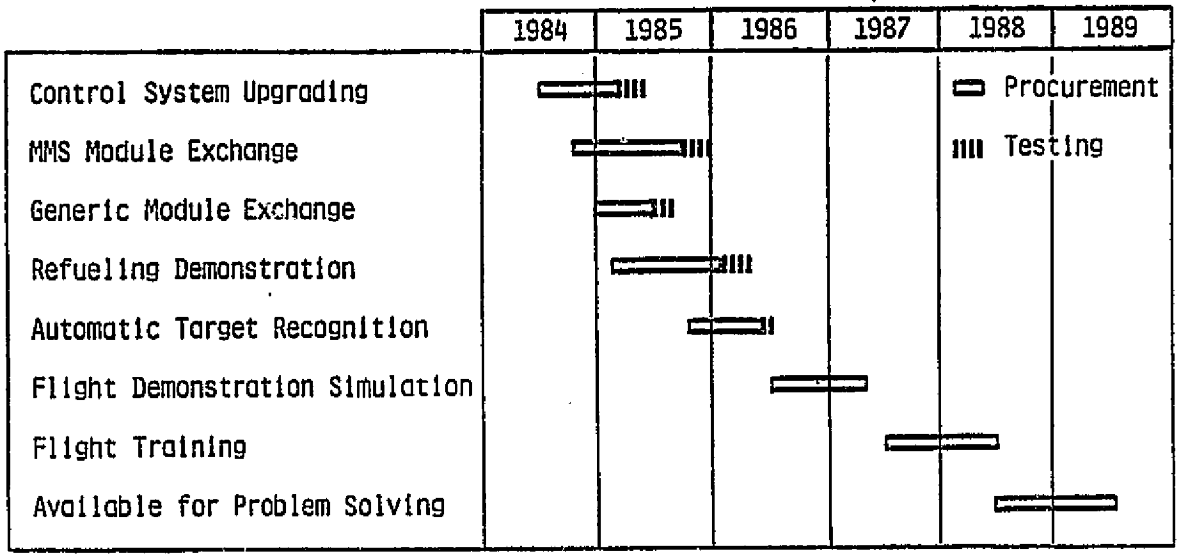

Figure 1-9 Ground Demonstrations Program Plan

1.5.9 Cargo-Bay Demonstrations

The following conclusions and recommendations were developed during the cargo-bay demonstration analyses:

1) A protoflight quality servicer mechanism should be built for use on the two cargo-bay demonstration flights;
2) The orbiter Remote Manipulator System docking arrangement should be used (Fig. 1-10);
3) The servicer should be exercised in all three control modes;
4) The servicer control station location should be further evaluated. It was selected to be on the ground for costing purposes;
5) The characteristics of the recommended servicer cargo-bay demonstration are:
   - Satellite mockup unstow and stow by RMS,
   - Supply of power, attitude control, and thermal control by orbiter,
   - Two-way communications links to ground through orbiter and Tracking and Data Relay Satellite System (TDRSS),
   - Servicer control station at OMV ground control station,
   - Docking rigidization by servicer docking probe,

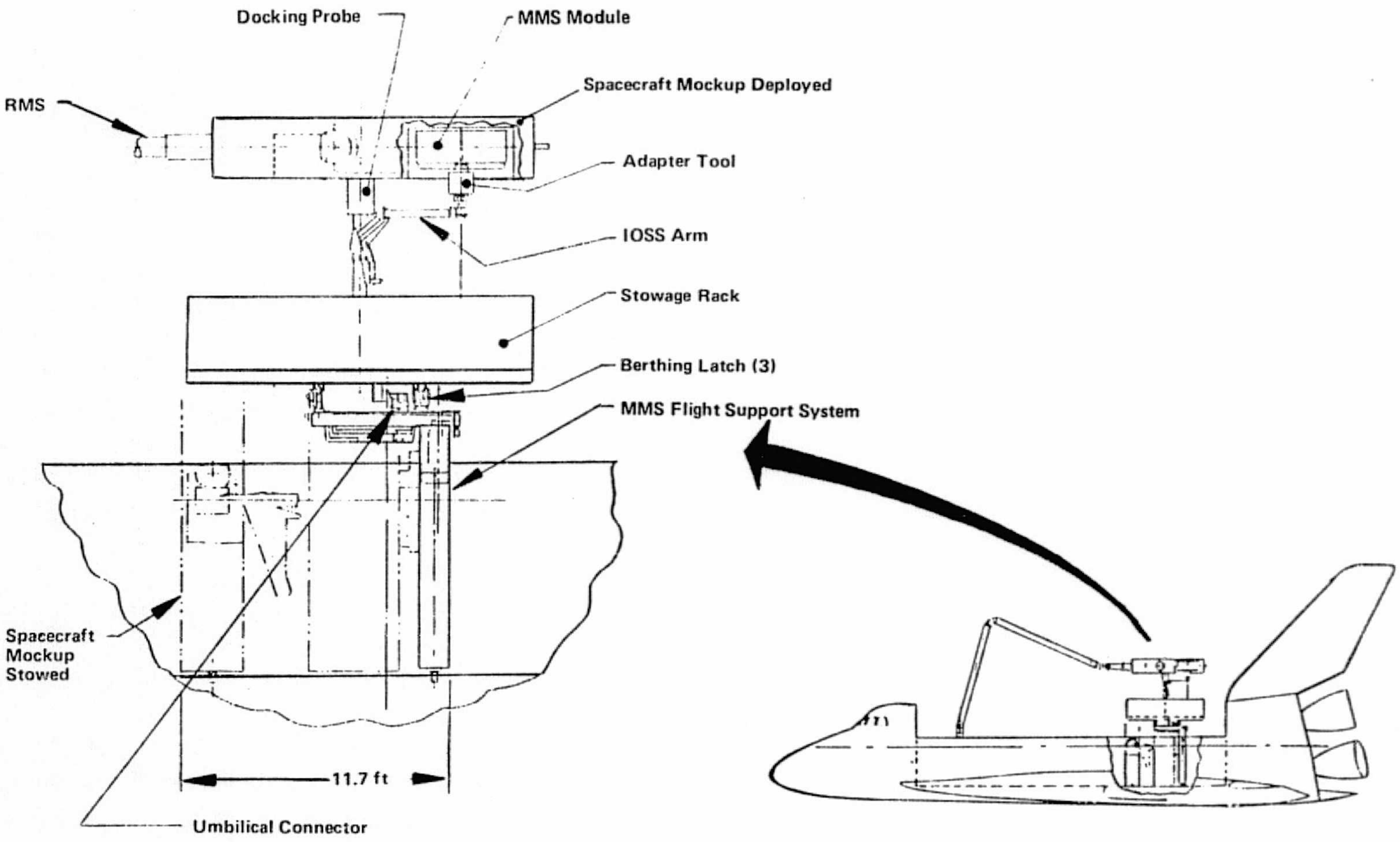

Figure 1-10 Use of RMS for Docking Arrangement

- Module exchange demonstration,
- Refueling demonstration,
- Servicing equipment performance demonstration,
- Control modes evaluation,
- Man-machine interaction evaluations,
- Compliance with orbiter system safety requirements,
- Deployment of stowage rack in orbiter by MMS flight support system,
- Use of representative servicing operational equipment,
- Operator training;

6) The hardware for the refueling demonstrations should be obtained from the ongoing Johnson Space Center refueling demonstration flight program;

7) The first cargo-bay demonstration flight can be completed by late 1988 (Fig. 1-11);

8) The recommended activities for the first test flight are:
   - A Multimission Modular Spacecraft type module using an MMS module servicing tool, incorporating an electrical connector, and mounted so that the module moves axially with one latch near the docking probe and one far away,
   - Battery module on a lightweight side interface mechanism using an electrical connector and with a near-radial module motion direction,
   - Hinged access door mounted so that the servicer end effector is attached in a radial direction;

9) The recommended activities for the second test flight are:
   - A multiple line propellant resupply probe with a fluid connector translation device and a hose management device mounted in a far-axial direction,
   - A propellant resupply module on a lightweight side interface mechanism using a propellant connection drive and mounted in a near-radial direction,
   - An access door treated as a module on a lightweight side interface mechanism and mounted in the near-axial position.

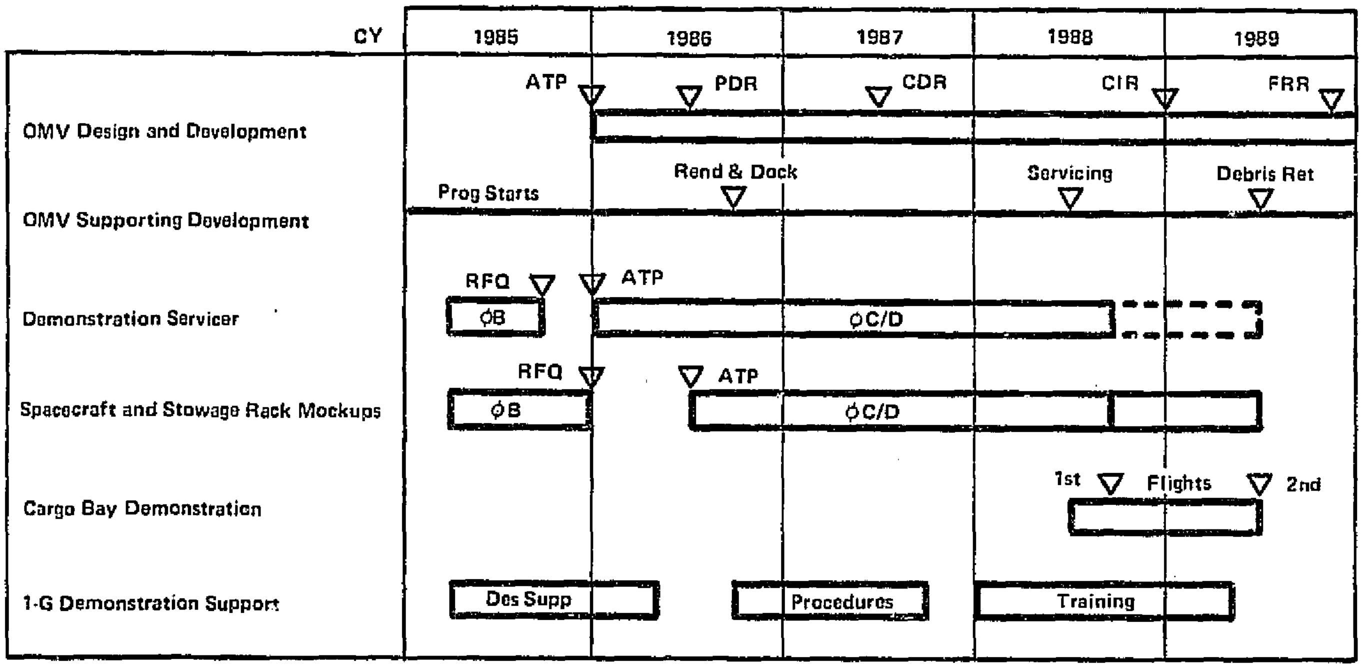

Figure 1-11 Servicer Cargo-Bay Demonstration Schedule

1.5.10 Free-Flight Verification

The following conclusions and recommendations were developed during the free-flight verification analysis:

1) A fully operational servicer system that has been qualified and documented should be built for use in the free-flight verification activity;
2) The Orbital Maneuvering Vehicle should be the servicer carrier vehicle;
3) The servicer control modes should be selected based on the cargo-bay demonstration results;
4) A spacecraft bus, such as the SPAS-01, should be rented rather than a new spacecraft being built for this one-time application;
5) The characteristics of the recommended servicer free-flight verification are:
   - Serviceable satellite mockup supported by a rented spacecraft bus,
   - Supply of power, attitude control, thermal protection and control of the servicer from the OMV,

- Use of OMV for rendezvous and docking of servicer to the serviceable spacecraft mockup,
- Two way communication links to ground through TDRSS,
- Servicer control station at OMV ground control station,
- Docking rigidization by servicer docking probe,
- Deployment of stowed servicer mechanism and docking probe,
- MMS module exchange demonstration,
- Refueling demonstration,
- Servicing equipment performance verification,
- Control mode verification,
- Operator training;

6) The recommended flight verification activities are:
   - Exchange of MMS module,
   - Exchange of representative modules,
   - Propellant transfer;

7) The free-flight verification of an operational servicer can be completed by mid 1992 (Fig. 1-12).

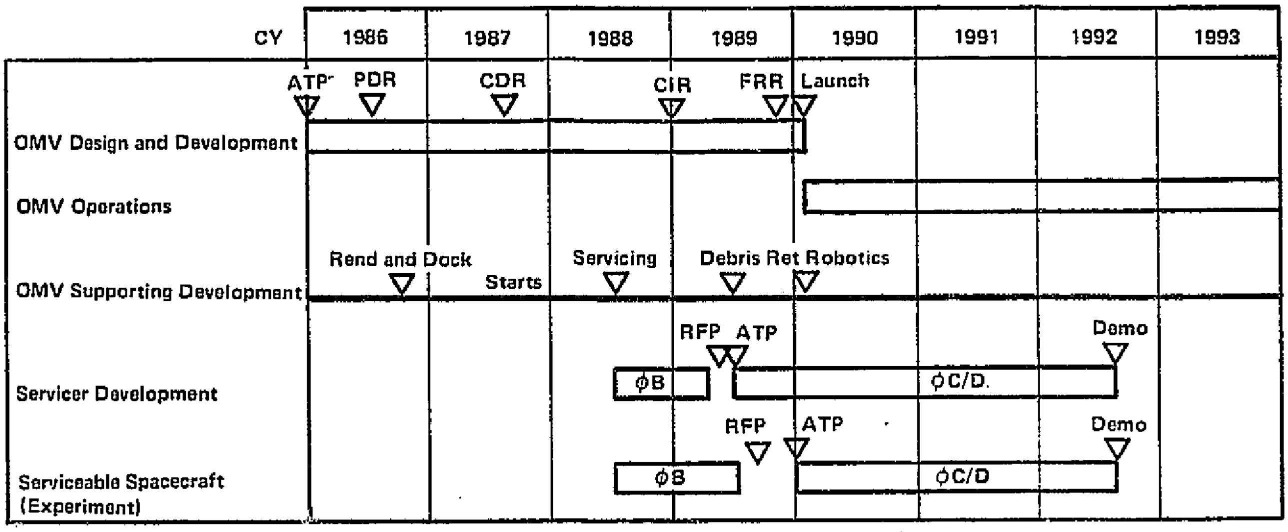

Figure 1-12 Free-Flight Demonstration Schedule

1.6 SUGGESTED ADDITIONAL EFFORT

A review of the study efforts and conclusions identified a number of areas that merit consideration for additional effort. In addition to the items listed below, it is assumed that the TDRSS program and the OMV program including a docking system, payload rigidization system, and ground control station will continue.

1.6.1 Servicing Tasks

The following additional efforts are related to servicing tasks and in particular to the Multimission Modular Spacecraft, refueling demonstrations, and representative satellite modules:

1) The module servicing tool and the ETU end effector should be adapted to work together for the exchange of MMS modules;
2) Lightweight MMS module mockups with standard MMS attachment fixtures and connectors should be used for ground demonstration;
3) The refueling interface should be standardized;
4) The refueling demonstration equipment should be based on the NASA-JSC standardization effort;
5) Thermal cover removal/replace mechanisms and sensors for fastener and attach interface status need to be developed;
6) A small, light interface mechanism or a tool adapter to remove conventional captive fasteners should be developed.

1.6.2 Servicing Mechanism

The following additional efforts are related to the servicing mechanism and particularly to the end effector, servicer mechanism selection, and Engineering Test Unit condition:

1) The interface between the servicer end effector and the interface mechanism, tools, and adapters should be standardized;
2) An electrical disconnect should be added to the ETU end effector;
3) Special adapters should be developed as required for other types of modules or servicing tasks;
4) Specific detail recommendations for upgrading the ETU are provided in Sections 4.4 and 4.5.

1.6.3 Demonstrations

The following additional efforts are related to the ground and cargo-bay demonstrations or to the free-flight verification:

1) The control system software of the MSFC servicing demonstration facility should be upgraded;

2) Refueling and resupply hardware should be developed and the process demonstrated;

3) An automatic target recognition and error correction system should be developed and demonstrated;

4) Additional development areas include:

   - Special refueling disconnects for cryogenics or high pressures, and self aligning conical electrical connectors,
   - Development of in-line fluid couplings for replacement of tanks and other propulsion system components,
   - Demonstration of other servicing tasks specific to space station operations;

5) Demonstration of the mating of the servicer stowage rack to the OMV should be a part of the space station technology development missions.

2.0 INTRODUCTION AND BACKGROUND

The study and planning activity that this final report documents was performed because of many prior studies that indicated the strong economic benefits of on-orbit servicing. It has been clearly shown that orbital maintenance functions can be supported by the Space Transportation System (STS) to effect large reductions in the cost of spacecraft programs. This was found to be true both in geosynchronous and low earth orbits. These economic benefits were augmented by significant operational benefits, the totality of which implied that the development of an on-orbit servicing capability should be undertaken by the NASA. Orbital servicing has a number of applications. The servicer and the Orbital Maneuvering Vehicle (OMV) can be carried to geosynchronous earth orbit (GEO) on an Orbital Transfer Vehicle (OTV). Communications satellites are typical geosynchronous spacecraft that can realize cost benefits from servicing. In low earth orbit the OMV can be used as the carrier vehicle for the servicer system. Where contamination or thruster impingement effects are a concern, the cold gas kit for the OMV could be used. For spacecraft in different orbits (altitude or inclination) the larger propulsive capability versions of the OMV are appropriate. The servicer system can also be deployed in the orbiter cargo bay and the failed spacecraft docked to it using the Remote Manipulator System (RMS).

To minimize servicer system development costs, it was recommended that a single servicer system having the capability to accommodate both low and high earth orbit applications should be evolved. This requirement has been satisfied effectively by the servicer mechanization (Fig. 2-1) conceptualized during the Integrated Orbital Servicing System (IOSS) studies. The single design is compatible with maintenance of most spacecraft of the STS era. Adapters are used to accommodate support structure differences across the applications. An effective interface between the spacecraft and the servicer was defined and breadboarded. The interface mechanism provides a logical and cost effective method of incorporating orbital replaceable units (ORU) for module exchange in all spacecraft.

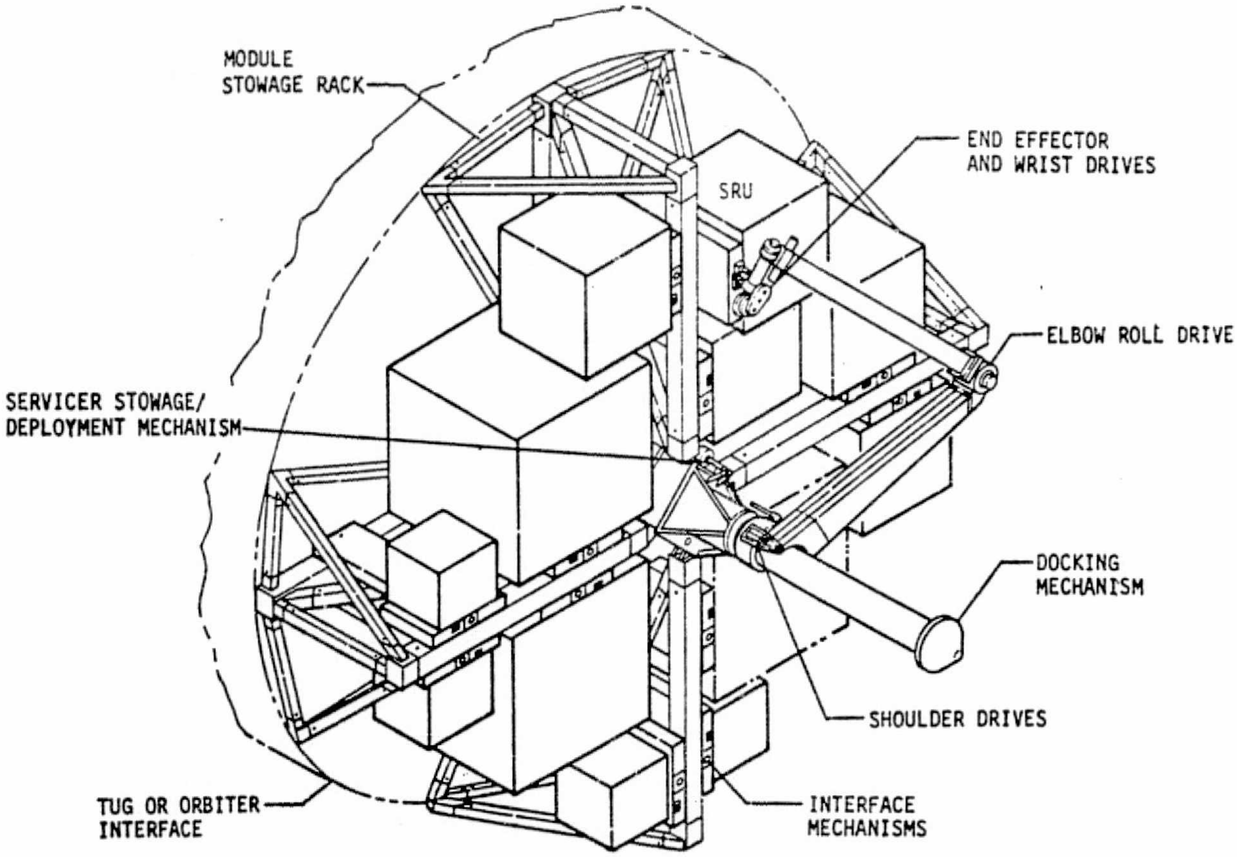

Figure 2-1 On-orbit Servicer Configuration

Considerable interest in spacecraft maintenance was expressed by both the Department of Defense and the commercial sector, however, the general tenor of their support was that a demonstration of orbital maintenance must be conducted prior to any commitment on their part. A flight demonstration of the all-up maintenance capability is also a NASA requirement prior to wholesale commitment to the concept. However, a reduced capability test that exercises the basic concept and exchanger capability can and should be demonstrated prior to the time when the full capability will exist. With this background material in hand, and with renewed interest by the space flight community, it was appropriate to perform a study that would define a path to culminate in the demonstration of the servicing capability. The objective of this study was to provide a single unified development program for both servicing implementors and users to utilize to guide their future development and operational plans for this important technology.

2.1 STUDY OBJECTIVES

The objectives of this Spacecraft Servicing Demonstration Plan study are to identify all major elements and characteristics of an on-orbit servicing development program and to integrate them into a coherent set of demonstrations. The goal of the development program is to produce and verify an on-orbit servicing capability so as to increase user acceptance of on-orbit repair by module exchange and refueling/resupply. A major emphasis of the study was to screen the elements and characteristics of the development program to identify activities to be demonstrated in the orbiter cargo bay that will convince the user and implementation community that orbital servicing is an available commodity. A ground demonstration plan is also envisioned that will provide confidence for the development and operation of the on-orbit system. These objectives are summarized in Table 2-1. It was later decided, as a result of the study, to add a free-flyer (Orbital Maneuvering Vehicle) demonstration to the plan as a way of verifying the capabilities of an operational servicer.

Table 2-1 - Study Objectives

To identify and integrate the major characteristics of a Spacecraft Servicing Demonstration Plan
Major plan elements
Ground demonstrations
Orbiter cargo-bay demonstrations
Free-flight verification

This study was performed to provide implementors and users with a single development approach that will culminate in orbital servicing operations. The study is necessary at this time because only by providing a planned development program will both development and user support be focused on the servicing issue. Current planning for the OMV is such that servicer development must be started soon if a servicing capability is to exist shortly after the OMV reaches an operational status. Verification of a servicing capability with the OMV will result in a well proven system being available for use with the space station. Many prior and current studies have addressed individual elements of servicing. Many tools and support hardware elements have been defined that will aid a future servicing program. These efforts, however, have not culminated in a general move on the part of the user community to incorporate serviceability into their spacecraft designs. It is only through the implementation of a development program that produces a demonstrated on-orbit servicing capability that the benefits of this program will be realized in future spacecraft operations. The servicing demonstration plan described in this report was prepared to satisfy this need.

2.2 BACKGROUND

Servicing development activities were initiated in the early 1970s and continue through the present time. Table 2-2 provides a list of related efforts that have been performed in this field. Studies and development work have been performed by many government agencies and contractors. Prior study results have concluded that on-orbit servicing is a more cost-effective approach than ground refurbishment of satellites. Recommendations included spacecraft designed for servicing and modular exchange concepts as the most cost-effective method of implementing servicing.

The majority of the studies listed in Table 2-2 were performed prior to or during 1978. There were only a few studies performed during the 1979-1982 time period as NASA's efforts were directed towards getting the STS into an operational status. The Multimission Modular Spacecraft (MMS) is an important operating spacecraft designed for on-orbit servicing and the Space Telescope (ST) has been designed for extravehicular security (EVA) servicing. While the MMS was initially designed for remotely manned module exchange, the only MMS repair mission (Solar Maximum Mission) was accomplished by astronauts on EVA.

The success of the SMM repair mission has increased interest in manned repair of spacecraft. However, the limitations of astronauts on EVA are such that there is clearly a place for remote module exchange and refueling/resupply in the space program. This aspect is supported by the recent activities of the US Air Force in spacecraft servicing and the design of spacecraft for on-orbit repair.

Table 2-2 Servicer Related Efforts

Unmanned Orbital Platform - MSFC/RI	
Payload Supporting Studies for Tug Assessment-MSFC Inhouse	1973
In-Space Servicing of a DSP Satellite - SAMSO/TRW	Mar 1974
Payload Utilization of Tug - MSFC/MDAC, GE, and Fairchild	May 1974
Operations Analysis- NASA/Aerospace	Jul 1974
Servicing the DSCS-II with the STS-SAMSO/TRW	Mar 1975
Multimission Support Equipment (Launch Site) MSFC/MMA	Jun 1975
Orbital Assembly and Maintenance - JSC/MMA	Aug 1975
Integrated Orbital Servicing and Payloads Study - MSFC/MMA (COMSAT)	Sep 1975
Earth Observatory Satellite System - GSFC/Inhouse and Contracted	1976
Study to Evaluate the effect of EVA on Payload Systems - Ames/RI	1976
Earth Orbital Teleoperator Systems Concepts and Analysis - MSFC/MMA	Apr 1976
Design, Development, Fabrication, and Testing of a Fluid Disconnect for Space Operation Systems - MSFC/Fairchild Stratos	Sep 1976
Analytical Study of Electrical Disconnect Systems for Use on Manned and Unmanned Missions - MSFC/MMA	Jan 1977
Orbital Construction Support Equipment - JSC/MMA	Jun 1977
Proto-Flight Manipulator Arm Assembly - MSFC/MMA	Apr 1977
Integrated Orbital Servicing System Study Follow-on MSFC/MMA and TRW	Apr 1978
Multimission Modular Spacecraft Inorbit Refueling Study - GSFC/RI	1980
Reuse/Resupply Component Study - AFRPL/MMA	1982
Satellite Services System Analysis Study GAC/LMSC	1980-1982

The IOSS study initially used the 1973 NASA mission model as a basis for establishing cost benefits. The model included the 47 NASA satellite programs for which maintenance is applicable. Applicability of maintenance was based on: spacecraft fleet size on orbit, program lifetime, and need for equipment replacement.

If a satellite program was short, or the spacecraft value was low, then maintenance was not attempted. Cost comparisons were made between:

1) Expendable spacecraft;

2) Return to the ground for refurbishment;
3) Return to the orbiter for refurbishment;
4) Module exchange in the operational orbit.

Generally, module exchange in the operational orbit was most cost effective. If spacecraft are cheap, then it is cost effective to expend them. The costs of returning a spacecraft to the ground and relaunching were high enough to rule out ground refurbishment. Orbit phasing effects and the launch costs related to propellant usage in bringing spacecraft, especially geosynchronous spacecraft, back to the orbiter ruled out maintenance at the orbiter. However, the orbits of some spacecraft make this an acceptable approach. There were significant cost savings from repair by module exchange in the spacecraft's operational orbit. These savings are larger than the cost of servicer system development. The same results were obtained using much smaller mission models. These study results are applicable to current-day situations. Some specific satellite programs have changed since these study results were generated, however, the conclusions on cost effectiveness are as applicable to today's satellite programs as they were to the program projected in 1973.

Previous studies also evaluated the seven alternative servicing system approaches shown in Figure 2-2. Approach e), the pivoting arm system, was initially selected for further development.

The selection of a servicer mechanism configuration was combined with an analysis of serviceable spacecraft designs that had been prepared prior to 1975. Of particular interest was the location of the replaceable modules with respect to the on-orbit servicer docking axis. It was found that most modules could be removed in a direction parallel to the docking axis and this was called axial module removal. There was another popular configuration where the modules were arranged in a donut fashion (approach d) of Figure 2-2) and could be removed in a direction perpendicular to the docking axis. This direction was

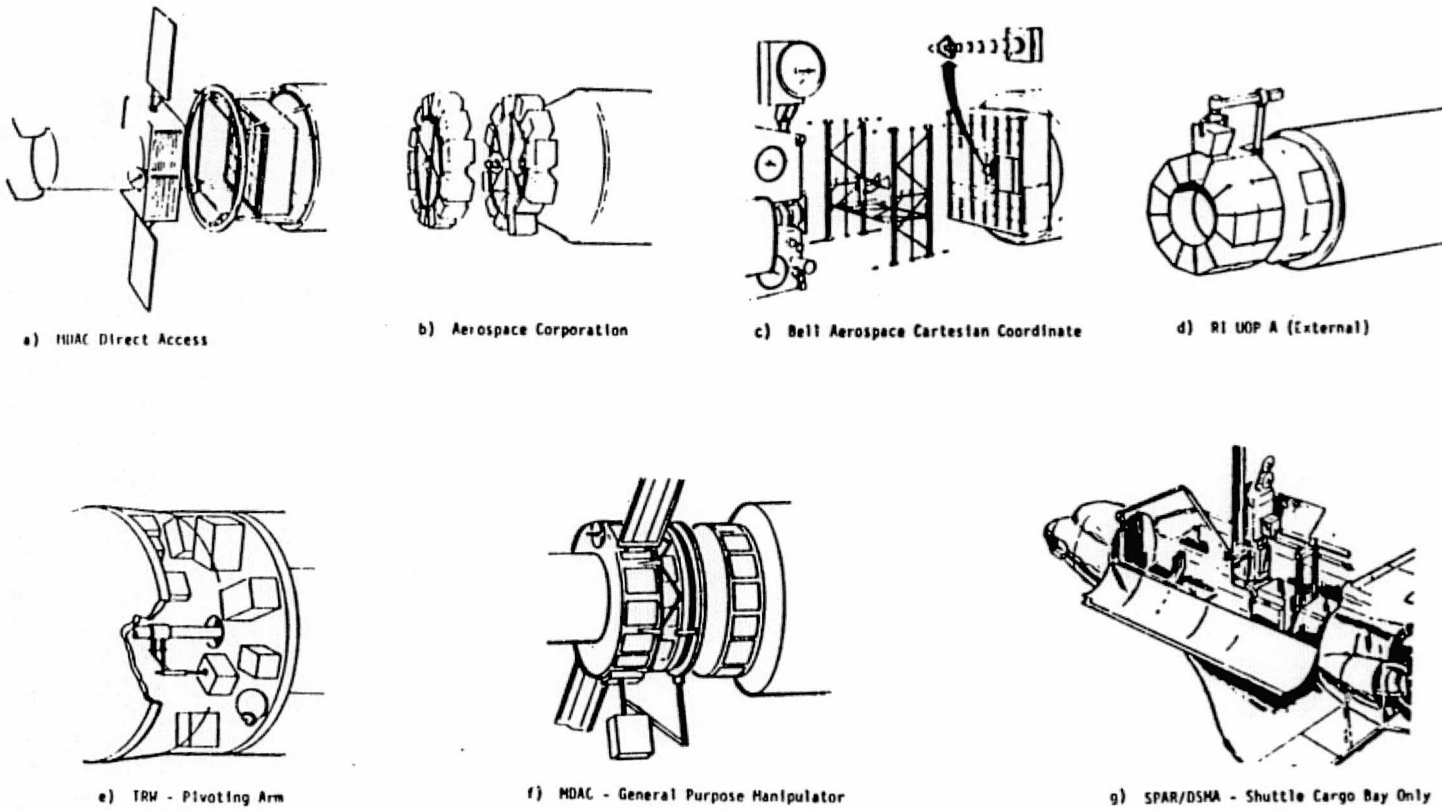

Figure 2-2 Alternative On-Orbit Servicer Concepts

called radial. In some cases, two tiers or layers of modules were arranged for radial module removal. It was concluded that most repairable spacecraft could be configured so their modules could be exchanged axially or radially in one or two tiers. In some cases it might be necessary to provide more than one docking port.

A careful examination of approach e) of Figure 2-2, the pivoting arm, shows that it can readily accommodate axial module removal. However, it was necessary to extend its size and number of joints to accommodate one tier of radial module exchange. The resulting configuration is shown in Figure 2-1. The spare module stowage rack configuration was addressed along with the repairable spacecraft configurations. It was decided that an axial configuration for the stowage rack would be best. The servicer configuration of Figure 2-1 can be extended to two radial tiers if its arm and wrist lengths are increased. However, it was decided to use the one-tier capability until the need for the second tier was clearly demonstrated.

The value of demonstrations in furthering on-orbit servicing development was recognized in the decision to build a 1-g version of the Integrated Orbital Servicing System of Figure 2-1. The result is the Engineering Test unit (ETU) shown in the photograph of Figure 2-3. This unit was built and delivered to MSFC in 1978. It has been used for over 250 demonstrations during the intervening six years. The ETU has shorter segment lengths than the IOSS as it was designed initially for axial module exchange only. The later addition of a sixth degree of freedom extended the ETU's capability to radial module removal, albeit at a radius less than that of the orbiter cargo bay.

To date, satellite systems in general have not been designed and built with the capability of changeout of subsystem or component modules. The only satellite which is currently in use that has a module exchange capability is the Goddard Space Flight Center's Multimission Modular Spacecraft. This satellite is in operation in several programs and is

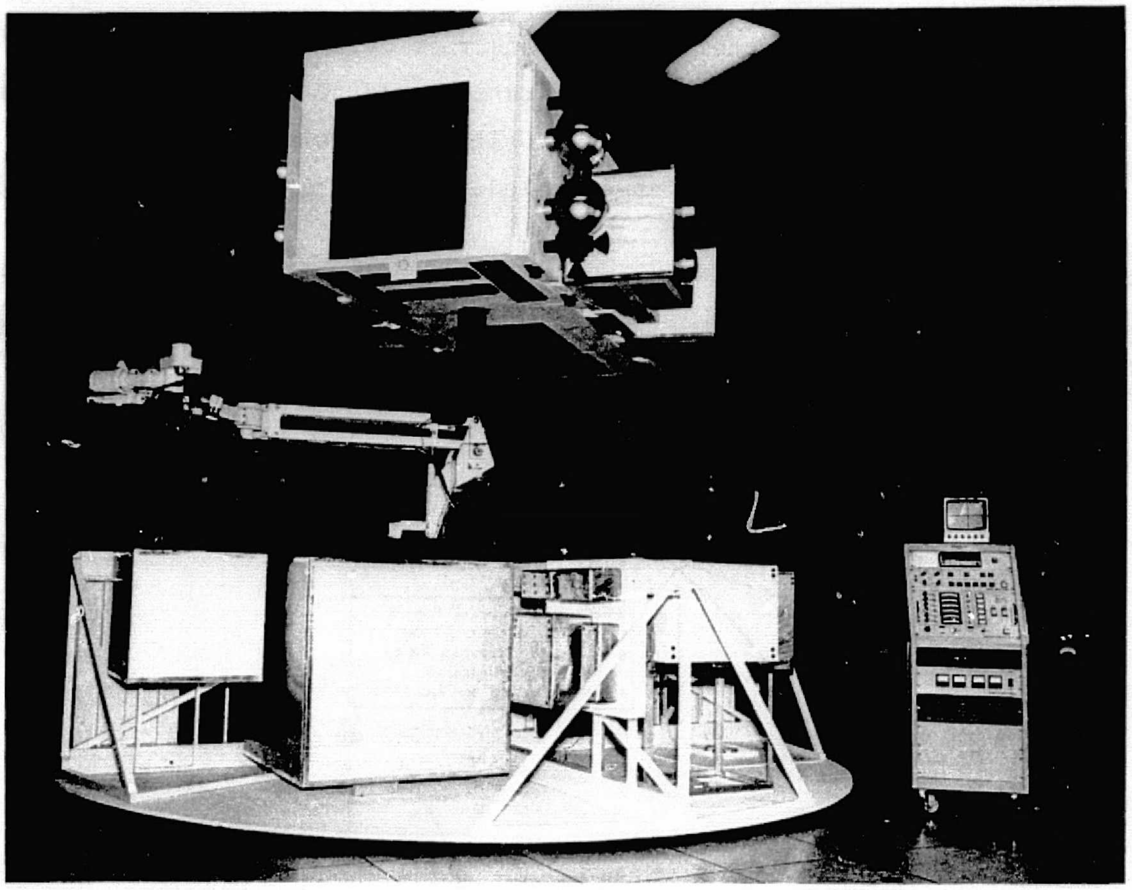

Figure 2-3 Engineering Test Unit

projected for continued use throughout the remainder of this century. The Marshall Space Flight Center's Space Telescope has been designed for on-orbit repair by an astronaut on EVA and is expected to fly soon. The US Air Force has also shown interest in the design of serviceable spacecraft, although the particulars are not known.

Several demonstrations and investigations of on-orbit refueling capability are currently being planned. These efforts will include definition and demonstration of connect/disconnect devices in support of the transfer of fluids. Electrical umbilicals and connectors have been developed in conjunction with the MMS subsystem modules as well as on other programs.

The various activities conducted as part of the IOSS contract work from 1974 to 1978 are shown in Figure 2-4. The first IOSS activity was basically a study to review past work and to do sufficient comparative cost analysis that the form of spacecraft maintenance having the greatest potential value to cost ratio could be identified. The best was clearly on-orbit maintenance using module exchange remote from the orbiter where module exchange includes refueling and resupply.

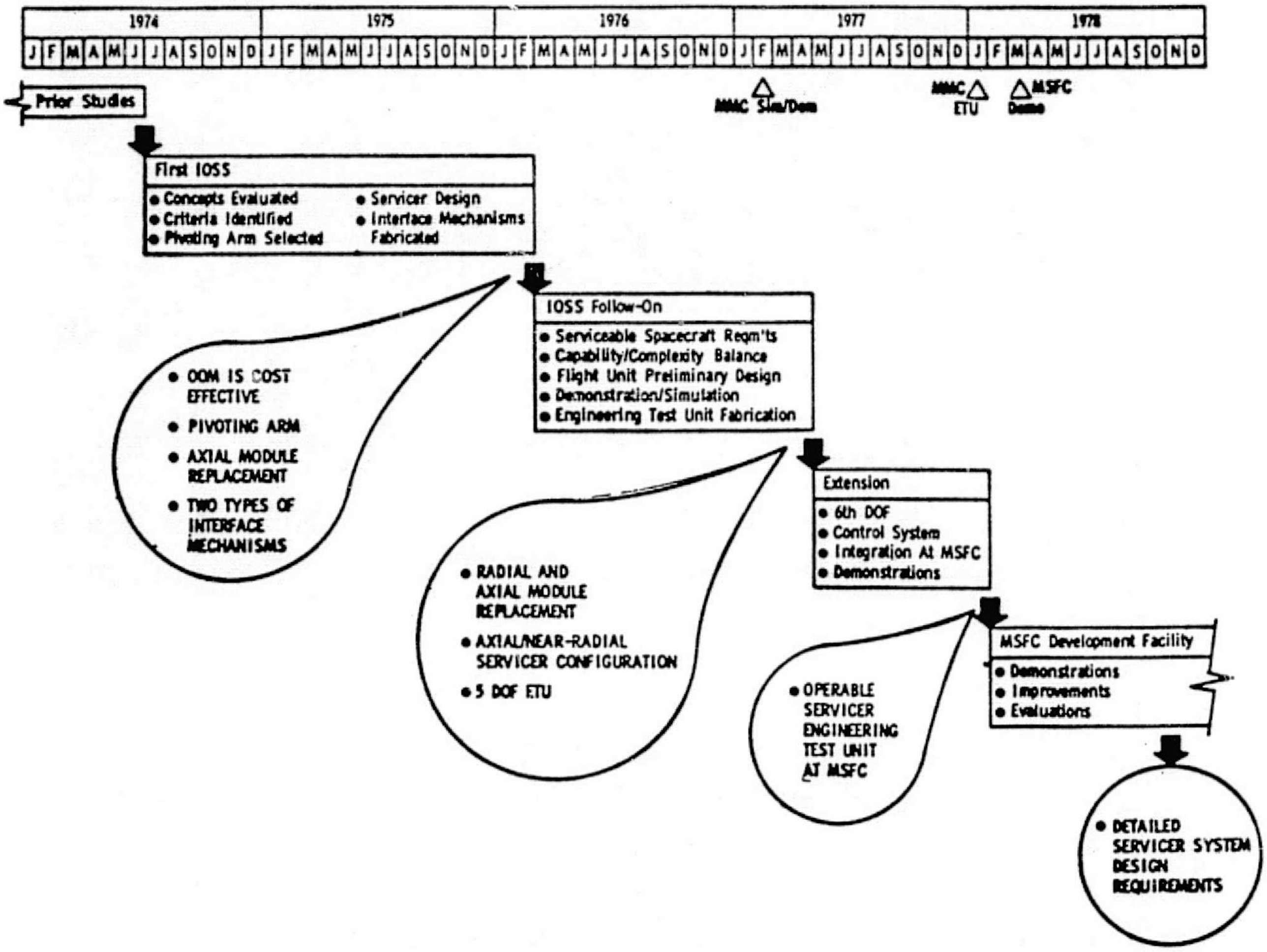

Figure 2-4 Servicer System Design Evolution

The second phase was the IOSS follow-on that examined the serviceable spacecraft requirements and the best servicer configuration (discussed in support of Fig. 2-2). The work resulted in the design of the servicer configuration shown in Figure 2-1 and the beginning of fabrication of the ETU. Several representative serviceable spacecraft were also conceptualized. The extension phase of the work in late 1977 resulted in adding a sixth degree of freedom to the ETU, development of three alternative control systems and demonstration of the ETU at MSFC. This was followed by a series of demonstrations at MSFC by MSFC personnel. The recommendations for future activities in 1978 was similar to that being recommended now. There would be a series of in-flight experiments supported by ground demonstrations using the ETU at MSFC. This activity would lead to an operational servicing system capability that complemented the basic Space Transportation System capability. However, the plan elements associated with in-flight experiments and servicing flight verification have not yet been implemented.

The major elements of the orbital servicing background are listed and summarized in Table 2-3. This background shows overwhelming benefits resulting from an on-orbit servicing capability. An extensive set of servicing system hardware and components have been defined. The next logical step in the progression of servicing development is to bring these elements together into an integrated servicing development program. Incorporation of servicing capability into future spacecraft and vehicles can best be promoted by initiating a flight and ground demonstration program. This document defines and proposes such a development plan.

Table 2-3 Major Results of Prior Orbital Servicing Studies

Cost benefits of unmanned on-orbit satellite servicing are high
Development activities were initiated in the early 1970s
A variety of servicing system concepts have been defined and evaluated
Module exchange is a major servicing activity
The Integrated Orbital Servicing System study identified a promising servicer mechanism configuration
The plan included the build of a servicer Engineering Test Unit

2.3 STUDY APPROACH

The major study tasks and their interrelationship are shown in Figure 2-5. The tasks consist of preparing both a flight and ground demonstration plan. The flight demonstration plan is the key element in that it will provide the basis for the satellite servicing capability that will exist in the future. This capability will be utilized by future spacecraft designers in establishing servicing concepts for their space vehicles. Both the ground and flight demonstration plan tasks include a review and selection of servicer designs, module changeout elements to be demonstrated, and types of servicing to be demonstrated. This latter item includes hardware exchange, fluid transfer service, and connection of electrical umbilicals. The ground demonstration plan task is directly keyed to the flight demonstration task. The objective of the ground demonstrations is to reduce risk and verify the approach for the flight demonstration planning.

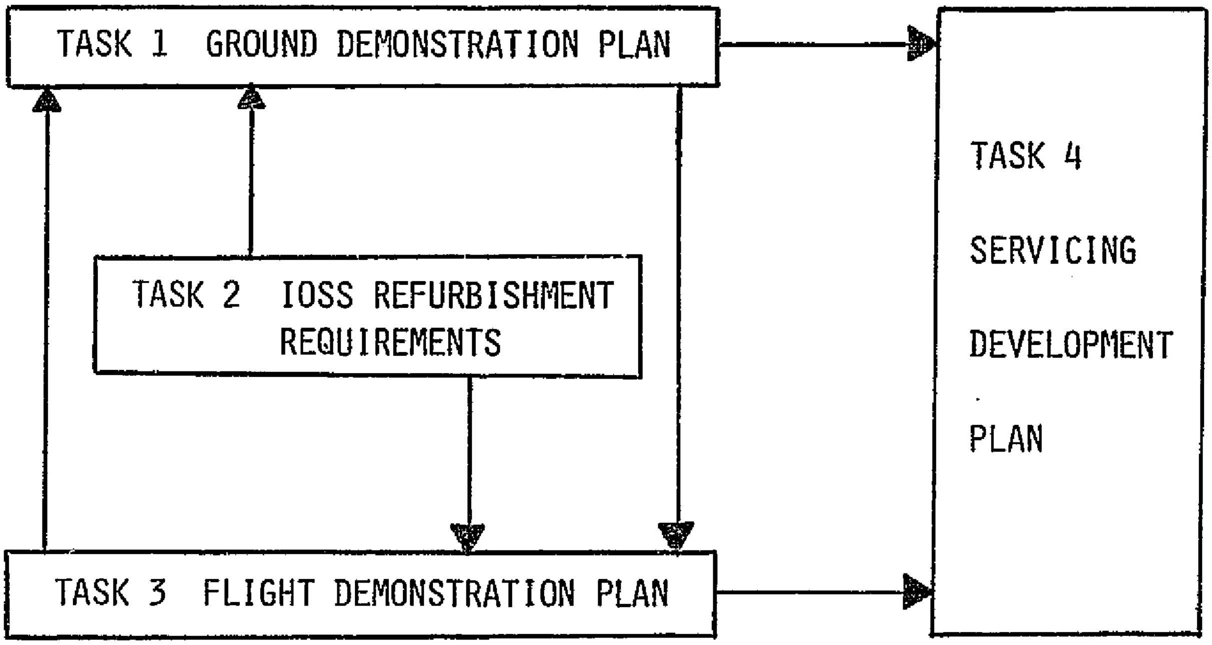

Figure 2-5 Task Logic Flow Diagram

An additional task associated with this study was to review the current status of the Engineering Test Unit from the IOSS studies. The review established, to a degree, the ETU's capability to be utilized in the ground demonstration planning.

With the major elements of both the ground and flight demonstration plan established, a development plan with overall cost estimates was prepared as part of the study.

3.0 GROUND DEMONSTRATION PLAN

The objective of this phase of the study was to define the type of servicer and the servicer hardware design for the ground demonstration operations. Analyses were performed to determine the type and the size of the modules to be used, as well as the refueling/resupply hardware, the type of end effector and the special adapters to be incorporated in the ground demonstration system. An evaluation of existing designs of servicer mechanisms was performed to select the type of servicer to be used. The Integrated Orbital Servicing System (IOSS), the Proto-Flight Manipulator Arm (PFMA), the Remote Orbital Servicing System (ROSS), the Remote Manipulator System (RMS) and other servicer mechanisms and systems were analyzed and traded off against the requirements for the ground demonstrations. This analysis was performed in parallel with the servicer system selection for the flight demonstrations, described in Section 5.0, so that commonality of design, hardware and procurement could be achieved. The results of the inspection of the Engineering Test Unit (ETU) of the IOSS, shown in Section 4.0, were considered in the final selection and recommendation of the type of servicer to be used for ground demonstrations of satellite servicing.

A modified IOSS servicer was recommended, capable of demonstrating MMS type module changeout and refueling in addition to the existing cube modules with side attachment interface mechanism. The flexibility of the servicing system was enhanced by using special adapters and modular refueling units. Other servicing task demonstrations were proposed, such as thermal cover removal and changeout of component level modules, communications satellite module, OMV module and AXAF module, to further demonstrate the flexibility of the system. Cost estimates were given for each of these demonstrations. A ground demonstration plan was developed, including a recommended schedule for the design and development of the necessary hardware.

3.1 SUPPORTING ANALYSES

Before the servicer mechanism was selected, a series of supporting analyses were performed, as listed in Table 3.1-1. The ground and flight demonstration system requirements were identified for each of these elements. Various candidate solutions were traded off in each supporting analysis to select the elements that best meet the requirements for the ground demonstrations and are compatible with, or easily adaptable to, the flight demonstration requirements.

Table 3.1-1 Supporting Analyses

Multimission Modular Spacecraft type module exchange analysis.
Servicing interface selection for refueling/resupply and electrical connections
Representative satellite module selection
End effector selection

The identification of these system elements helped define the requirements of the servicer mechanism and assisted in its final selection.

3.1.1 Multimission Modular Spacecraft Module Exchange Analysis

The Space Telescope (ST) and the Multimission Modular Spacecraft (MMS) are the two major spacecraft systems designed for on-orbit maintenance and repair using module exchange by EVA. Of the two systems, the MMS is more amenable to adaptation for remote on-orbit maintenance using a servicer. Several MMS type satellites are presently operational and many more are projected to be used in the future. Satellites like Solar Maximum Mission (Figures 3.1.1-1 and 3.1.1-2), Landsat-D (Figure 3.1.1-3), Leasecraft and some defense systems utilize the MMS concept. In addition, other spacecraft concepts, presently being developed like the Advanced X-Ray Astrophysics Facility (AXAF) may incorporate MMS type modules for on-orbit servicing capability. One of the best ways to advance the satellite servicing technology, using module exchange techniques, is to demonstrate a MMS module exchange.

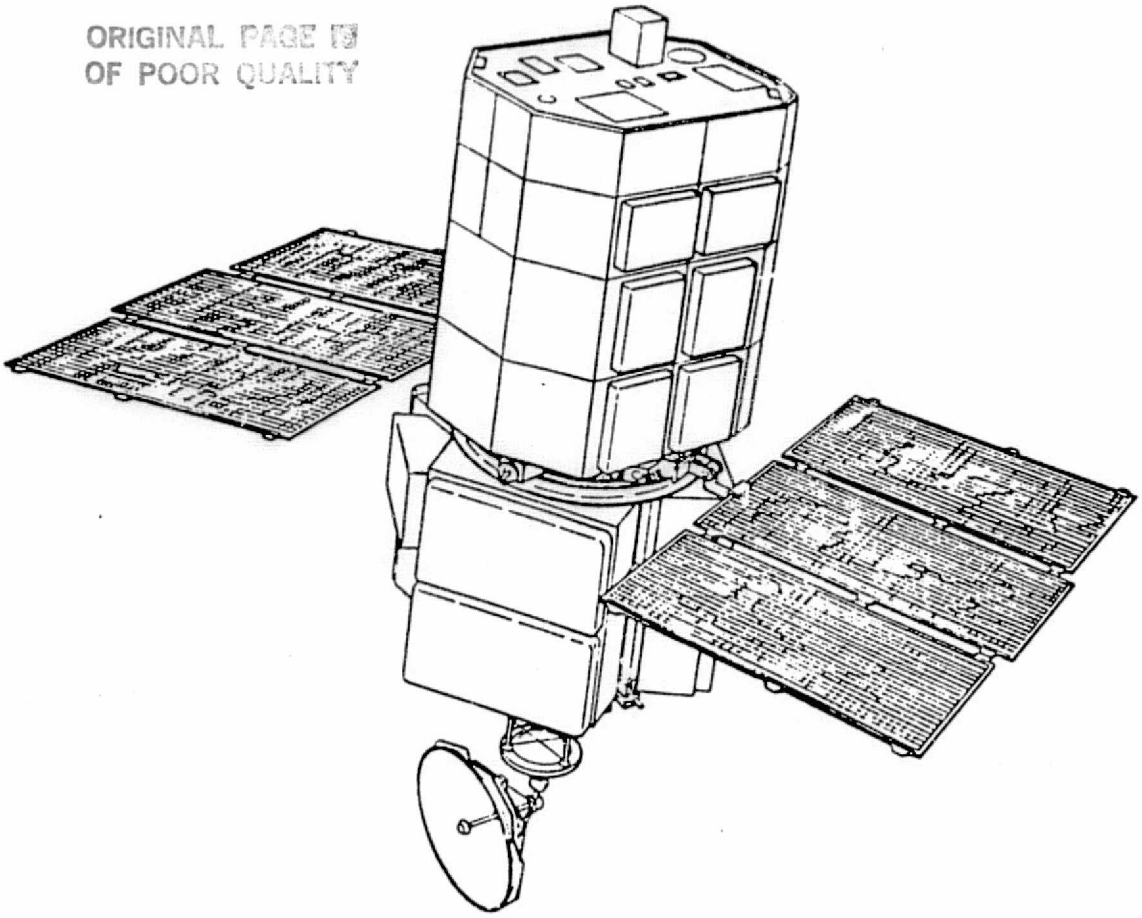

Figure 3.1.1-1 Solar Maximum Mission Version of MMS

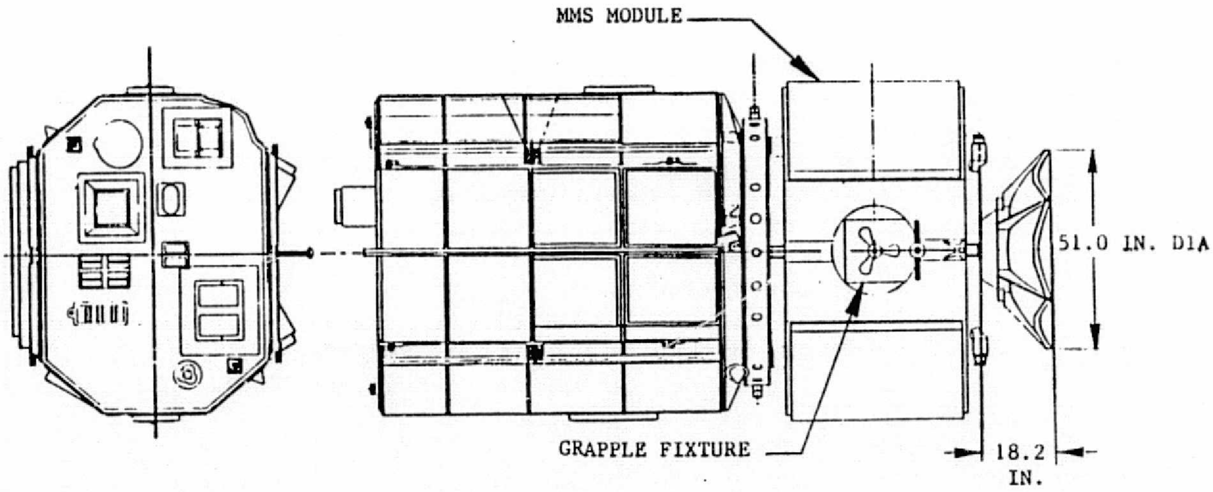

Figure 3.1.1-2 Solar Maximum Mission Return Configuration

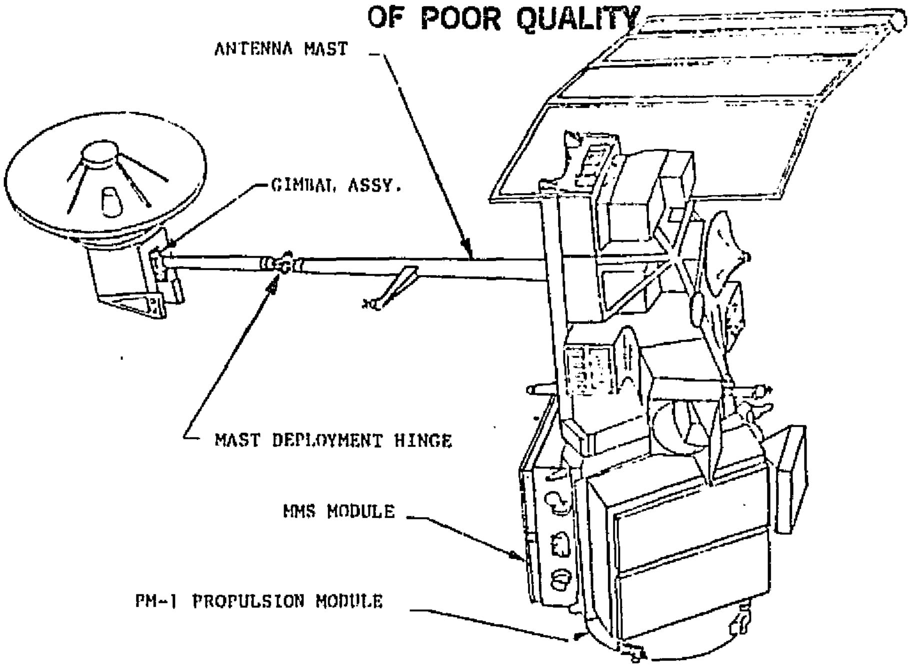

Figure 3.1.1-3 Landsat-D Spacecraft, Orbital Configuration

The MMS is a fully developed, operational system. Therefore, MMS design changes to accommodate servicer existing interfaces or other servicer requirements affect existing hardware and tooling and their implementation is expensive. This cost element was considered in defining a servicer system capable of exchanging MMS modules.

The servicing methods listed in Table 3.1.1-1 were considered for the MMS module exchange study.

Table 3.1.1-1 MMS Module Servicing Method Alternatives

Axial Docking Methods
1) Modified servicer end effector and specialized adapter tool
2) Use of existing side interface mechanism
3) Use of alternative interface mechanisms
    a) Single power takeoff
    b) Dual power takeoff
    c) Latches directly actuated with electric motors
4) Use of one latch mechanism in back of modified MMS module
5) Use of one active latch at bottom of modified MMS module and a passive hook-up point at the top
Lateral Docking Methods
6) Use of an offset docking probe adapter and tool adapter
7) Use of straight docking probe adapter, tool adapter and modified stowage rack

As a conclusion of this analysis, method 7) using lateral docking with a straight docking probe adapter, tool adapter and modified servicer stowage rack has the least impact on the spacecraft and servicer and is recommended for on-orbit exchange of MMS modules.

The basic MMS spacecraft (Figure 3.1.1-4) consists of three standard spacecraft subsystem modules and a mechanical structure which supports the spacecraft subsystem modules. The structure also provides the support for the instrument (payload) module, which is not part of the MMS. The standard spacecraft subsystem modules are a communications and data handling (C&DH) module, an attitude control subsystem (ACS), and a modular power subsystem (MPS). The instrument module, which includes the payload instruments and other mission unique equipment (such as solar arrays, high-gain antennas, etc.), attaches to a transition adapter ring on the forward end of the MMS. A propulsion module (PM) or a high gain antenna may be added to the aft end of the MMS as a mission option. A signal conditioning and control unit (SC&CU) and the electrical interconnecting harness complete the basic MMS.

The mechanical system of the basic MMS consists of the module support structure (MSS), the transition adapter (TA), and the structural enclosures for the three standard spacecraft modules (see Figures 3.1.1-4 and -5).

Satellites employing the MMS have been launched into low earth orbits by the Delta conventional launch vehicle, such as the 2910 and 3910, and the Shuttle of the Space Transportation System (STS).

In certain cases, the satellite will be designed to be capable of being recaptured by the orbiter, for on-orbit servicing and resupply, or for return to the ground.

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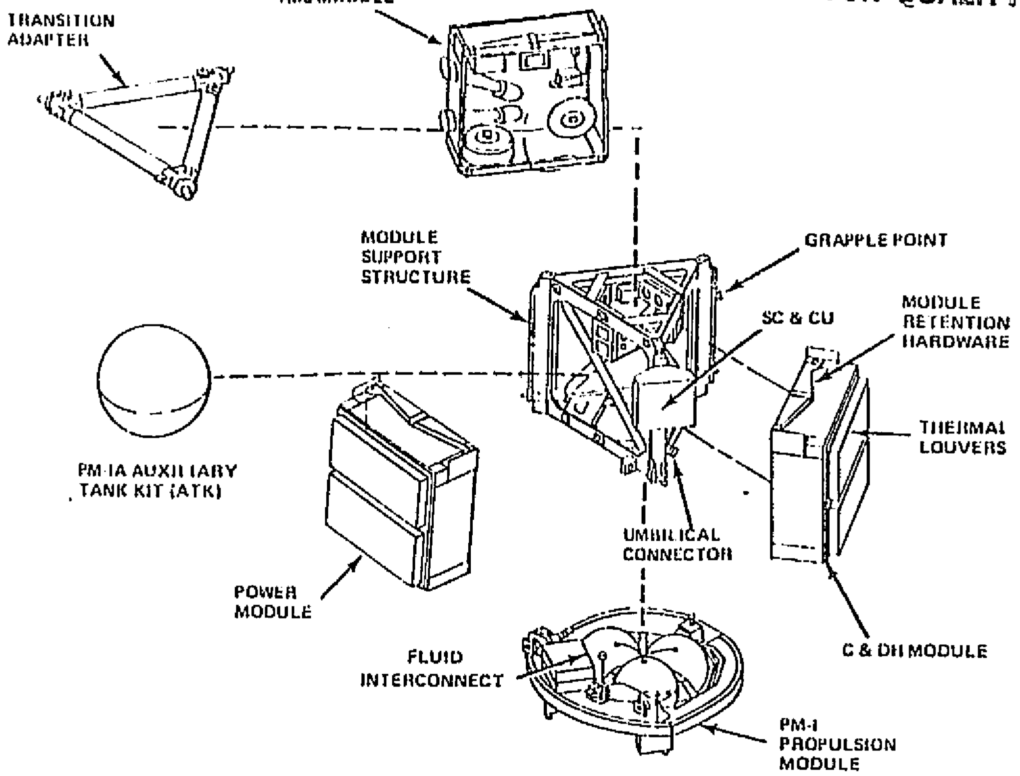

Figure 3.1.1-4 Multimission Modular Spacecraft Mechanical System

The maximum weight of an MMS module (design goal) is 500 lbs and the module structure (frame, cover, module retention system (MRS) and thermal hardware) weighs approximately 95 lbs. For ground demonstrations the module retention system and the electrical connectors may be used in a MMS module mockup weighing approximately 20 lbs.

A module servicing tool (MST) was designed and built as a battery powered EVA hand tool (See Figure 3.1.1-6). It was designed to loosen and tighten the MMS module retention hardware to predetermined torques of up to 160 ft-lb. It provides a means for locking onto the modules in a manner which avoids reaction torques on the crew member. Power is supplied by a battery housed in the tool assembly.

The MST can be provided with a servicer standard interface and can be used as an adapter for exchanging the MMS modules using the servicer.

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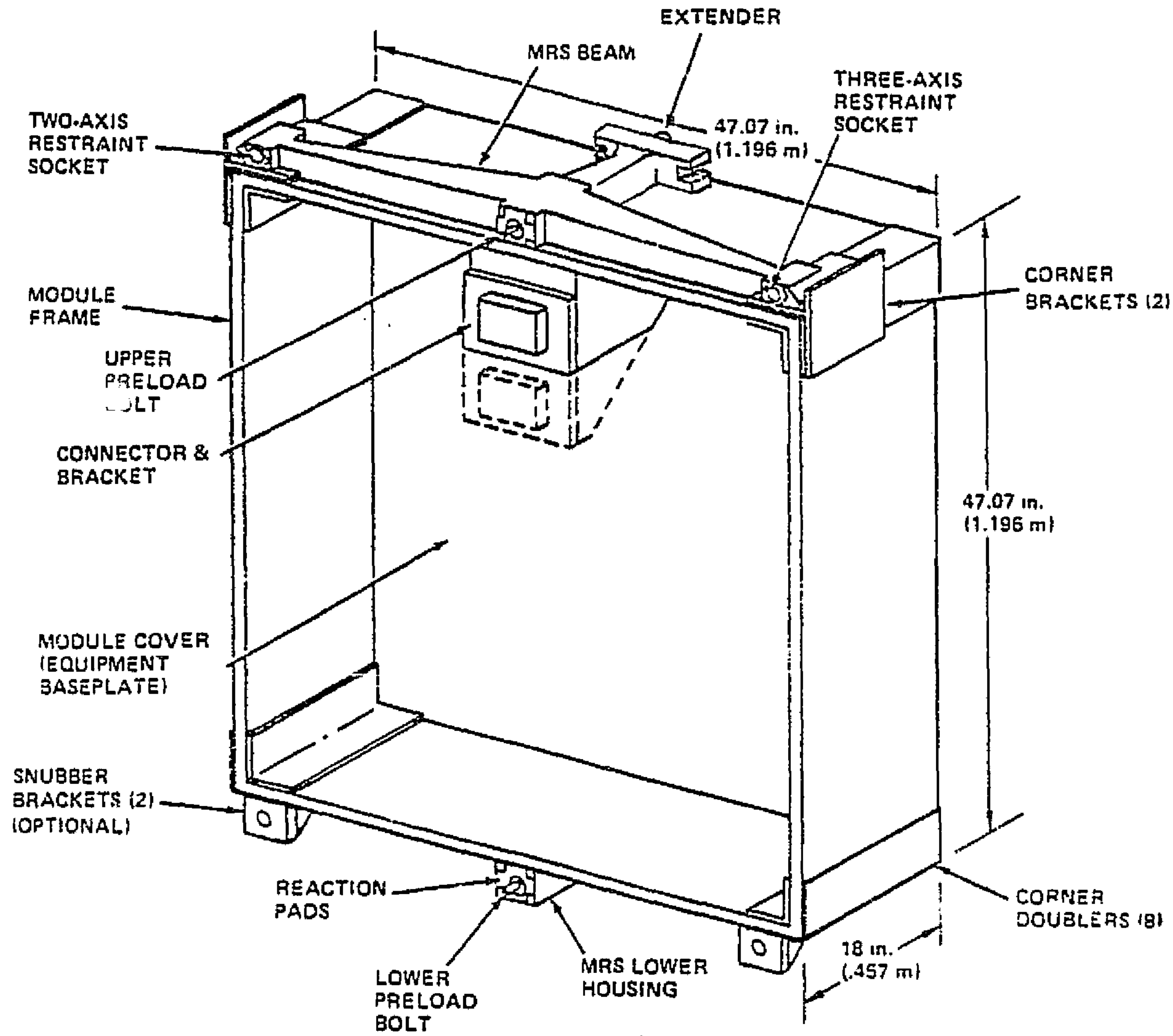

Figure 3.1.1-5 Standard Subsystem Module Structure with Module Retention System

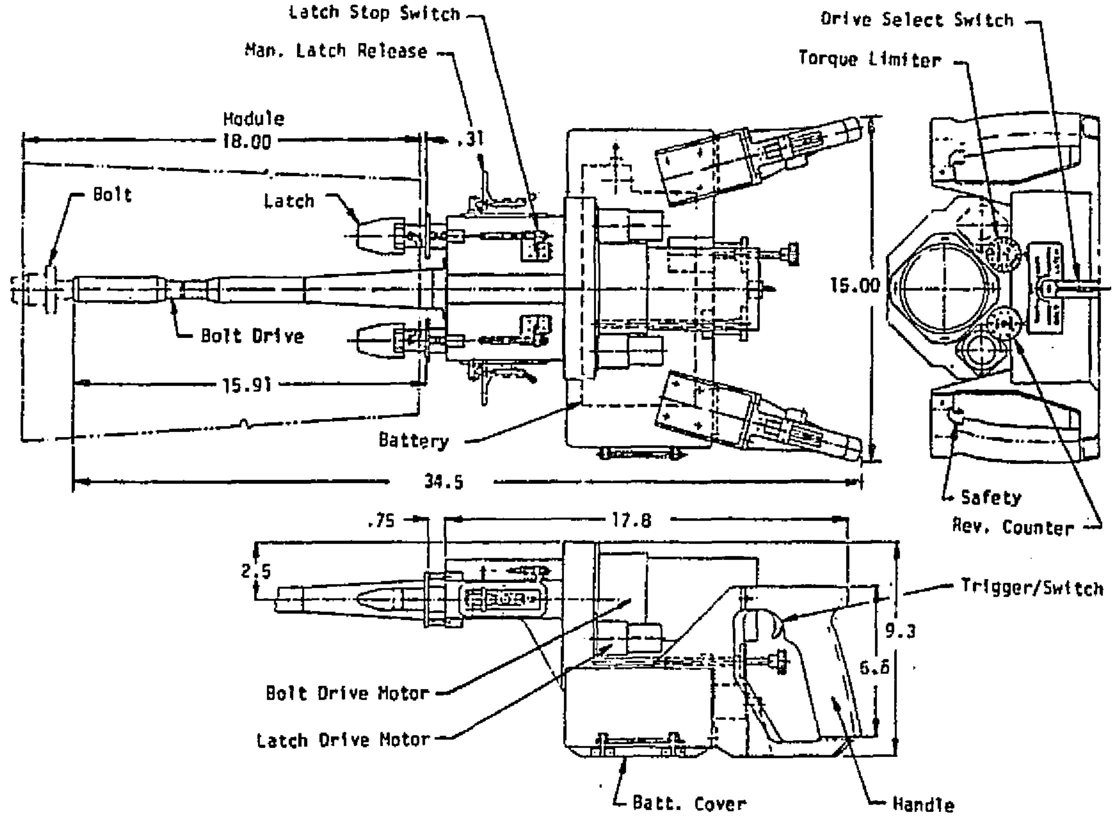

Figure 3.1.1-6 MMS Module Servicing Tool

Requirements

The following requirements apply both to ground and flight servicer demonstrations:

1) Minimum modification of the present configuration of the MMS module and/or module support structure;
2) Servicer interface with the MMS module shall be the same as for servicing other spacecraft or an adapter shall be used;
3) The method of removal/attachment of the MMS module shall be compatible with the demating/mating of the existing electrical connector(s) situated in the back of the module;
4) Adequate clearance shall be provided at all times between module and satellite structure or other components;

5) The servicer shall clear the propulsion module or high gain antenna at the lower end of MMS support structure. A clearance envelope of 19.2 in. by 53 in. diameter is required for the high gain antenna in the stowed position or for the PM-1 propulsion module. For satellites using the Mark II propulsion system (See Figure 3.1.1-7), a much larger clearance of 86 in. by 103 in. diameter is required;
6) The number of times the servicer engages with the MMS module in order to detach or attach its connections with the satellite structure, shall be kept to a minimum;
7) The accuracy in positioning the servicer for module engagement shall be within the capture envelope of the attachment mechanism. When using adapters, their design shall be such as to minimize the errors and the softness of the coupling at the interface;
8) The servicer docking TV camera and lights shall be suitably located for docking both with and without an adapter.

The following requirements apply only to the ground demonstration servicing system:

1) A MMS module mockup will be used, fitted with the actual attachment mechanism and electrical connectors if appropriate;
2) The maximum weight of the MMS module mockup shall be 20 lbs;
3) The increased end effector load due to MMS module mockup, tool adapter and other servicer modifications shall not exceed the servicer design capability. An engineering analysis of all affected components shall be conducted to ensure their safety and integrity;
4) The total height of the ground demonstration unit, including the docking probe and MMS mockups shall not exceed 20 ft;
5) A non-functional mockup of the docking probe shall be used in the ground demonstrations with the same envelope and general configuration as the actual flight hardware.

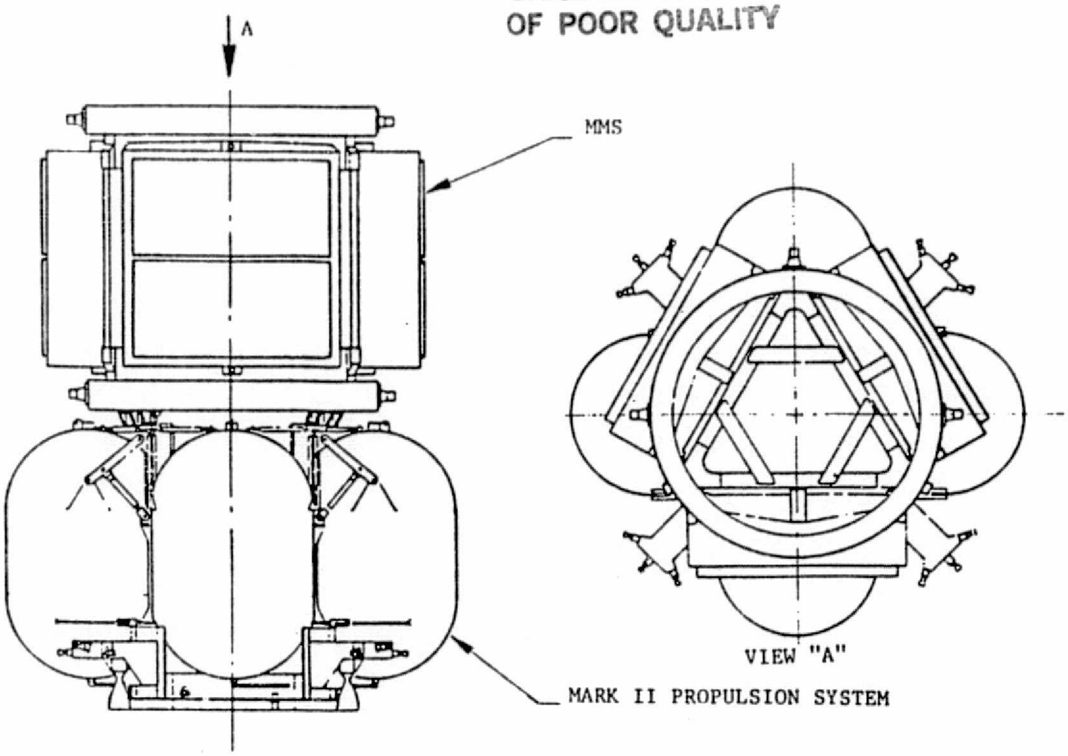

Figure 3.1.1-7 Mark II Propulsion Module Attached to Standard MMS

Candidate MMS Module Servicing Methods

In describing the following candidate servicing methods it was assumed that the servicer mechanism used is the Engineering Test Unit of the Integrated Orbital Servicing System. However, the general arrangement would be similar if another servicer mechanism were used. The candidate servicing methods 1) through 5) use axial docking of the servicer with the three berthing pins of the MMS.

1) Modified Servicer End Effector and Specialized Adapter Tool.
The end effector is replaced by a special adapter bar carrying the module servicing tool, TV camera and lights at one end and a counterbalance weight at the other (see Figure 3.1.1-8). The MST has the batteries and the controls removed and it is powered by the servicer. In order to reach the upper fastener of the MMS module from an axial docking position, the wrist segment of the servicer, between the Y and Z joints, is lengthened from 5.14 in. to 47.5 in.

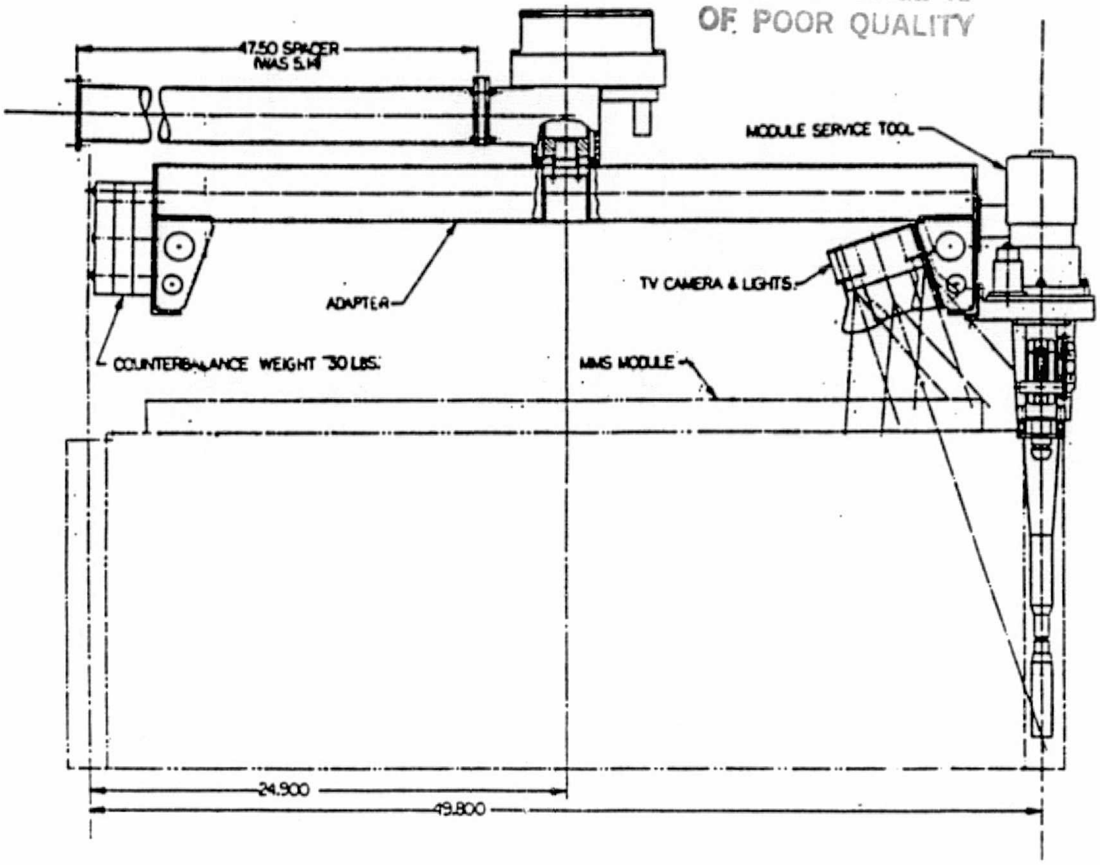

Figure 3.1.1-8 Adapter for MMS Servicing

A counterbalance weight and bracket are added to the wrist arm and the weight is increased on the other two counterbalance brackets. The docking probe has three prongs with latches, engaging the three berthing pins situated at the aft end of the MMS support structure. The docking probe clears the high gain antenna or PM-1 propulsion module envelope of 19.2 in. deep and 53 in. in diameter.

The general arrangement of the servicer is shown in Figure 3.1.1-9 and 3.1.1-10.

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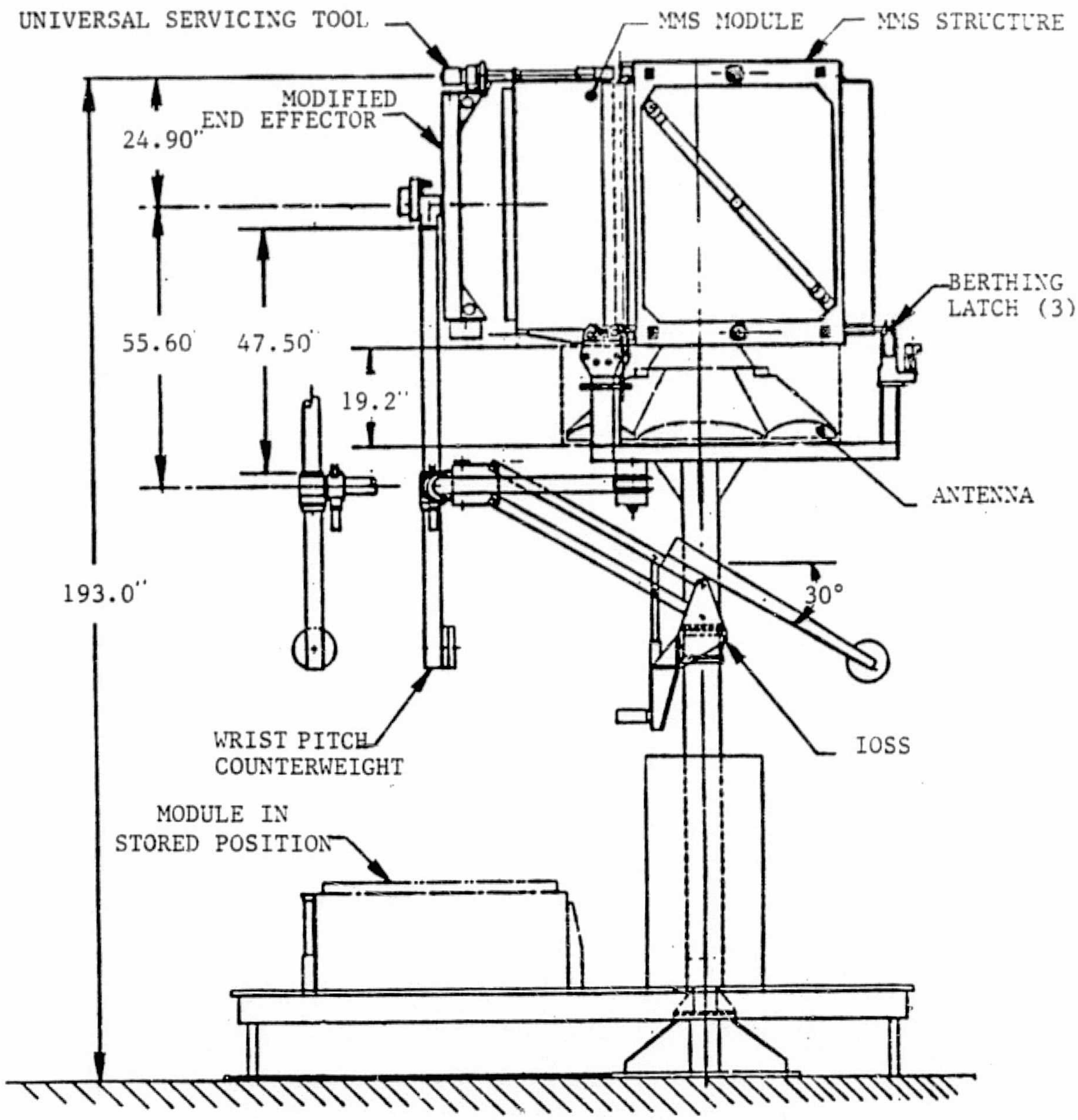

Figure 3.1.1-9 Servicer General Arrangement, Elevation View

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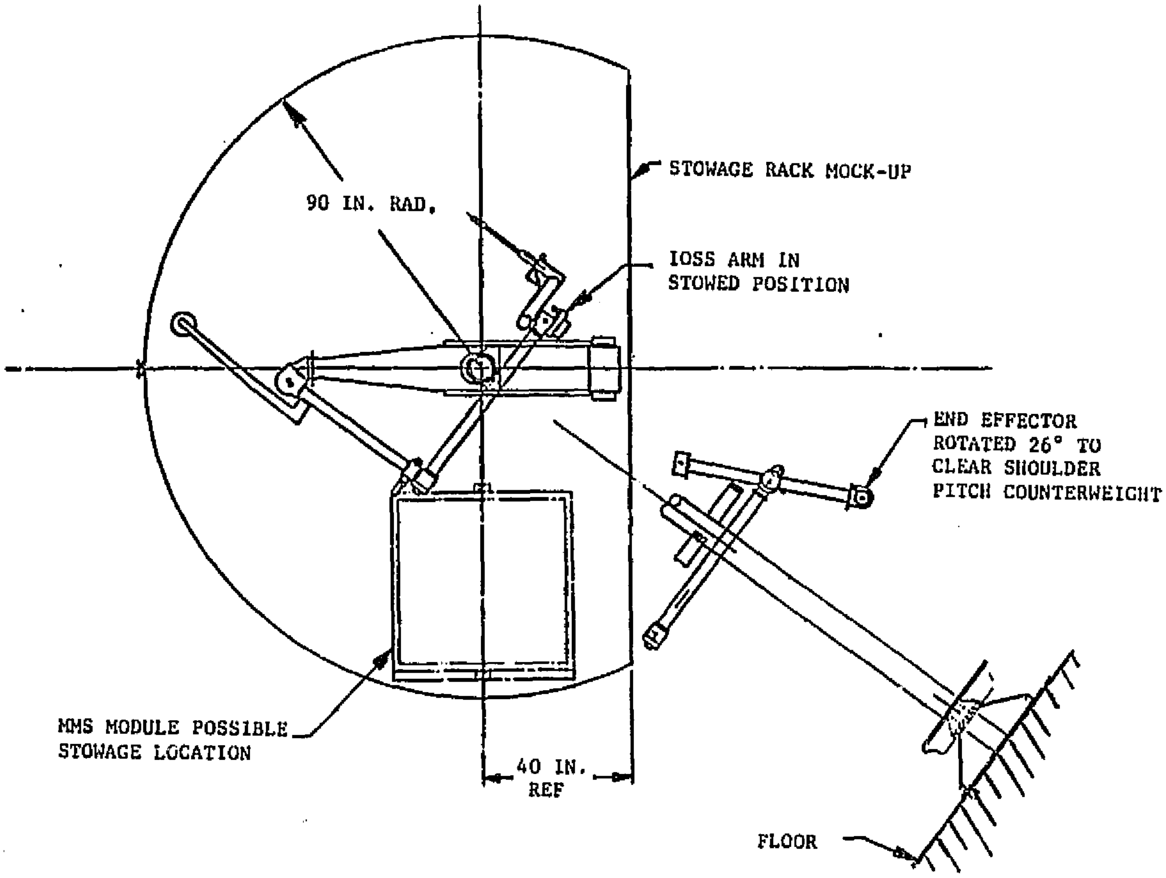

Figure 3.1.1-10 Servicer in Stowed Position, Top view

The MMS module changeout is accomplished in the following sequence (See Figure 3.1.1-11): (1) The adapter tool approaches the MMS module radially, engages and detaches one of the two attachment fasteners. During fastener detachment the two latches of the MST are engaged to cancel the tool reaction torque. The latches are then released, the tool is retracted radially to clear the module and the adapter is rotated 180° using the Z joint of the servicer. The second fastener is then engaged in a similar manner except that the latches are not released so that the module remains attached to the servicer. (2) The module is moved outward radially approximately 2 ft., using the T, V and Y joints at the same time. (3) The MMS module is rotated 90° using the Y joint. (4) The module is brought in a horizontal position by a 90° flip using the W joint.

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(5) The module is held in horizontal position and is lowered toward the stowage rack using the U joint. (6) The module is moved in a horizontal plane using rotations of the T, V and Y joints until it lines up with the fastener receptacles attached to the stowage rack. Then the module is lowered to contact the stowage rack and the two fasteners are attached in reverse order of their detachment. The servicer moves to the location of the replacement module on the stowage rack. The transfer of this module from the stowage rack to the spacecraft mockup follows the reverse order of the steps (1) through (6) described above.

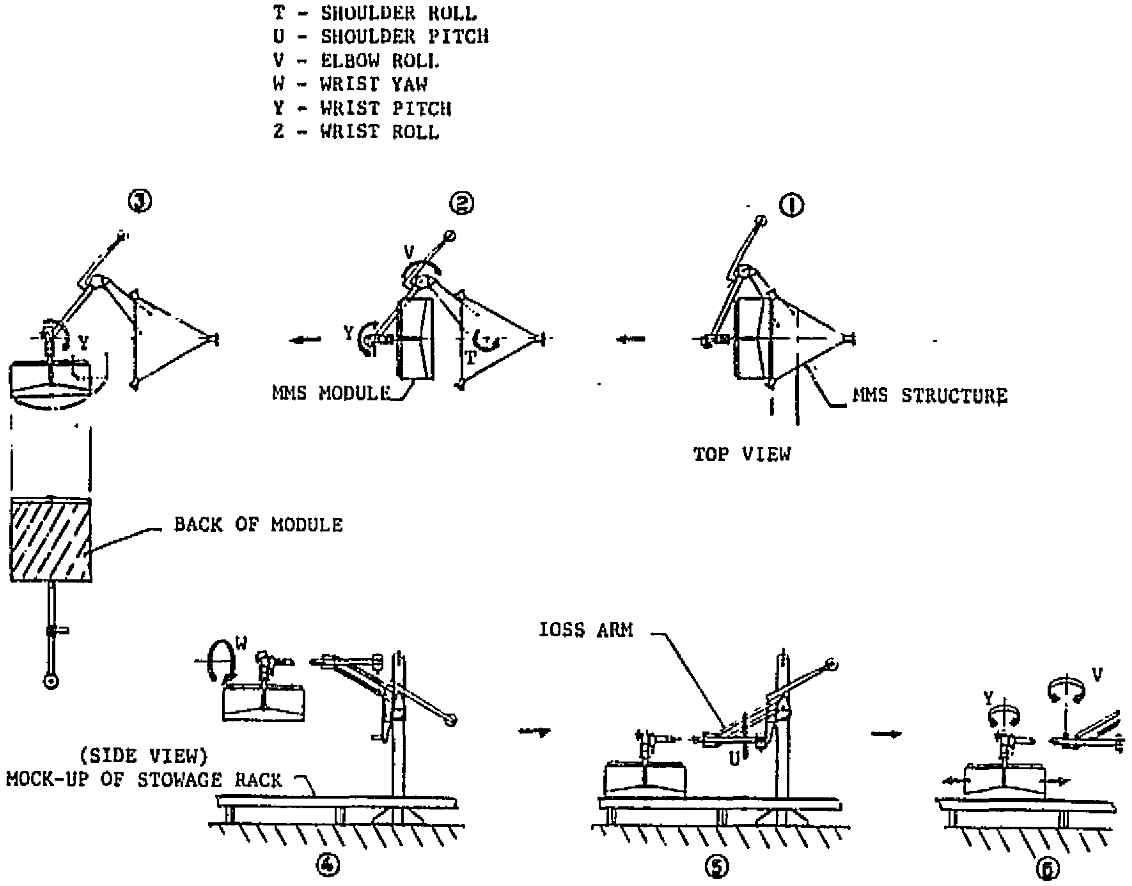

Figure 3.1.1-11 MMS Module Removal Scenario

The advantages and disadvantages of candidate solution 1) are:

Advantages:

- No change in the present configuration of the MMS.
- Uses the existing module servicing tool for module changeout.
- Can be used to service MMS type satellites already in orbit.
- Modifications to the IOSS arm are relatively inexpensive.
- Engagement of the second fastener of the module is relatively simple, requires 180° flip of the adapter.
- Engagement of both fasteners at the same time is possible by providing the adapter with two module servicing tools, one of them having a compliant attachment.

Disadvantages:

- MMS fitted with Mark II propulsion system (see Figure 3.1.1-7) cannot be serviced because of excessive length of wrist segment required (approximately 11 ft.).
- Docking simultaneously with three berthing pins is difficult, requires viewing of all three pins during docking while there is a high probability of damaging the antenna or the propulsion module.
- Longer wrist segment of the arm means less accuracy and heavier arm (33 lbs. heavier for 47.5 in. long wrist segment).
- Requires two engagements with the module or two tools.
- The servicer no longer has a standard end effector interface. However, it is possible to retain the standard end effector and attach it to the cross bar carrying the module servicing tools(s). This optional configuration would be heavier, has less stiffness and the wrist counterbalance would need to be removed when the cross bar adapter is not in use.
- 1-g demonstration requires large counterbalance weights and stiffeners for the arm segments.
- Less dexterity. Because the wrist segment is not compact, it is difficult to maneuver in tight spots or to reach around a corner.

Impact on MMS design:

- None.

Impact on servicer:

- Relatively simple modifications required - no joints changed.
- Non-standard servicer interface.
- Lower stiffness.
- Three point docking probe.

2) Use of the Existing Side Interface Mechanism.

The existing MMS module retention system is replaced by an existing side interface mechanism, mounted at the aft end of the module for easier reach (see Figure 3.1.1-12). The MMS support structure is extensively modified to receive the guides of the side interface mechanisms, as shown in Figure 3.1.1-13.

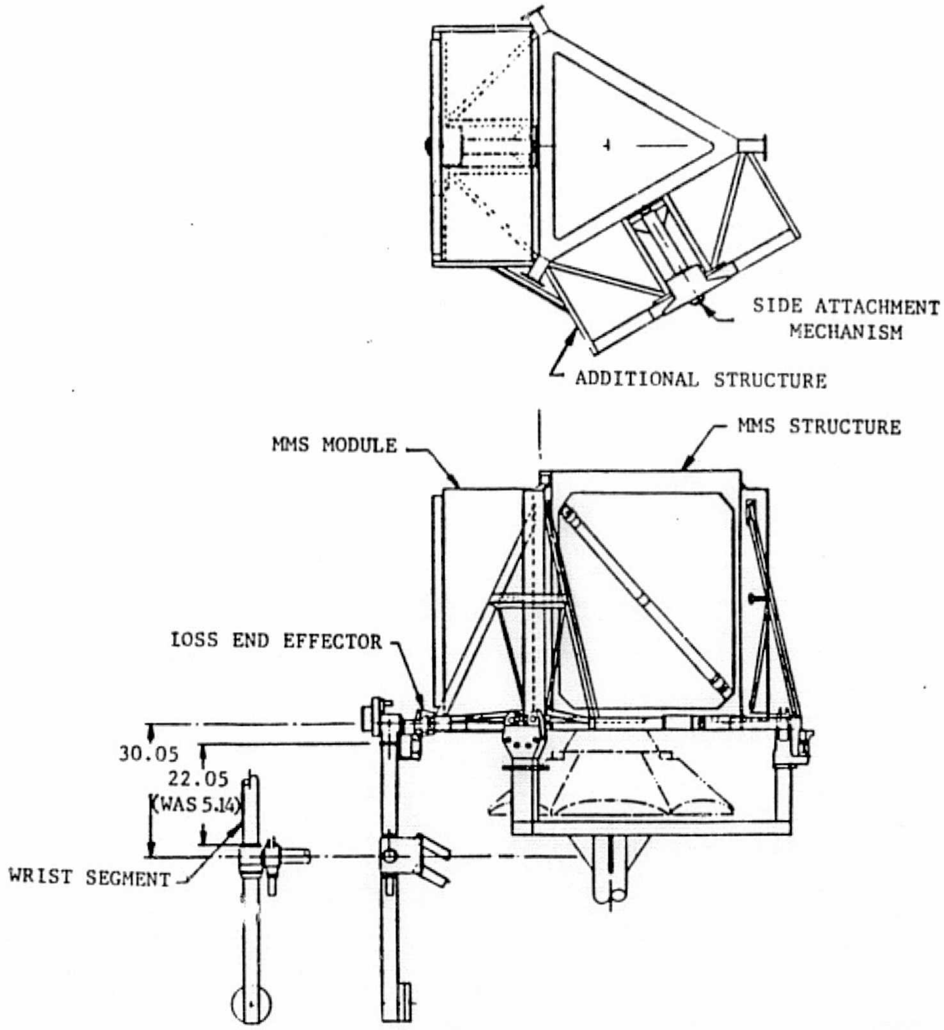

Figure 3.1.1-12 Use of the Existing Side Interface Mechanism

The wrist segment of the servicer arm is lengthened from 5.14 in. to 22.05 in. and a counterbalance weight and bracket are added to the wrist pitch joint (Y), as shown in Figure 3.1.1-12.

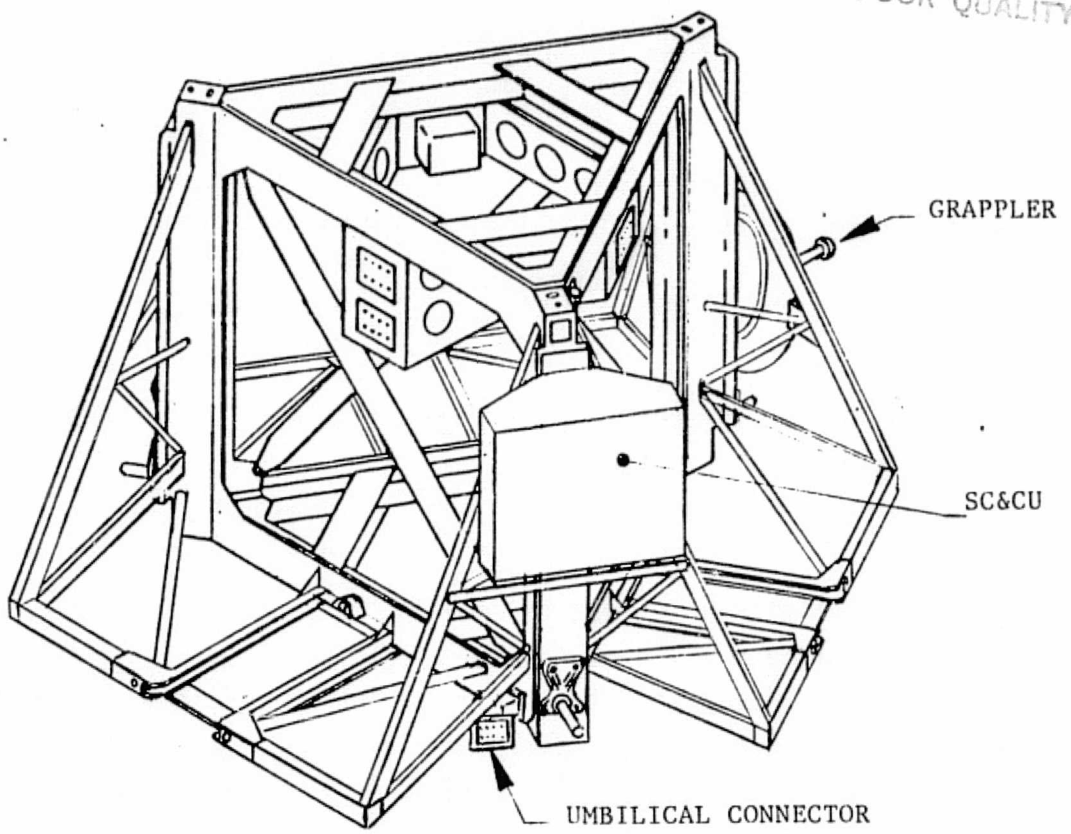

Figure 3.1.1-13 MMS Support Structure Modifications

Advantages:

- Standard servicer interface (end effector/module).
- Uses a latch mechanism of proven design.
- Only one engagement of the servicer with the module is required for changeout.

Disadvantages:

- Extensive modification of the MMS module and support structure.
- Increase of MMS weight by 80 lbs.

- The wrist segment of the servicer mechanism is 17 in. longer - less compact wrist.
- Wrist counterbalance weight is needed for 1-g demonstrations.
- Possible vibrations at the top of the MMS module, affecting the electrical connectors.
- Servicing MMS with Mark II propulsion system not practical.
- Requires three point docking.

Impact on MMS:
- Extensive modifications of the module and support structure.
- Weight increase.

Impact on servicer:
- Little change to the servicer mechanism: Longer wrist segment and wrist counterbalance weight.
- Three point docking probe.

3a) <u>Use of Alternative Interface Mechanisms - Single Power Takeoff</u>

Two interface mechanisms of a new design (Figure 3.1.1-14) replace the existing module retention system of the MMS module. They are located at the top and bottom of the module and are driven from a standard servicer power takeoff through a differential mechanism, miter gears and torque shafts (see Figure 3.1.1-15A). The new interface mechanism is derived from the existing FSS berthing latch. It is a scaled down version (approximately 1:2 scale) with direct drive to the ball screw. The jaws of the latch are curved to act as cams. In conjunction with the inside surfaces of the receptacle, the cams provide a push-out capability of approximately 1.7 in. The latch provides the force and the engagement stroke required for mating and demating of the electrical connector(s). The differential mechanism of the drive allows complete closing/opening of both latches before stalling the power takeoff motor. The drive shaft passes through the MMS module. The two latches and the differential mechanism are attached to the outside of the module.

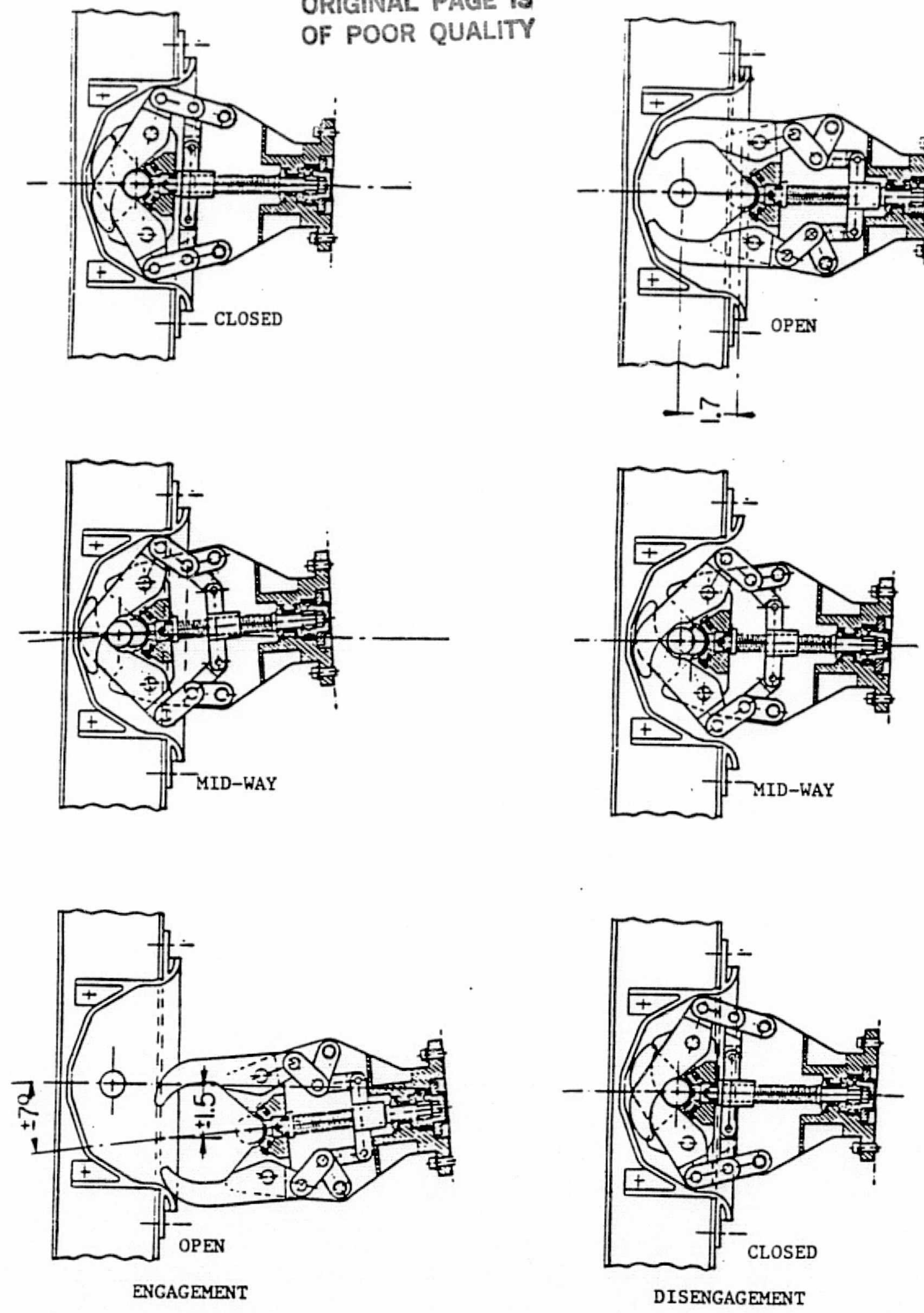

Figure 3.1.1-14 Modified Berthing Latch Mechanism with Push-Out Capability

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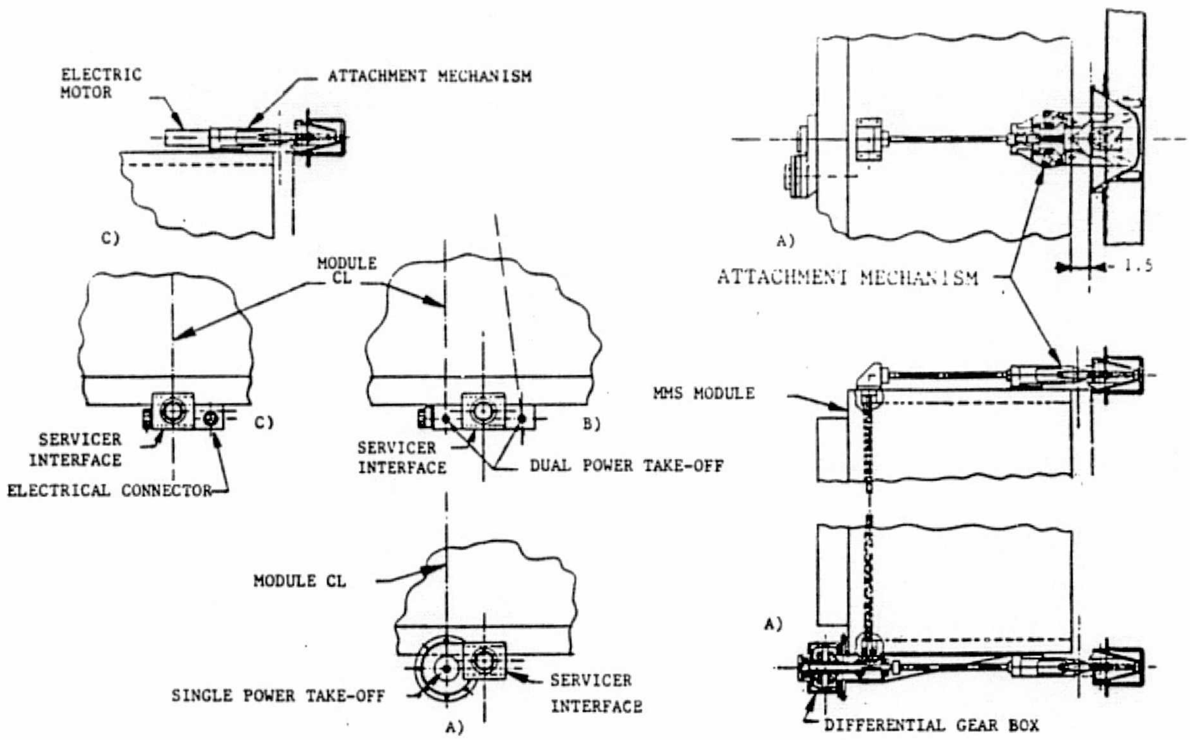

Figure 3.1.1-15 Use of Alternative Interface Mechanism

Advantages:

- Uses standard servicer interface.
- Only one engagement of the servicer with the module is required for changeout.
- Minimal modification of the MMS support structure. The nut assemblies are replaced by two receptacles with cross pins.
- The latch design is similar to the berthing latch, an existing, proven design.
- No change in the operation of the existing MMS electrical connectors.

Disadvantages:

- Mechanically complex.
- Extensive modification of MMS module. Drive shaft routed through the module.
- Increase of MMS weight estimated to be 30 lbs.
- The wrist segment of the servicer is 17 in. longer.
- Wrist counterbalance weight required for 1-g demonstrations.
- Servicing of MMS having Mark II propulsion system is not practical.
- Requires three point docking.

Impact on MMS:

- Modification of modules and support structure.
- Moderate weight increase.

Impact on servicer:

- Little change to servicer mechanism. It requires a longer wrist segment and added wrist counterbalance weight.
- Three point docking probe.

3b) Use of Alternative Interface Mechanisms - Dual Power Takeoff

This candidate solution is similar to 3a), except that the two interface mechanisms are independently actuated. The servicer standard interface is modified by adding a second power takeoff as shown on Figure 3.1.1-15B. The second power takeoff motor is attached to the end effector in a position symmetrical to the existing one. Another option would be to use the existing power takeoff motor for both locations by engaging the end effector to the module twice, with a 180° turn of the Z joint in between.

Advantages:

- Same as candidate solution 3a).

Disadvantages:

- Second power take-off added to the existing standard interface.
- Extensive modification of MMS module.
- Increase of MMS weight, estimated to be 22 lbs.
- The wrist segment is 17 in. longer.
- Wrist counterbalance weight required for 1-g demonstrations.
- Servicing of MMS with Mark II propulsion system is not practical.
- Requires three point docking.

Impact on MMS:

- Module and support structure modifications.
- Slight weight increase.

Impact on servicer:

- Longer wrist segment and added wrist counterbalance weight.
- Three point docking probe.

3c) Use of Alternative Interface Mechanisms - Electric Motor Actuation

This candidate solution is similar to 3a), except that the two interface mechanisms are fitted with electrical gearmotors and are independently controlled. An electrical disconnect is added to the existing servicing interface (see Figure 3.1.1-15C). Actuation of the electrical disconnect can be done with a translation mechanism attached to the module and powered by the existing end effector power takeoff.

Advantages:

- Same as candidate solution 3a).

Disadvantages:

- Electrical disconnect added to the existing standard servicing interface.
- Electrical motors permanently attached to the MMS module. Dual motor arrangement for each attachment is required for redundancy.
- Increase of the MMS weight, estimated to be 22 lbs.
- The wrist segment is 17 in. longer.
- Wrist counterbalance weight required for 1-g demonstrations.
- Servicing MMS with Mark II propulsion system is not practical.
- Requires three point docking

Impact on MMS:

- Module and support structure modifications.
- Slight weight increase.

Impact on servicer:

- Longer wrist segment and added wrist counterbalance weight.
- Three point docking probe
- Electrical disconnect added to the present end effector, plus the related controls changes.

4) Use of One Latch Mechanism in the Back of Modified MMS Module

The existing module retention system is replaced by a latch mechanism as shown on Figure 3.1.1-16. The latch mechanism is driven by the standard power takeoff of servicer end effector. It is attached to the bottom and the back of the MMS module and consists of a power takeoff shaft, a double reduction gear box and a double over-the-center link and push rod arrangement. The two push rods slide along the back of the module and engage two receptacles attached to the support structure. The two receptacles are shaped to provide structural support while correcting misalignment during engagement. The shear force in the plane parallel to the spacecraft centerline is taken by the existing two restraint sockets. The push rod for the upper attachment has a frame-like shape to prevent interference with the existing electrical connector(s) of the MMS module.

Advantages:

- Uses standard servicer interface.
- Only one engagement of the servicer with the module is required for changeout.
- Minimal modification of the MMS module and support structure.

Disadvantages:

- Increase of MMS weight, estimated at 12 lbs.
- The wrist segment of the servicer is 17 in. longer.
- Wrist counterbalance weight required for 1-g demonstrations.
- Servicing of MMS fitted with Mark II propulsion system is not practical.
- Requires three point docking.

Impact on MMS:

- Modification of MMS module and support structure.
- Small weight increase.

Impact on servicer:

- Longer wrist segment and added wrist counterbalance weight.
- Three point docking probe.

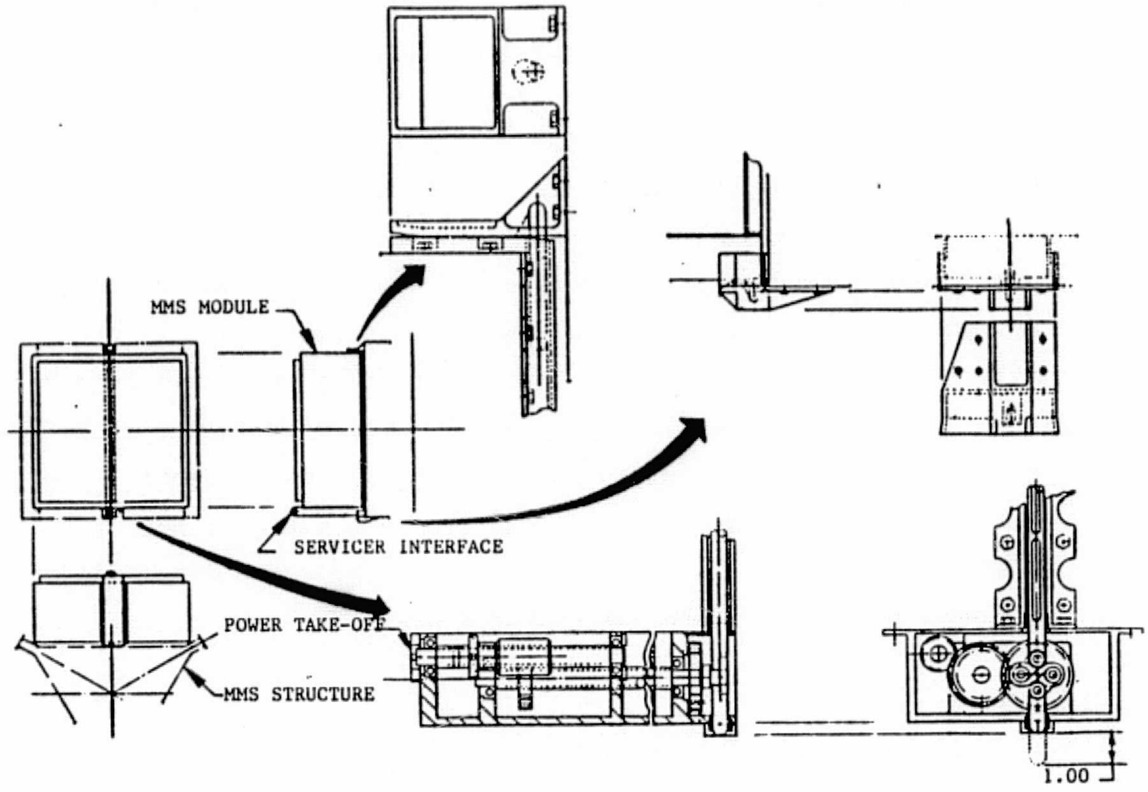

Figure 3.1.1-16 Use of One Latch Mechanism in Back of the MMS Module

5) <u>Use of One Active Latch at Bottom of Module and a Passive Hook-up Point at the Top</u>

The bottom of the module is provided with an attachment mechanism of new design, same as for candidate solution 3a), with a direct drive from a standard servicing interface, as shown in Figure 3.1.1-17. At the top of the module a passive hook-up point is installed. For attaching, the servicer holds the module and approaches the MMS at approximately a 20° angle so that the passive hook-up point engages the receptacle mounted on the upper bar of the support structure. The module is then rotated around the hook-up point reducing this angle. Engagement of the electrical connectors occurs at approximately a 10° angle which is within the misalignment capability of these connectors. Attachment is then completed by actuating the latch mechanism using the servicer power takeoff. Both the upper hook-up point and the lower latch mate with receptacles capable of correcting initial misalignment. The module trajectory is more complex and requires new equations for the supervisory mode of control. Special targets are required for the manual-direct or manual-augmented modes.

Advantages:

- Uses standard servicer interface.
- Only one engagement of the servicer with the module is required.
- Minimal modification of the MMS module and support structure for changeout.
- Relatively simple and light weight.

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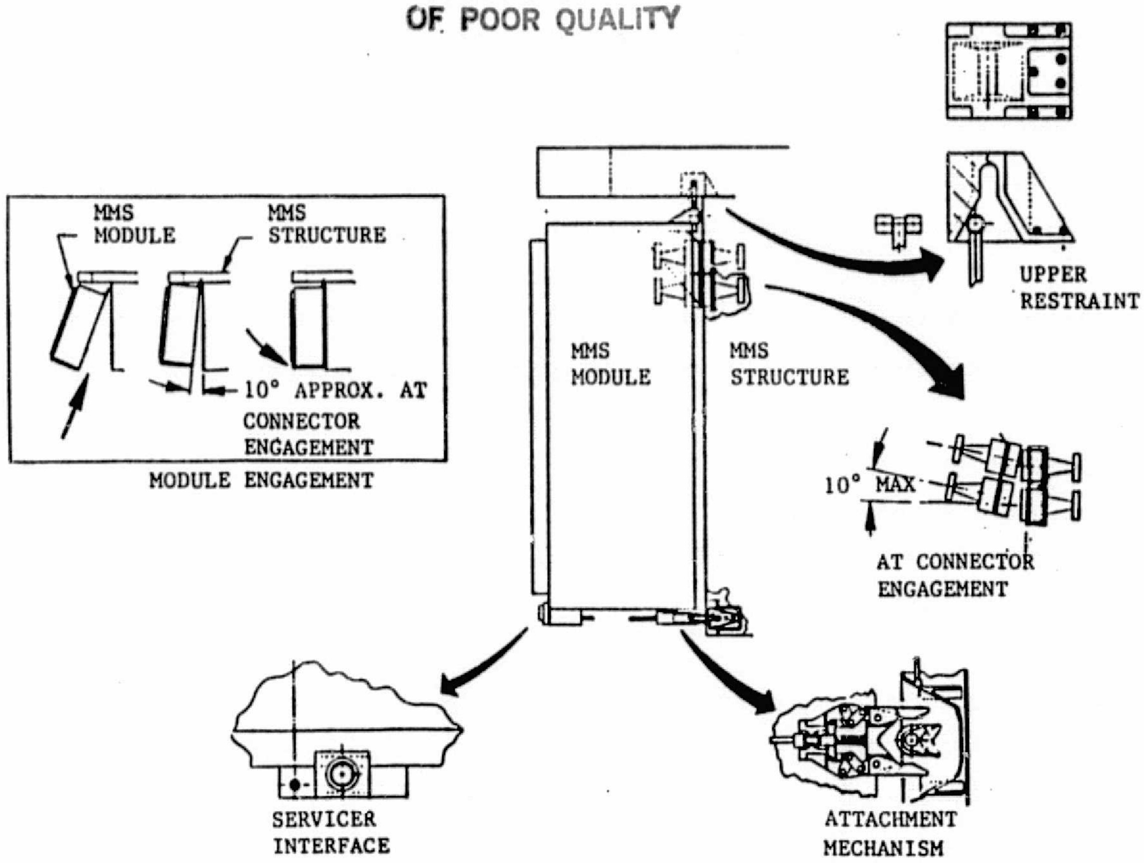

Figure 3.1.1-17 Use of One Active Latch at Bottom of MMS Module and a Passive Hook-up Point at the Top

Disadvantages:

- The wrist segment of the servicer is 17 in. longer.
- Electrical connectors start mating at 10° angle.
- More complex servicer movement to attach/detach module.
- Needs changes in control software and targets.
- Servicing MMS fitted with Mark II propulsion system is not practical.
- Requires three point docking.

Impact on MMS:

- Modification of MMS module and support structure.

Impact on servicer:

- Longer wrist segment.
- Software modifications.
- Three point docking.

The candidate solutions 6) and 7) use lateral docking of the servicer with the MMS. The servicer can dock on the existing MMS standard grapple fixture or on a special docking aid/berthing pin combination which will be designed to replace the existing berthing pins as shown in Figure 3.1.1-18. A docking probe adapter and a tool adapter will be used in conjunction with a servicer of standard configuration in order to service the MMS. Both adapters can be carried in a tool rack attached to the stowage rack.

6) <u>Use of an Offset Docking Probe Adapter and Tool Adapter</u>

The servicer is fitted with an offset docking probe adapter so that the stowage rack clears the MMS payload envelope defined as the space above the payload interface ring (see Figure 3.1.1-19) which may be occupied by solar arrays or other appendages (see Solar Maximum Mission, Figure 3.1.1-1). The adapter design is compatible with the servicer docking probe interface at one end and with the MMS docking aid interface at the other. A joint design similar to the other servicer joints is included in the docking probe adapter to allow tilting of the servicer with respect to the MMS

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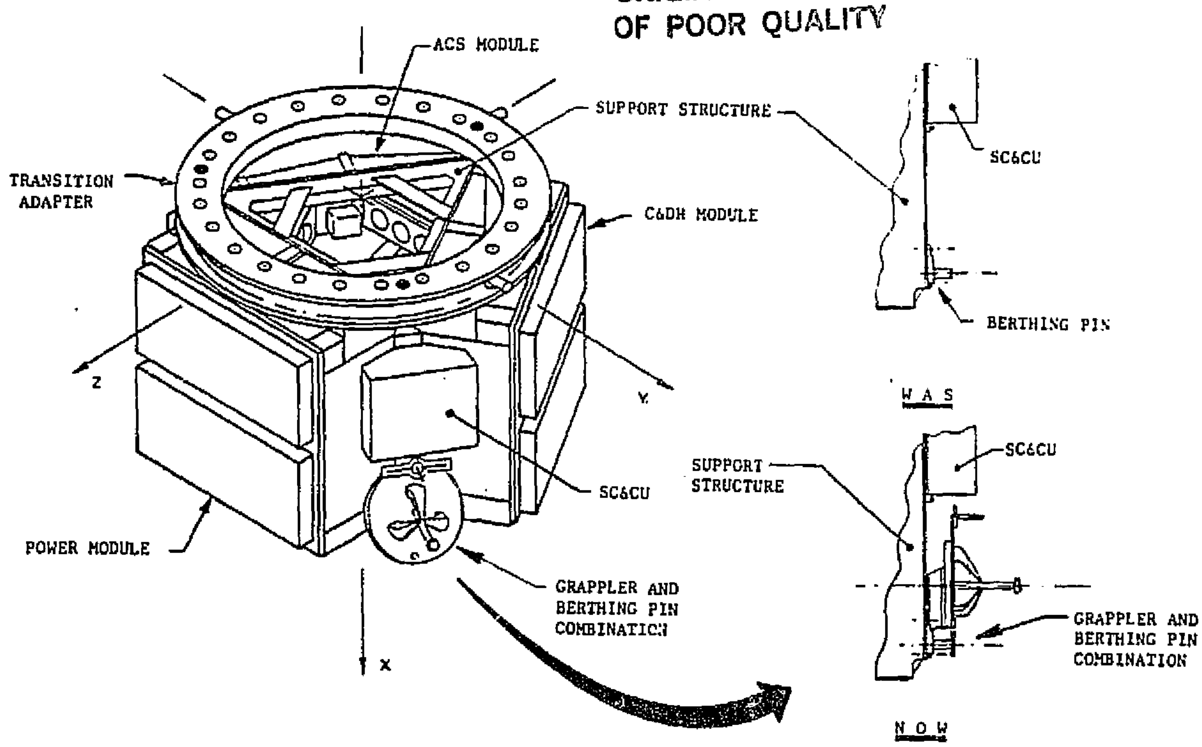

Figure 3.1.1-18 Docking and Berthing Pin Combined Design

after docking to bring the servicer mechanism into a plane parallel to the face of the module to be exchanged. The joint is powered through an electrical connection across the servicer docking interface. This feature simplifies the servicing operation without modifying the basic configuration of the servicer. No modifications of the MMS modules or module retention system are required. Instead, an adapter tool compatible with the existing MRS and with the servicer standard interface is used (see Figure 3.1.1-20).

The adapter tool is based on the existing MST (see Figure 3.1.1-6). The battery, battery case, EVA handles and controls are removed and a standard servicer interface including electrical disconnect is added. The servicer end effector power takeoff can be used to power a translation mechanism within the adapter tool, to actuate the mating or demating of the electrical disconnect.

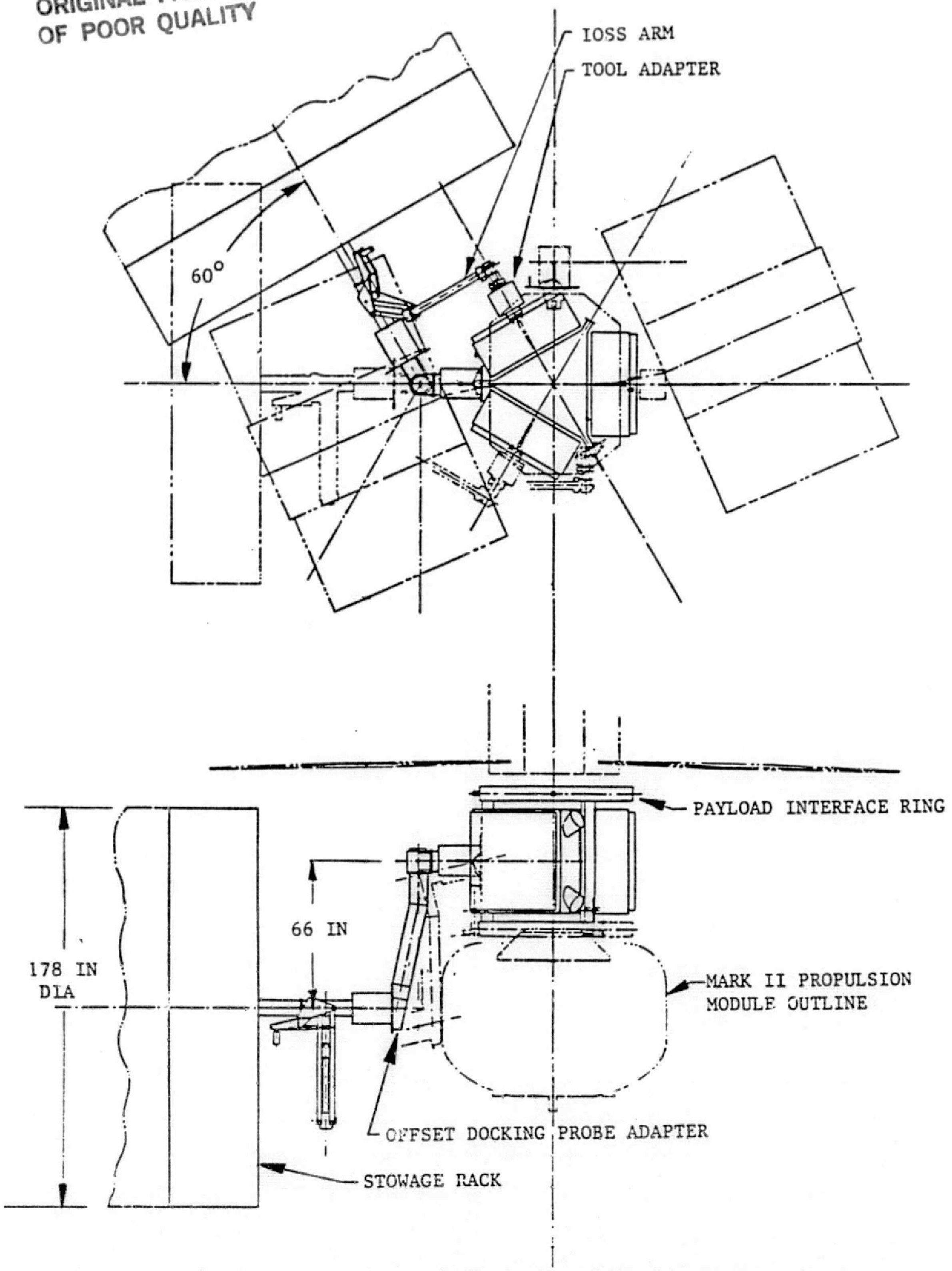

Figure 3.1.1-19 MMS Module Exchange Using Offset Docking Probe Adapter and Tool Adapter

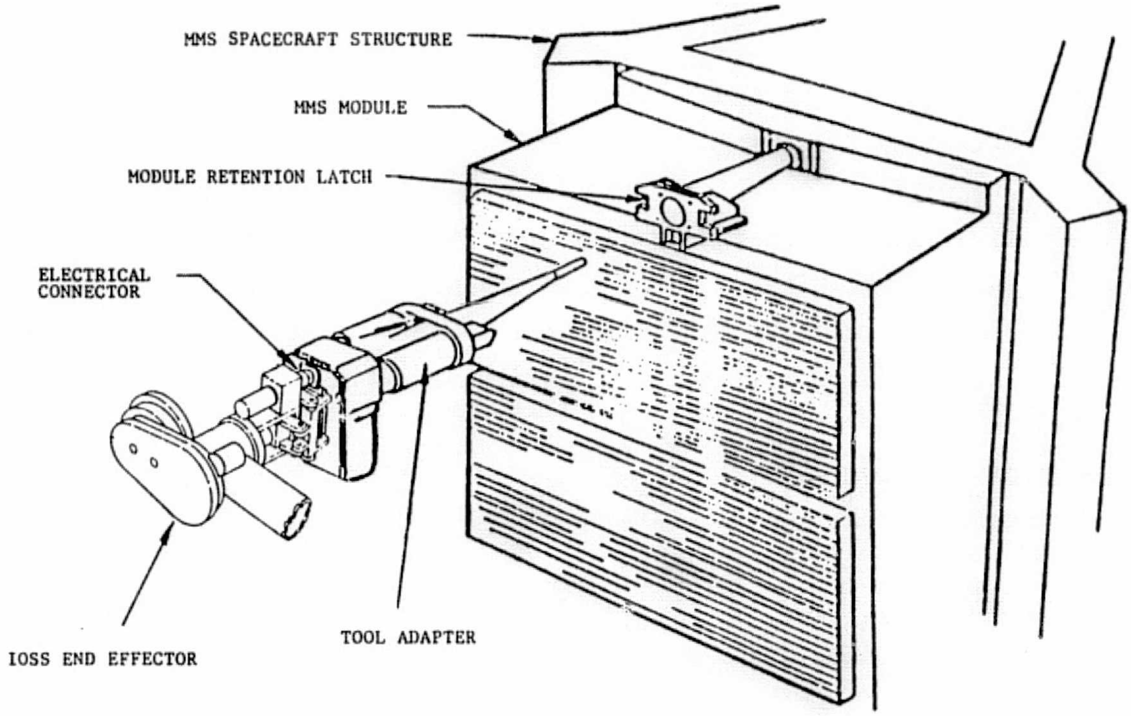

Figure 3.1.1-20 Adapter Tool for MMS Module Exchange

Advantages:

- No modification of the MMS.
- No modification of the servicer basic configuration.
- Capable of servicing all MMS including those fitted with the Mark II propulsion system.
- Capable of servicing all satellites provided with standard servicer interface.
- Single point docking.
- Simple servicer operation, using axial type module exchange.

Disadvantages:

- Requires docking probe adapter and tool adapter.
- Increased servicer weight and complexity.
- Reduced accuracy because of extra errors and softness introduced by adapters.
- Requires two engagements of the servicer with the module for attaching/detaching.
- The offset docking probe introduces extra softness in the coupling with the MMS and the attitude control system of the servicer carrier vehicle is required to act promptly after docking to prevent interference between the docking probe adapter and the Mark II propulsion system.

Impact on MMS:

- No modifications

Impact on servicer:

- Docking probe and tool adapters required.

7) <u>Use of Straight Docking Probe Adapter, Tool Adapter and Modified Stowage Rack.</u>

This candidate solution is similar to candidate 6). The main difference is that instead of the offset docking probe a straight one is employed, simplifying the docking procedure and reducing the risk. The docking probe design also incorporates a joint, which is used in a similar manner after docking to orient the servicer parallel to the front face of the MMS module to be exchanged (see Figure 3.1.1-21). In order to clear the appendages on some MMS, like the Solar Maximum Mission, the stowage rack is modified to have a built-in recess. This is not an important limitation on the servicer since the stowage rack structure is of modular design and the rack configuration and module loading is mission specific.

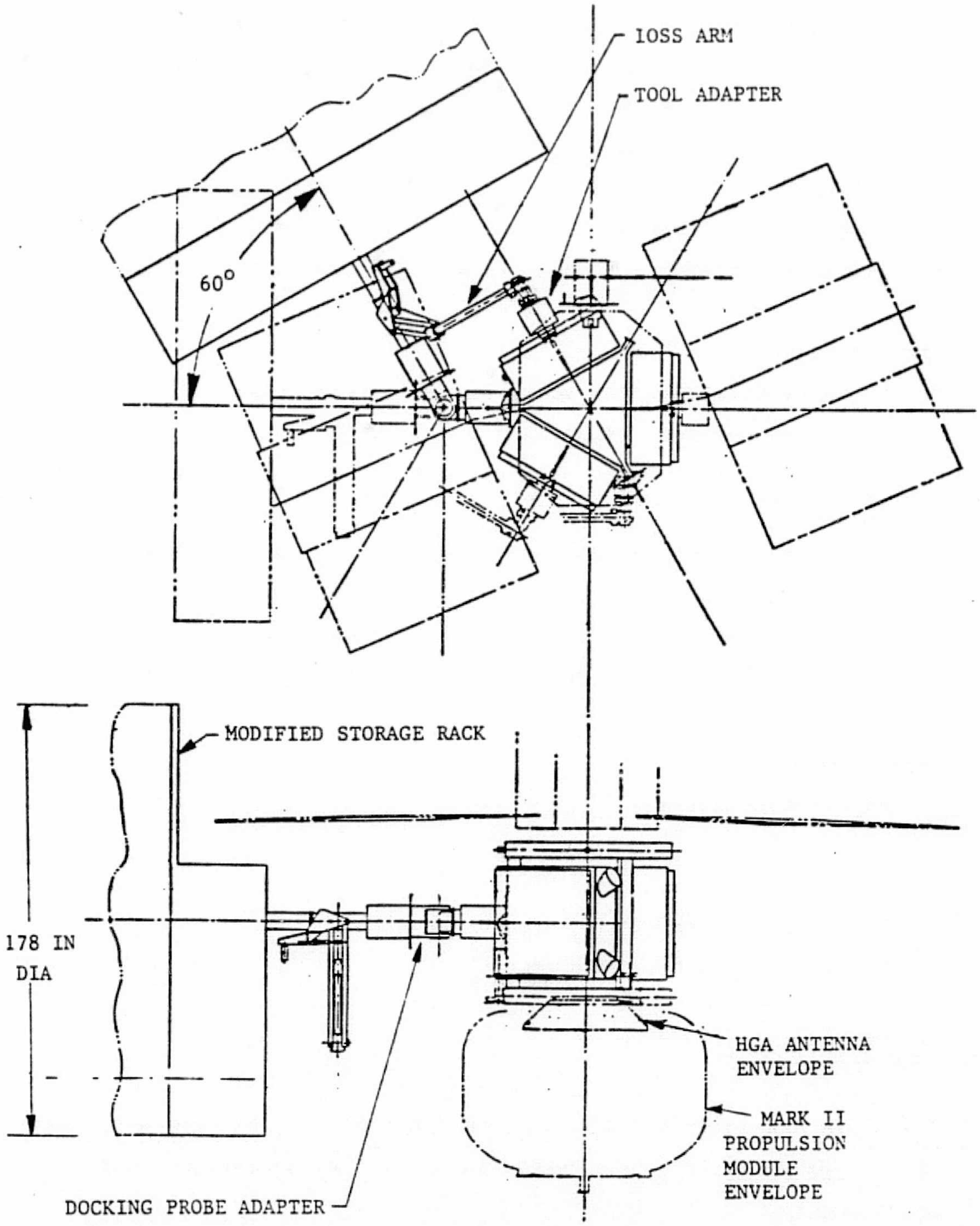

Figure 3.1.1-21 MMS Module Exchange Using Straight Docking Probe Adapter and Tool Adapter

Advantages:

- No modification of the MMS.
- No modification of the servicer basic configuration.
- Capable of servicing all MMS including those fitted with the Mark II propulsion system.
- Capable of servicing other satellites having standard servicing interface.
- Single point docking.
- Simple servicer operation, using axial type module exchange.

Disadvantages:

- Requires docking probe adapter and tool adapter.
- Increased servicer weight and complexity.
- Reduced accuracy because of extra errors and softness introduced by adapters.
- Requires two engagements of the servicer with the MMS module for attaching/detaching.
- Stowage rac' modifications required on some missions.
- Require more than one docking to exchange all three modules.

Impact on MMS:

- No modifications.

Impact on servicer:

- Docking probe and tool adapters required.
- Stowage rack modifications.

<u>Coarse screening</u>

The seven candidate solutions listed in Table 3.1.1-1 and described above were analyzed to determine how well they satisfy the system requirements.

The ability of the system to service all MMS satellites was considered a "must" requirement. It is expected that several MMS satellites with Mark II propulsion system will be built and would need servicing.

The candidate solutions 1), 2), 3a), 3b), 3c), 4) and 5) were eliminated because servicing of MMS with Mark II propulsion system was not practical and because of complexity of the design and the risk involved in the three point docking.

The remaining two candidates were traded off against the system requirements, as shown in Table 3.1.1-2:

Table 3.1.1-2 Comparison of Alternative Servicing Methods based on Lateral Docking

	CANDIDATE SOLUTIONS	
	6) Offset Docking Probe Adapter	7) Straight Docking Probe Adapter
Minimum modification of MMS	Yes	Yes
Use of standard servicer interface	Yes	Yes
Adequate clearance between servicer and MMS	No	Yes
Accuracy, stiffer adapter	No	Yes
Compatibility of docking probe with TV camera & lights	No	Yes

Recommendation

The recommended servicer configuration for servicing MMS is the candidate solution 7) "Use of straight docking probe adapter, tool adapter and modified stowage rack". For freeflight demonstration the servicer configuration is shown in Figure 3.1.1-21. For ground demonstration of the MMS module changeout, one possible configuration of the servicer is shown in Figure 3.1.1-22. The docking probe and docking probe adapter can be non-functional mockups. However, we recommend a more practical solution which is to add a MMS module mockup to the existing ETU spacecraft mockup, as shown in Figure 3.1.1-23, and

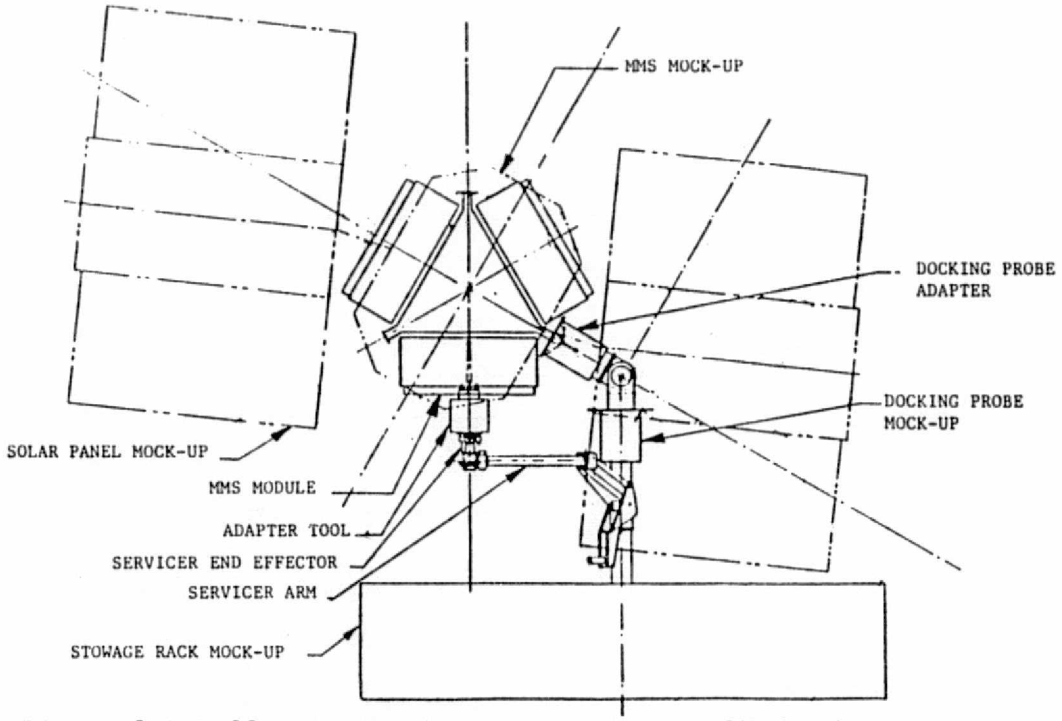

Figure 3.1.1-22 MMS Module Exchange 1-g Configuration

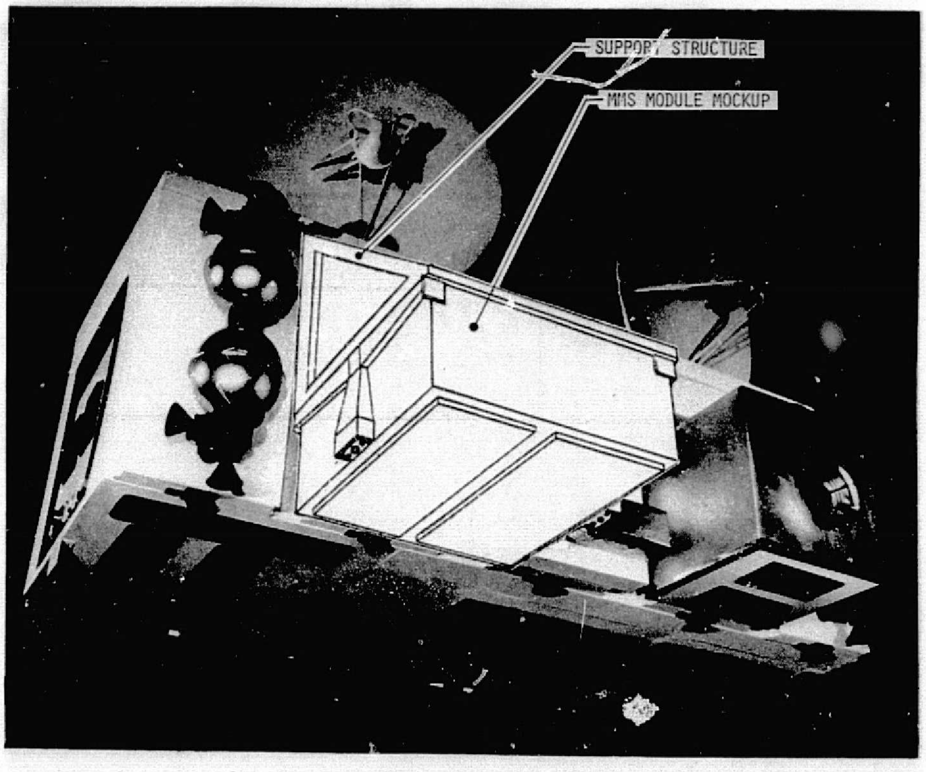

Figure 3.1.1-23 Spacecraft Mockup Modification to Add MMS Module

to add attachment hardware for MMS module mockup in two locations in the stowage rack mockup.

3.1.2 Servicing Interface Selection For Refueling/Resupply and Electrical Connections

A review of prior and current ongoing activities relative to refueling/resupply and electrical umbilical connection was performed and existing techniques for connecting fluid lines and electrical/RF cables, for on-orbit satellite servicing were identified.

The word refueling is used to denote the replenishment of any or all fulids involved in the spacecraft propulsion or attitude control system, while the word resupply is generally used to denote the replenishment of all other fluids, including cryogenics used for instruments.

The system requirements for satellite refueling demonstrations were defined for ground, cargo-bay and free-flight demonstrations.

Because the refueling operations require connecting electrical lines in addition to fuel lines in order to control valves, and perform monitoring functions within the serviced fuel system, a separate demonstration of connecting electrical cables only may be avoided.

The servicing interface and the servicer configuration was selected for the refueling/resupply demonstrations. Several different methods of connecting the refueling interface were considered, as shown in Table 3.1.2-1. They were traded against the system requirements in order to select and recommend the servicer configuration for refueling demonstrations.

Table 3.1.2-1 Alternative Refueling/Resupply Interface Concepts.

1) Refueling/resupply interface unit(s) attached to the docking probe
2) Refueling/resupply interface unit(s) stored on the stowage rack positioned and connected by the servicer arm
3) Refueling/resupply interface unit attached to the end effector of the servicer
4) Auxiliary servicer arm dedicated to refueling operations.

The method 2), refueling/resupply interface unit(s) on stowage rack, positioned and connected by the servicer arm best meets the requirements and is recommended for use in the refueling demonstrations.

3.1.2.1 Prior and Current Activities Relative to Satellite Refueling - Many studies and demonstrations during the past decade proved the cost benefits and the feasibility of spacecraft on-orbit refueling and resupply. A list of prior and current efforts in the area of satellite refueling is given in Table 3.1.2-2.

Table 3.1.2-2 Prior and Current Satellite Refueling Activities

NO	ACTIVITY	ORGANIZATION	CUSTOMER	STATUS
1	Fluid Disconnect for Space Transportation System. Design, Development, Fabrication and Testing.	Fairchild/Stratos W. P. Rigsby ER 76300-4,5	NASA/MSFC NAS 8-32806	1976-1980
2	Electrical Disconnect System. Analytical Study, for use in Manned and Unmanned Missions	Martin Marietta MCR-76-393	NASA/MSFC NAS 8-31971	Completed 1976
3	MMS In-Orbit Refueling Study. Analysis and Design	Rockwell International SSD 80-0175-1	NASA/GSFC NAS 5-26152	Completed 1980
4	Reuse/Resupply Component Study	Martin Marietta L. J. Rose MCR-83-600	AFRPL Mel Rogers F04611-82-C-0008	Completed 1983
5	Cryogenic Fluid Management Facility	Martin Marietta R. Eberhardt	NASA/LeRC Erick Kroeger	1984
6	Satellite Servicing Study (classified)	Martin Marietta A. Wudell	DoD	Ongoing
7	Satellite Services System Analysis Study	Lockheed-EMSCO LMSC D 798873A	NASA/JSC NAS 9-16121	Completed 1982
8	Space Based Laser and Escort Vehicle Servicing and Resupply	Martin Marietta D. Smerchek	DoD/SD Maj L. Young	Ongoing
9	Definition of Technology Development Missions for Early Space Stations. Satellite Servicing	Martin Marietta S. Schrock	NASA/MSFC NAS 8-35042 Bob Middleton	Ongoing

Table 3.1.2-2 Continued

NO	ACTIVITY	ORGANIZATION	CUSTOMER	STATUS
10	Space Operations Center Technology Identification Study	Boeing D180-26495-7	NASA/JSC	
11	Demonstration of Fluid Transfer Interface Tool (Landsat Type Tank)		NASA/JSC N.C. Elden	STS 11
12	Orbiter Mid Deck Transfer Experiment	Martin Marietta Z. Kirkland D-02R	NASA/JSC H. E. Benson and DoD/SD	STS 15 (41-F)
13	Mark II Refueling System Design and Development	Martin Marietta J. Haley	DoD	Trade Studies Complete
14	Demonstration of Fluid Transfer of Hydrazime Using EVA Techniques		NASA JSC/EB	STS 17
15	Development of a Quick Disconnect Fluid Transfer Coupling		NASA JSC/EP H. E. Benson	Start 1984
16	Tethered Orbital Refueling Study	Martin Marietta R. Rozycki	NASA/JSC Ken Kroll	Ongoing
17	STS Propellant Scavenging Study	Martin Marietta W. Gilmore	NASA/MSFC Milt Page	Ongoing
18	Development of Standardization of Fluid Transfer Interfaces		NASA JSC/EB	Start 1984
19	Orbital Refueling System Design and Analysis. Definition of System Requirements.		NASA JSC/EB	Start 1984
20	S^3 Fluid Transfer Interface Requirements Study	Lockheed-EMSCO Jack Wohl	NASA/JSC	Ongoing
21	Satellite Services Fluid Transfer Considerations Study	JPL Jim Lumsden		Ongoing

Table 3.1.2-2 Continued

NO	ACTIVITY	ORGANIZATION	CUSTOMER	STATUS
22	Fluid Transfer System Study	Grumman Ron Boyland		Ongoing
23	Satellite Services Fluid Transfer Interface Study	Boeing Keith Amy		Ongoing
24	Shuttle Infrared Telescope Facility-Instrument and Cryogen Replenishment Study		NASA/ /ARC R.A. Lavond	RFP 1984
25	Gamma Ray Observatory On Orbit Refueling Study	TRW Donald Mollgard	NASA/GSFC	1982
26	Design of Fluid Couplings for Automated Servicing Applications	Fairchild Robert Burns	Presented at Satellite Services Fluid Transfer Interface Requirements Workshop NASA -JSC Feb 15-17, 1984	Ongoing
27	Aft Propulsion System-Orbital Maneuvering System for Satellite Resupply	McDonnell Douglas V.A. Blythe		Ongoing
28	Satellite Services Fluid Transfer Interface Requirements Workshop Vol I & II		NASA-JSC JSC 19535	Feb 15-17 1984

3.1.2.2 Critical Components and Processes for the Design of the Propellant Transfer System - A reuse/resupply study (Ref 4 on Table 3.1.2-2) identified the critical components that need development work before an on-orbit refueling system can become operational. Table 3.1.2-3 lists these components and the requirements for their use in refueling/resupply systems.

Two types of components are used to connect fluid lines, depending on the application. For refueling/resupply, fluid disconnects are used for making a temporary line connection for the duration of the fluid transfer. They are normally backed by control valves on both sides. The in-line couplings are used to connect fluid lines when replacing components such as tanks or trusters. Leak prevention is more important for couplings and more difficult to achieve, because they stay under pressure at all times and may be subject to vibration and external loads.

Table 3.1.2-3 Refueling Components Requiring Development Work

Component/Subsystem	Requirements
Disconnects (for Refueling/Resupply)	Leak free Reliable, multiple cycles, long life Self aligning Automatic operation Standardized interface One half mates correctly with any opposite half. Compatible with the fluid to be transferred. Thermally protected
In-line Couplings (for Component Replacement)	Leak free, under continuous pressure, vibration, load Reliable, multiple cycles, long life Self aligning Standardized interface One half mates correctly with any opposite half Compatible with the fluid to be transferred.

Table 3.1.2-3 (continued)

Component/Subsystem	Requirements
Valves	Long life, 10^7 Cycles Low internal leakage, minimum increase with time.
Filters and Dessicants	Long life
Subsystem for Mating/Demating the Disconnects	Reliable, multiple cycles, long life Standard interface Single interface for all functions including electrical Applicable to free-flyer servicer
Subsystem For Mating and Demating In-Line Couplings	Reliable, multiple cycles, long life Standard interface Applicable to the free-flyer servicer
Propellant Management Device	Provide vapor free propellant feed Multiple fill/empty cycles Prevent propellant slosh Compatible with the fluid to be transferred.
Flexible Fluid Lines	Long life, multiple flexing cycles Minimum bend radius Compatible with the fluid handled.

Other requirements for the disconnect valves are listed below as design goals (Ref 3 and 28, Table 3.1.2-2):

1) Means must be provided for verifying leak integrity of the interface seal between the two disconnect halves before admitting propellant to the interface cavity. Warning indication of any propellant leakage during refueling, and automatic circuitry for correcting any resulting hazardous condition, shall also be provided;
2) Three mechanical inhibits shall be provided to prevent external leakage of propellant from each disconnect half. Leak rate (mated or demated) shall be less than 10cc/hr at 0-400 psi GN_2;

3) Means shall be provided for preventing any propellant leakage from entering the cargo bay or contaminating the free-flyer servicer. Maximum spill volume shall be 1cc;

4) Design of the disconnect and the refueling system shall be such that the presence of propellant vapor pockets or bubbles in the disconnect is minimized and their rate of pressure increase is limited to preclude detonation by adiabatic compressive heating of such vapor or vapor/gas mixtures;

5) The design of the disconnect shall minimize any possibility of jamming while connected, and failing to disengage under normal retraction forces;

6) Flowrate 1000 lbs/hr;

7) Pressure drop 50 psi @ rated flow;

8) Maximum required stroke 3.0 in;

9) Allowable lateral offset 1/16 in. ($\pm$ 3/4 in. for mate/demate subsystem);

10) Allowable misalignment $\pm$ 5 degrees ($\pm$15° for mate/demate subsystem);

11) The force required for mating/demating shall be kept to a minimum;

12) Maximum volume occupied by the disconnect valve(s) and the mate/demate mechanism shall be 12 in. cube of internal spacecraft volume;

13) Cycle life of 25 mate/demate cycles applies to the spacecraft side of disconnect and 5000 mate/demate to the servicer side;

14) Dust covers or other means shall prevent the mating surfaces from contamination at all times during the mission, except during the refueling operations;

15) The active mechanism of the mate/demate subsystem shall be provided on the servicer side, the satellite interface shall be passive;

16) Positive locking of the mate/demate subsystem shall be provided by the servicer;

17) Power/signals and monitoring capability during servicing shall be provided by the servicer or its carrier vehicle;

C - 2

18) The refueling/resupply interface shall be designed with commonality for all modes of servicing (free flying servicer, in cargo bay, FSS, RMS, etc.) Power/signal capability, if needed shall use the common interface;

19) EVA override or redundant actuation shall be provided for the demating of the mate/demate subsystem in contingent situations;

20 The mate/demate subsystem shall include an auto indexing feature to ensure the correct mating of disconnect halves.

The study listed under No 4 in Table 3.1.2-2 gives the status of the development work on each critical component and gives an extensive list of references. In addition to the critical components, critical technologies that are not state of the art but are essential to the refueling/resupply mission were identified. Among them, low-g fluid transfer, mass gaging and venting technologies need further development and experimental verification.

These critical components and technologies are being developed under various NASA, DoD and contractor activities and are expected to be available for use in the refueling demonstration as planned under this contract. The demonstration hardware selected should provide enough flexibility to accommodate further improvements. The ground and cargo-bay servicer demonstration units can be used in the future for the development and qualification of new refueling hardware.

Standardization of the refueling interface is an important issue affecting the economics and ultimately the success of the satellite servicing activities. An Interface Standardization Project is being pursued by NASA/JSC (Ref 18, Table 3.1.2-2). The program start is planned for the third quarter of 1984 and the objective is to develop a standard propellant servicing interface for all satellites. A committee will be formed consisting of the appropriate NASA elements, the DoD and those industrial firms active in the design and fabrication of satellite propulsion stages. This committee will define the fluid interconnects, mechanical attachment hardware, isolation philosophy, data format requirements, and instrumentation and control interfaces consistent with safety requirements and minimization of crew time lines.

The program objectives are to develop and certify a standardized disconnect design for on-orbit resupply of earth storable, gaseous and cryogenic fluids and to provide earth storable fluid disconnect flight hardware for the Gamma Ray Observatory (GRO) by March 1986.

The design of the refueling demonstration unit should be coordinated with this standardization effort in order to gain industry wide acceptance.

3.1.2.3 Existing Techniques for Connecting Fuel Lines - Previous studies (Ref 1 and 3, Table 3.1.2-2) conducted an extensive search of industry and government sources of technical data in order to identify available space-qualified hardware usable as disconnect valves in orbital refueling systems (ORS). Fairchild/Stratos Division (FSD) was the only firm that had extensive experience in developing and manufacturing small, remotely actuated, space-rated disconnects for storable propellant service.

A NASA prototype disconnect valve (see Figure 3.1.2-1 and Table 3.1.2-4) was developed by FSD during a Space Transportation System Disconnect Program under contract number NAS8-32806 to MSFC. The program was started in September 1976 and ended in March 1980 (Ref 1, Table 3.1.2-2). During the first phase of the program FSD designed and successfully tested the medium duty (300 psi) NASA prototype disconnect following a thorough search of industry and government sources which failed to locate an existing off-the-shelf design suitable for the orbital servicing concept. This design features an external swivel with semi-balanced sleeve/poppet that provides relatively low pressure induced separation forces (approximately 1/3 standard unbalanced design), only one close tolerance sealing diameter, relatively short engagement and reasonable low interface volume. In February 1978 FSD installed the backup prototype NASA disconnect on the Engineering Test Unit built by Martin Marietta Aerospace for demonstrations to NASA personnel during the ETU Performance Design Review, in Denver. FSD also installed the second NASA prototype disconnect on the ETU following its delivery to MSFC for further fluid transfer demonstrations and evaluation.

In the second phase of this program starting in April 1978 FSD studied modifications of the NASA prototype disconnect and of an existing Jet Propulsion Laboratory (JPL) disconnect (see Figure 3.1.2-2 and Table 3.1.2-4) for use as a medium duty Freon 21 disconnect for the 25 Kw Power Module. The JPL disconnect, was previously flight qualified by the Jet Propulsion Laboratory and used in the Mariner Space Vehicle. The design of this disconnect with internal swivels offers simplicity and hard line installation that may make it compatible with the ORS.

The third disconnect valve design applicable to the ORS, also developed by FSD, is the LEM disconnect (see Figure 3.1.2-3 and Table 3.1.2-5). Of the three designs, the LEM disconnect is considered to be the best choice, chiefly because it already incorporates a pressure test port that can be used to verify leak integrity of the interface seal prior to admitting propellant between the engaged halves. Also this disconnect was subjected to a very stringent qualification test program because it was a critical component for separation and return of the LEM ascent stage from the Moon to the Apollo Command Module. Although designed for glycol and gaseous oxygen service, the supplier states that it can be readily adapted for hydrazine service by the use of appropriate seal materials.

The pressure drop estimated for the LEM disconnect at a flow of 1.0 lb/sec of hydrazine is 150 psi. For the case of refueling a Mark II spacecraft with 5000 lbs of propellant, at an initial ullage pressure of 70 psia, with a regulated supply pressure of 370 psia in the ORS tank, propellant transfer could be accomplished in approximately 120 minutes. This assumes a 1/2 in. refueling line size in the ORS and spacecraft and the use of only one refueling disconnect.

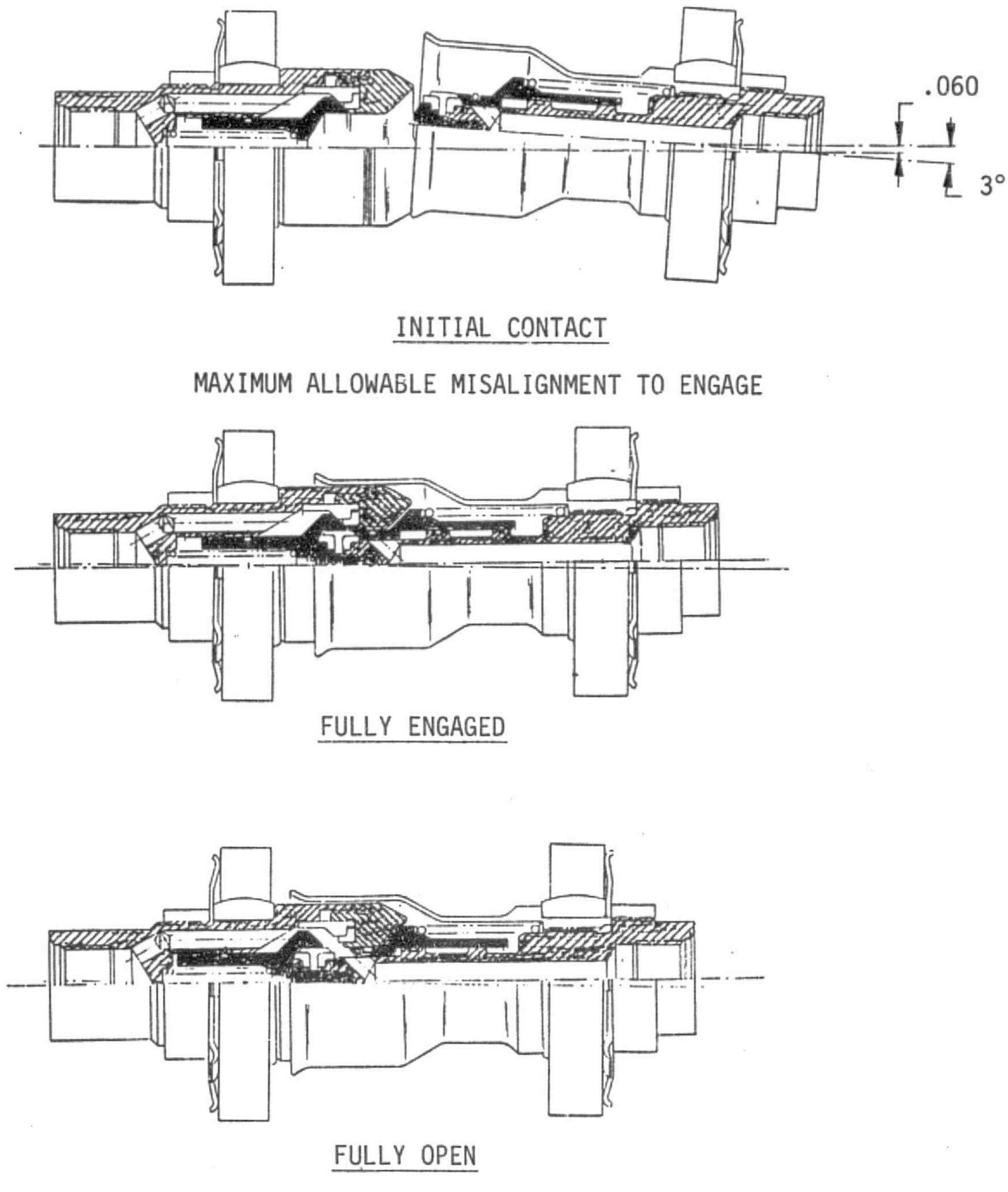

Figure 3.1.2-1 Fairchild Stratos NASA Disconnect (P/N 76300002)

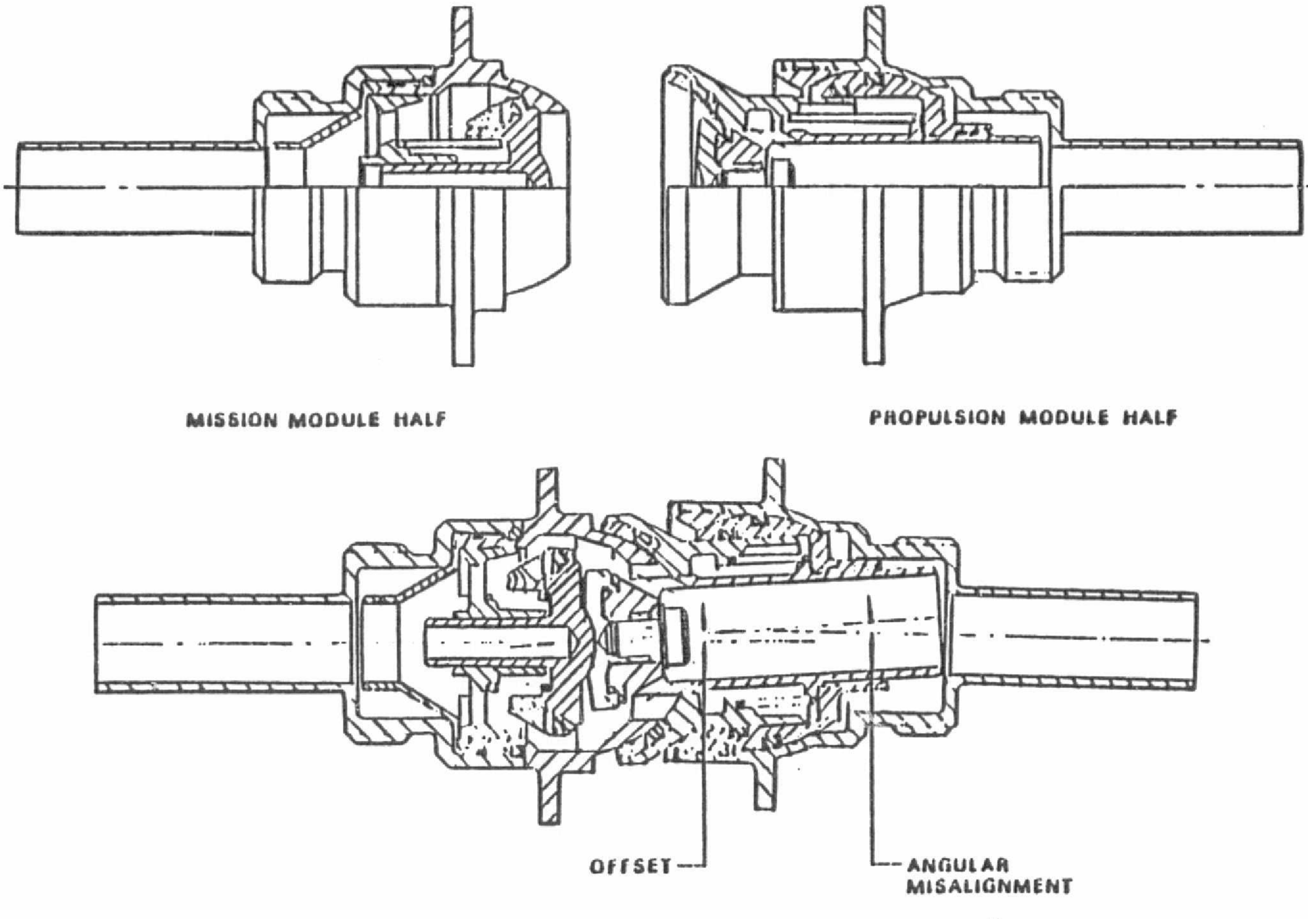

Figure 3.1.2-2 Fairchild-Stratos JPL Disconnect Unbalanced Design
(P/N 76366004)

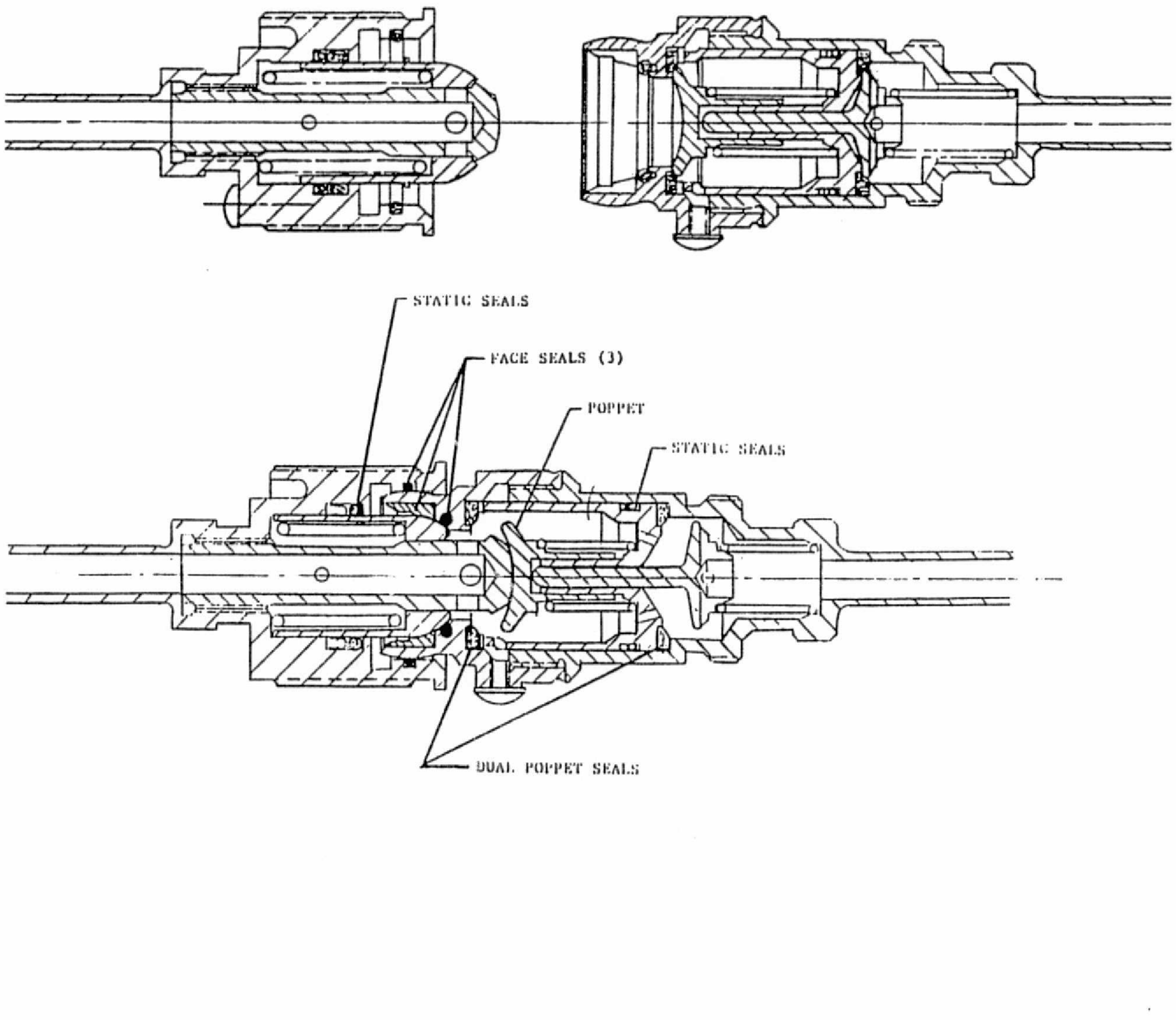

Figure 3.1.2-3 Fairchild-Stratos LEM Disconnect Valve P/N 553004

Table 3.1.2-4 Fairchild-Stratos Disconnects Design Requirements - NASA vs. JPL

DESCRIPTION	NASA DISCONNECT P/N 76300002	JPL DISCONNECT P/N 74366004
Application	Flight Interface	Flight Interface
Tube Size	0.5 in.	0.5 in.
Type of Disconnect	Breakaway-Self Sealing External Swivel Balanced Design	Breakaway - Self Sealing Internal Swivel Unbalanced Design
Attachment Method	Flexhose	Hardline
Alignment Offset Angulation	 0.06-in. $\pm 5°$ conical	 0.03-in. $\pm 5°$ conical
Operating Fluid	N_2H_4, MMH or Freon 21	MMH or Freon 21
Operating Pressure	0 - 300 psig	0 - 456 psig
Proof Pressure Mated Unmated	 440 psig 440 psig	 1650 psig 930 psig
Burst Pressure Mated Unmated	 1200 psig 1200 psig	 2200 psig 1860 psig
Operating Temp. Leakage Mated Unmated	-50 to +225°F 1×10^{-4} sccs GHe 1×10^{-4} sccs GHe	+10 to +150°F 1×10^{-3} sccs GHe(1) 5 scch GN_2(1)
Flow /ΔP MMH @ 1.1 lb_m/sec	 28 psid	 6.0 psid
Engagement Force	82 lb_f @ 300 psi	260 lb_f @ 300 psi
Spillage Volume	0.14 ml/cycle	1.0 ml/cycle
Life	500 cycles, 10 yrs.	200 cycles, 2 yrs.
Random Vibration	11.43 GRMS	11.39 GRMS
Weight	2.3 lb max.	1.2 lb max

(1) Qualification results were 1×10^{-7} sccs of He.

TABLE 3.1.2-5 Fairchild Stratos LEM Disconnect

DESCRIPTION AND APPLICATION

The interstage disconnect is a matched set of half-disconnect assemblies consisting of a dual series redundant, poppet-type ascent half disconnect and the coupling and actuating descent half. The interstage disconnect has been designed to maintain a flow of gaseous oxygen or ethylene glycol between the ascent and descent stages of a lunar module during launch and boost, orbit, descent, and lunar stay. Upon ascent, the disconnect halves separate and then act as a leakproof seal of the fluid in the lines.

DISCONNECT TYPE:	Breakaway
PROGRAM:	Apollo Lunar Module
DESIGN DATE:	April 5, 1968

SPECIFICATIONS

Operating Fluid:	Gaseous oxygen at 0 to + 100°F or liquid glycol
Ambient Temperature Range:	Connected, 0 to + 160°F; disconnected, ± 260°F (continuous) and +400°F (for 5 minutes)
Oxygen Operating Pressure:	1575 psia
Proof Pressure:	2100 psia
Burst Pressure:	3150 psia
Glycol Operating Pressure:	50 psia
Proof Pressure:	100 psia
Burst Pressure:	150 psia
Flow Rates:	
Oxygen:	10.0 lbs/min. at 1000 psia inlet; 30 psi (maximum) pressure drop
Glycol:	75 to 150 lbs/hour; 0.5 psi (max.) pressure drop.
Mating Seal Leakage:	10^{-4} scc/sec. (max total leakage) helium
Breakaway Force:	7.0 lbs (maximum required)
Mounting Provisions:	(See Installation Drawing)
Weight:	1.18 lbs (maximum, design)

All three valves have internal poppets for opening and closing. They can be redesigned to incorporate the third seal to satisfy the STS safety requirements and also can be fitted with leak sensing and gas purging piping. Fairchild is now developing new types of disconnect valves (Ref 26, Table 3.1.2-2) suitable for on-orbit refueling.

Another concept, proposed by Martin Marietta technology specialists, simplifies the disconnect design by eliminating the internal poppets and uses three in-line control valves on each side. The two halves of the disconnect would be self aligning and would contain triple redundant seals. Provision for leak testing with nitrogen or other inert gas after mating prior to opening the control valves, as well as for gas purge prior to demating can be easily incorporated. The design would utilize existing, proven control valves and would offer higher reliability, lower engagement/separation force, lower pressure drop and important cost savings.

For the preliminary design of the ground and flight servicer demonstration unit the use of the LEM disconnect (Figure 3.1.2-3) is recommended and should alternative designs be selected later, they would fit within the same envelope.

There is no space qualified design available for an in-line fluid coupling to be used for on-orbit component replacement (including tank). Some basic couplings used in ground equipment can be used, such as A/N threaded fittings. A special mechanism to engage the two halves, to torque the coupling and to test for leaks needs to be developed.

3.1.2.4 Existing Electrical Connectors - Among the flight qualified electrical/RF cable quick disconnects, the 882 series developed for MMS by G&H Technology Inc. (see Figure 3.1.2-4) can accommodate the highest number of wires (225 #16 for the 882-003). Engagement can be accomplished within a 20 degree cone with 0.12 in. misalignment.

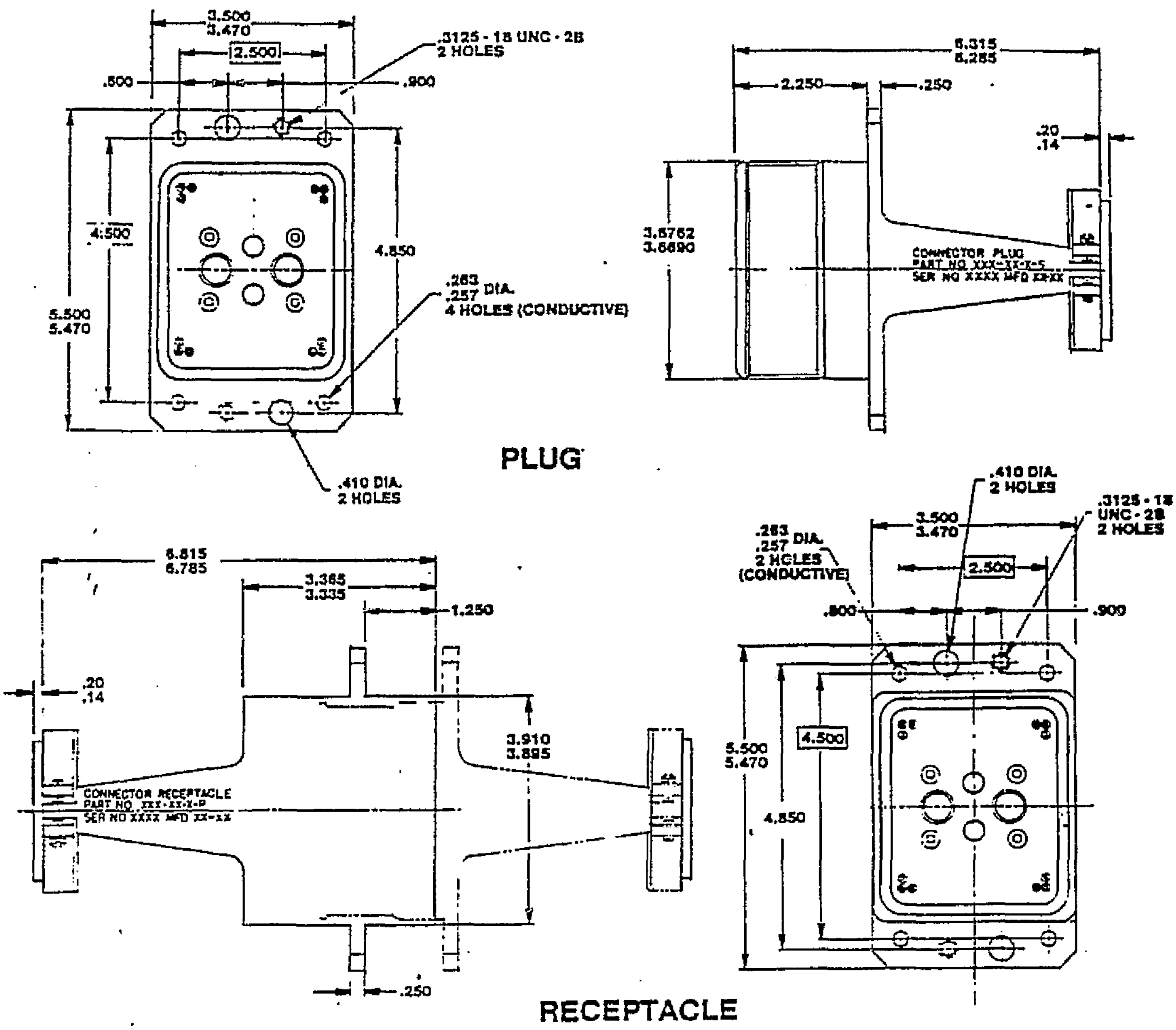

Figure 3.1.2-4 MMS type Electrical Connector

However its relatively high mate/demate force of 250 lbs and its weight in excess of 3 lbs may limit its application.

Another space qualified, self aligning electrical connector which can be used for the refueling interface is the Deutsch RSM09 series rack and panel subminiature connector (see Figure 2.1.2-5). It is produced in different sizes, has up to 91 wires, and features low mate-demate forces and a self-aligning feature with up to 1/16 in. misalignment capability.

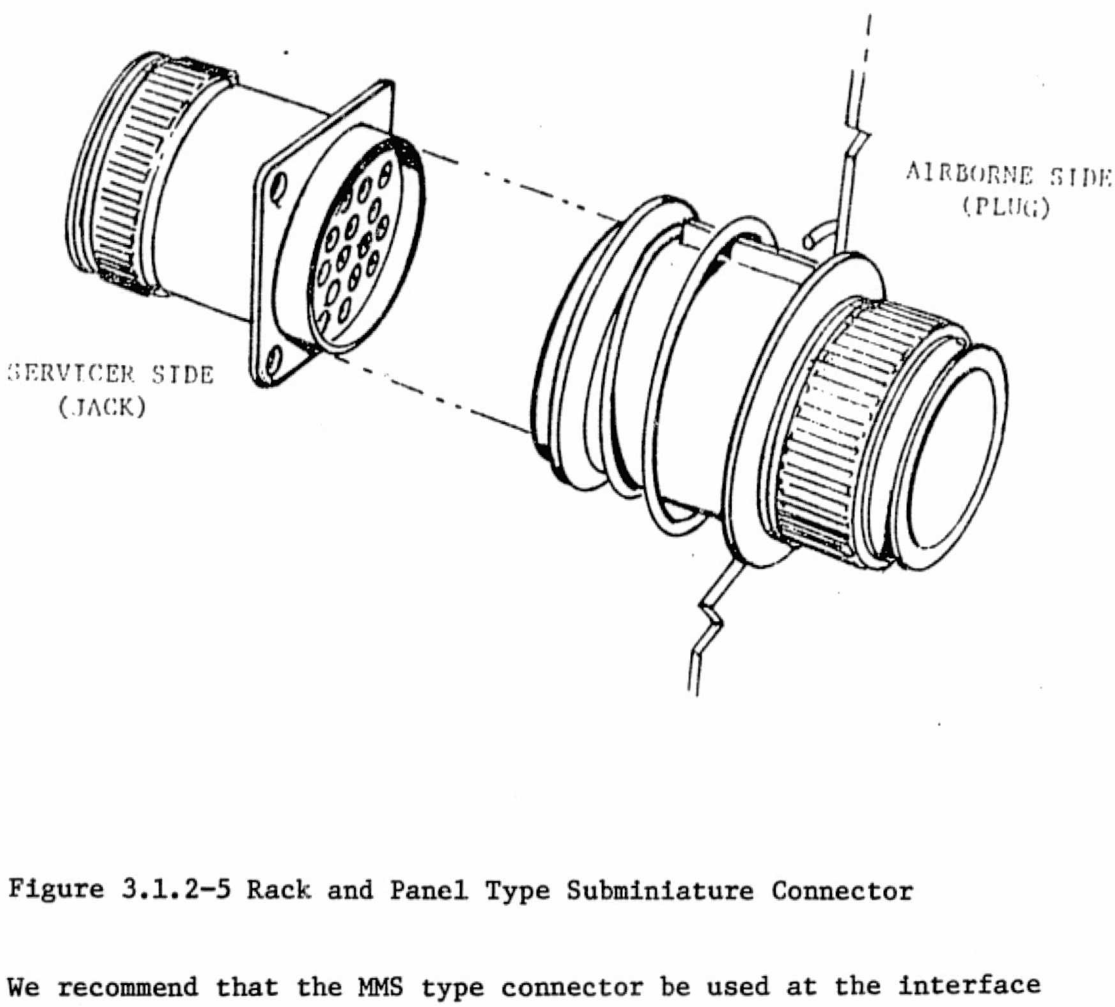

Figure 3.1.2-5 Rack and Panel Type Subminiature Connector

We recommend that the MMS type connector be used at the interface between module mockup and support structure in connection with MMS module exchange demonstrations and that the Deutsch connector be used as part of the refueling interface between servicer and serviced satellite.

3.1.2.5 System Requirements For Satellite Refueling Demonstrations - The requirements of the refueling/resupply interface (disconnects and mate/demate subsystem) are given in Section 3.1.2.2. Additional system requirements for satellite refueling systems are given in this section grouped by major areas of concern and with emphasis on those specific to the flight and ground demonstration units.

The requirements for the servicer configuration are:

1) Servicer shall be designed so that different types of servicing operations can be performed during the same mission, such as refueling/resupply and module exchange;
2) Servicer configuration shall allow minimizing the mission duration. One way of accomplishing this is by performing more than one task at a time, such as resupplying more than one fluid at a time or module exchange while refueling;
3) A solid docking interface between spacecraft and servicer is required. Mating and demating of the disconnect(s) shall be performed while the servicer is hard-docked to the spacecraft;
4) High fidelity of the refueling/resupply servicer demonstration shall be ensured by using real flight hardware or accurately duplicated equipment for the servicing interface.

The requirements for the fuel lines and electrical cables management system are:

1) The length of the fluid transfer/electrical lines shall be kept to a minimum;
2) The fluid lines and the electrical cables shall be prevented from tangling, abrading each other, or interfering with the arm, docking probe, stowage rack or other equipment or structures of the servicer or of the spacecraft;
3) The number of bends in the fluid lines shall be kept to a minimum. The line management system shall ensure a suitable minimum bend radius of the lines;
4) The line/cable management system shall assure servicing of all required locations (different spacecraft and/or multiple servicing locations);
5) The line/cable management system shall be simple and reliable.

The requirements for selection of a fluid type for the demonstrations include:

1) Initial ground and flight demonstrations may use water and air instead of the actual propellant and pressurant gases in order to minimize risk and cost;
2) In a subsequent phase of ground and flight demonstrations, as the disconnect valves, flexible lines and other specific hardware becomes available, the following fluids should be demonstrated:
    a) Earth storable propellants (N_2H_4, MMH, N_2O_4),
    b) Pressurant gases (GHe, GN_2),
    c) Cryogenic fluids,
        Propellants (LH_2, LO_2),
        Coolants (LHe, SfHe, LH_2, etc., see Table 3.1.2-6);
3) Commonality of design concepts and of servicing interfaces shall be emphasized while the disconnects will be specifically developed and designed for each type of fluid.

The thermal protection requirements include:

1) The design of the disconnects, mate/demate subsystem and the line management system shall provide adequate thermal protection to prevent freezing or overheating of the fluids to be transferred;
2) The refueling system shall condition the earth storable propellants to 70 ± 20°F.

The spillage requirement is:

1) Disconnect valve purge lines shall be connected to a vent "catch" system to prevent spillage.

The command and control requirements include:

1) The following real time control functions of the refueling/resupply servicer shall be provided from the ground control station (GCS) through the communication link of the carrier vehicle (or from the control console for the ground demonstration):

    a) Control of disconnects mate, demate, leak test and purge functions,
    b) Control of flow rate(s),
    c) Control of liquid and gas pressures,
    d) Control of valve on/off sequencing. Provide interlocks for critical functions;

2) The following measurements and monitoring of the refueling/resupply sevicer functions are required:
    a) Mass gauging (1/2 % accuracy) for fluids in spacecraft and servicer tanks,
    b) Critical pressures and temperatures in spacecraft and servicer systems,
    c) Valve position indication,
    d) Status monitoring of spacecraft and servicer systems.

The data requirements are:

1) The existing data system of the servicing ground demonstration, of the FSS and orbiter for cargo-bay demonstrations and of the carrier vehicle for free-flyer servicer should be used to fullest extent possible.

The software requirements are:

1) The software required for operating the refueling/resupply functions of the servicer shall be integrated with the servicer software in all configurations (ground servicing demonstration, cargo-bay servicing demonstration and free-flyer servicer).

The non-propulsion cryogenic requirements are:

1) Fluids to be transferred and their characteristics are shown in Table 3.1.2-6;

2) The following requirements apply to the disconnect valves:
    a) Low pressure (see Table 3.1.2-6),
    b) Low to zero leakage,

c) Spillage-minimize, but it is not a design driver,

d) Counter flow chiller for liquid helium,

e) Minimum thermal mass,

f) Remotely located/thermally insulated from propellant disconnects,

g) Ensure fluid/material compatibility,

h) Replaceable, insulated cover door,

i) Internal pressure relief for trapped cryogens,

j) Similar alignment requirements as for propellant/gas disconnects;

3) The following requirements apply to the transfer lines:

a) Counter flow chiller for liquid helium,

b) Insulated line for other liquids,

c) Minimum thermal mass,

d) Minimum length;

4) Operational requirements:

a) Provisions to be made for prechilling transfer lines to transfer temperatures,

b) Chill down gas to be routed to a safe disposal area,

c) Spillage-minimize, but it is not a design driver,

d) Transfer time nominally 8 hrs for a prechilled receiver,

e) Electrical connection needed across the servicing interface for valve actuation and status monitoring.

Table 3.1.2-6 Non-Propulsion Cryogenic Requirements

Cryogenic Liquids	Service Volume	Transfer Temperature	Transfer Pressure-Torr
Superfluid Helium	5000 Liters	1.8°K	20
Normal Helium	9000 Liters	4.2°K	760
*Hydrogen	3000 Liters	20°K	760
Nitrogen	TBD	77°K	760
*Argon	TBD	TBD	760
Oxygen	TBD	90°K	760
*Methane	TBD	TBD	760
*Neon	TBD	36°K	760
*Carbon Dioxide	TBD	TBD	TBD
*Ammonia	TBD	TBD	TBD

*Transferred as liquid and converted to a solid

The system failure prevention requirements are:

1) The electrical data equipment will not have a single point failure that could preclude the success of the refueling mission. Equipment that has a failure mode will fail safe or adequate means will be provided to detect the failure and take corrective action in sufficient time to prevent a hazardous situation.

The safety requirements are:

1) The disconnects for reactive fluids shall be separated and dissimilar/keyed fittings shall be used;
2) Material compatibility shall be a major design concern, to ensure long life and low potential corrosion rate;
3) Provide for automatic detection of hazardous conditions such as propellant leaks and overpressure, valve operation and/or interlock failure;
4) Control pressures and temperatures to eliminate adiabatic compression detonation potential;
5) Design structures and resupply systems for maximum pressures and accelerations with appropriate factors;
6) Minimize leakage/spillage due to component failure by using redundant seals and isolation valves, leak detectors and collection/containment/neutralization provisions;
7) Provide mechanical and electrical redundancy;
8) Use remote vents on servicer or spacecraft with plume directed away from sensitive surfaces. Control venting reaction forces;
9) Verification of the system status and safety shall be performed before starting resupply;
10 Provide EVA override or remote redundant system for disconnect demate. Use existing tools for EVA override actuation. Provide personnel protection (suit covers and gloves) and decontamination procedures;
11) Provide visibility/TV viewing for connection status monitoring;

12) Disconnects carrying hazardous fluids should incorporate appropriate caution flags, markers or plates for both ground and flight crew recognition;
13) In case of emergency, the cargo-bay servicer demonstration system shall be safed and demated in one hour maximum.

3.1.2.6 Refueling/Resupply Servicing Interface Selection - The main purpose of the refueling/resupply interface is to provide a connect/disconnect function for fluid and electrical lines between the servicer and the spacecraft to be serviced. In order to perform this function a mechanical connection is first established across the servicer/spacecraft interface to index and align the disconnect halves in proper positional relationship. Then the two halves of each disconnect are brought in contact and coupled in a translating motion provided by a mating/demating subsystem. Other functions of the refueling/resupply interface are automatic removal of dust covers prior to disconnect engagement, leak testing of the external seals before starting the fluid transfer, gas purge before disconnecting to minimize spillage, thermal protection to prevent freezing or overheating and EVA override, or remote redundant system, for contingent situations.

In order to minimize the impact on the spacecraft design the active side of the mating/demating subsystem should be located on the servicer side with only a smaller, self aligning, passive attachment and positioning device on the spacecraft side.

One conceptual design of a refueling/resupply interface unit is shown in a Satellite Services Analysis Study prepared by Lockheed Missiles and Space Co. for NASA/JSC (Ref 7, Table 3.1.2-2). It was designed for EVA operation in the orbiter cargo bay in connection with a hydrazine propellant transfer system (PTS) (See Figures 3.1.2-6 and-7).

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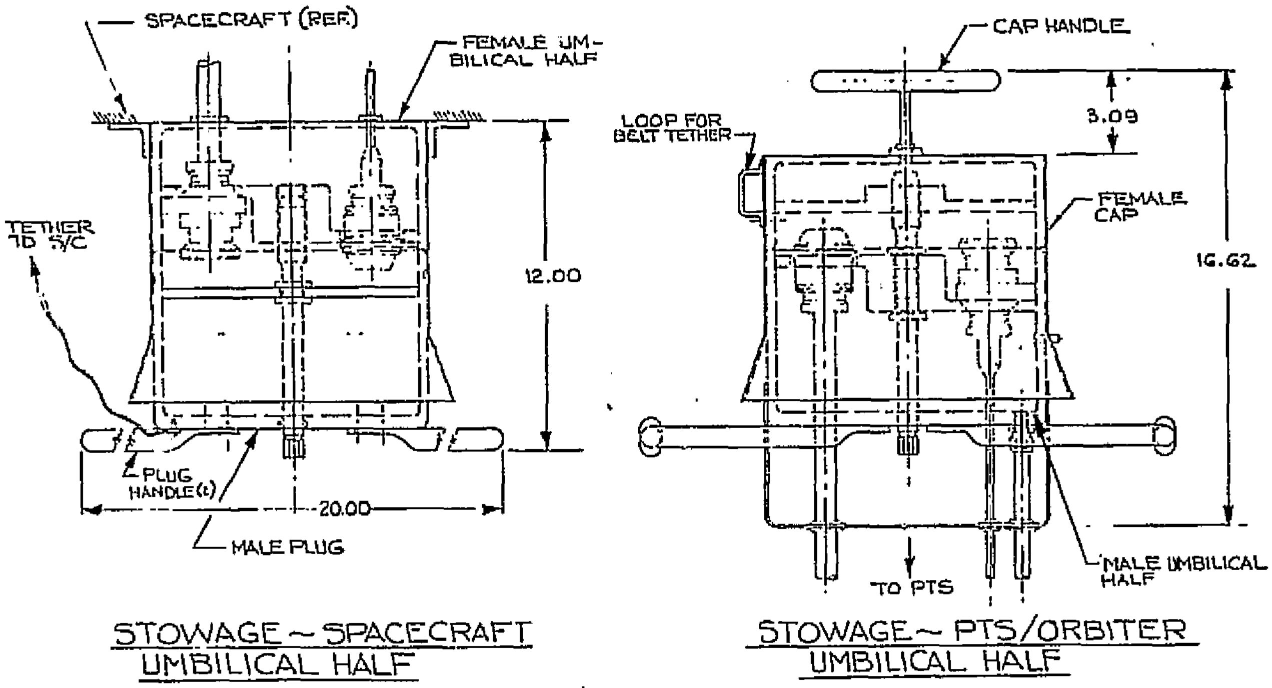

Figure 3.1.2-6 EVA Operated Refueling Interface Unit.

The unit is mated/demated by a jack screw that is EVA operated by an hexagonal socket wrench. It includes two disconnect valves, one for hydrazine and one for pressurant gas and has no electrical connections. In actual operation, the astronaut will manually remove the dust covers from the satellite disconnect halves, insert the PTS receptacle and crank it into the connected position.

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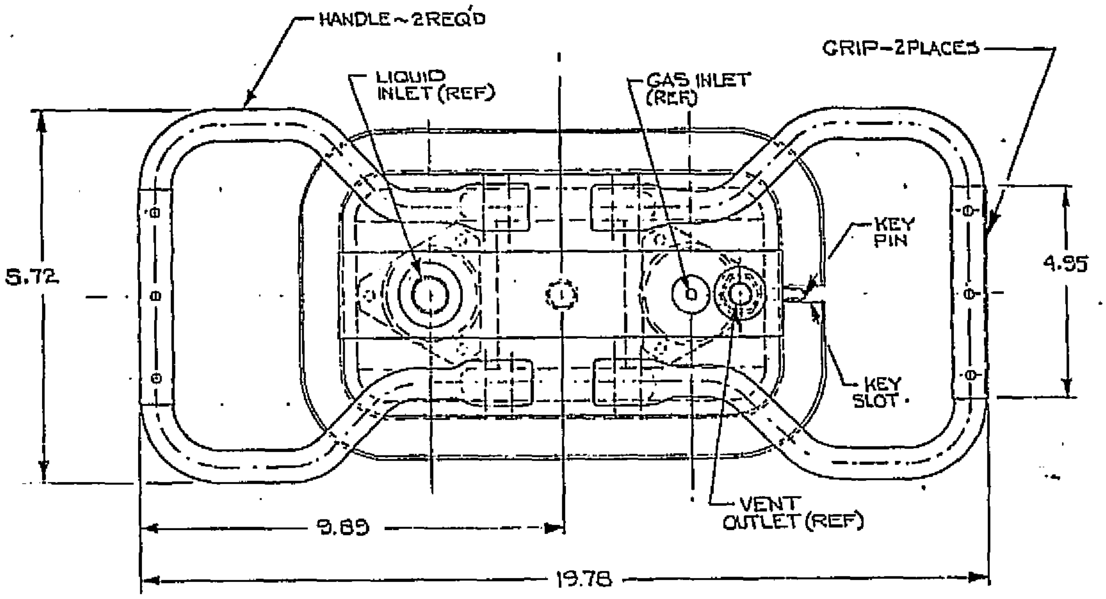

Figure 3.1.2-7 EVA Operated Refueling Interface Unit
View from the Orbiter Side

Mating in a wrong position is prevented by an indexing key in the housing and by having a male half and a female half of the two disconnects on one side of the unit. Color coding is also suggested. The mated housings are sealed to render them leak tight. If there is any leakage within the housing, a vent line is provided to allow leak detection and safe venting of the leak. The unit envelope is 19.78 x 8.72 x 16.62 in. including the EVA handles.

Among the advantages of the design are its simplicity and the indexing and leak detection/venting features. However, it does not include an electrical connector, is relatively bulky and does not have a test for leak after engaging the valves and prior to fuel transfer. It also requires extensive redesign in order to be adapted to remote operation by a servicer.

A refueling/resupply interface unit designed by Martin Marietta Aerospace jointly under a DoD contract and an internal research and development task, is shown in Figure 3.1.2-8.

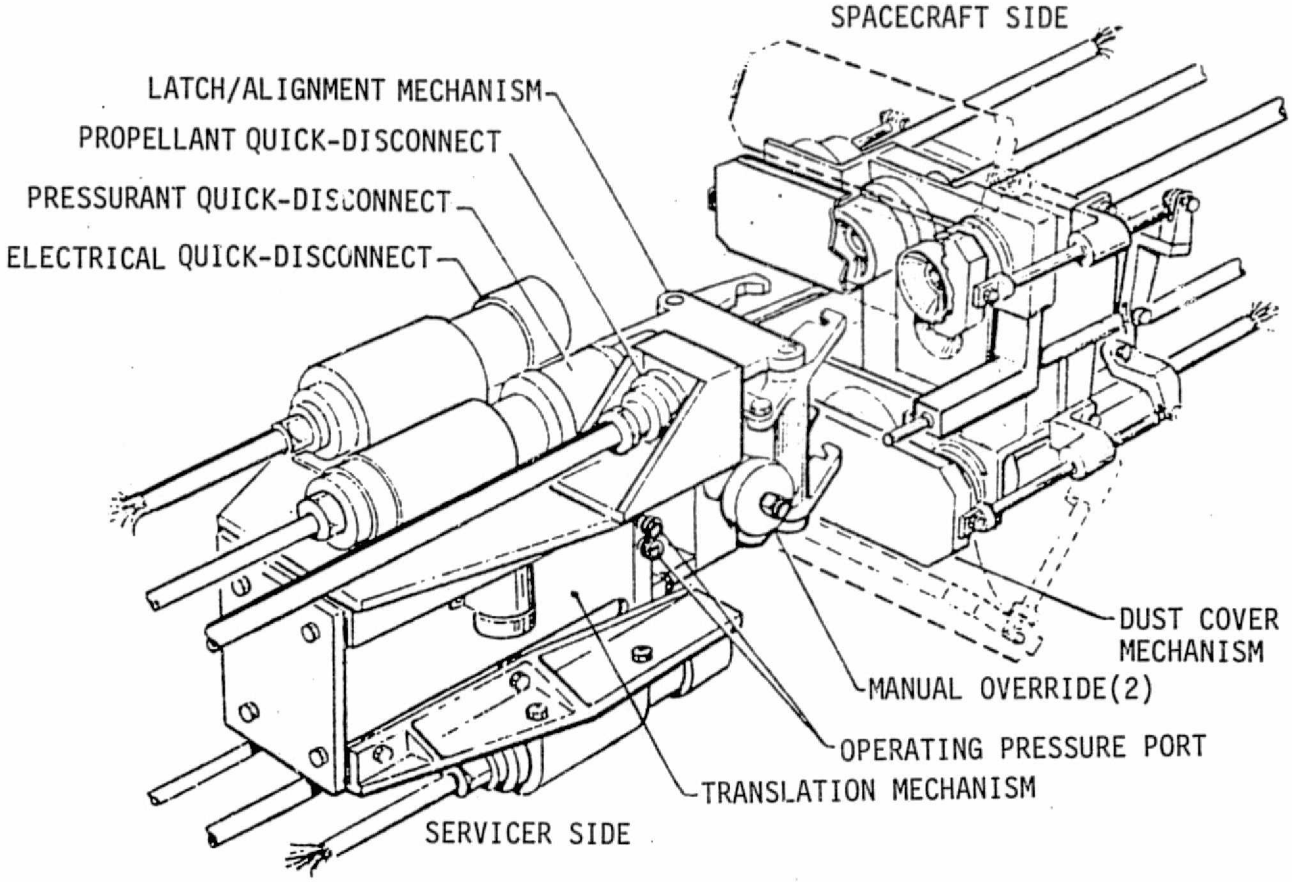

Figure 3.1.2-8 Refueling/Resupply Interface Unit

The unit is comprised of a mate/demate subsystem which can connect various sets of disconnect valves and electrical connectors depending on the application. The mate/demate subsystem includes a latch/alignment mechanism based on the IOSS end effector design (Fig. 3.1.2-9), a translation mechanism (Fig. 3.1.2-10), and a dust cover removal mechanism (Fig. 3.1.2-11).

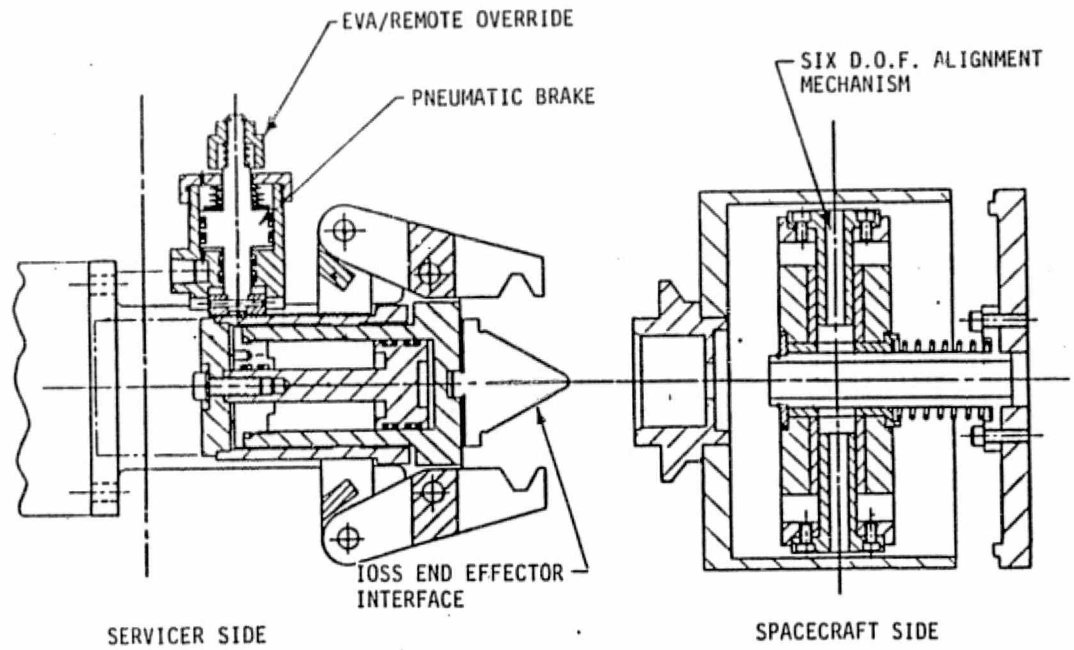

Figure 3.1.2-9 Latch/Alignment Mechanism

Both the latch/alignment and the translation mechanism are pneumatically actuated. The speed of actuation is controlled with flow controls and a brake on each mechanism is applied whenever the actuation pressure is relieved. Each brake has an override that can be actuated either remotely or by EVA for demating.

An electrical mate/demate and latching mechanism was preliminarily designed and traded against the pneumatic one. The pneumatic mechanism is preferred because is smaller, lighter and has higher reliability. It uses pressurized gas, which is available in almost any refueling/resupply systems, and has relatively low gas consumption. The translation mechanism can be stopped in intermediate positions, using limit switches. By relieving the pressure, the brake is automatically applied. This feature can be used for leak test before fluid transfer, or for gas purging, prior to demating.

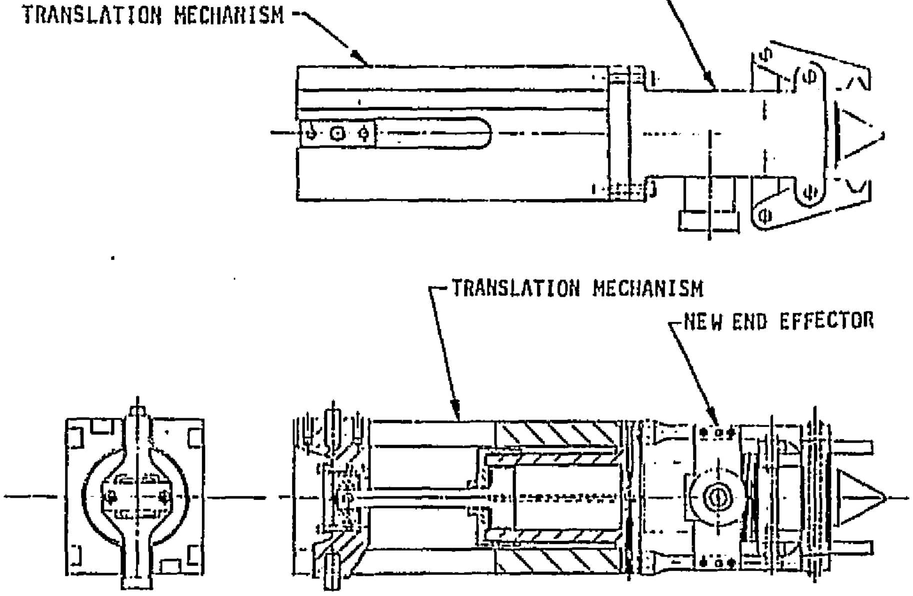

Figure 3.1.2-10 Translation Mechanism

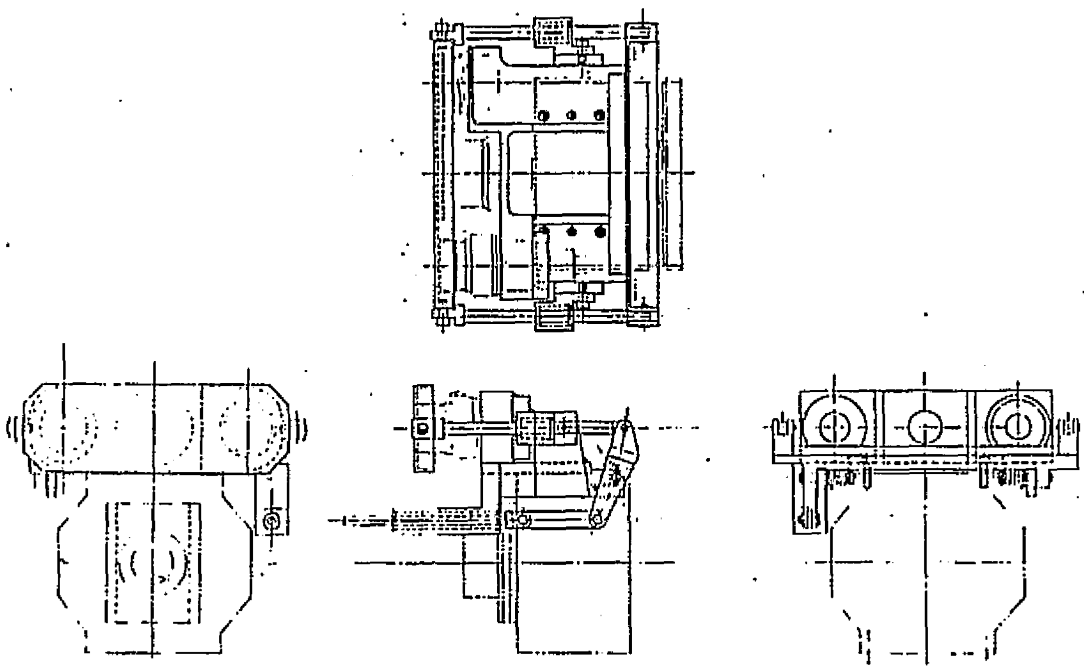

Figure 3.1.2-11 Dust Cover Mechanism

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Various disconnect configurations can be accommodated on the two mounting plates of the translation mechanism allowing flexibility and redundancy (Figs. 3.1.2-12 and 3.1.2-13).

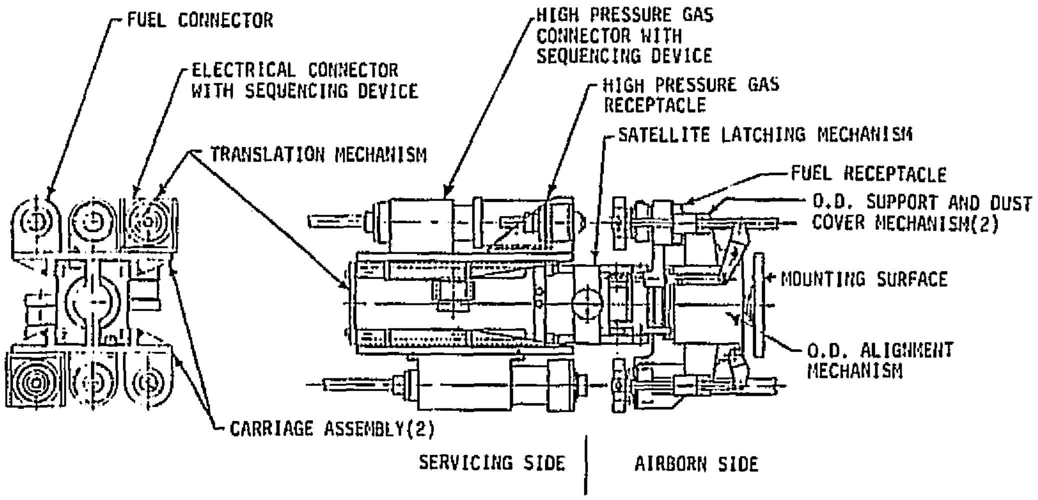

Figure 3.1.2-12 Refueling/Resupply Interface Unit

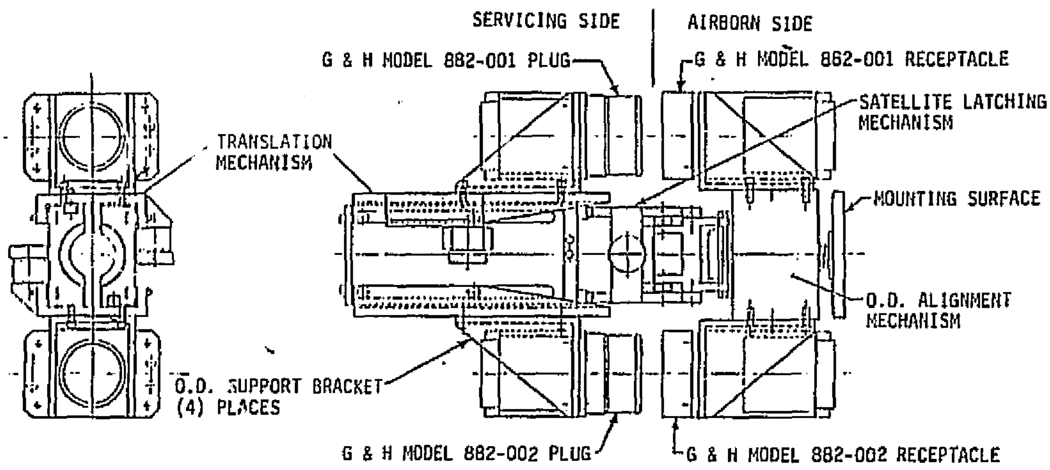

Figure 3.1.2-13 Dual Electrical Disconnect Interface Unit

The latch mechanism is a proven design from the IOSS. It makes possible a high degree of commonality for all servicing interfaces, including the one used for module exchange. The unit can be used at the end of a servicer arm or as a separate umbilical for refueling/resupply or for electrical cable connection.

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This design satisfies all the requirements and is recommended for use as refueling/resupply and electrical remote umbilicals in connection with the servicer.

3.1.2.7 Servicer Configuration Selection for Refueling/Resupply Demonstrations - Four candidate solutions were considered (Table 3.1.2-1) and are described below together with a discussion of their advantages and disadvantages and their effects on the spacecraft and servicer.

1) Refueling/Resupply Interface Unit Attached to the Docking Probe. One or more units can be attached to the docking probe of the servicer vehicle as shown in Figure 3.1.2-14. The multiple fluid lines and electrical cables pass through the center of the shoulder joint of the servicer. The corresponding disconnects on the spacecraft are located around the docking area. The servicer may have a two arm configuration as shown or a single arm like the IOSS. If the IOSS arm is used, the shoulder joint must be redesigned to allow fluid lines and cables to pass through its center or if the fluid lines are routed on the outside the reach envelope of the servicer will be reduced.

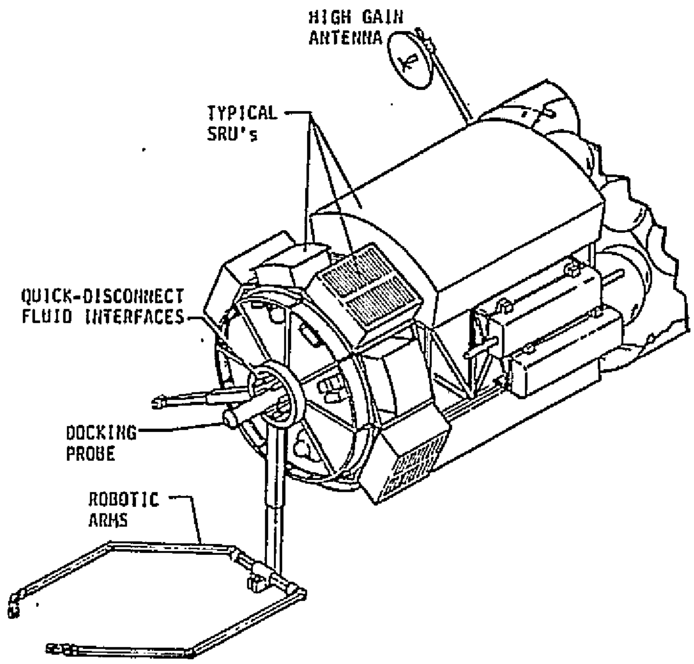

Figure 3.1.2-14 Refueling/Resupply Units Attached to the Docking Probe

Advantages:

- The servicer arm is free to perform other tasks during the refueling/resupply operations (module changeout, removal of dust covers, actuation of overrides on refueling/resupply units, inspection of the mated disconnects, etc).
- Simpler line/cable management system.
- Shorter fluid lines.
- No latch/alignment mechanism needed, the function is performed by the docking probe.
- Simpler controls.

Disadvantages:

- Potential risk of damaging disconnect valves during docking. Impact speeds up to 0.5 fps and/or up to $\pm 15°$ misalignment are possible during docking and therefore impact shields are required to protect the disconnects.
- Requires redesign of shoulder joint to allow the cables and fluid lines to pass through its center, or if routed on the outside of the shoulder joint, they will reduce the reach envelope of the servicer arm.
- Less flexibility in the design of the spacecraft, all disconnects must be close to the docking area.
- Separation of the disconnects for reactive fluids is more difficult to achieve.
- Requires a dust cover removal mechanism for the servicer disconnect halves.

Impact on spacecraft:

- Less flexibility in design.
- Shields required to protect disconnect valves from docking impact.

Impact on servicer:

- Shoulder joint redesign.
- Docking probe and stowage rack customized for the mission.

2) Refueling/Resupply Interface Unit(s) Stored on Stowage Rack and Deployed by the Servicer Arm.

For each type of fluid and its pressurant (when applicable), a module is mounted on the stowage rack, containing the tanks, support structure, plumbing, valves, monitoring instrumentation and controls, flexible lines, cables and their management system, thermal protection and refueling/resupply interface unit, as shown in Figure 3.1.2-15.

In the stowed position, the module is flush with the front face of the stowage rack allowing free movement of the servicer arm. The fluid line/cable management system is secured with latches during launch. It may consist of a commercially available metal flexible conduit which limits the minimum bend radius, while protecting and containing the cables and the flexible lines, or it may consist of a folding mechanism with straight support bars and joints and a means of attaching the fluid lines and cables and of controlling their minimum bend radius.

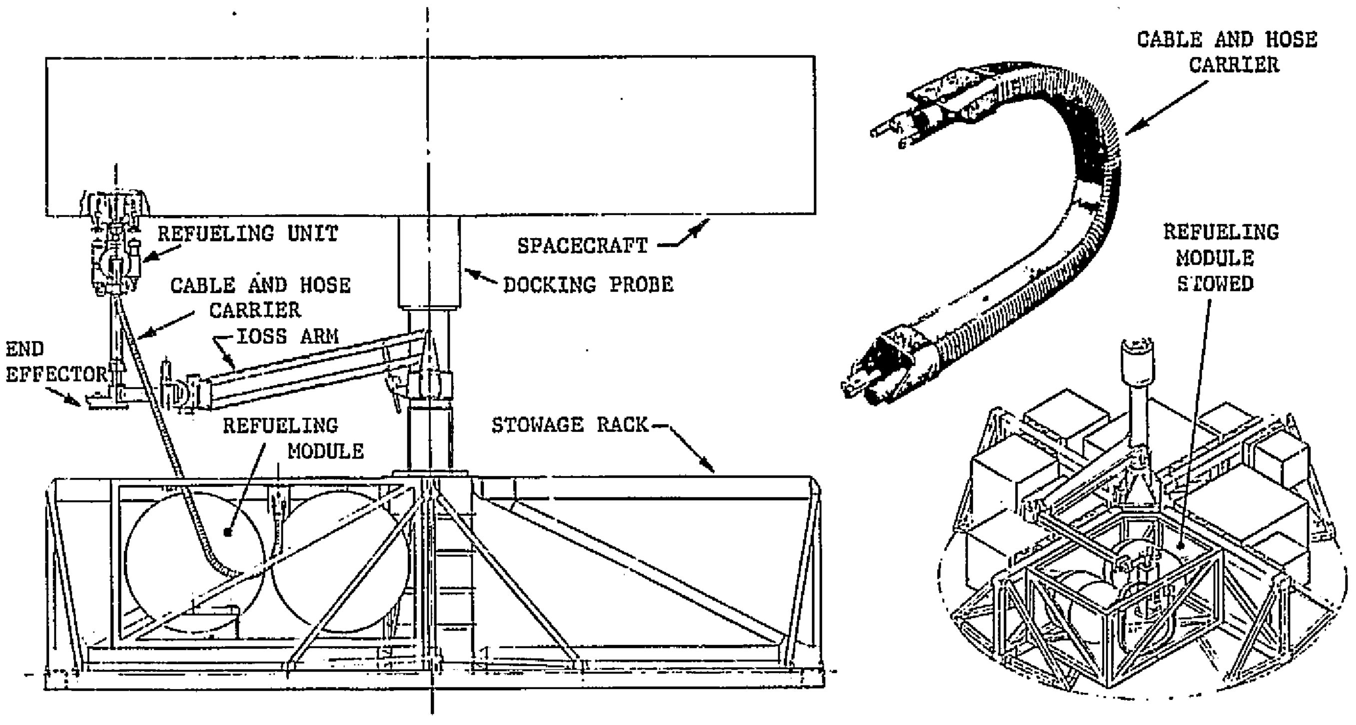

Figure 3.1.2-15 Refueling/Resupply Module on Stowage Rack

The interface between the refueling/resupply module and the servicer is a simple, mechanical fastening system and electrical connections for control and monitoring functions. Integration of the module(s) with the servicer is a simple operation which can be performed at the space station, at the orbiter cargo bay or on the ground, allowing a large degree of operational flexibility.

The servicer arm is used to deploy the refueling/resupply interface unit and to attach it to the disconnects on the spacecraft, anywhere within its reach envelope. More than one fluid can be transferred at the same time by connecting additional refueling/resupply interface units from additional modules. During the fluid transfer operations the servicer arm can be used for other tasks, such as equipment module changeout, inspection, etc. The arm reach envelope would be somewhat limited. However, the arm may be able to pass under the connected refueling lines.

Advantages:

- Servicer arm is free during long periods of fluid transfer operations to do to other tasks.
- Servicer arm can be used for actuating overrides, dust cover removal, inspection of mating of disconnects.
- No modifications of the servicer arm are required.
- Modular refueling/resupply system is easy to integrate with the servicer.
- Separation of disconnects for reactive fluids is easy.
- Flexibility for spacecraft design-fewer constraints in locating the disconnects.
- Stationary dust covers on the servicer side are easy to accommodate on the module.
- Easier to integrate with ETU for ground demonstrations (lower cost).

Disadvantages:

- Needs fluid line/cable management system(s) - one for each refueling/resupply module.
- Attach/alignment mechanism(s) needed, in addition to the arm end effector.
- Servicer envelope for equipment module changeout is somewhat limited during refueling/resupply.

Impact on spacecraft:

- Minimal - provision for compatible refueling/resupply interface with the servicer is required.

Impact on servicer:

- No impact on arm.
- Fluid line/cable management system needed.

3) <u>Refueling/Resupply Interface Unit Attached to the End Effector of the Servicer</u>.

The servicer arm end effector is replaced by the refueling/resupply interface unit, using an offset wrist segment between the Y and Z drives (Fig. 3.1.2-16 and -17). A power takeoff, TV camera and lights are also provided. No management system for the fluid lines and cables is required. They are attached to the servicer arm. Adequate loops must be provided at each joint, including the shoulder roll joint, to allow free movement of the arm.

Advantages:

- No fluid line/cable management system required - the servicer arm performs this function.
- Only one end effector to build.
- Flexibility in spacecraft design - Disconnects can be located anywhere within the reach envelope of the arm.
- Full reach envelope of the servicer is available.
- Dust covers on the servicer side can be stationary by mounting them on the stowage rack

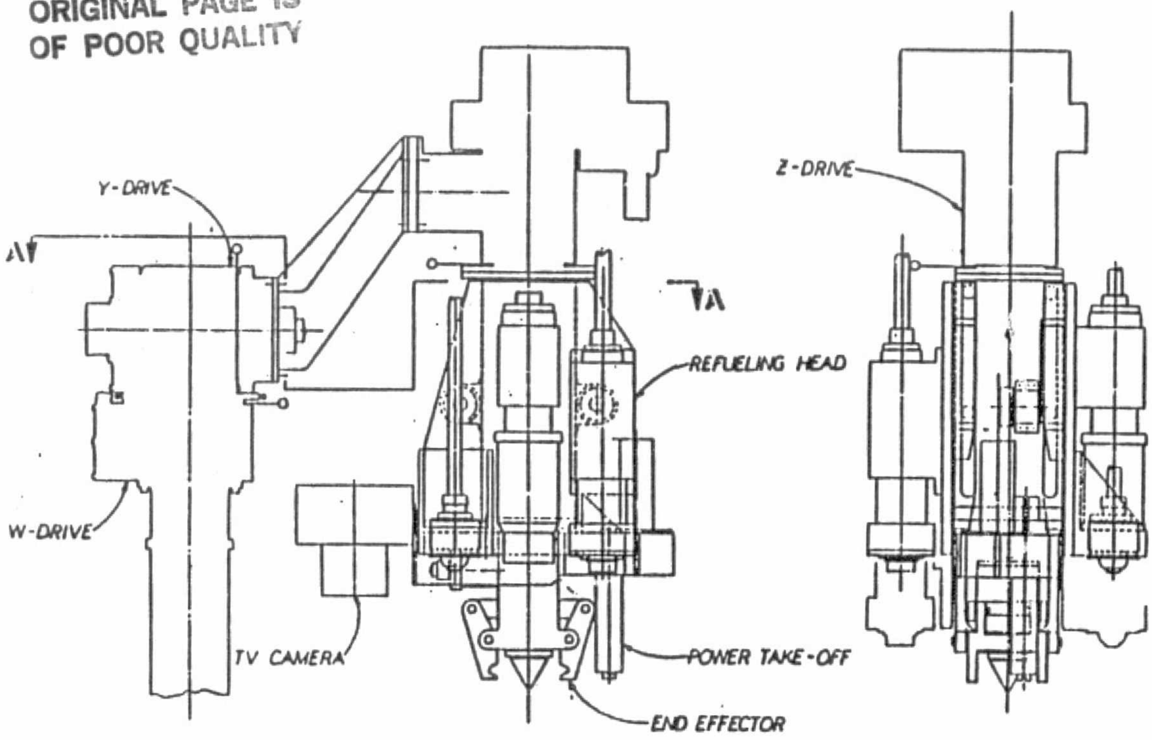

Figure 3.1.2-16 Refueling/Resupply Interface Unit Used as End Effector
(For View A-A, see Fig. 3.1.2-17)

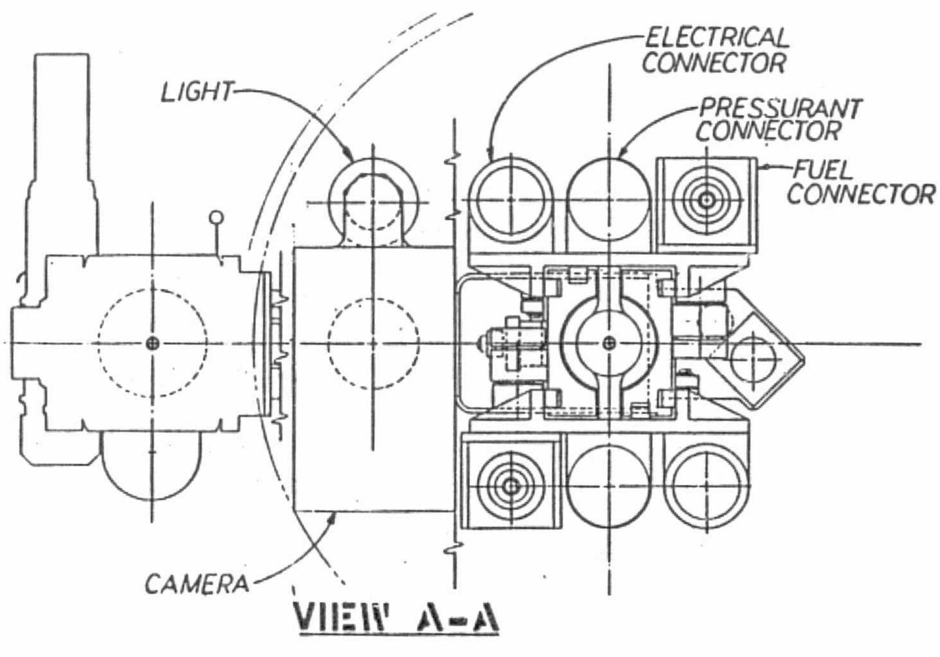

Figure 3.1.2-17 Refueling/Resupply Interface Unit Used as End Effector - View A-A of Figure 3.1.2-16

Disadvantages:

- Servicer arm not available for other uses during long refueling/resupply operations.
- Multiple fluid lines attached along the arm may limit its movement
- Heavier arm - less load capability for 1-g demonstration.
- More cycles of fluid line flexing - less life.
- End effector less compact - arm cannot operate in tight spots.
- Servicer arm cannot be used to actuate overrides - special, redundant mechanism needed.
- Integration of mission specific refueling/resupply hardware with the servicer is difficult. Disassembly of harness and end effector is required for each different mission.
- Integration with ETU for ground demonstration requires new counterbalancing.
- Separation of disconnects for reactive fluids is difficult.

Impact on spacecraft:

- Minimal impact - compatible disconnects required.

Impact on servicer:

- Modification of end effector and wrist segment of the arm is required.
- Modifications of the fluid lines along the arm needed between two different refueling/resupply missions.
- Decreased load capacity in 1-g.

4) <u>Auxiliary Servicer Arm(s) Dedicated to Refueling/Resupply Operations</u>.

Refueling/resupply operations are performed at the same time with other servicing tasks such as equipment module changeout by using one or more dedicated servicer arms for fluid transfer in addition to the main servicer arm. For each type of fluid to be transferred, the servicer is fitted with a modular refueling/resupply system comprised of tanks, support structure, plumbing, valves, monitoring instruments, controls, thermal protection system and an arm of the same type as the servicer main arm, or a simplified version, fitted with fluid lines, electrical cables and a refueling/resupply interface unit as an end effector (Fig. 3.1.2-18). Latches are provided on the module structure to support the arm in a stowed position. A simple interface with the servicer includes mechanical fasteners and electrical disconnects for monitoring and control functions. Integration of the mission specific refueling/resupply module with the servicer is simple and can be performed at the space station, in the cargo bay or on the ground allowing considerable operational flexibility.

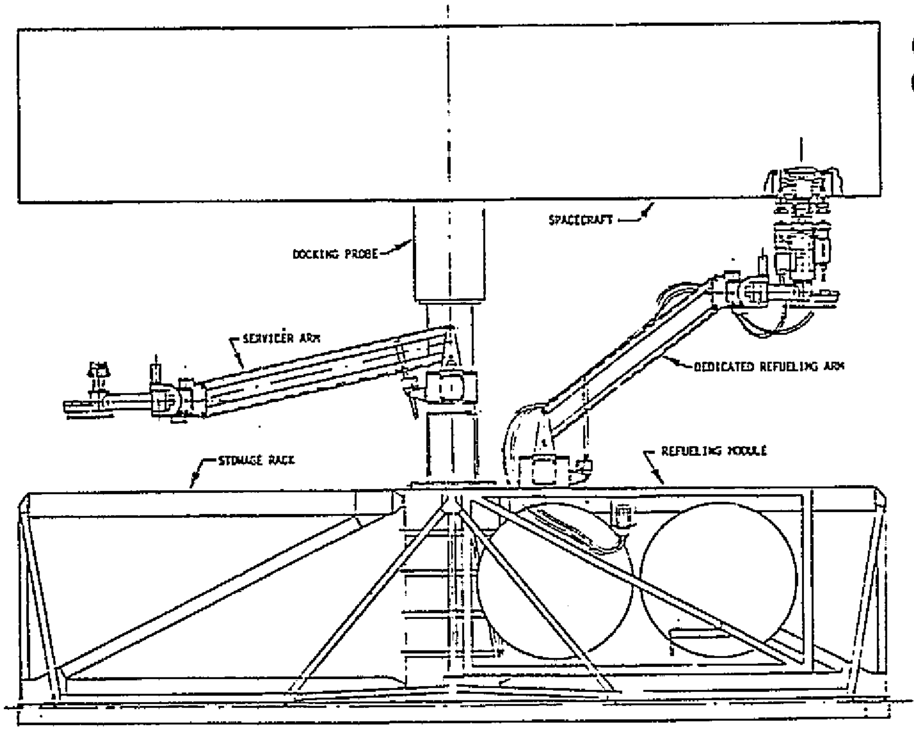

Figure 3.1.2-18 Refueling/Resupply Using Dedicated Servicer Arm

Advantages:

- The main servicer arm is free to do other tasks during long fluid transfer operations.
- Servicer arm can be used to actuate overrides on the refueling/resupply interface unit, inspection of disconnect mating, dust cover removal, etc.
- More flexibility in the design of the spacecraft, fewer constraints on the location of disconnects.

Advantages (Continued):

- Separation of disconnects for reactive fluids is easy by using separate modular units.
- Modular refueling/resupply system is easy to integrate with the servicer.
- Dust covers on the servicer side are easy to accommodate in the module, where the interface unit is stowed - no mechanism required.

Disadvantages:

- Increased complexity, weight and cost - multiple arms.
- More complex controls, require coordination and/or collision avoidance between two or more arms.
- Reduced reach envelope for the main servicer arm.

Impact on spacecraft:

- Minimal, compatible disconnects required

Impact on servicer:

- No impact on the main servicer arm design
- Reduced envelope for module changeout

In a coarse screening process, methods 1) and 4) (see Table 3.1.2-1) were eliminated because of mechanical complexity, high cost and high risk level due to docking impact or multiple arm coordination.

The remaining two candidate solutions were traded off based on criteria derived from the requirements. In comparing the two candidates for each criterion, a (+) was assigned for an advantage and a (-) for a disadvantage for Method 3). Weighting factors were assigned to various criteria and a total weighted score was computed as shown in Table 3.1.2-7.

Table 3.1.2-7 Refueling/Resupply Servicer Configuration Tradeoff

Criteria	Weight	Method 2): Refueling Units on Stowage Rack	Method 3): Refueling Unit Replacing Arm End Effector	
SYSTEM EFFECTIVENESS				
- Simultaneous Servicing Operations Performed	10	Yes	No, Single Arm	(-)
- Separation of Reactive Fluids	8	Easy	Difficult	(-)
- Modular System, Easy to Integrate with Servicer	10	Yes	No	(-)
- Servicer Arm can be Used for Override Actuation	5	Yes	No	(-)
RELIABILITY				
- Number of Flexing Cycles of Fluid Lines	10	Low, Dedicated Units	High, Lines Bend Whenever Arm Moves	(-)
RISK				
- Development Work Required; Impact on Schedule	9	Line/Cable Management System Development	Line/Cable Harness Attached to Arm	(+)
- Margin of Safety; Performance Estimates vs Requirements	3	Servicer Arm Reach Envelope Limited During Fluid Transfer	Full Arm Reach Envelope Available	(+)
COST				
- System Complexity and Modularity Affecting Cost	5	More Cost Due to Modular Design and Separate Line/Cable Management System(s) and End Effector(s)	Simpler System, But Less Flexible	(+)

(+) Advantage

(-) Disadvantage
Net 26 Negatives
(Weighted Score)

(0) Approximately the same

In conclusion, method 2) of connecting the refueling/resupply interface, using modular units attached to the stowage rack better satisfies the system requirements and is recommended for ground and flight refueling/resupply demonstrations.

3.1.3 Representative Satellite Modules and their Attachment Mechanisms

Existing designs of equipment modules suitable for on-orbit satellite servicing were reviewed and the requirements for the modules to be used in the ground and flight servicer demonstrations were defined. A set of modules was selected and recommended for the ground and flight servicer demonstrations (see Table 3.1.3-1).

The Space Telescope was designed for on-orbit servicing through module exchange by EVA. The module retention system and the equipment position on the spacecraft are such that the Space Telescope is less adaptable for remote on-orbit maintenance and repair using the servicer.

The Multimission Modular Spacecraft (MMS) was also designed for remote on-orbit subsystem module changeout. The attachment system of the MMS module is not compatible with any of the existing remote manipulator arm designs. However, the IOSS servicer can change out MMS type modules by using special adapters. A description of the MMS module and the changeout method is given in Section 3.1.1.

Table 3.1.3-1 Types of Modules to be Demonstrated

– MMS Type Module and MMS/IOSS Tool Adapter
– 24 in. IOSS Cube Module with Side Mounting Interface Mechanism
– Communications Satellite Module (Design TBD)
– AXAF Focal Plane Instrument Module (Design TBD)
– Smaller Modules, Component Level (Design TBD)
– "Thermal Cover" Module Removal or Hinge/Latch Actuation (Design TBD)

The Integrated Orbital Servicing System Study* analyzed 683 modules from 30 different serviceable spacecraft in order to determine the requirements for the size and weight of the IOSS modules as well as position and direction of removal of the modules. Following are the conclusions of that study:

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* Martin Marietta Integrated Orbital Servicing Study Follow-On, Final Report April 1978, Vol. II MCR-77-246 Contract NAS8-30820 SA-5

1) An axial and near-radial module changeout capability of the servicer is required;
2) Changeout of modules on the stowage rack should be axial only;
3) Interface mechanism and module size and weight as shown in Table 3.1.3-2, for the flight unit.

Table 3.1.3-2 Replaceable Module Characteristics from the IOSS Study

Module Max. Size (cube)	Module Max. Weight (lb)	Interface Mechanism Size	Interface Mechanism Weights (lb)			
			Bottom-Mounting		Side-Mounting	
			Receptacle	Baseplate	Receptacle	Baseplate
17 in.	75	17 in.	2.6	12.8	3.4	9.0
26 in.	200	26 in.	3.5	17.0	4.5	12.0
40 in.	400	40 in.	5.3	25.5	6.8	18.0

The Engineering Test Unit (ETU) of the IOSS was designed to accommodate servicing of a one-tier spacecraft with module exchange being in the axial or radial directions. The servicer mechanism can replace modules in off-axis directions also.

The module interface mechanisms provide the structural attachment between a module and the spacecraft or the stowage rack. It also provides the alignment and mating/demating forces for the connectors. The interface mechanism has two parts--a baseplate that is fastened to the module and a baseplate receptacle that is fastened to the spacecraft or to the stowage rack. The baseplate receptacle is passive. The baseplate has the linkages, cams, and rollers that latch the baseplate into the receptacle. The baseplate mechanism is mechanically driven from the servicer end effector. The interfaces of this mechanism are with the modules, the servicer end effector, the spacecraft, and the stowage rack.

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As the interfaces between the interface mechanism and the module and the spacecraft both seem to lie within the spacecraft designer's usual responsibilities, it would be possible to leave these design aspects up to the spacecraft designer. However, the interface with the servicer mechanism end effector and its mechanical drive system would have to be standardized across all interface mechanisms. Similarly, the method for attaching the interface mechanism baseplate receptacle alternatives into the stowage rack would also have to be standardized. In this way, a single--or few--stowage rack designs could be used for all missions.

Two types of interface mechanisms were used in the module exchange demonstrations with the ETU. The side mounting interface mechanism (see Figure 3.1.3-1) and the base mounting interface mechanism (see Figure 3.1.3-2). The two interface mechanisms are functionally equivalent. They have the same interface with the servicer end effector, can handle equivalent size modules, can incorporate the same connectors and use the same type of status indicators. Either concept can be used, depending on the spacecraft application.

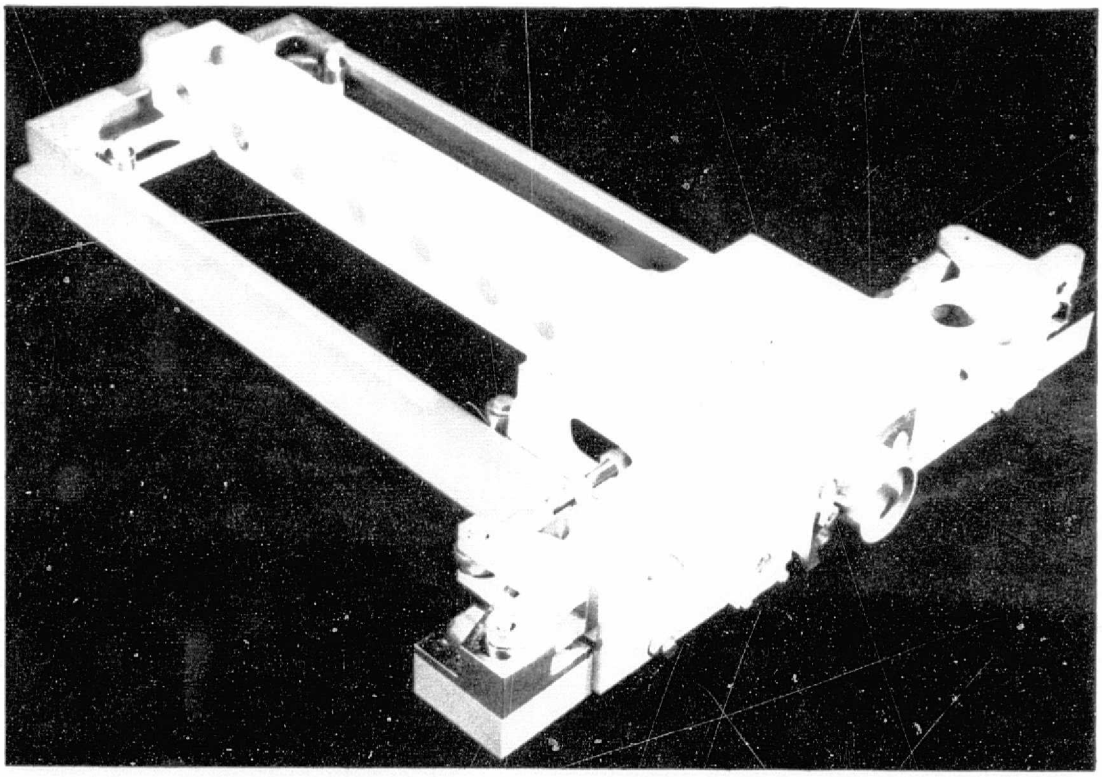

Figure 3.1.3-1 Side Mounting Interface Mechanism

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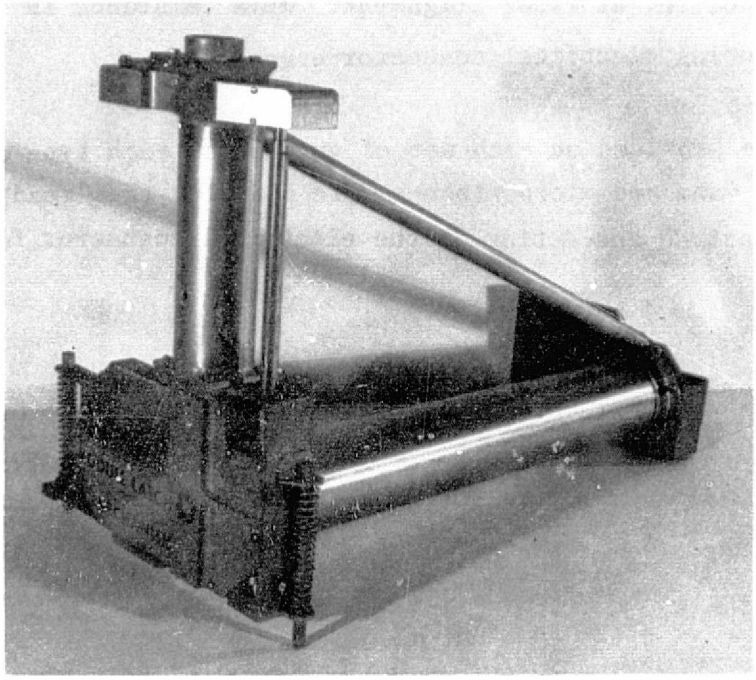

Figure 3.1.3-2 Base Mounting Interface Mechanism

The side interface mechanism assembled with a cubic module is shown in Figure 3.1.3-3. The module used in the demonstration was a 24 in. cardboard cube configured for minimum weight with adequate strength. The mechanism uses a three point, nonredundant, attachment system so spacecraft thermal and structural loads do not pass through the module. The bell crank linkage is driven via a worm and gear from a motor on the end effector. A spring-loaded self-aligning tongue in a slot accomplishes the mechanical interface. The linkage starts engagement with a low force that gradually increases to 200 lb as the links approach an over-center position. Total travel is 1-3/4 inches.

The attach cone has a ±3/4 in. capture volume, while the baseplate-to-guide capture volume is ±1/2 in. This large capture

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volume is gradually reduced during engagement by the shape of the guide rails to less than 0.001 in. at final alignment. This tolerance is less than 0.005 in. during electrical connector engagement.

Status indicators were provided on each set of guides for each type of interface mechanism. Cams and microswitches were used for the "ready" and "unlatched" indications and mating of the electrical connector for the "latched" signal.

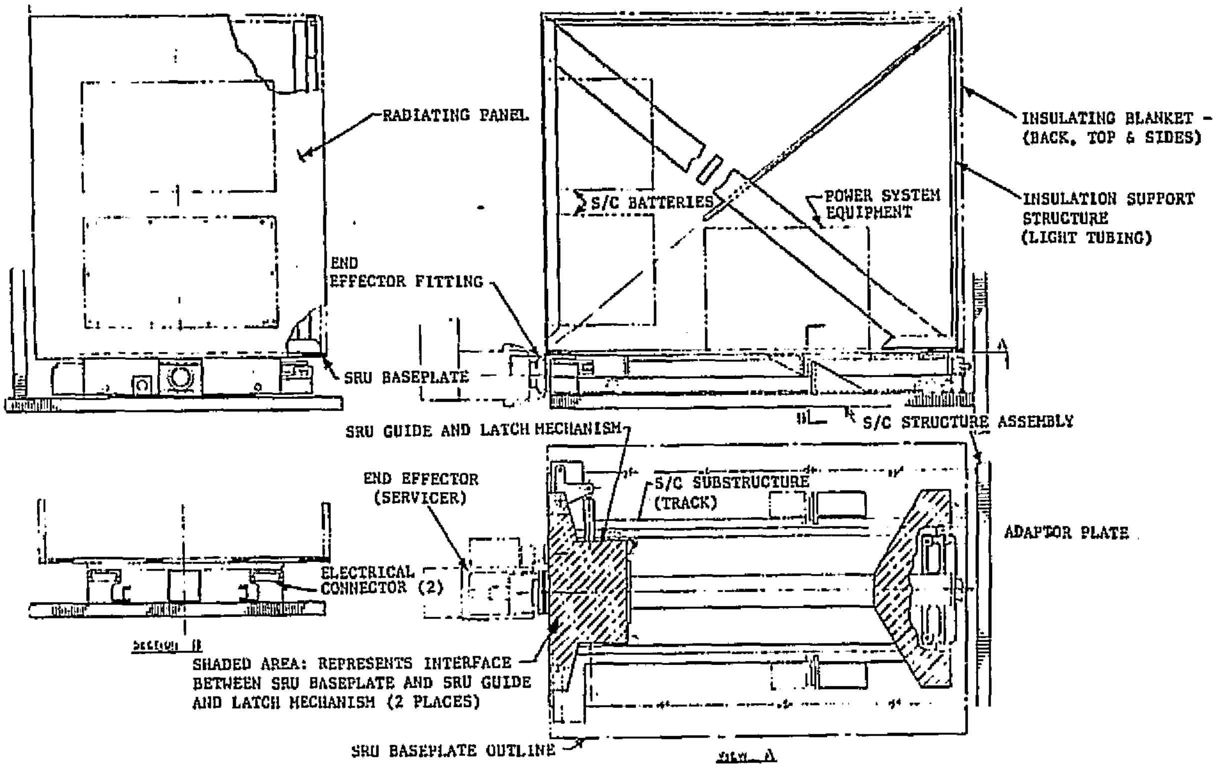

Figure 3.1.3-3 IOSS Cube Module with Side Mounting Interface Mechanism

The base mounting interface mechanism shown in Figure 3.1.3-2, is also shown assembled with a module mockup in Figure 3.1.3-4. The module is an 18 x 24 x 26 in. foam-core representation that was configured for minimum weight. The base interface mechanism is heavier and has a more adverse c.g. location than the side unit and thus requires higher motor torques to support and turn (see also Table 3.1.3-2).

A diagonal brace passing through the module is needed to transfer the gear box weight loads to the end effector attach points. It adds to the weight of the interface mechanism and potentially prevents full utilization of the available internal space of the module. However these problems can be eliminated by redesigning the unit if there is a need for a base mounting interface mechanism on a spacecraft.

In order to demonstrate the flexibility of the IOSS and its direct application to a wide range of spacecraft designs, in addition to the MMS and IOSS cube equipment modules, other module configurations and interface mechanisms should be demonstrated. The advent of the space station and of the OTV and OMV, will make possible servicing communication satellites and other spacecraft in geosynchronous orbit with a reusable, remotely controlled servicer.

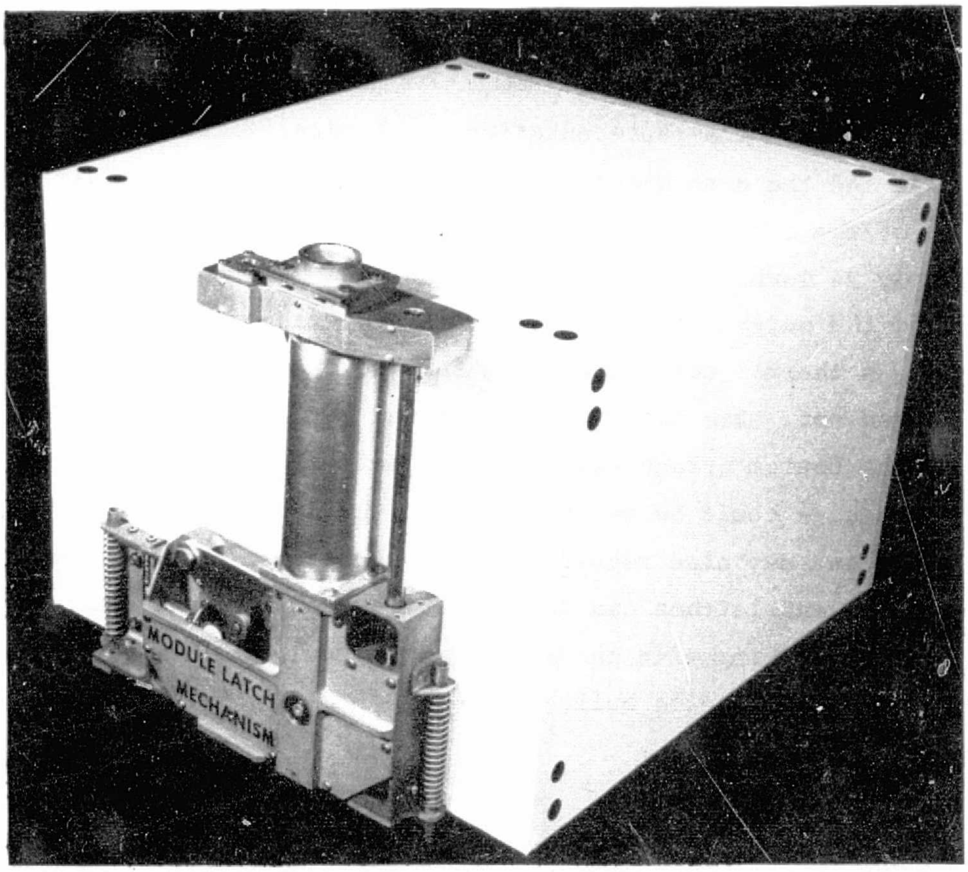

Figure 3.1.3-4 Module Representation with Base Mounting Interface Mechanism

A communications satellite module mock-up and a realistic attachment interface mechanism should be designed and built in cooperation with a contractor specializing in the design and manufacture of this type of spacecraft. The ground and flight demonstrations of changing out such a module will help define the specific requirements of communication satellite servicing and develop a flight qualified serviceable module, ready for use in future designs. The attachment interface mechanism may be either one of the two IOSS types described above, or an MMS type or it may be of new design.

The servicer demonstrations should also show changeout of modules representative for other types of spacecraft, such as AXAF. As in the case of communications satellites, development of the hardware for such demonstrations should be conducted in close cooperation with the respective project office of NASA, to become an integral part of the spacecraft design and development effort.

A conceptual design of a representative focal plane instrument module for the AXAF and the attachment interface is shown in Figure 3.1.3-5. The length of the module is 45 in. and the other two dimensions are approximately 24 inches. The nominal weight is 384 lbs maximum. Both the size and the weight of the AXAF module are within the IOSS servicer capability. A thermal cover has to be removed/opened before the module can be changed out. The attachment mechanism could be a modified version of the bottom attachment interface mechanism described above. Other AXAF modules could be the MMS type or the IOSS cube type or of other designs and may also require opening a cover prior to changeout. The cover hinge and latches can be also actuated by the servicer end effector, after docking with the AXAF, providing that a compatible power take-off interface is built in.

In order to show operational flexibility the ground and flight servicer demonstrations should also include changeout of smaller, "component level" modules, approximately 10-12 in. cube in size, for which a small, weight effective interface mechanism should be developed or a tool adapter will be used to remove conventional captive fasteners. Thermal cover removal/opening mechanisms and sensors for fastener/attach interface status need to be developed.

The proposed set of modules to be demonstrated are shown in Table 3.1.3-1. Not all of them need to be operational at the same time. The system can be reconfigured for different types of demonstrations, simulating actual servicing missions. It is not necessary, nor is it recommended, that all the servicing configurations be fully developed and demonstrated in 1-g before the flight demonstrations can begin. On the contrary, demonstrating on-orbit servicing using the already developed systems as soon as possible will speed the development of other servicing hardware and its application to new satellite designs.

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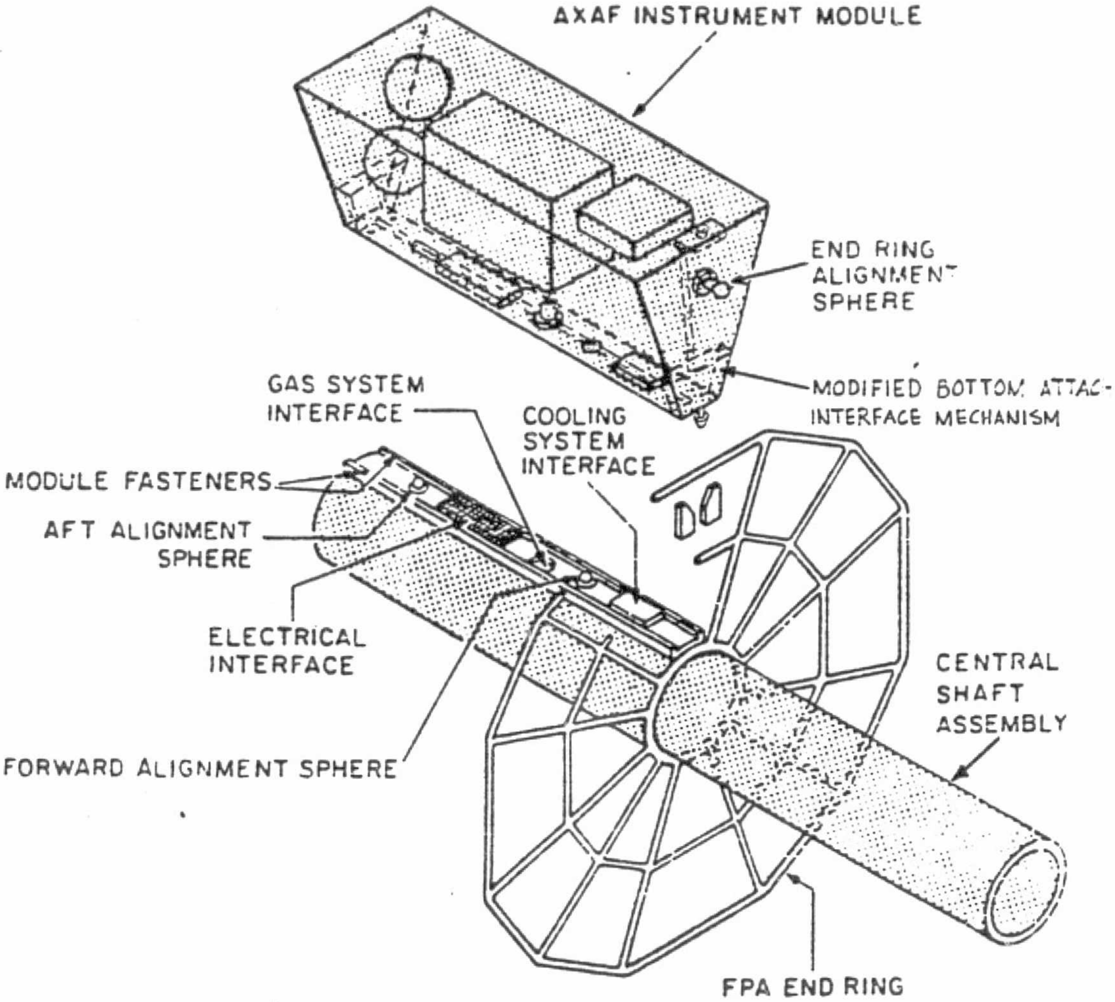

Figure 3.1.3-5 AXAF Focal Plane Instrument Module and Attachment Interface

3.1.4 End Effector Selection

A review of existing end effectors was performed to determine their applicability and feasibility for use in the ground demonstrations program. The interfaces of the end effector with different module attachment systems, with refueling/resupply hardware and with other tools and adapters for performing the required servicing tasks were analyzed. In conclusion, the IOSS end effector complemented by a series of tools and adapters was recommended for the ground demonstration servicer.

3.1.4.1 End Effector Requirements Definition - The majority of the tasks to be performed by the satellite servicer fall in the module changeout or refueling/resupply categories for which a simple, compact and very rigid end effector interface is required. Other servicing/maintenance tasks, planned or unplanned can be performed, as required, with appropriate adapters using the same, standard end effector interface. In this case the rigidity and the compactness of the end effector interface is also very important in order to maintain the required positional accuracy and the ability to operate in volume limited regions. The end effector interface is also required to provide rotating shaft actuation (power takeoff) and electrical disconnect capability.

The requirements for the end effector of the ground servicer demonstration system are as follows:

1) The end effector shall provide at least 200 lb grip force at the jaw tip level and be non-backdriveable up to 500 lb applied load, in closed position;
2) The end effector shall provide engagement with and alignment of the mating interface within an attachment envelope of $\pm$ 0.75 in. (radially);
3) The end effector shall have a positioning accuracy of the mating interface of less than $\pm$ 0.010 in. in all directions and an angular positioning accuracy of at least 0.2° after engagement;

4) The end effector shall be as compact as possible to allow access to volume limited regions;

5) The end effector shall have a single, standardized interface compatible with all module interface mechanisms, refueling/resupply interface units and adapters. It shall provide a standard power takeoff capable of at least 8 in-lb torque at an operational speed of approximately 100 rpm and a stall torque of 33 $\pm$ 3 in-lb. It shall provide electrical disconnect capability for TBD electrical wires of TBD gage. A mate/demate mechanism for the electrical disconnect shall be provided either on the end effector or on the mating interface, being actuated by the power takeoff;

6) Adequate dexterity/versatility of the end effector shall be assured by using adapters for specialized functions, as required, such as unlimited rotation, special fastener actuation, special tools operation, fingers and thumb adapter for special handling, force feedback sensor, tactile sensors, etc;

7) The operating life of the basic end effector interface shall be in excess of 10,000 open/close cycles without refurbishment;

8) The end effector controls shall be easy to integrate with the servicer control system;

9) The following monitoring sensors shall be provided for the end effector:

   a) Engagement status (ready to close),

   b) Closed/open status,

   c) TV camera and lights,

   d) Other sensors, through special adapters, shall be developed as required;

10) Manual/EVA override or adequate redundancy shall be provided for demating of the end effector and electrical disconnect. This requirement applies to the flight unit but the ground demonstration unit shall provide the envelope and other features required for an easy adaptation of such override/redundancy capability, in order to achieve hardware commonality with the flight unit.

3.1.4.2 Existing End Effector Designs - Several end effectors and adapters are described and their advantages and disadvantages are discussed.

1) The IOSS end effector was designed to mate with the side and bottom attachment interface mechanisms (see Figures 3.1.4-1 and -2). It accomplishes two things: 1) it attaches the servicer to the module, refueling/resupply interface unit, or other adapters; and 2) it operates the latching mechanism from the power takeoff. The end effector attachment is accomplished by two closing jaws grasping a rectangle-shaped baseplate grip.

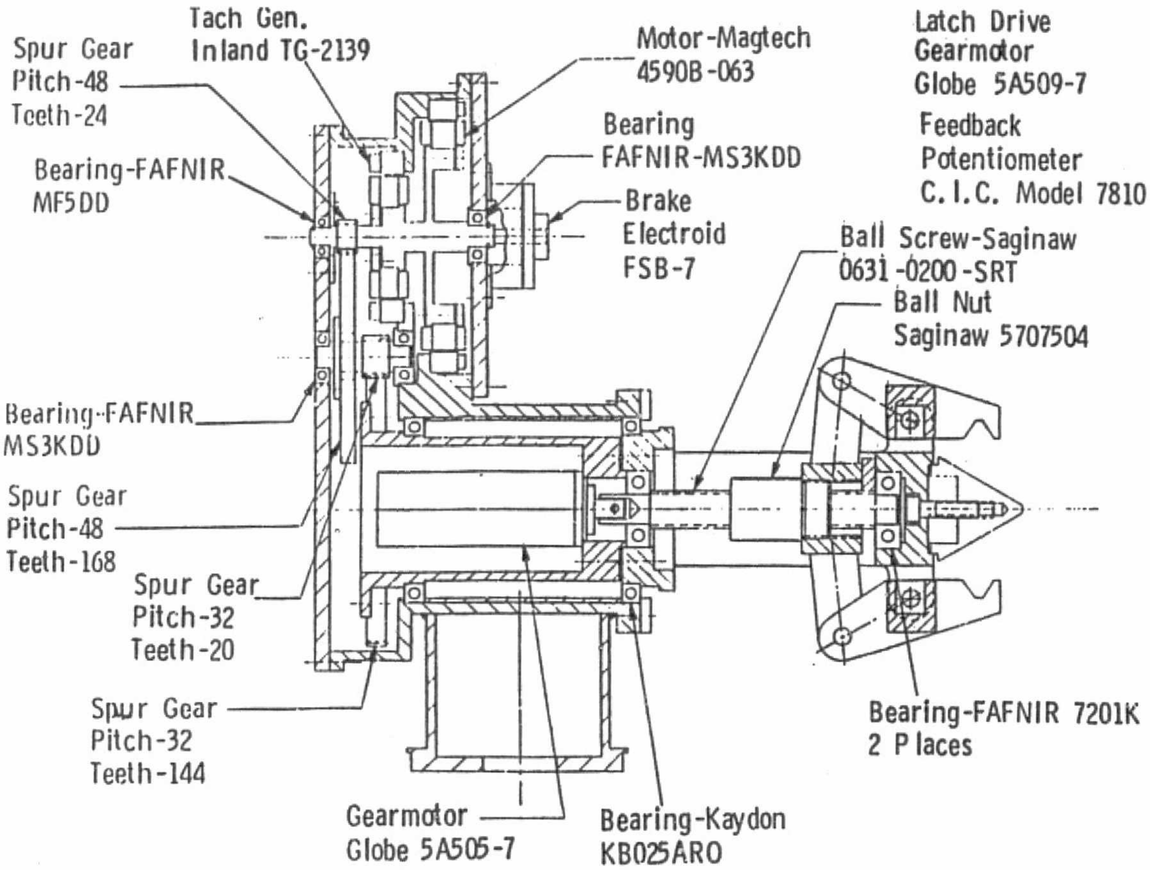

Figure 3.1.4-1 IOSS End Effector and Wrist Roll Drive

The closing force is supplied by a motor-driven ball screw drive. This drive applies a low initial closing force when radial alignment is taking place and a very high final closing force when module handling is taking place. This high force occurs because the jaw links are approaching an overcenter position with respect to the ball screw carriage.

The interface mechanism power takeoff is an integral part of the end effector. It is operated by an electric motor through a gear head. The motor and gear train are designed to produce an operating torque of 8 in-lb. with a stall torque of 33 in-lb.

Installation of the TV camera and end effector lights are shown in Figure 3.1.4-2. The camera is a General Electric 4TN2000A1 side lens solid state video camera which uses a charge injection device image. The sensing region is 244 x 188 pixels and the camera is fully compatible with a standard monitor. The camera is fitted with an auto-iris lens which changes its light admitting characteristics to keep the output video at a useable level. As the camera gets closer to the target, the reflected light gets stronger and the lens iris closes down. This in turn increases the depth of field and permits operation over the full target range with one focus setting.

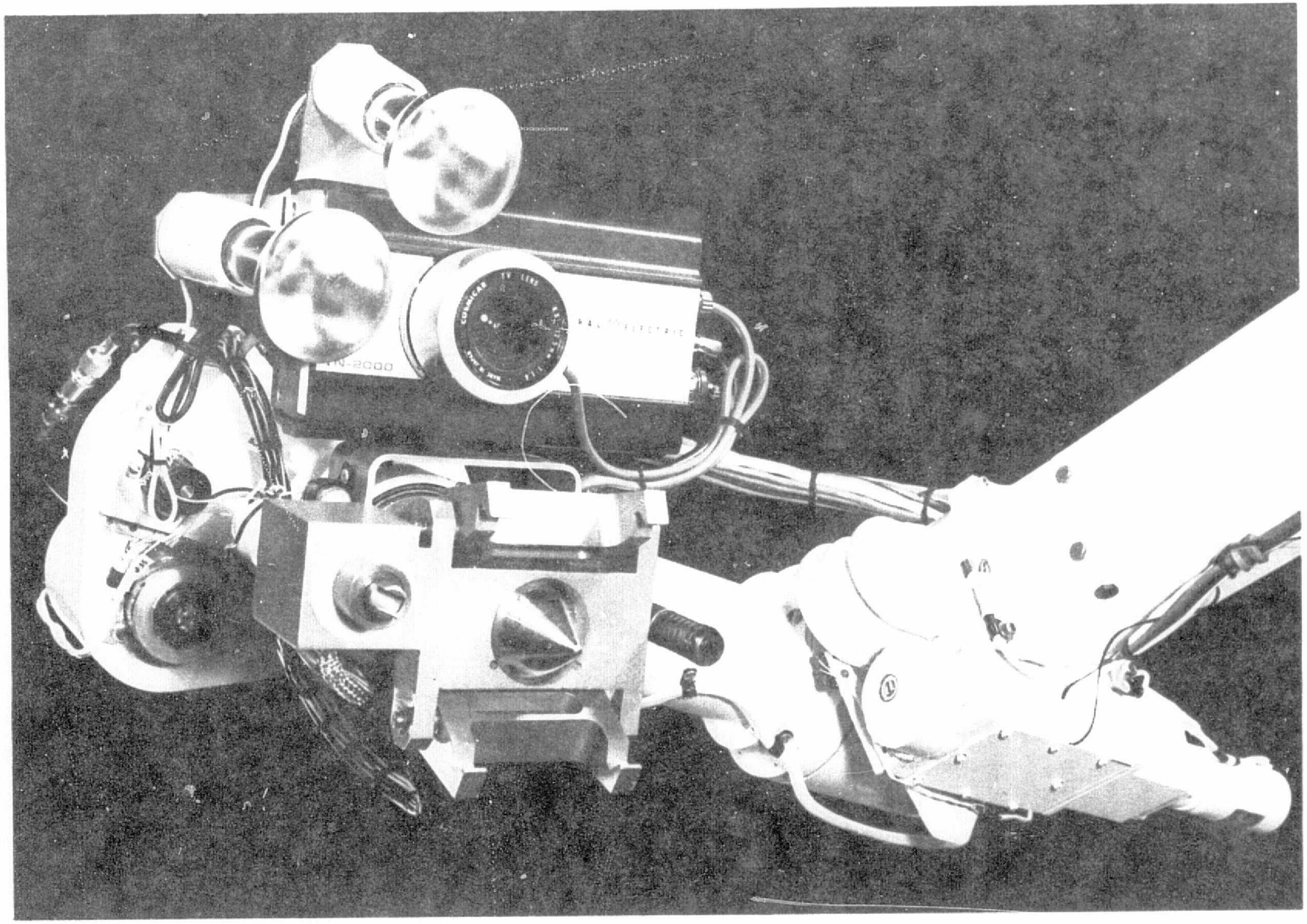

Figure 3.1.4-2 ETU End Effector and Wrist Drives

A limit switch senses the ready to close (engagement) position. The capture envelope for engagement is ± 0.75 in. radially, and the final alignment after closing is provided by a cone which mates with a conical opening in the grip plate. An electrical disconnect can be easily adapted on the side of the end effector opposite to the power takeoff.

The strong gripping mechanism and the accurate cone positioning system is also ideal for interfacing with a variety of adapters, which can be actuated from the power takeoff. A conceptual design of a simple gripper adapter is shown in Figure 3.1.4-3. The parallel jaw mechanism and the shape of the jaws make gripping a variety of round, flat, or irregular-shaped objects possible. Other jaw configurations have also been proposed.

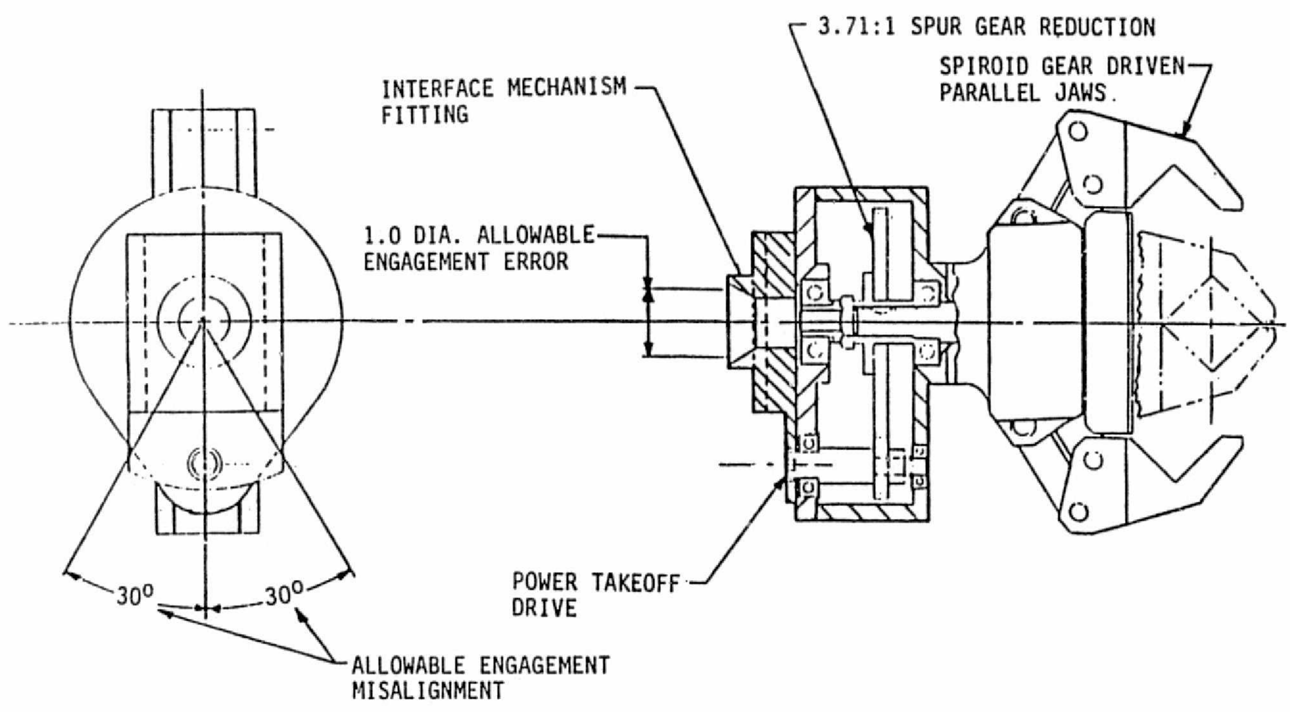

Figure 3.1.4-3 Gripper Adapter for IOSS End Effector

Adapter tools, like the socket wrench adapter shown in Figure 3.1.4-4 can be used in conjunction with the IOSS end effector to enhance the versatility of the system. The power takeoff can be used for their activation.

The IOSS arm configuration and the joint ordering are natural for module changeout and refueling/resupply with minimum separation between the servicer and the serviced spacecraft. When designing adapters for performing various tasks using the IOSS, the limitations of the kinematics of the servicer arm and the size of the end effector fitted with the TV camera and lights must be considered.

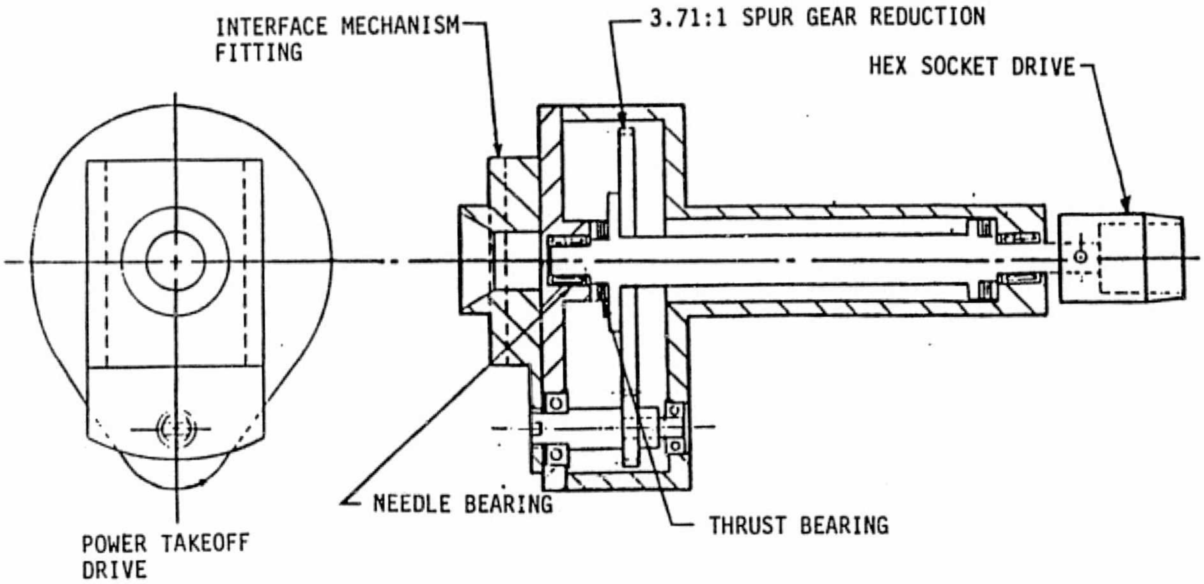

Figure 3.1.4-4 IOSS Adapter Tool - Socket Wrench

Advantages and disadvantages of using the IOSS end effector in the ground servicing demonstrations are listed below. Hardware commonality with the flight demonstrations was considered in this evaluation.

Advantages:

- Satisfies all the requirements, provides adequate gripping force and accuracy.
- Proven, reliable design.
- Supplier available.
- Commonality of design with the refueling/resupply interface unit.
- Has power takeoff.

Disadvantages:

- Wrist roll joint, TV camera and lights are close to the end effector, limiting its use in tight spots. The problem can be alleviated by using adapters.

2) The PFMA end effector, as shown in Figures 3.1.4-5 through 3.1.4-7, is powered by a pancake torque motor, which drives a spiroid gear set, having a gear ratio of 31:1. This special gearing provides a parallel jaw motion and is nonbackdriveable. The jaws are serrated for improved gripping and have a square recess for specialized gripping. The maximum jaw opening is 3.5 in. The closing/opening rate and grip force are controllable for rates of 0.1-1.5 in/sec, and forces of 10-90 lbs.

The end effector can be controlled with a variable voltage (0-31V dc) input and an incrementally adjustable current (0-4.5 amps) limiter.

Figure 3.1.4-5

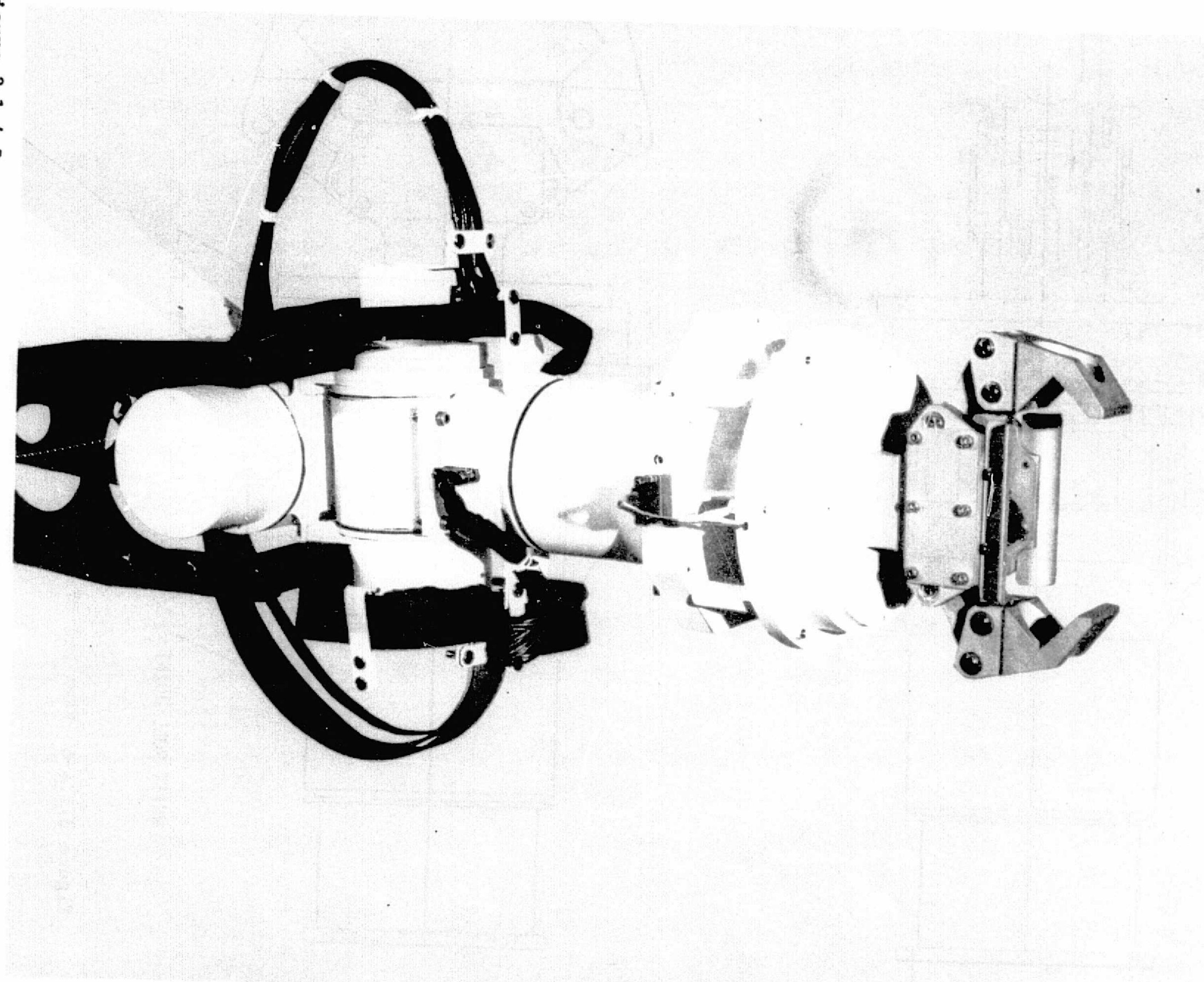

Figure 3.1.4-5 PFMA End Effector and Wrist Joints

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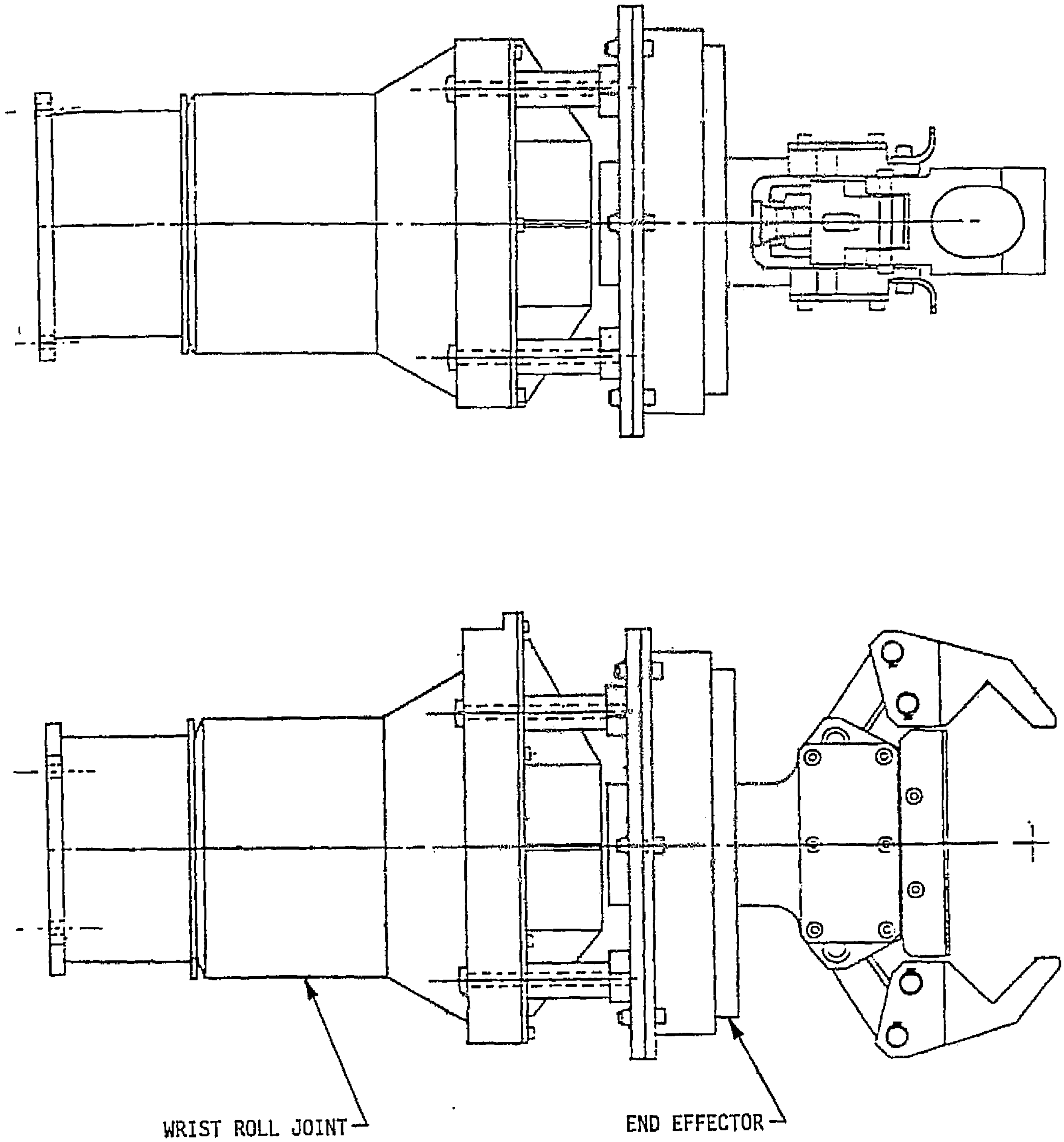

Figure 3.1.4-6 PFMA End Effector and Wrist Roll Drive

Since the closing rate is always at no-load speed, the operator may vary this speed from 0.1-1.5 in/sec. After securing the object, the motor voltage and current limiter may be adjusted upward to attain the desired grip force. This type of control prevents the crushing of fragile objects, but provide a secure grip on objects having high inertial loads.

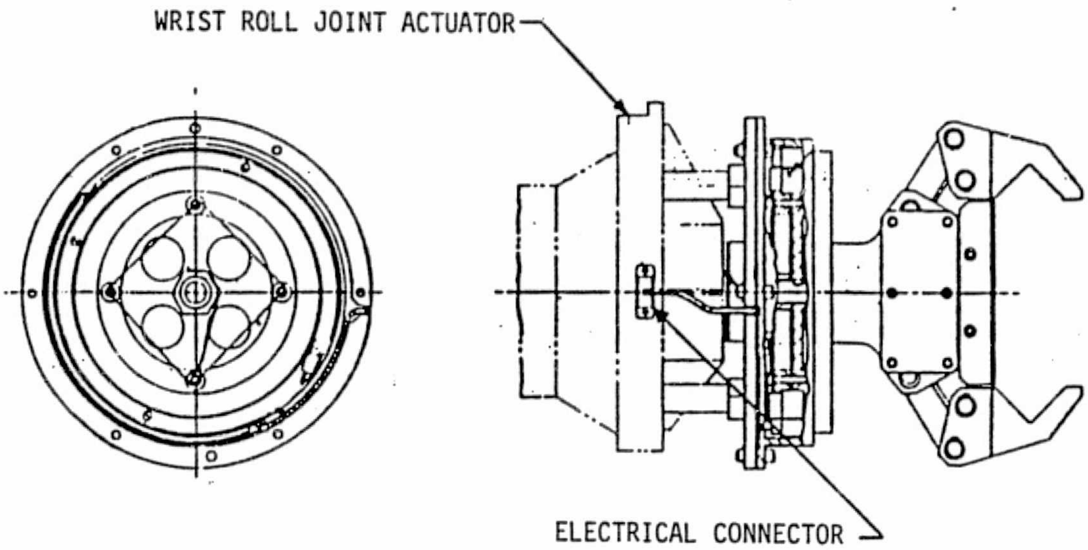

Figure 3.1.4-7 PFMA End Effector

An alternative PFMA end effector design is under development at the MSFC Information and Electronics Laboratory. It is a parallel jaw mechanism similar to the original PFMA end effector except for a new jaw concept, as shown in Figure 3.1.4-8. Each jaw is made of a series of parallel plates joined at the attachment base. When the end effector is closed the plates of one jaw slide between the plates of the opposite jaw. The profile of each jaw has a "V" notch. While the end effector is closing, they form a rectangular opening of diminishing size. This special feature enables the end effector to pick up objects of various shapes (a golf ball, a small rock, a round or square bar, even a welding rod). The jaws are dipped in an elastomeric material to improve the grip and to prevent damage to the object being handled.

As in the case of the original PFMA end effector, the grip force can be controlled. A prototype of this new end effector has been built and its capabilities were demonstrated. Preliminary investigations for adding a force feedback feature have started. A series of adapters to work with this new end effector are being developed by the MSFC Information and Electronics Laboratory. One conceptual design for a grasping tool is shown in Figure 3.1.4-9. Electrical power for actuation is provided by a self-aligning connector of a special, conical design. This concept of electrical connector, (see Figure 3.1.4-10) has been proposed by the MSFC Information and Electronics Laboratory. Its conical shape allows for large initial misalignment, has large area contacts and does not need indexing. The mate/demate force is expected to be low. The number of wires that can be connected is relatively small and the current must be interrupted some other place in the system during insertion to prevent short circuits or wrong contacts. The concept is being developed in cooperation with Columbia University. One potential application is to replace the centering cone of the IOSS end effector with such a connector. Thus, a simple electrical interface would be added for various monitoring or control functions without the need for a special mate/demate mechanism.

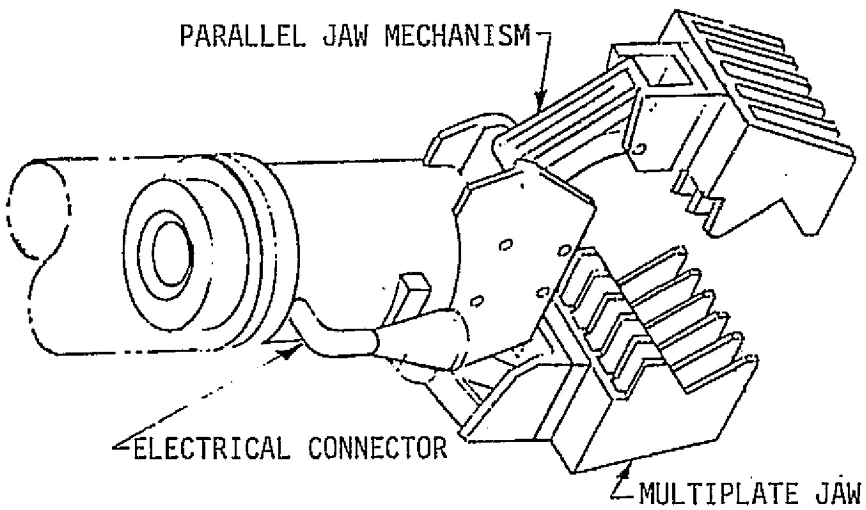

Figure 3.1.4-8 PFMA New End Effector with Multiple Plate Jaws

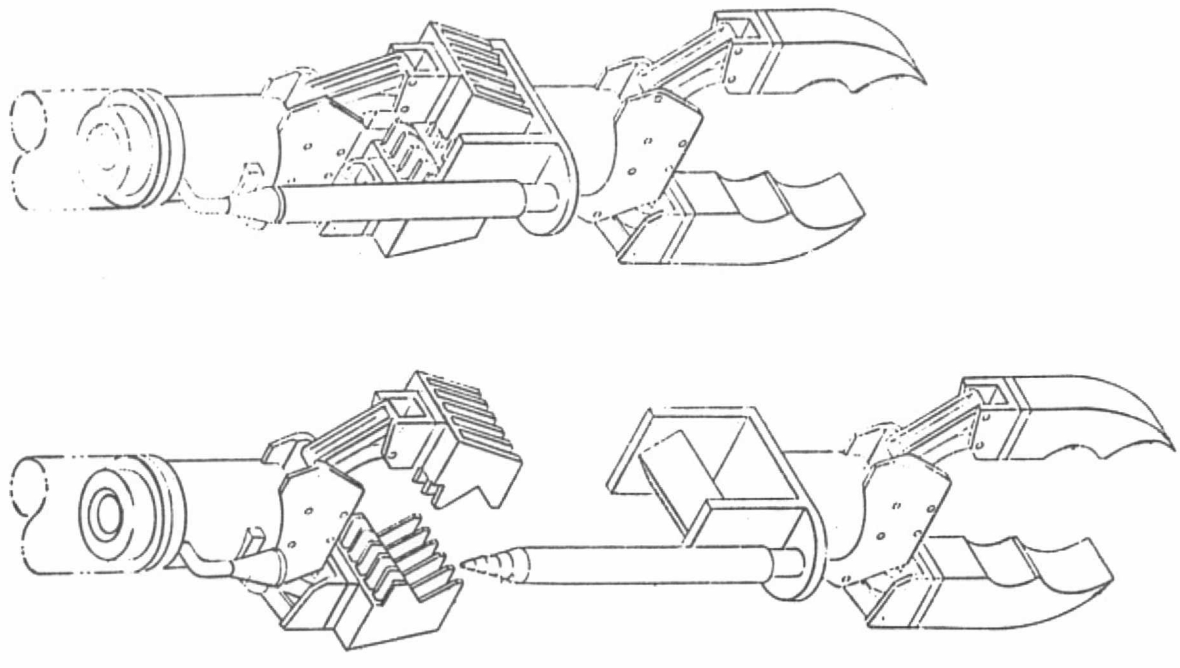

Figure 3.1.4-9 Adapter Tool for the New PFMA End Effector

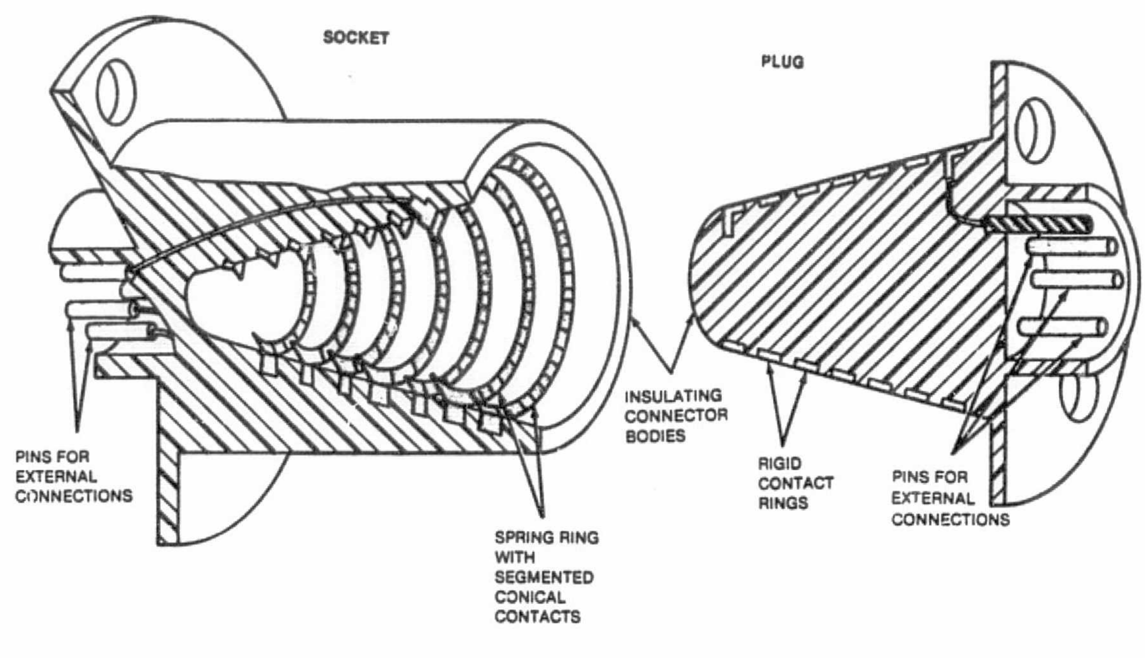

Figure 3.1.4-10 Self-Aligning Electrical Connector Concept

Another concept of an adapter to be used with the new PFMA end effector is a fluid disconnect actuation device (see Figure 3.1.4-11). The end effector holds the device by a handle-like bracket, square in cross section and made of two spring loaded halves. The end effector squeezes the two halves and through a series of cams opens two locking jaws against spring pressure.

The initial engagement of the two disconnect halves is made using the arm joints to achieve the correct relative position. The force for final mating of the disconnect is provided by the two locking jaws under spring pressure when the bracket squeeze is released. The force available for mating is very limited considering the end effector capability and the losses in the multiple cam mechanism. Demating could be actuated by the locking jaws, although no provisions are shown. The available demating force is also very limited. Leak testing prior to fluid transfer and purging prior to disconnecting is difficult to provide. No electrical connections are available for fluid transfer control and monitoring and due to limited mate/demate force available only one low pressure disconnect per each adapter can be accommodated. For each adapter a separate flexible line management system is required.

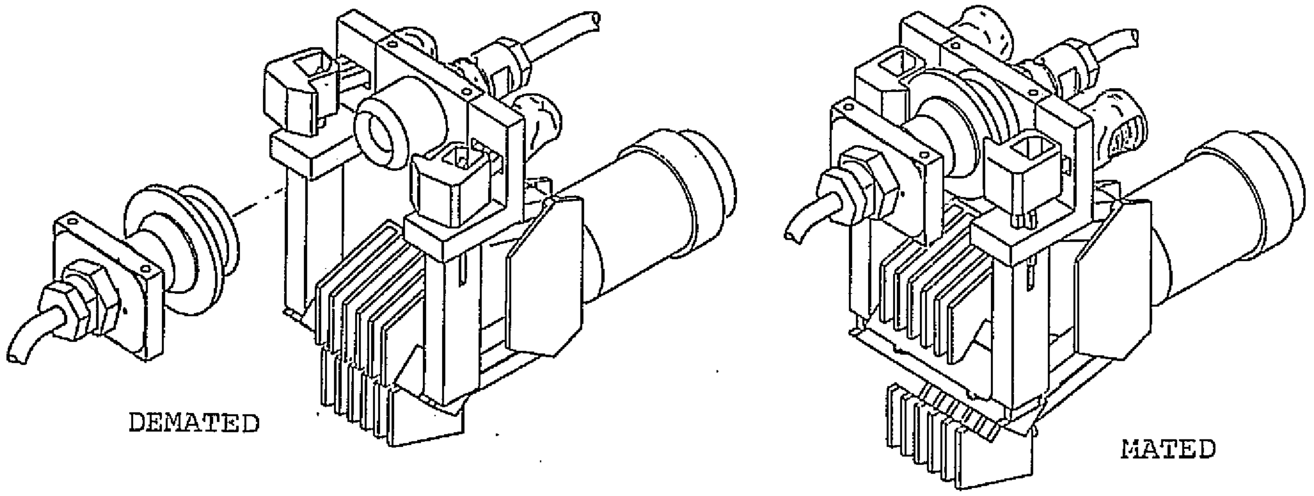

Figure 3.1.4-11 Fluid Disconnect Adapter for PFMA End Effector

Neither the new nor the original PFMA end effectors have provisions for accurately locating the mating interface or power takeoffs to actuate a module latch mechanism. If an electrical interface is provided, the motors would be located on the module, increasing the complexity and the weight of the spacecraft.

Advantages and disadvantages of using the PFMA end effector in the ground servicer demonstrations are shown below.

Advantages:

- The basic mechanism of the end effector is a reliable, proven design.
- Supplier available.
- Controllable grip force, it can handle light grasping jobs without adapters.

Disadvantages:

- Low grip force available, insufficient for module changeout.
- Special grip plate interface needed to achieve $\pm$ 0.75 inch capture envelope.
- Difficult to achieve required positioning accuracy of modules or adapters.
- Does not have provisions for power takeoff.
- Does not have provisions for TV camera and lights.
- End effector becomes bulky if TV camera, lights, power takeoff and electrical disconnect are added. Difficult to operate in volume limited regions.
- The refueling/resupply adapter does not meet the requirements defined in Section 3.1.2.

3) The Advanced Servomanipulator System (ASMS) end effector Shown in Figure 3.1.4-12 is a conceptual design developed by Martin Marietta Aerospace under a DoE contract for nuclear powerplant hot cell maintenance applications.

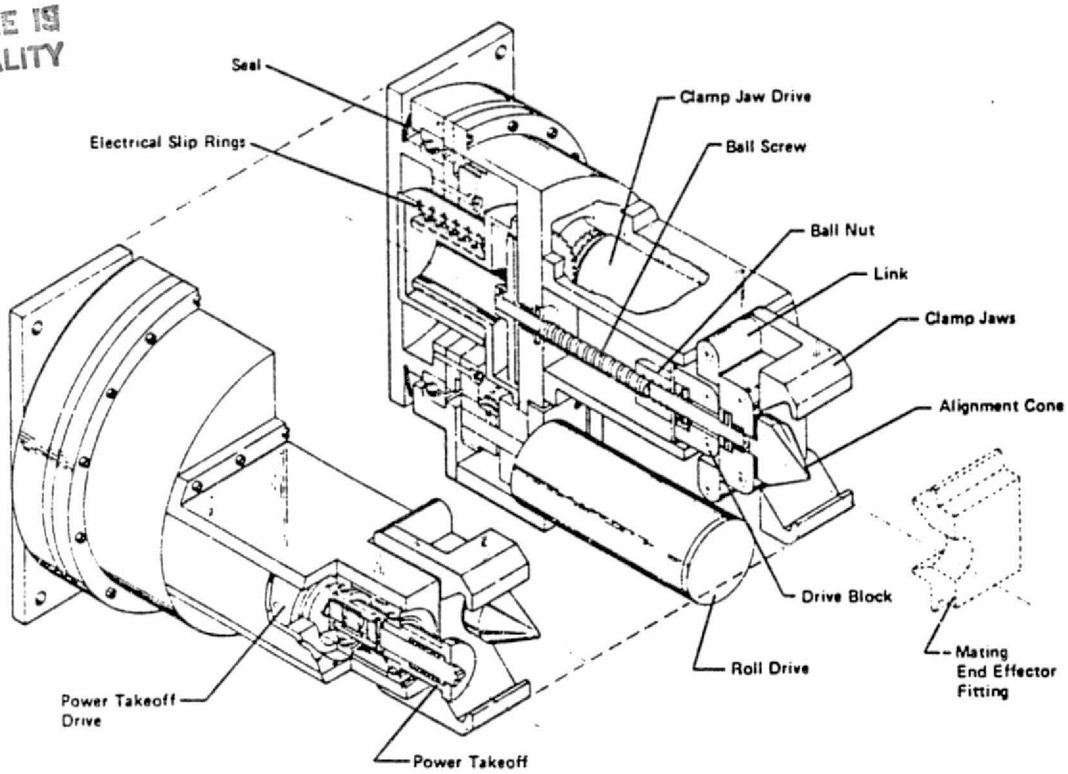

Figure 3.1.4-12 The ASMS End Effector

The end effector concept has the same grip mechanism as the IOSS and the same power takeoff, although in a slightly different position. The main difference is in the wrist roll drive that is integrated with the end effector in a more compact arrangement. The wrist roll joint also includes a multiple slip ring assembly allowing unlimited rotation. All the adapters for the IOSS can be modified for use with this end effector because the interface is almost identical.

The ASMS end effector has all the advantages of the IOSS end effector but was never built and demonstrated. The wrist roll/end effector superior compactness is shown in Figure 3.1.4-13.

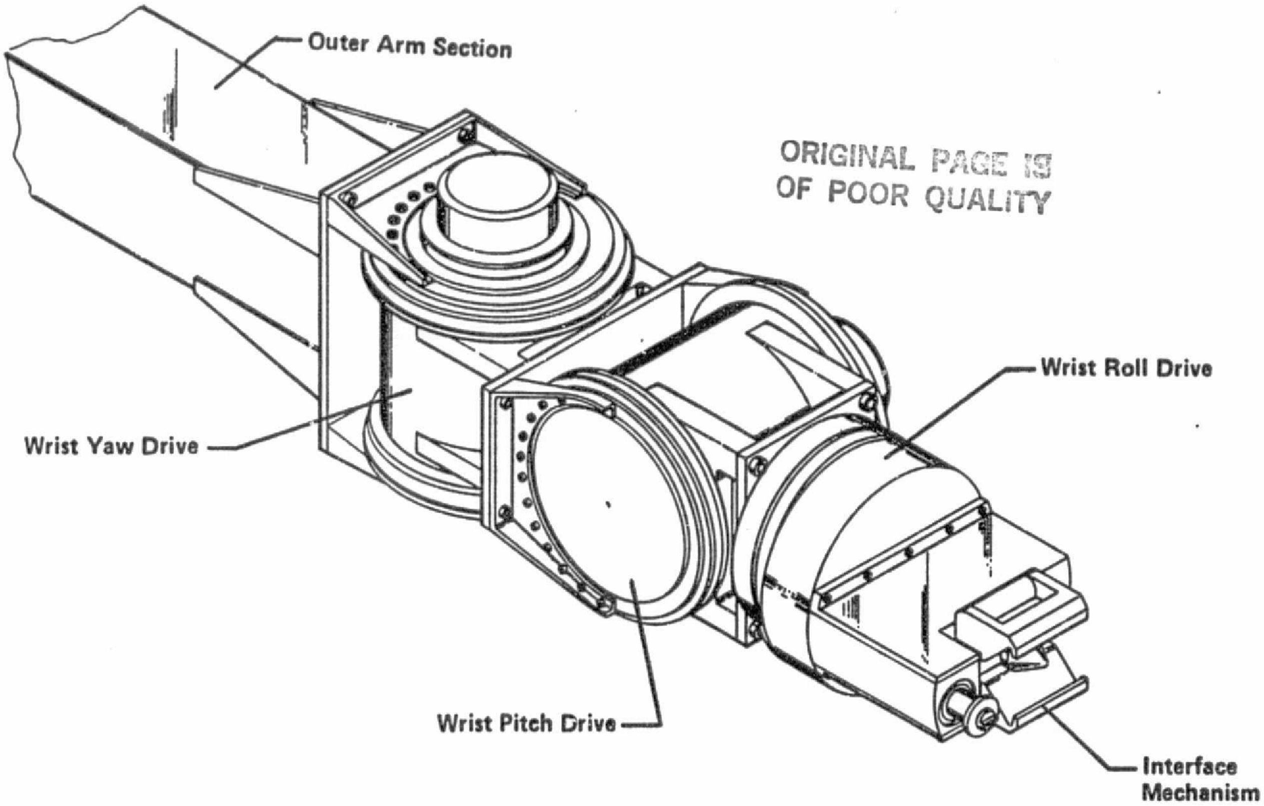

Figure 3.1.4-13 ASMS End Effector and Wrist Roll, Pitch and Yaw Joints

This end effector design as well as the PFMA end effector, and their adapters described in this section were proposed to be used in the Remote Orbital Servicing System. This was a conceptual design developed by Martin Marietta Aerospace for NASA Langley Research Center.

4) The Remote Manipulator System end effector was developed by SPAR for NASA, JSC. It is space qualified equipment and was operated in space during several orbiter flights (see Figure 3.1.4-14). The standard end effector (SEE) is a hollow, light-gauge aluminum cylinder that contains a remotely controlled motor drive assembly and three wire snares. The SEE drive system provides the ability both to capture and release and to rigidize a payload.

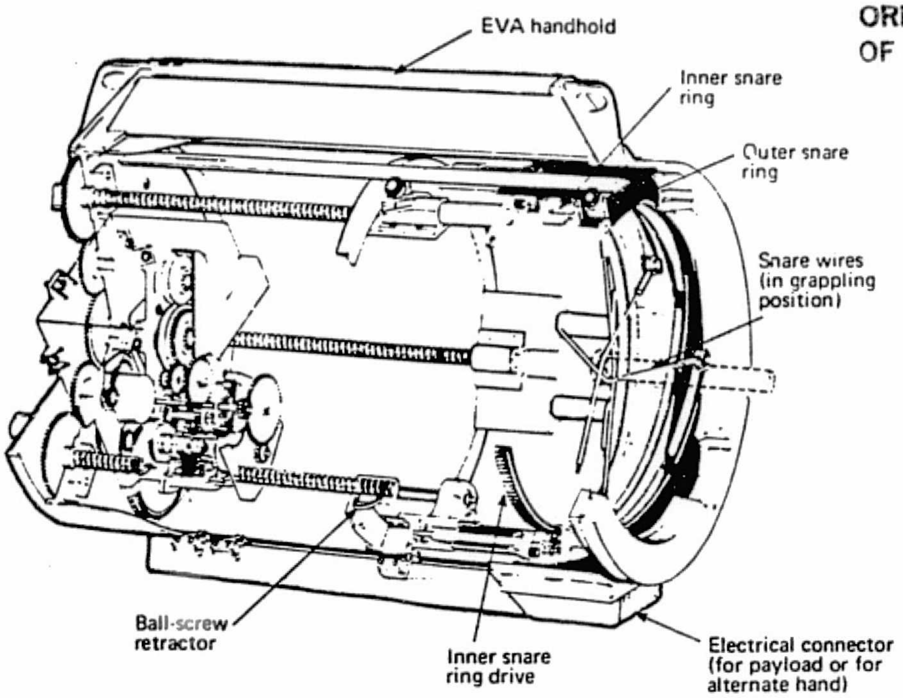

Figure 3.1.4-14 Standard End Effector for the RMS

The capture/release function is achieved by rotating rings at the end of the unit which open or close the wire snares around the payload-mounted grapple fixture. The captured payload is rigidized when the snare assembly is with drawn into the end of the end effector, pulling the payload into full contact with it. The SEE is controlled from the RMS control panel in the aft flight deck of the orbiter.

The length of the SEE is 21.5 in., the outside diameter is 13.6 in., and the weight is 65 lbs.

A standard grapple fixture is attached to the payload half of the Remote Manipulator System interface and is grappled by the SEE, allowing the payload to be manipulated by the RMS.

The standard grapple fixture consists of a rigid shaft, three alignment cam arms, and a target fixture (see Figure 3.1.4-15). The rigid shaft, which is grappled by the SEE, provides the structural support between the payload and the RMS. The grapple target fixture is sighted by the RMS wrist camera and is used to align the SEE with the grapple fixture prior to capture. When the grapple fixture is within the capture envelope, the snares of the SEE are closed about the rigid shaft and are withdrawn to the end of the end effector until a firm connection is made. The grapple fixture cams are fitted into corresponding slots in the SEE to rigidize the payload during manipulation.

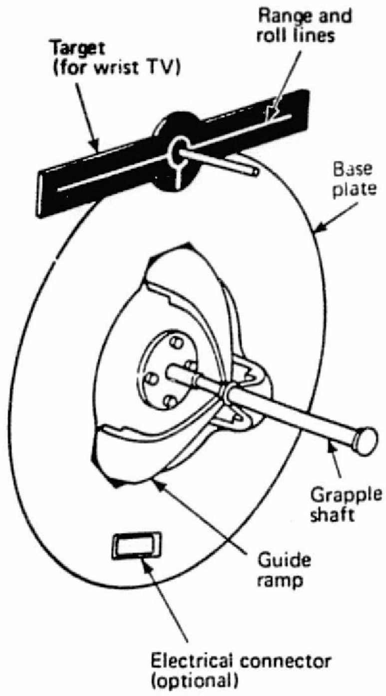

Figure 3.1.4-15 The RMS Standard Grapple Fixture

Specifications:

Maximum weight: 22 lbs.

Torsional moment about longitudinal axis of SEE: 450 lb-ft.

Bending moment to SEE: 1,200 lb-ft.

Shear force associated with bending moment: 50 lbs.

Maximum payload weight: 32,000 lbs.

A series of adapters for the RMS end effector are being developed by NASA/JSC (see Figure 3.1.4-16).

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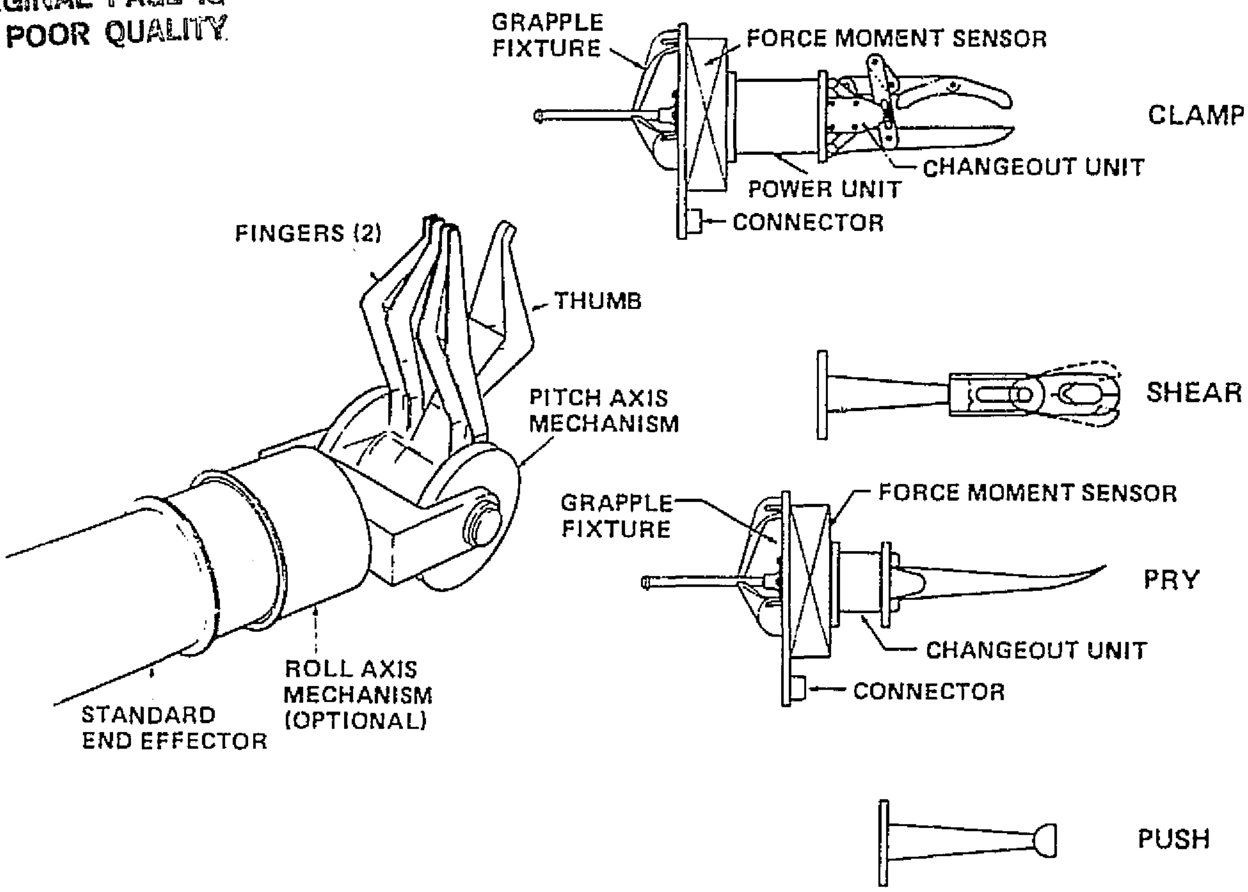

Figure 3.1.4-16 Special Adapter Tools for RMS End Effector

The end effector is an interface for the on-orbit changeout of adapter tools. Electrical power and data as well as fluid transfer can be provided to the payload across the interface between the end effector and the grapple fixture. Load and moment sensors can be added as part of a force feedback system linked to the RMS hand controllers.

The distance between the servicer and spacecraft during on-orbit servicing needs to be kept to approximately 60 in. in order to minimize the length of the docking probe, for accurate positioning while allowing enough room for module changeout. This condition limits the size of the servicer arm elements, particularly the size of the end effector. The RMS end effector is too bulky to be used in the servicer system. A scaled down version can be designed, but important advantages are lost in the process. It will need extensive development work and the smaller grappler fixture will be no longer a standard interface. The set of adapters needs to be redesigned also.

Advantages:

- Light weight high performance.
- Proven design.

Disadvantages:

- Needs redesign for scaling down too bulky as it is.
- Does not have power takeoff.
- The grappler fixture tends to be too large even after scale down.
- Needs provisions for TV camera and lights (presently located on the wrist).
- No commonality of design/hardware with the refueling/resupply interface unit.

5) Other End Effector Designs. A literature survey was performed to find other end effector designs and assess their applicability to the satellite servicing system. The rapid advances in the robotics field in the past few years generated a series of innovative designs of end effectors for general purpose and specialized manipulators. Some of these designs are shown in Figure 3.1.4-17. Many of them may be used in the future as adapters for specialized tasks. However, for the satellite servicer end effector a simple, rigid interface, capable of transmitting large forces, accurately positioning equipment modules and accommodating a multitude of interchangeable adapters would provide the best system flexibility. The IOSS end effector best meets these requirements.

Special sensors for end effectors and other robotic applications are being developed through intense research effort by many universities, government agencies and industry, both in this country and abroad. The areas of research applicable to satellite servicing include telepresence and artificial intelligence. Telepresence represents a man controlled robotic capability with the ability to sense and to affect a remote environment. It involves the development of force feedback systems and tactile sensors capable of detecting shape, surface texture and temperature and relay the information to the operator in a simple, meaningful way. It also involves development of stereoscopic vision systems and ways of minimizing the transmission time delay within the communication link between the servicer and the manned control station. Some sensors are in more advanced stages of development than others and the development and design refinement could span decades. The realistic approach would be to build enough flexibility into the satellite servicing system to be able to test, develop and incorporate new sensors, new end effectors, as interchangeable adapters, using a simple, standard interface. The IOSS end effector is ideally suited for supporting such development work.

Ultimately, the advances in the artificial intelligence field will make possible on-orbit unplanned maintenance and repair of spacecraft, using a robotic servicer, without man's intervention. A new generation of satellite servicing systems will evolve. However, the evolution process is likely to be gradual, building upon the experience gained with simpler systems, using them for testing features as they are developed and having available a satellite servicing capability while developing more sophisticated systems.

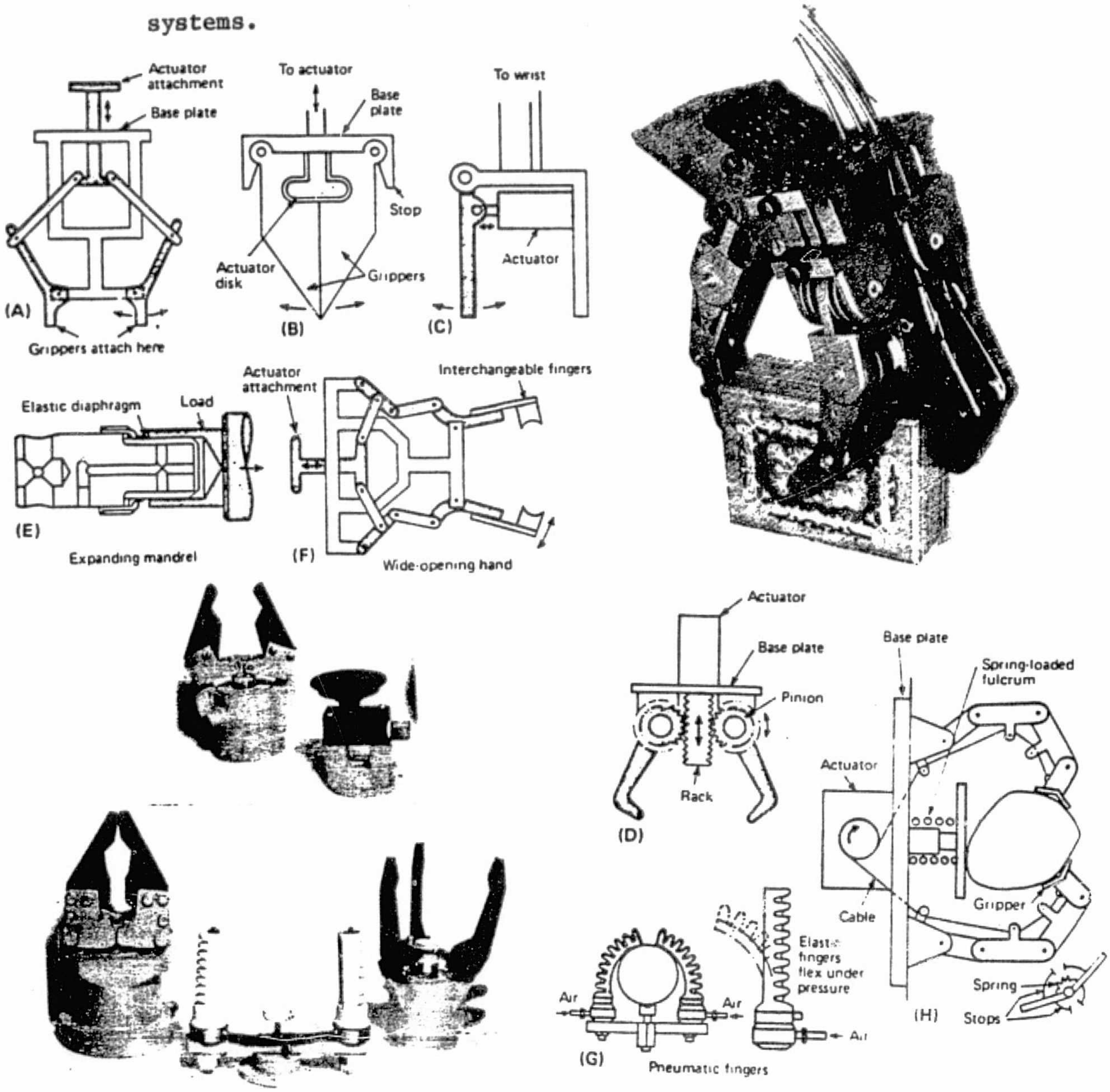

Figure 3.1.4-17 Other End Effector Designs

3.1.4.3 <u>Conclusion and Recommendation</u> - The IOSS end effector, meets all the requirements and when complemented by a series of adapters, can perform all the servicing tasks required. The extensive experience accumulated during the ETU demonstrations minimizes the risk, the amount of the development required and the cost. The use of the IOSS end effector is recommended for the ground and flight servicing demonstrations. An electrical disconnect should be added to the servicer interface and an adapter tool should be designed to interface with the existing MMS module retention system. Other special adapters should be developed as required for other types of modules or servicing tasks.

3.2 SERVICER MECHANISM SELECTION

A trade study was performed to select the type of servicer to be utilized in the ground demonstrations program. The trade study approach is summarized in Table 3.2-1.

Table 3.2-1 Trade Study Approach

Identify system requirements
Identify potential candidates
Analyze and evaluate candidates
Coarse screen candidates to eliminate unacceptable ones
Evaluate remaining candidates as to system effectiveness, supplemental costs and risks
Recommend specific candidate and summarize rationale

Based on the conclusions of this study the Engineering Test Unit (ETU) of the IOSS was selected for the 1-g servicer demonstrations. Several modifications of the existing hardware were proposed in order to demonstrate MMS module change out, refueling and other servicing tasks.

3.2.1 Servicer Mechanism Requirements

The applicable requirements for the servicer mechanism to be used in ground demonstrations were defined, and are summarized in Table 3.2.1-1. In parallel with this activity the definition of the flight demonstration requirements was performed as described in Section 5.0, to assure hardware commonality and to provide high fidelity of the ground demonstrations to the proposed flight operations. The requirements in the "Must" category refer to the basic functions of the servicing system and were later used to screen out unacceptable candidates. The "Want" category of requirements were further used to compare the remaining candidates for making the final selection. The "Want" requirements were grouped into five different criteria: high fidelity, accuracy, versatility, reliability and cost. A risk analysis was conducted prior to the final selection.

Table 3.2.1-1 Servicer Ground Demonstration Requirements

MUST:		
	1)	Able to perform the basic operations of module exchange (axial and near-radial)
	2)	Less than 75 in. axial clearance
	3)	Proven design (hardware and software)
WANT:		
HIGH FIDELITY:	1)	Efficiently perform representative satellite servicing operations (manual and automated control)
		a) Module exchange: axial and near-radial
		b) MMS module exchange
		c) Refueling interconnections
		d) Electrical connections
	2)	Use configuration similar to flight demonstration
		a) Minimum impact because of 1-g operation
		b) Similar controls, sensors, software
ACCURACY:	1)	Minimum number of joints
	2)	Minimum length of arm segments
	3)	Minimum length of docking probe
VERSATILITY	1)	Full reach envelope
		a) Axial: 360°; 1 ft to 7.5 ft radius
		b) Near Radial: 1 ft depth at 7.5 ft radius
		c) Additional envelope allowance for module removal/ insertion
	2)	Compact wrist/end effector configuration
	3)	Capable of use in conjunction with adapters
RELIABILITY:	1)	Operational experience - hardware, controls, software
	2)	Margin of performance estimates over requirements
		a) Load capability
		b) Critical clearances
	3)	System complexity - mechanisms, controls
COST:	1)	Maximum use of existing controls and software
	2)	Minimum development work required
	3)	Producibility
	4)	Minimize software complexity

"Must" Requirements

The following requirements are considered a "Must" for a servicer mechanism in order to be used for ground demonstrations of remote satellite servicing:

1) The servicer mechanism shall be able to perform the basic operations of module exchange:
   a) Types of modules as defined in Table 3.1.3-1,
   b) Axial and near-radial module exchange;

2) The servicer mechanism shall be compact in order to minimize weight and achieve acceptable levels of accuracy through use of a short docking probe and short arms. The axial clearance betweem the stowage rack and the serviced spacecraft shall be less than 75 in.;

3) The servicer mechanism shall be of proven design. Existing, proven hardware and software shall be used in order to control the risk and minimize the development cost.

"Want" Requirements

The following requirements of the ground demonstrations servicing mechanism may be satisfied only to a certain degree by a particular candidate. The selected candidate should be a servicing mechanism satisfying more requirements and to a higher degree. These requirements were grouped into five sets.

High Fidelity:

1) The servicer mechanism shall be able to efficiently perform ground demonstrations of representative satellite servicing operations in manual and automated modes of control. As a minimum the following servicing operations shall be demonstrated:

a) Module exchange, IOSS type with side attachment interface mechanism, both in axial and near-radial directions,

b) MMS module exchange,

c) Refueling interconnections,

d) Electrical connection;

2) High fidelity of the ground demonstrations of satellite servicing as compared to actual remote servicing operations is required in order to be convincing:

a) Departure from the flight configuration due to 1-g operation shall be kept to a minimum. Counterbalancing shall enable demonstration of all or a variety of required servicing tasks without reconfiguration,

b) Similar mechanisms and structures, sensors, controls and software, as proposed for flight operations, shall be used in the ground demonstrations. The control system structure shall simulate the distribution of sensors, actuators, data processing units and controls between spacecraft, servicer and ground control station. Transmission time delays shall be simulated. The flight servicer requirement that minimum constraints are to be imposed on the spacecraft design in order to be on-orbit serviceable applies also to the ground servicer to the extent of desired commonality of hardware, conducive to high fidelity ground demonstrations:

- The servicing interface on the spacecraft side shall be kept as simple as possible,
- Minimum constraints on the design of the module attachment mechanism. Use of a standard servicer interface is recommended. The servicer shall use adapters for other interfaces,
- The added weight and complexity on the spacecraft side for equipment modularization, for attachment mechanisms, for sensors and controls and for docking shall be kept to a minimum.

Accuracy:

1) The servicer mechanism shall have the minimum number of joints for maximum accuracy;
2) The servicer mechanism shall have the minimum length of arm segments to improve stiffness and reduce the required angular accuracy of the joints;
3) A minimum length docking probe is required in order to maintain an adequate accuracy level. In computing the maximum cumulative error, an allowance shall be made for the docking mechanism softness and for addition of the docking probe adapter or tool adapter. The cumulative error of the mechanical systems of the servicer, docking probe and spacecraft as well as of controls and sensors shall be less than the capture envelope of the end effector. Optical targets for use in conjunction with the video system shall be designed for minimizing the end effector positional error in manual modes of operation. In the automated mode, an automatic target recognition and error correction system should be used.

Versatility:

1) The servicer mechanism shall have the following reach envelope:
   a) For axial servicing: 360°, from 1 ft to 7.5 ft radius,
   b) For near-radial servicing: 1 ft depth at 7.5 ft radius,
   c) Additional envelope allowance shall be made for module removal and insertion;
2) The wrist/end effector configuration shall be as compact as possible to minimize clearance requirements and enhance the versatility of operations. Adequate clearance shall be provided between the servicer docking system and the arm operating envelope;
3) The servicer mechanism shall be designed to demonstrate a variety of servicing tasks in its basic configuration. It should also be capable of using adapters (adapter tools and/or docking probe adapters),

a) As a minimum, exchange of two types of modules and refueling shall be demonstrated without configuration changes,

b) Radial (single tier) as well as axial module removal shall be demonstrated,

c) A variety of interface mechanisms for module and tank attachment shall be demonstrated,

d) Servicer controls shall be operable in three modes:
   - Automatic, performing preprogrammed servicing operations,
   - Manual-augmented, using two hand controllers and video feedback,
   - Manual-joint by joint.

Reliability:

1) The servicer mechanism selected for ground demonstrations shall be a proven design, which has been in operation in the same or similar use. This requirement applies to hardware, controls and software;

2) Adequate margins of performance estimates over the requirements shall be provided in the following areas:
   a) Load capability,
   b) Critical clearances;

3) The servicer system complexity (mechanisms and controls) shall be kept to a minimum. The number of joints (degrees of freedom) of the arm shall be kept to the minimum necessary for performing all the required servicing tasks. The arm configuration shall be selected so that the number of joints being operated at the same time in coordination shall be kept to a minimum to reduce the controls complexity and improve accuracy. Operating the servicer controls shall be simple in all modes, requiring a minimum of training. The control station shall be easy to understand and operate. Human factors shall be a major consideration in the design of the control station.

Cost:

1) Existing controls and software shall be used to the maximum extent possible. Space qualified hardware to be used later in flight demonstrations shall be simulated, using less expensive components if the functional requirements for the ground servicer are met;
2) The development of new hardware, controls and software shall be kept to a minimum;
3) All components of the servicer shall be produced with the most efficient manufacturing methods. Supplier(s) availability, capability and experience shall be considered when selecting the servicer mechanism;
4) Controls software complexity shall be minimized. Ease of adaptation for performing different servicing tasks shall be a prime concern.

3.2.2 Servicer Mechanism Candidates

The candidates considered for the servicer mechanism are listed in Table 3.2.2-1 and are followed by a description of each candidate.

Table 3.2.2-1 Servicer Mechanism Candidates

1) Integrated Orbital Servicing System (IOSS)
2) Proto-Flight Manipulator Arm (PFMA)
3) Remote Manipulator System (RMS)
4) Remote Orbital Servicing System (ROSS)
5) Slave Manipulator Arm (SMA)
6) Advanced Servomanipulator System (ASMS)

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1) The Integrated Orbital Servicing System (IOSS) was developed by Martin Marietta Aerospace and NASA Marshall Space Flight Center. After two phases of study, an Engineering Test Unit (ETU) was designed and built and was delivered to MSFC in March 1978. The ETU has been in operation for over six years and was used in a comprehensive program of servicing demonstrations, system evaluations and improvement, with the objective of detailed definition of the servicer system design requirements. The IOSS design evolution is shown in Figure 3.2.2-1. A wealth of experimental data has been accumulated during this servicer demonstration and development program and constitutes the basis for the next step in the development of on-orbit satellite servicing capability, a phase of ground and flight servicing demonstrations.

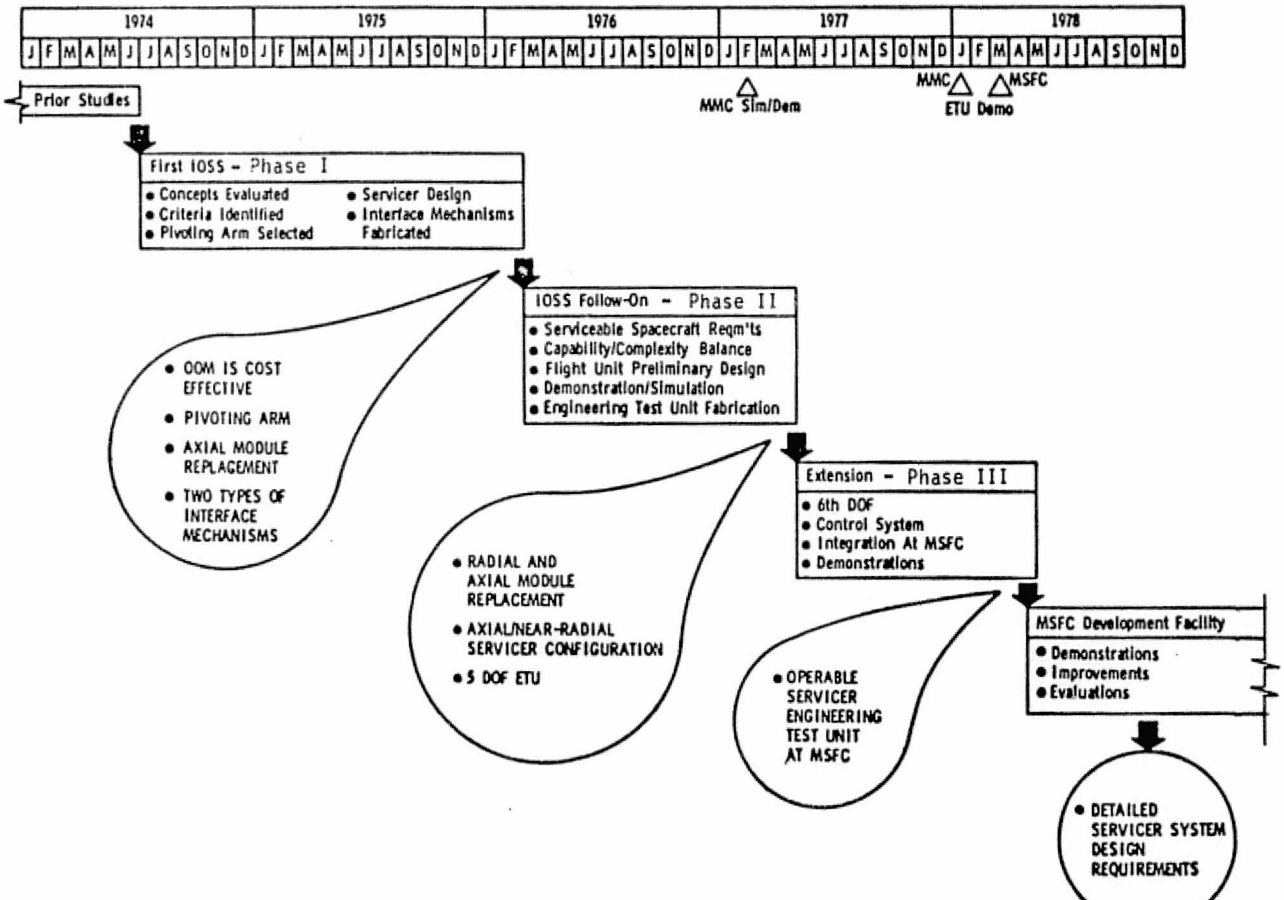

Figure 3.2.2-1 The IOSS Design Evolution

Operational experience with the ETU and the refurbishment that would be required in order to continue to use it in the servicer ground demonstrations are discussed in Section 4.0.

The main elements of the IOSS are shown in Figure 3.2.2-2 and they are followed by a description of the Engineering Test Unit of the IOSS. The ETU provides a functional representation of a serviceable spacecraft design, servicer mechanism, stowage rack and control console.

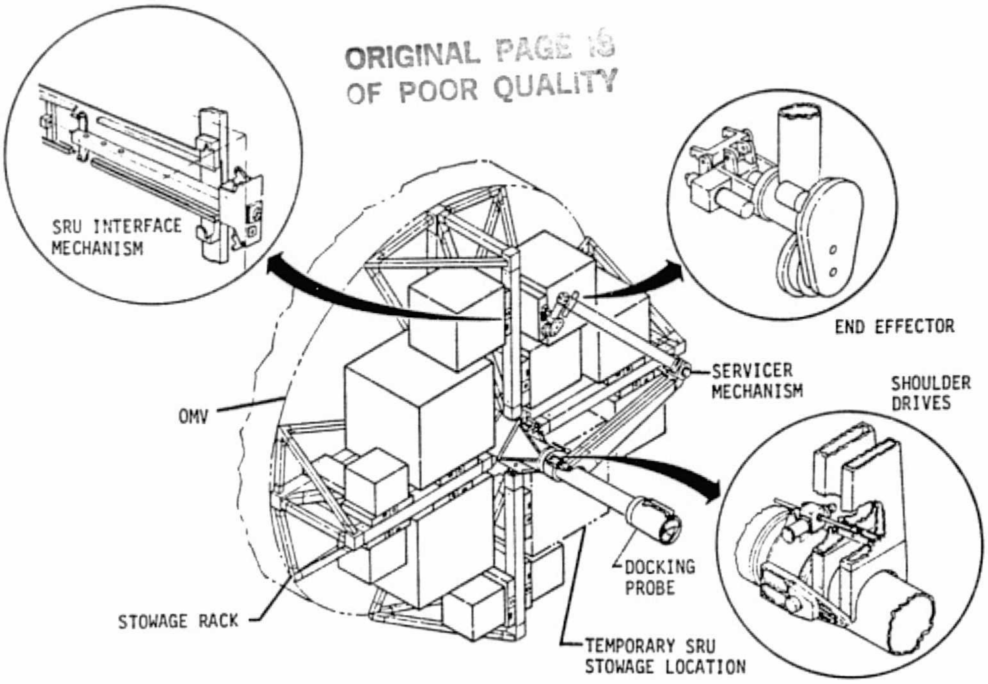

Figure 3.2.2-2 Integrated Orbital Servicing System (IOSS)

The relationships of the representative elements of the facility are shown in figure 3.2.2-3. The full scale spacecraft mockup is shown in docked configuration with the stowage rack. The separation of spacecraft to stowage rack is 60 in. and the docking axis is vertical.

The docking axis has been offset so that axial module exchange can take place at the maximum expected radius of 80 in. and radial module exchange can take place on the short end of the spacecraft. The third module location is also axial and was selected to be near the minimum radius of 20 in. The module locations can handle either side or base interface mechanisms.

The servicer mechanism mounts on the docking probe, half-way between the stowage rack and spacecraft mockups.

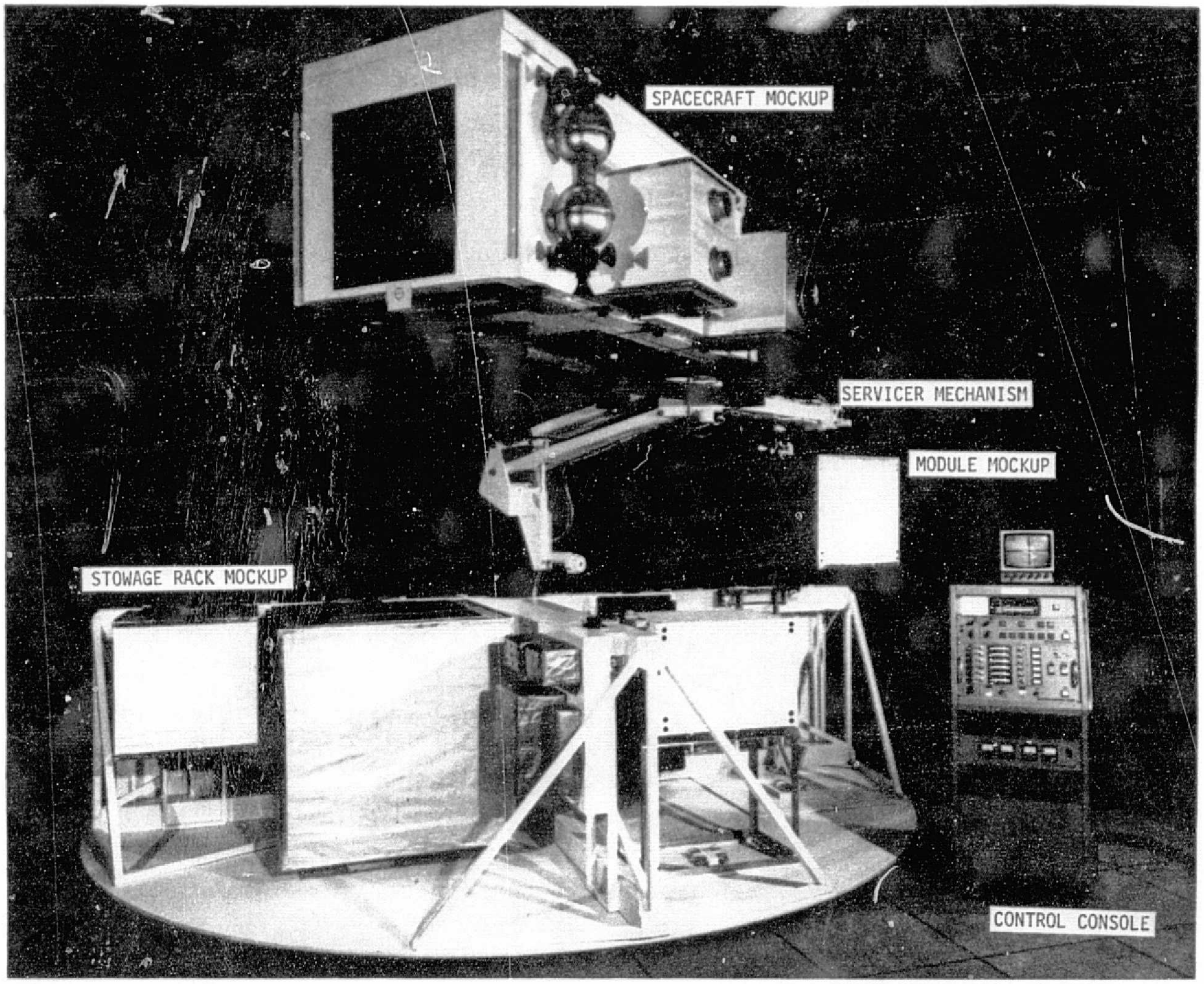

Figure 3.2.2-3 Engineering Test Unit of the IOSS

The ETU servicer mechanism (see Figure 3.2.2-4) is a high quality precision manipulator arm. Its configuration was designed to accommodate servicing a one-tier spacecraft with module exchange being in the axial or radial directions. The servicer mechanism can remove modules in off-axis directions also. Modules can be located anywhere on the end surface of the spacecraft or stowage rack mockups, and both side and bottom mount interface mechanisms can be accommodated. The axial removal interface mechanism attachment points can be located anywhere within a 20-in. to an 81-in. radius of the central docking axis. Modules can be located inboard or outboard of these radii if desired. Radial module removal can be effected for spacecraft radii up to 43 in.

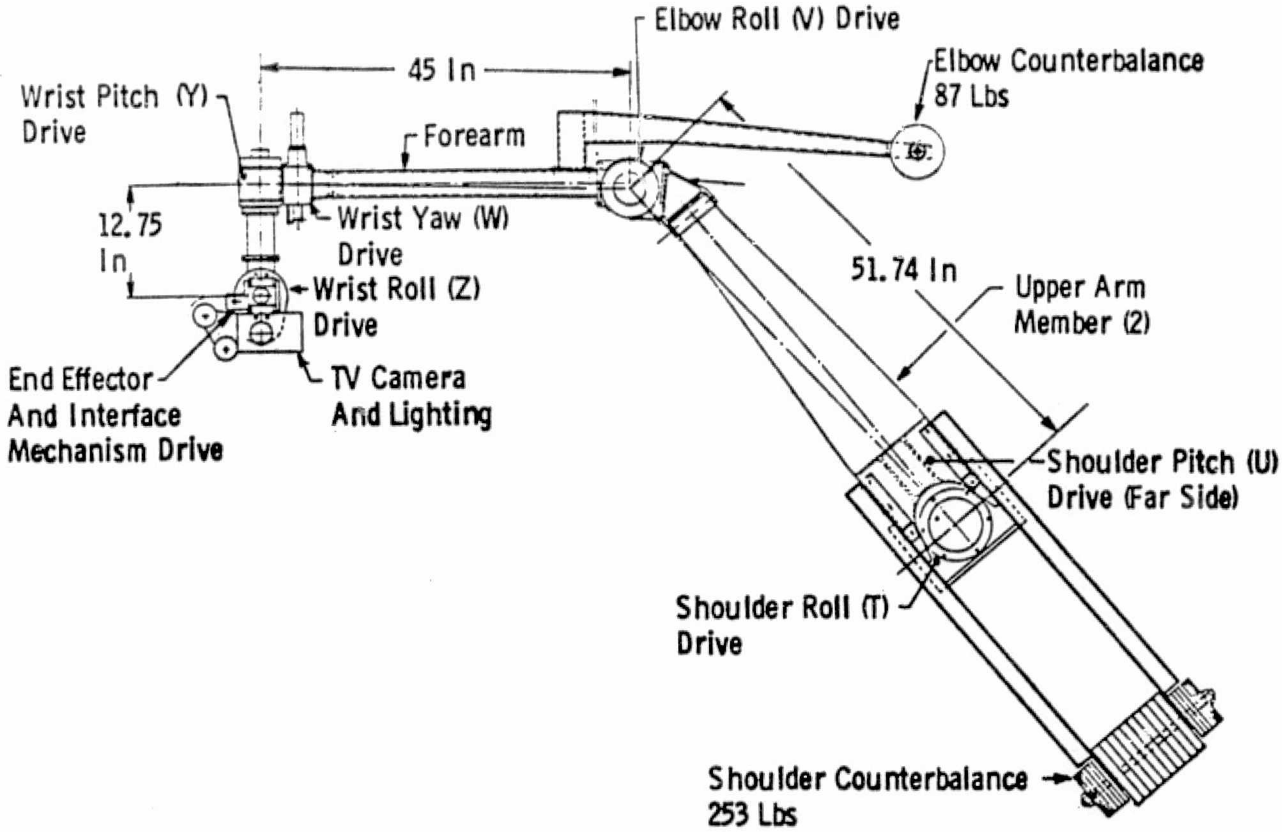

Figure 3.2.2-4 The ETU Servicer Mechanism - Top View

A significant value resulting from use of the selected configuration is its ready adaptation to counterbalancing. Three of the joints normally have their motion axes parallel to the local vertical axis. These are shoulder roll, elbow roll, and wrist roll for axial motion, or wrist pitch for radial motion. If a joint axis is kept vertical at all times, then it need not be counterbalanced. The bearings must be strong and rigid enough to take the unbalanced moments, but the motor will not see any unbalanced torques. The shoulder translation drive must be counterbalanced and it was made extra strong so variations in degree of counterbalance due to picking up interface mechanisms and modules can be accepted. The three wrist drives are not counterbalanced to ensure a compact wrist/end effector and because a wide range of gravity moments are applied. These drives are designed with high capacities to handle the range of unbalanced moments expected.

Advantages:

- Meets all requirements
- Minimal development required - low risk
- Reliable, proven technique
- Existing, reusable hardware and software
- Compact design
- Good accuracy
- High fidelity maintained between 1-g and flight demonstration through superior arrangement of joints
- Adequate torque and load capability
- Parallelogram mechanism allows simple control system

Disadvantages:

- Wrist not compact enough, requires tool adapters for limited volume regions

2) The Proto-Flight Manipulator Arm was designed and built by Martin Marietta Aerospace for NASA Marshall Space Flight Center, under Contract NAS8-31487. It was delivered to the MSFC Information and Electronics Laboratory in March 1977. The remote controls were developed and integrated by NASA-MSFC. The PFMA is a seven-degree-of-freedom general-purpose manipulator arm capable of being remotely operated in an earth orbital environment (see Figure 3.2.2-5). A counterbalance system permits the manipulator to perform useful tasks in 1-g, during laboratory testing and evaluations.

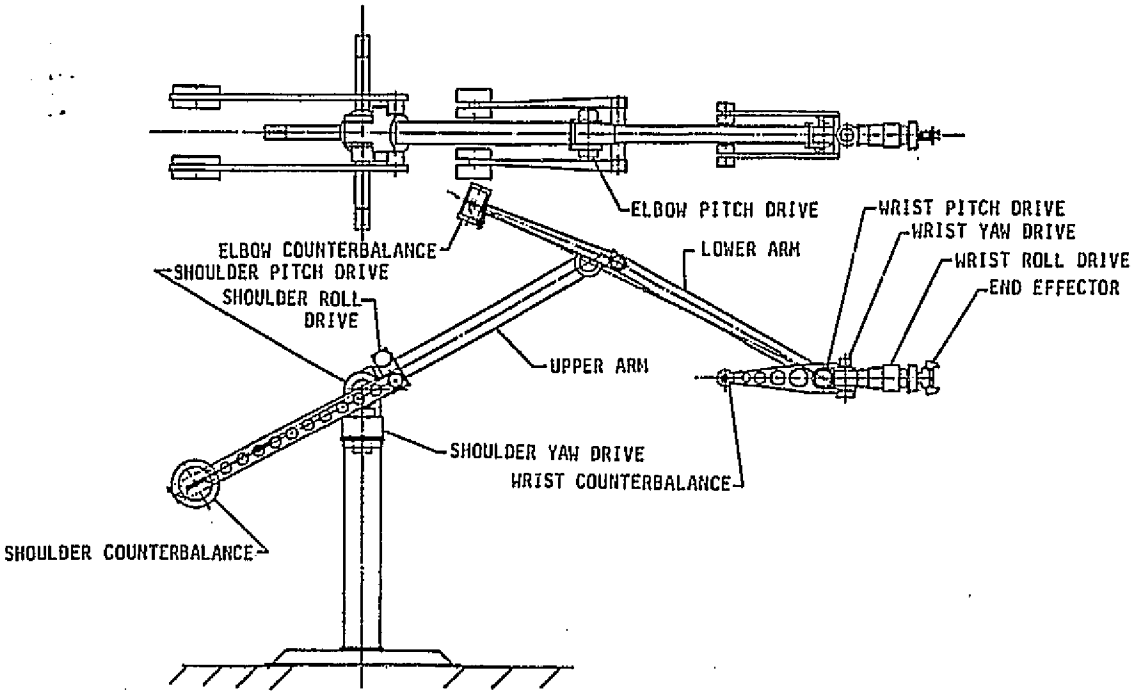

Figure 3.2.2-5 Protoflight Manipulator Arm (PFMA) 1-g Configuration

This counterbalance system can be unbolted and removed to provide the flight configuration of the PFMA (see Figure 3.2.2-6) The arm has space qualified joints and it was designed and built per NASA-MSFC 50M23186 and 50M02442, Rev W specifications. The unit was designed for stiffness and precise motion, which were accomplished by the proportional sizing of the drive joints and intermediate arm members, and the unique design of the drive gearing to minimize gear backlash. The arm develops tip forces at the end effector of up to 13 lbs in directions normal to the arm length, and can develop forces of up to 25 lbs in the extend/retract axis. The end effector can develop grip forces of 10-90 lbs and rotational torques up to 16ft-lbs in either direction through the wrist roll actuator.

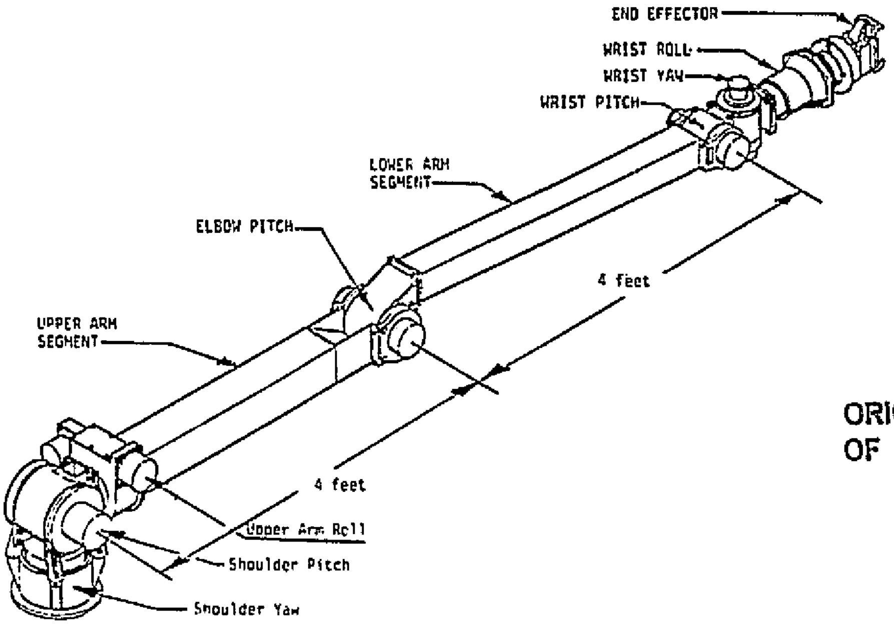

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Figure 3.2.2-6 PFMA Flight Configuration

The PFMA drives were based on the design and experience developed by Martin Marietta Aerospace during the development of a 12-foot arm, the Slave Manipulator Arm (SMA) that was an internally funded effort during the period of 1973-74. The SMA has been used as a laboratory tool to develop various control modes and to evaluate orbital assembly operating techniques. Design improvements that were identified by this earlier experience were incorporated into the PFMA. Specific improvements included precision gearing, high quality motors and tachometer generators, improved position feedback transducers (brushless sine-cosine resolvers), and supplier-adjusted fail-safe brakes. The PFMA also has the following special flightworthy provisions incorporated in the design:

1) Thermal coatings for passive thermal control in earth orbital operations;
2) Low outgassing, flat viscosity index wet lubricant compatible with earth orbital environments;
3) Space-compatible materials and processes;
4) Demonstration of the drive design under thermal vacuum conditions.

Formal acceptance tests were performed on all drive joints to verify operational performance prior to final assembly of the PFMA. These tests included torque and velocity performance, position accuracy measurements, and maximum travel. After final assembly of the manipulator, the acceptance tests included maximum reach, effective tip forces, electrical resistance and continuity, and end effector performance. A thermal vacuum test was conducted on one drive joint that demonstrated the operational performance capabilities at the temperature extremes of -100°F and +200°F, as well as 93 hours of continuous operation.

Six of the seven drives (shoulder pitch and yaw, elbow pitch, and wrist pitch, yaw, and roll) are all of one typical design, but sized for different torques and speeds. They are backdriveable and have fail-safe brakes and limit switches for end of travel indications (except wrist roll drive which has unlimited travel through the use of a slip ring assembly). All six drives are provided with position resolvers and heaters.

The shoulder roll drive that is used only for positional indexing is a worm drive with the resolver worm and the motor on the same shaft. The worm drive provides a nonbackdriveable condition and therefore no brake is required. The limit switches and heater serve the same functions as in the other drives.

The PFMA drives are high quality precision mechanisms and were very successful in operation. When the Engineering Test Unit of the IOSS was designed, three of its drives were adapted from the PFMA and the other three were designed for the specific application, by the same engineering team. Other servicer arm conceptual designs by Martin Marietta Aerospace such as the Advanced Servomanipulator System (ASMS) and the Remote Orbital Servicing System (ROSS) incorporate these high performance drives of the PFMA.

The PFMA is a general purpose manipulator arm which requires further development work in order to perform axial and radial module exchange in a satellite servicing system. Adaptation to the stowage rack/docking probe by redesigning the shoulder yaw joint is necessary, or if the arm or the docking probe is offset the reach envelope will be reduced. The counterbalance system is less efficient than that of the IOSS, when integrated into a 1-g demonstration system, with a vertical docking probe. Gravity moments induce variable motor loads and there are interferences with the stowage rack and spacecraft mockup. The load lifting capability is approximately half of the capability of the IOSS ETU.

The desired distance between stowage rack and the spacecraft is 60 in., for minimizing the length of the docking probe and refueling lines while allowing enough room for module exchange. In order to perform axial module exchange within this spacing of 60 in. (see Figure 3.2.2-7) all seven joints need to be actuated at the same time in proper synchronization. The control system becomes more complex than for IOSS, while the accuracy of the arm is reduced. The wrist/end effector portion of the PFMA is less compact compared to IOSS.

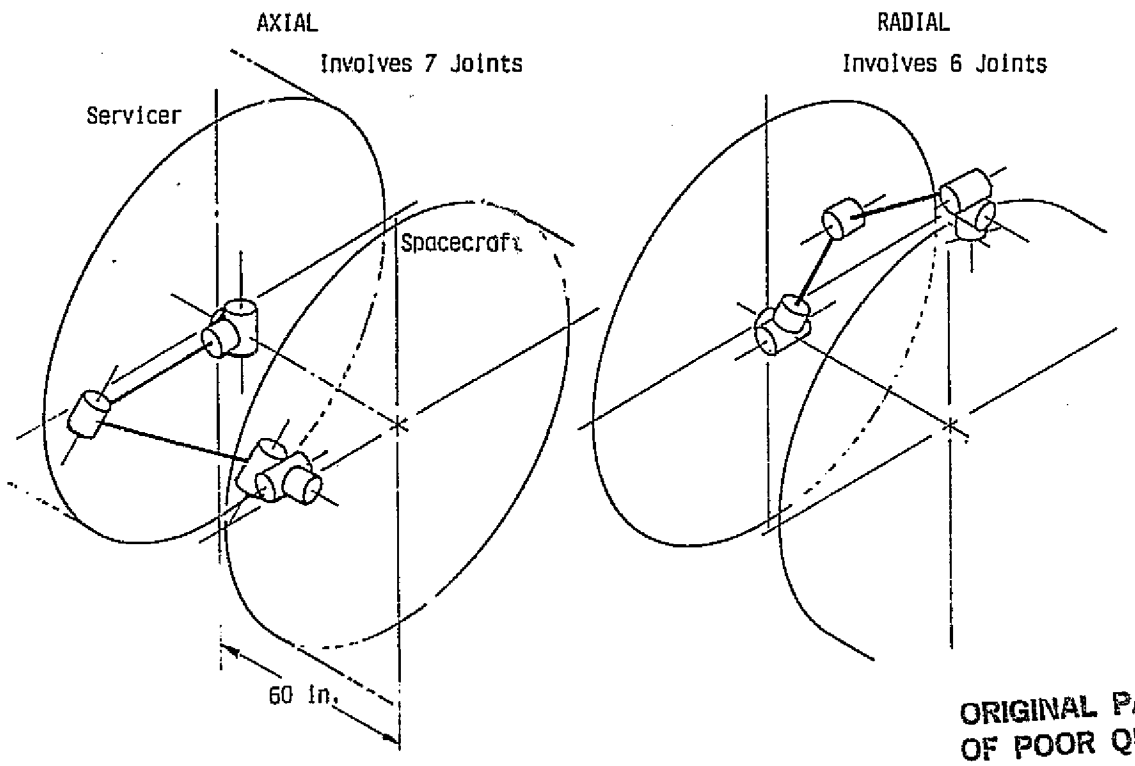


Figure 3.2.2-7 Use of PFMA as a Servicer

Additional development work is necessary in order to adapt the IOSS end effector to the PFMA arm.

The advantages and disadvantages of using the PFMA as a servicer mechanism are:

Advantages:

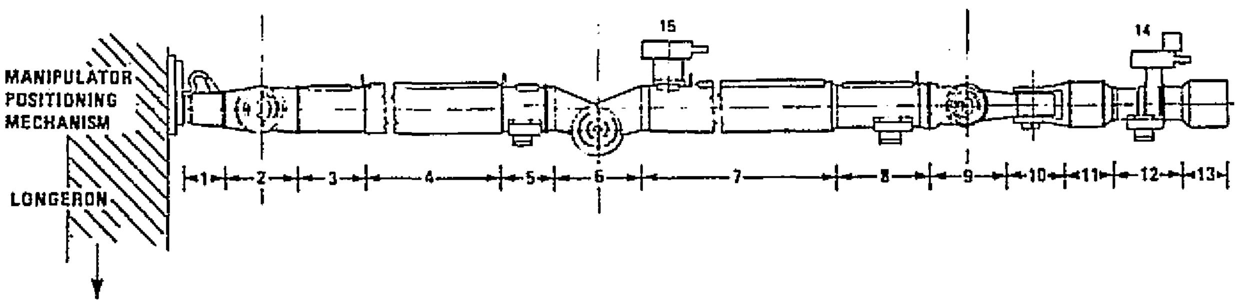
- Space qualified joints
- Reliable, proven technique
- Suppliers and expertise available

Disadvantages:

- Requires development work for adaptation to stowage rack/docking probe
- Offset arm or docking probe - reduced reach envelope
- Complicated controls, requires coordination of many joints
- Lower accuracy - one extra joint at shoulder and longer arm required
- Lower fidelity between 1-g and flight servicer
- Lower load and moment capability
- Wrist less compact

3) The Remote Manipulator System is a mechanical arm that augments the orbiter systems in performing the deployment and/or retrieval of a payload. In addition, the RMS may be used to perform other tasks in support of satellite servicing or to assist in extravehicular activities.

The manipulator arm (see Figures 3.2.2-8 through-10) consists of six joints connected by structural members to a payload-capturing device called an end effector. The movement of the arm is controlled by an operator using a display and control panel and two three-degree-of-freedom hand controllers. The operator also has visual access through the windows in the aft flight deck. The manipulator arm is anthropomorphic by design, comprising shoulder pitch, shoulder yaw, and elbow pitch joints (mainly providing end-point translation) plus wrist pitch, yaw, and roll joints (providing rotation of the end effector). For specifications see Table 3.2.2-2



COMPONENTS

1. SHOULDER YAW JOINT
2. SHOULDER PITCH
3. SHOULDER ELECT. COMP.
4. UPPER ARM BOOM
5. ELBOW ELECT. COMP.
6. ELBOW JOINT
7. LOWER ARM BOOM
8. WRIST FWD. ELECT. COMP.
9. WRIST PITCH JOINT
10. WRIST YAW JOINT
11. WRIST AFT ELECT. COMP.
12. WRIST ROLL
13. STANDARD END EFFECTOR
14. CCTV CAMERA AND VIEWING LIGHT - WRIST LOCATION
15. CCTV CAMERA WITH PAN/TILT UNIT - ELBOW LOCATION

Figure 3.2.2-8 The Remote Manipulator System Components.

Table 3.2.2-2 RMS Specifications

Length: 50 ft
Weight: 905 lbs (additional 28 lb for elbow camera)
Positioning accuracy (within reach envelope): $\pm$ 2 in. $\pm$ 1°
Design limit load:
Torsional moment about longitudinal axis of end effector: 750 ft-lb
Shear force associated with bending moment: 50 lb
Bending moment to end effector 1200 ft-lb
Payload characteristics:
Maximum size: 15 ft diameter by 60 ft long cylinder
Maximum nominal payload weight: 32,000 lb
Maximum contingent payload weight: 65,000 lb

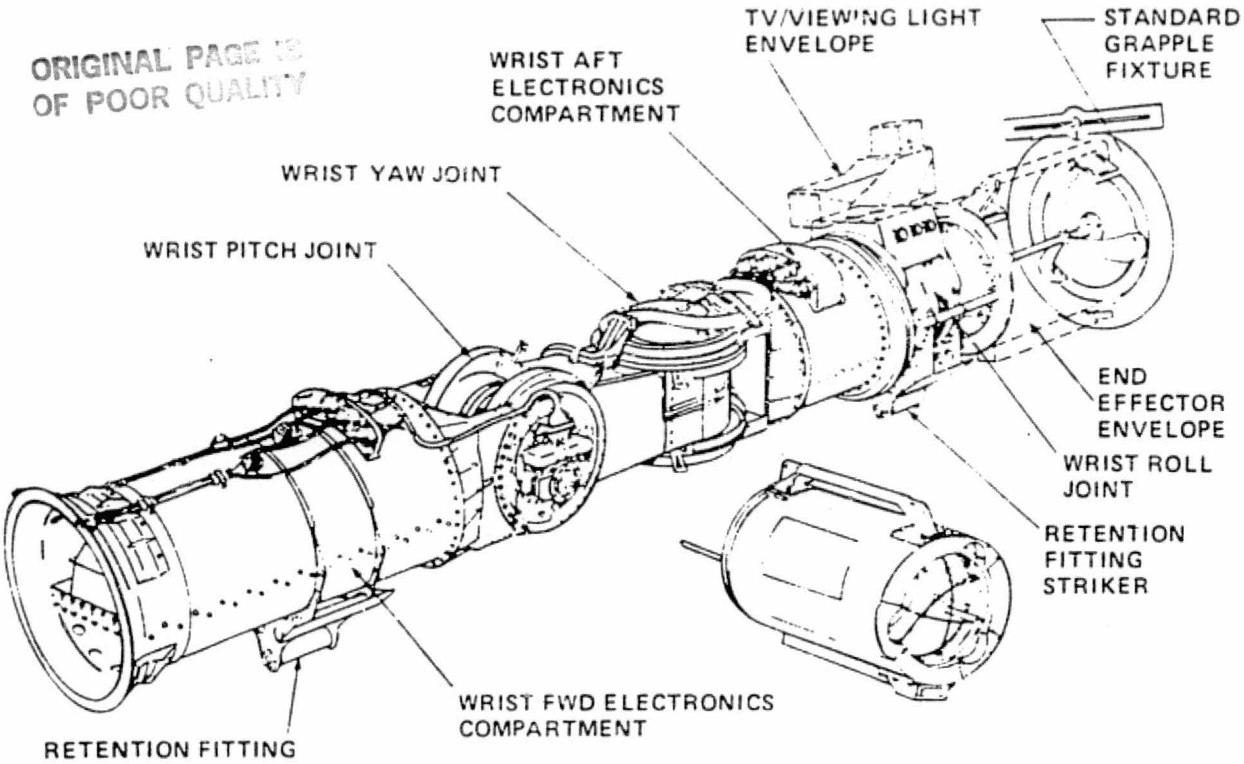


Figure 3.2.2-9 RMS Wrist Joints and End Effector

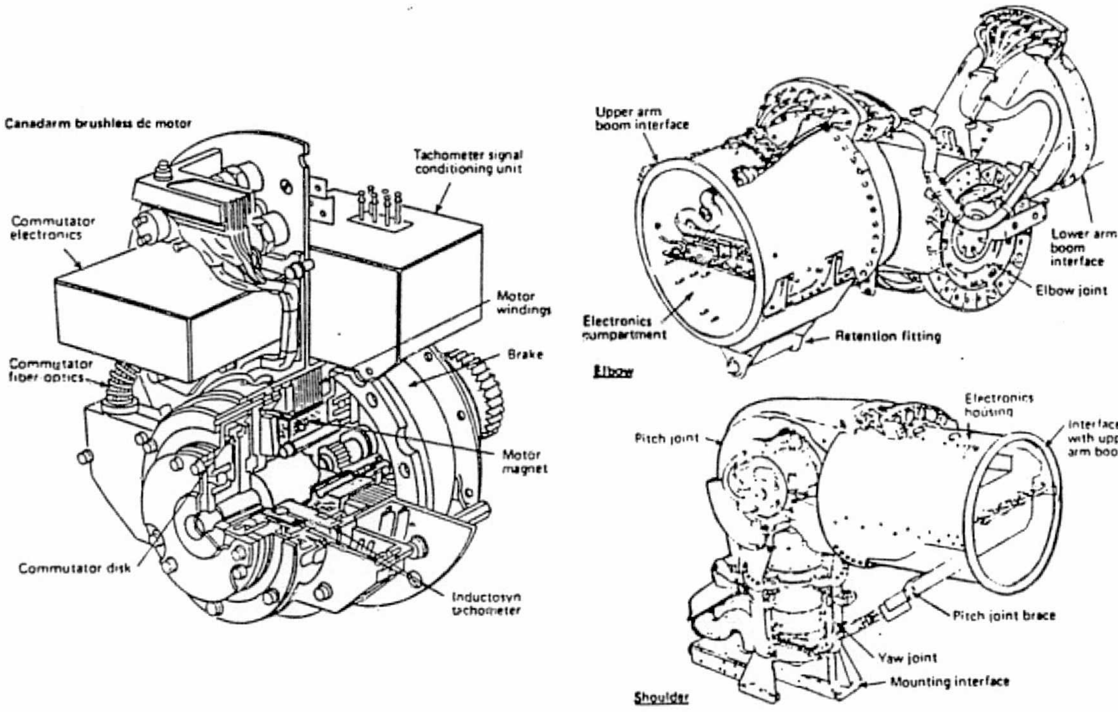


Figure 3.2.2-10 RMS Shoulder and Elbow Drives and Typical Joint Detail

The RMS is space qualified and has been in operation during several orbiter flights. It was designed and built by SPAR of Canada. The joints have brushless dc motors, brakes, tachometers and optical encoders. The electronics for the control of each joint are located in adjacent compartments within the arm (see Figure 3.2.2-11).

Control of the RMS is effected by an operator from the RMS panel in the aft flight deck. The operator has access to four prime control modes, in which he has varying degrees of software support, and a backup mode that completely by-passes the control and display software. The control modes that can be selected by the operator are as follows:

a. Manual Augmented Mode - The operator issues commands through two three-degrees of freedom hand controllers for commanding resolved rates for the six degrees of freedom of the arm. The rotational controller provides for resolved roll, pitch, and yaw without inducing translation of the end effector. The translation controller provides for resolved up/down, left/right, fore/aft translation without inducing rotation.

b. Automatic Mode - The manipulator arm movement can be controlled automatically along a prespecified trajectory. This trajectory is defined by a series of predefined positions and orientations stored in the orbiter general purpose computer. The operator can select up to four preprogrammed automatic trajectories. Also, an operator commanded auto sequence mode can be initiated by input of the required position and orientation of the end effector or payload. A straight line trajectory is then performed from the current position and orientation to the desired position and orientation.

c. Single Joint Drive Mode - The operator commands, through panel switches, movements of individual arm joints. These commands are made through the RMS software, which controls the position of all joints, limits drive speeds, provides joint position displays, and indicates when joint angle limits are encountered.

d. Direct Drive Mode - Direct drive control of the RMS is by operator command of individual joints, using hardwired commands from the control panel. This is a contingency mode that by-passes the software when driving the motors (software data is normally displayed).

e. Backup Drive Mode - Backup control of individual joints by operator commands through unique hardwired channels. No position data is displayed.

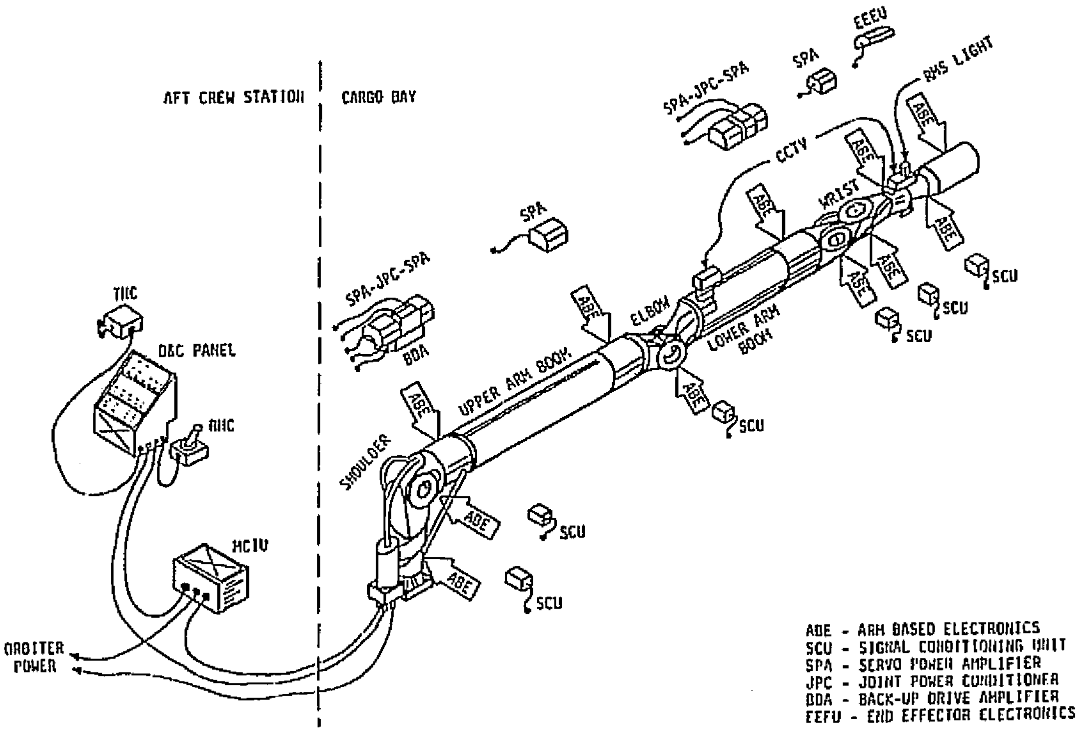


Figure 3.2.2-11 RMS Controls System - Component Location

The RMS arm dimensions and joint angle limits are shown in Figure 3.2.2-12.

Figure 3.2.2-12

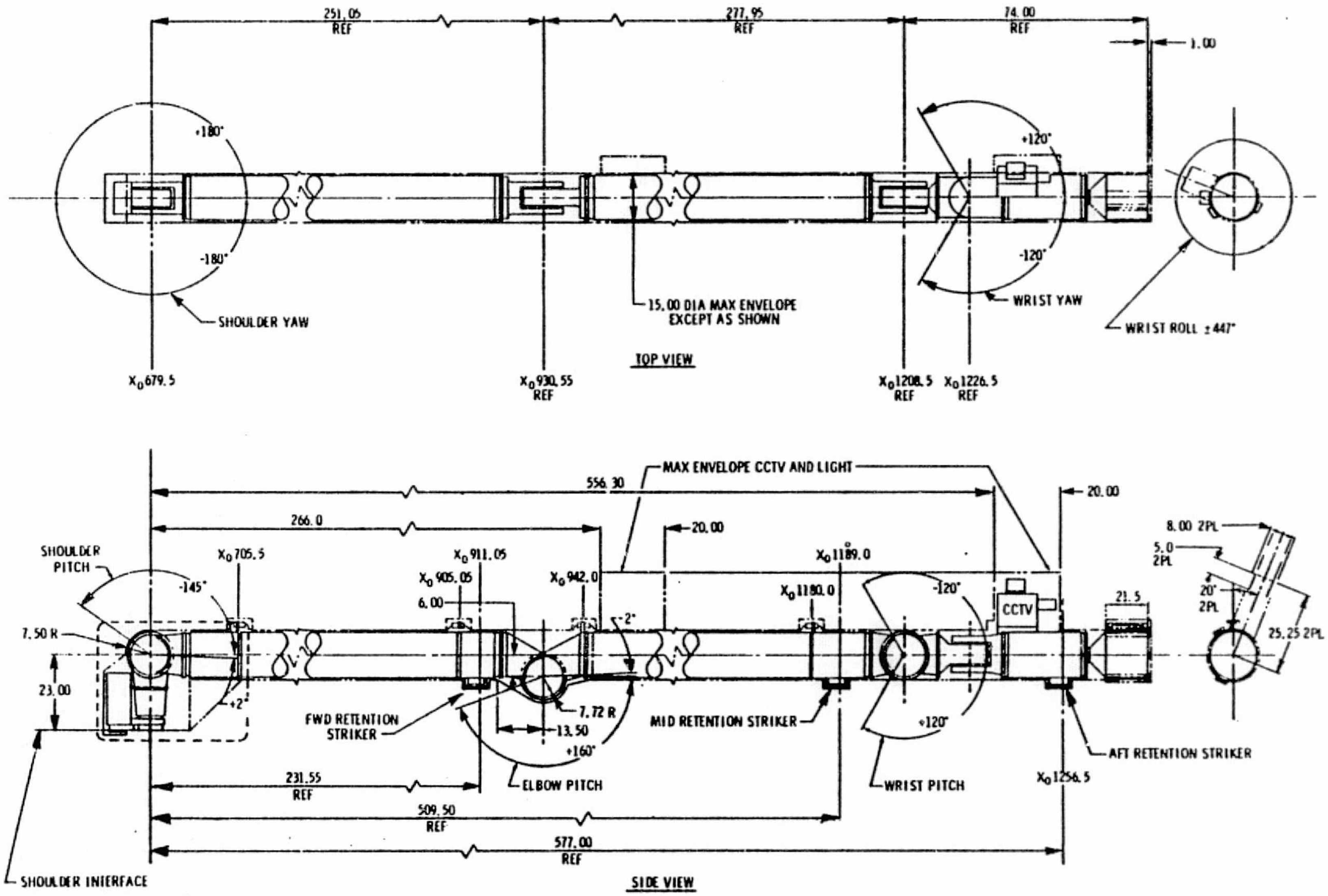


Figure 3.2.2-12 RMS Arm Dimensions and Joint Angle Limits

In order to adapt the RMS arm and the control system to a free-flyer servicer, performing axial and near-radial module exchange and refueling, considerable development work is required. A scaled down version needs to be built and the electronics compartments within the arm need to be relocated to the servicer control modules, on the carrier vehicle (OMV). The shoulder joint design should be modified to accommodate the docking probe. If instead, the docking probe is mounted in an offset position the servicer reach envelope will be reduced. The use of a scaled down version of the RMS as a servicer is shown in Figure 3.2.2-13, for axial or radial module exchange.

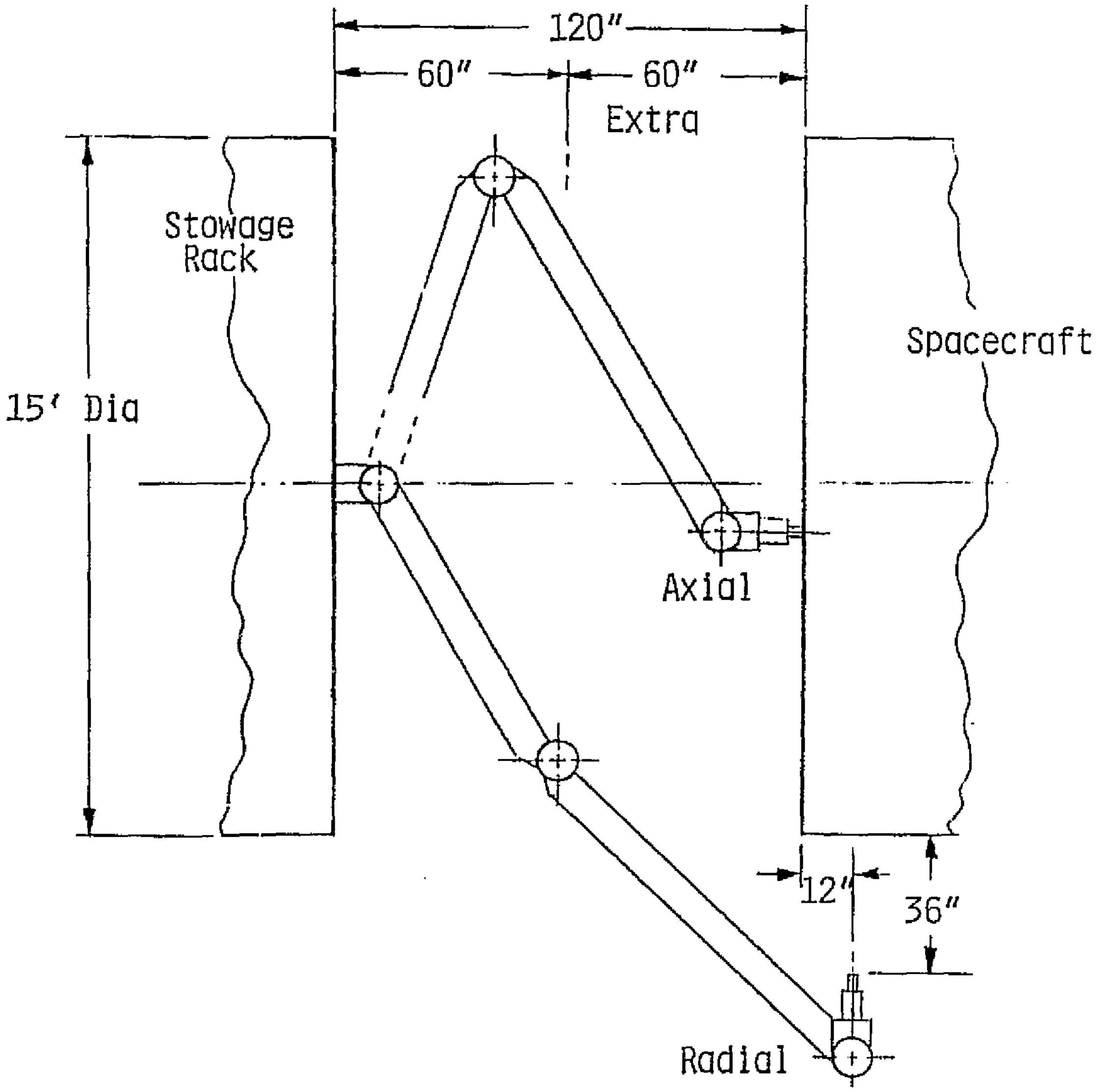


Figure 3.2.2-13 Use of RMS as a Servicer

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Using the existing joint order and angle limits, in order to perform near-radial module exchange and axial module placement in the stowage rack and allow arm stowage within the 15 ft diameter envelope, the distance between stowage rack and spacecraft must be increased to 120 in., which is 100% longer than for the IOSS. The longer docking probe and arm segments mean less accuracy. A counterbalance system must be developed for the ground servicer demonstrations. With the docking probe vertical and the present joint order, the gravity moments will affect the load on the drive motors and interference between counterbalance weights and stowage rack and spacecraft mockups is difficult to prevent.

Following is a summary of the advantages and disadvantages of using the RMS as a servicer.

Advantages:

- Space qualified hardware
- Proven technique
- Suppliers and expertise available

Disadvantages:

- Considerable development work required
    + Scale down necessary
    + Adaptation of IOSS end effector
- Controls need total rework, presently attached to the arm
- Offset arm or docking probe - reduced reach envelope
- Wrist is not compact
- Less accuracy, docking probe 100% longer than IOSS

4) <u>The Remote Orbital Servicing System</u> is a conceptual design of a satellite servicing system, proposed by Martin Marietta Aerospace. The analysis was performed for NASA/Langley Research Center under Contract NAS1-16759. The general configuration of ROSS is shown in Figure 3.2.2-14.

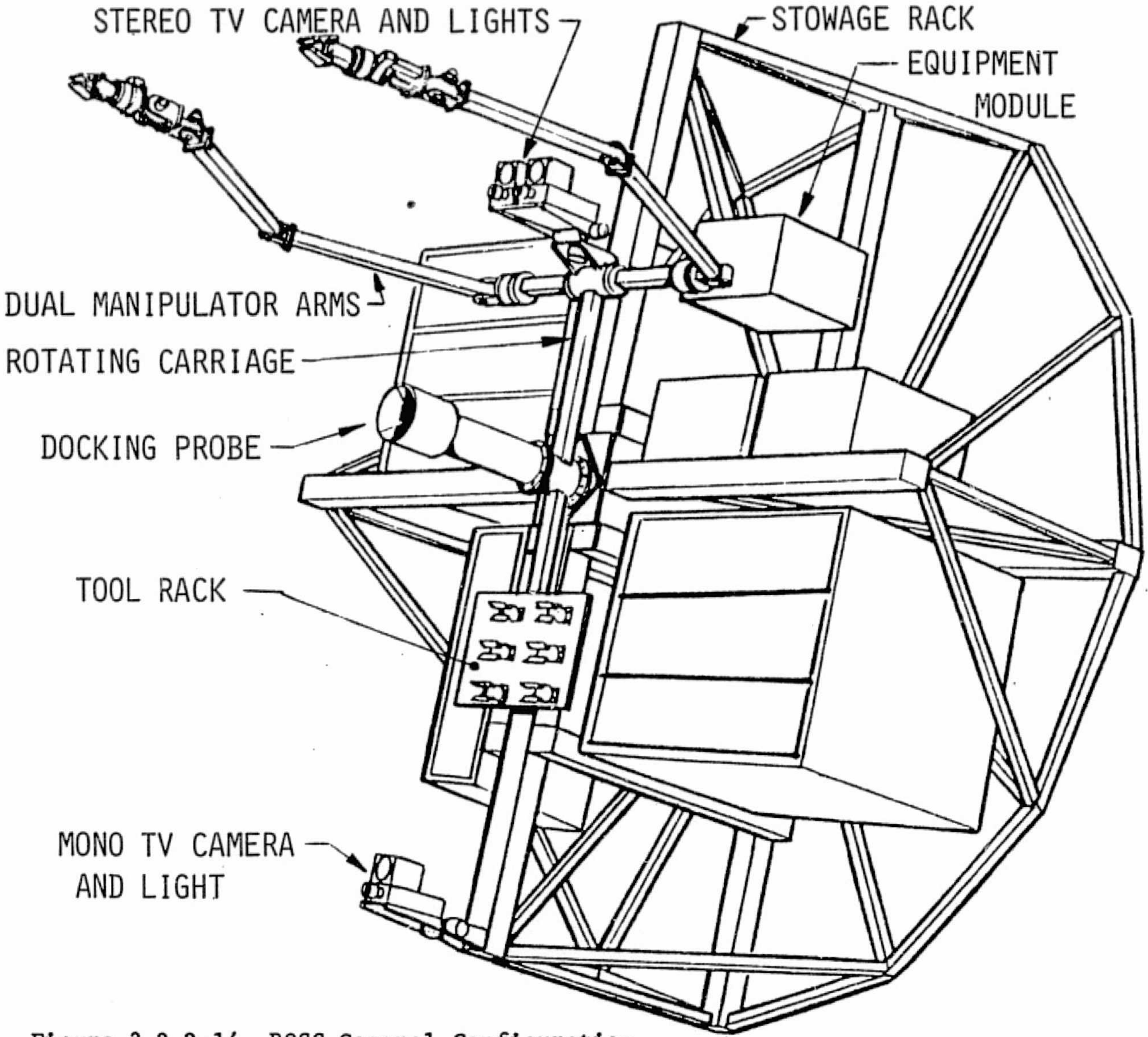

Figure 3.2.2-14 ROSS General Configuration

Like the IOSS, the ROSS servicer is to be attached to a carrier vehicle, such as OMV, which provides power, attitude control, communications link, control and data handling (including video processing), propulsion, docking capability and structural support.

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The stowage rack concept is the one developed for the IOSS. The servicer mechanism is comprised of two manipulator arms (see Figure 3.2.2-15) attached to a rotating carriage, pivoting 360° around a telescoping docking probe.

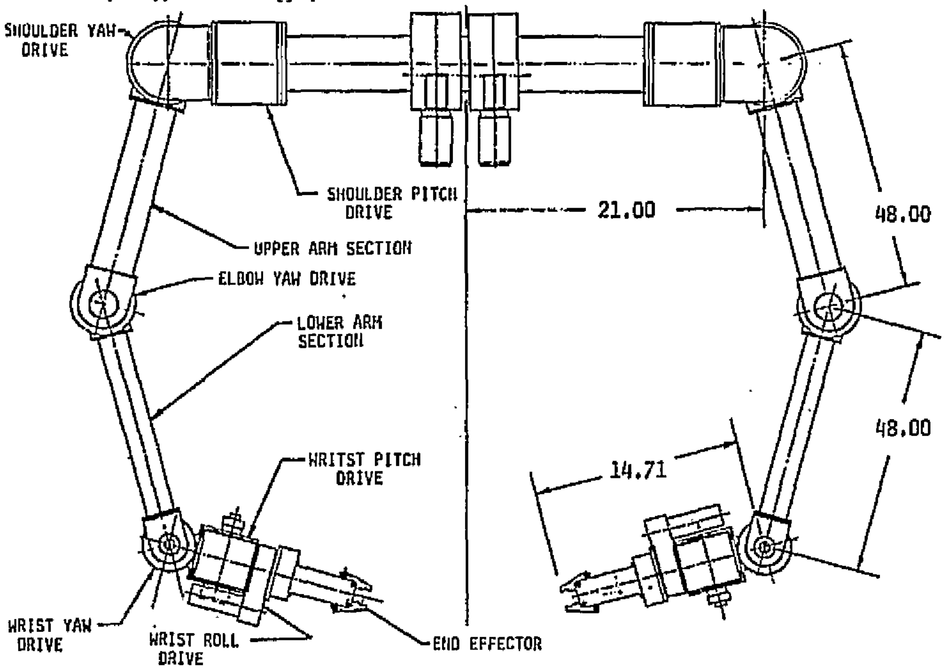

Figure 3.2.2-15 ROSS Dual Manipulator System

Each manipulator arm has the same joint order as PFMA (shown before, in Figure 3.3.3-6) but without the shoulder roll joint. The same PFMA joint design is to be used. The rotating carriage uses the IOSS shoulder roll joint. The two-arms, the carriage "T" section, and the stereoscopic TV camera with lights and pan/tilt mechanism form an anthropomorphic configuration. In addition to module exchange, the ROSS is intended to perform EVA type tasks through remote control. The two-arm configuration is to be used for certain servicing functions such as holding an access door open with one arm while replacing modules with the other arm. Other functions involving two arms are the movement of packages while simultaneously removing/reconnecting connectors in areas of limited accessibility, reorienting a package held with a non-rigid attachment by one of the arms prior to installation in spacecraft or stowage rack, etc. The second arm also provides a backup for the many operations requiring only one arm.

The end effector is similar to IOSS design modified to add force feedback sensors and controllable grip force. It is used for module exchange like the IOSS and for other tasks can use adapter tools stored on a rack attached to the rotating carriage.

The stereo video system provides depth perception in performing arm manipulation activities. It uses two monitors, two imaging lenses and a Fresnel display screen to direct the right and left images to the corresponding eyes of the viewer. This concept has been built and tested at Martin Marietta Aerospace for various simulations (see Figure 3.2.2-16). A mono TV camera on a pan/tilt mechanism is mounted on the rotating carriage (or on the periphery of the stowage rack), to give a different view angle for docking and monitoring hardware removal or insertion in the stowage rack. Two mono TV cameras and lights can be mounted, one near each end effector, for viewing operations within a confined or partially enclosed volume. Another mono TV camera can be mounted, depending on the mission, anywhere on the servicer for missions with viewing requirements exceeding the capabilities of the basic system.

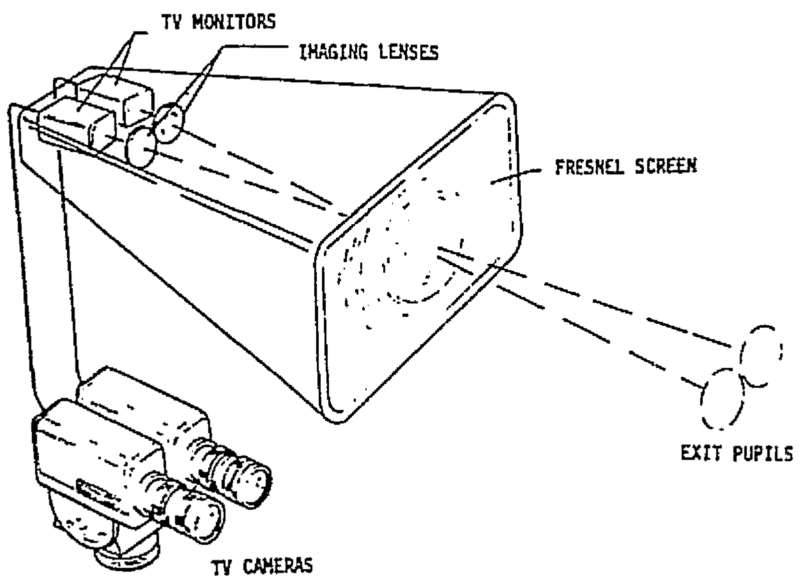

Figure 3.2.2-16 Stereo Video System Conceptual Design

The control of the two manipulator arms is to be done with two 6 degree-of-freedom controllers (man-in the loop mode) or semi-autonomous with pre-stored sequences. The communications link time delay is 1-2 seconds for round trip transmission between the ground control station and the servicer. Because of the time delay an on-board, dedicated processing capability is to be provided for immediate safety shutdown.

Except for redundancy, only one arm is needed for performing all the servicing tasks in the requirements. The access door can be removed as a module in a separate sequence and stored on the stowage rack or the power takeoff of the end effector can power a cover unlatching/opening mechanism that will hold the cover in the open position. The single arm servicer then performs the module exchange the normal way. The electrical disconnect function between spacecraft and the exchanged equipment is performed by the module attachment mechanism using the power takeoff of the end effector. Only one arm is required. A failure analysis and reliability study must be performed to determine the redundancy required for the servicer. Dual motors on drives and dual control circuits may provide the required reliability.

The anthropomorphic configuration of the servicer is not a requirement without a true telepresence capability. Sophisticated, high dexterity end effectors, with tactile sensors and force feedback (simulating the human hand) need to be developed, to supplement the vision system. Simply adding more viewing cameras is unlikely to solve the problem. The end effector may obscure the object, and it is difficult for the operator to view more than one screen display at a time. Development of improved communication links is also required to achieve a significant reduction in transmission time delays considering the increased volume of data from sophisticated sensors and/or additional video circuits.

A complicated control system capable of coordinating the motion of the two manipulator arms needs to be developed before a dual arm servicer can operate in the automatic mode.

Development of sophisticated artificial intelligence capability is needed before unplanned servicing tasks can be performed in an automated mode.

Significant research and development is presently being done in all areas mentioned above and important, but gradual, progress is expected in the not-too-distant future. The anthropomorphic configuration may be required for the future generations of satellite servicing systems, such as ROSS.

A single-arm servicer mechanism, with a simple end effector interface and supplemented by specialized adapters and interface mechanisms, like the IOSS, can be built today with the present technology. It will provide the much needed satellite servicing capability now and the ability to test and develop the elements of future generation servicers.

A rotating carriage with only one arm was considered as a candidate for the servicer mechanism (see Figure 3.2.2-17).

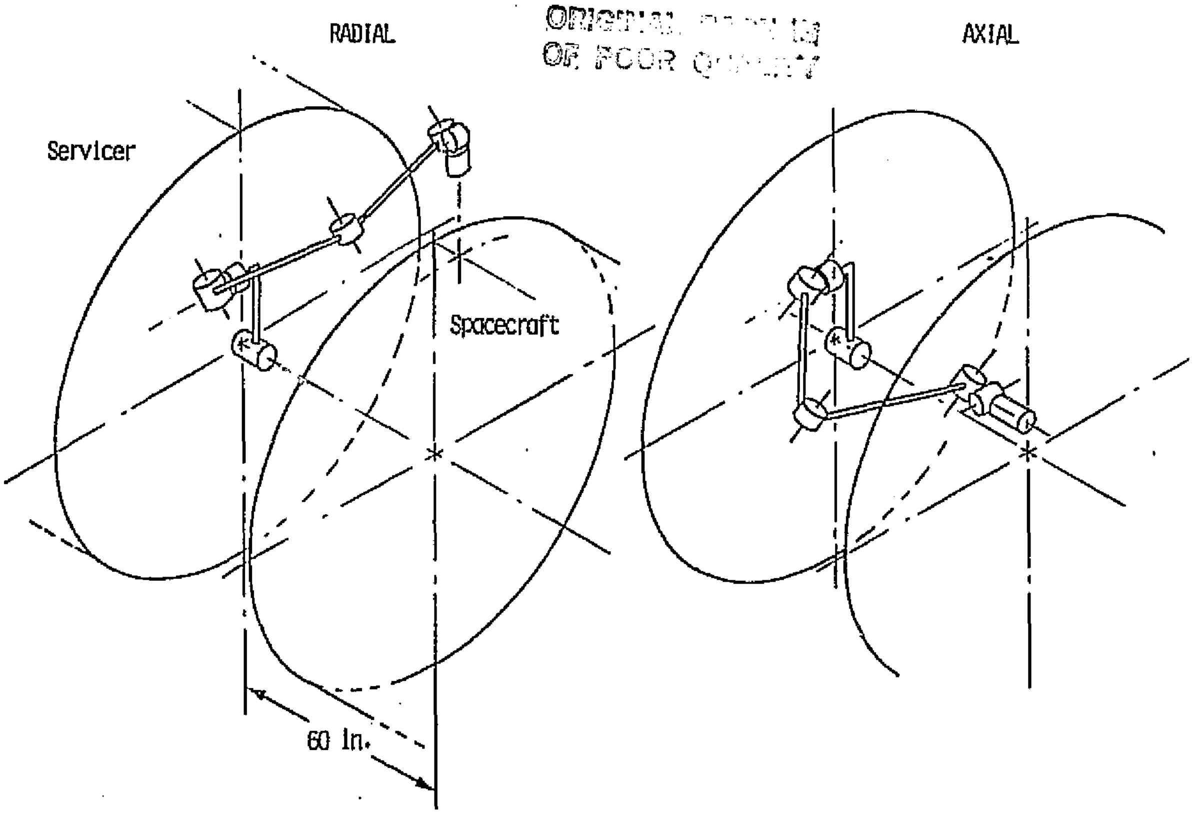

Figure 3.2.2-17 Use of ROSS as a Servicer - Single Arm Configuration

Performing near-radial and axial module exchange requires synchronization of six joints increasing the complexity of the controls system and reducing the accuracy, as compared to the IOSS.

For the ground demonstration servicer, development work would be required to adapt the rotating carriage to the docking probe and stowage rack mockups and for the counterbalance system. With the docking probe in vertical position, because of the joint position the counterbalance system would be inefficient. It will produce variable extra load on the drive motors. Other positions for the docking probe would require rotation of the entire stowage rack and spacecraft assembly mockup in order to service more than one location. This increases the complexity and the cost of the system. Regardless of the orientation of the docking probe, interference of the counterbalance system with the stowage rack and the spacecraft is difficult to avoid. The load capability of the arm (like the PFMA) is lower than the IOSS.

In conclusion, the advantages and disadvantages of using the ROSS configuration for the servicer mechanism are as follows:

Advantages:

- Redundant design - two arms
- Suppliers available
- Capable of doing two operations at the same time
- Variable length docking probe

Disadvantages:

- Unit has not been built
- Requires adaptation to stowage rack
- Lower accuracy - one extra joint at shoulder
- Two arms - mechanical complexity and increased cost (one arm configuration can accomplish module exchange)
- Complex control system
- Lower fidelity, 1-g vs. flight
- Lower load and moment capability

5) The Slave Manipulator Arm was developed by Martin Marietta Aerospace as an internally funded effort during the period of 1973–74. It is a six degrees of freedom mechanism (see Figure 3.2.2-18). The arm has been used in laboratory simulations of orbital assembly techniques and to develop the requirements for the orbiter Remote Manipulator System. It has two modes of control, proportional rate and position control. Force feedback on each drive is sensed by measuring the motor current or the servo error signal and it is reflected to the operator either through torquers on a 6 DOF hand controller or through meter displays.

The SMA has an articulated counterbalance system at shoulder level. In order to be used for ground servicing demonstrations, adaptation to the stowage rack is necessary. Constraints are similar to those for the RMS (see Figure 3.2.2-13). The wrist/end effector is not compact and adaptation of the IOSS end effector would require additional development.

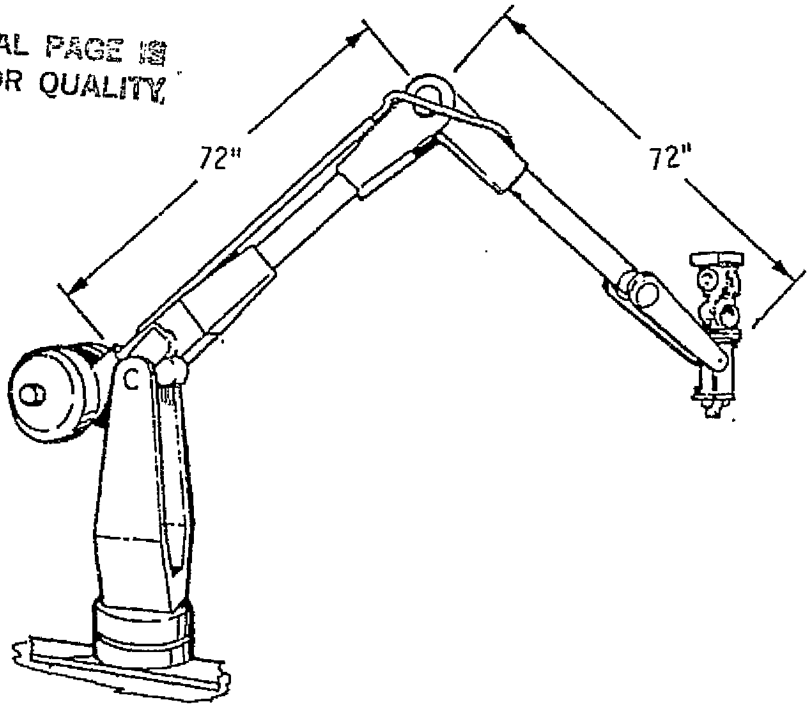

Figure 3.2.2-18 The Slave Manipulator Arm

Advantages:

- Reliable, proven design
- Expertise and suppliers available

Disadvantages:

- Requires development work for adaptation to stowage rack
- Offset arm or docking probe - reduced reach envelope
- Wrist not compact
- Lower fidelity 1-g vs. flight unit
- Long docking probe - 100% longer than IOSS

6) The Advanced Servomanipulator System is a conceptual design resulting from a study performed by Martin Marietta Aerospace for DOE Oak Ridge National Laboratory. A dual arm manipulator system concept design (see Figure 3.2.2-19) was developed for performing maintenance tasks in the radiation contaminated environment of nuclear power plants. It has two arms, each with six degrees of freedom. All joints, except the shoulder elevation drive are similar to the PFMA drives. Three of the drives have vertical axes: the shoulder yaw, the elbow yaw and the wrist yaw. The overhanging loads are reacted through the drive bearings rather than through motor torque or counterballancing weights. The bulkiness of mechanical counterbalancing is avoided, a major source of motor heating is removed and the result is a much lighter weight system. This advantageous joint orientation is also used in the Engineering Test Unit of the IOSS, in the shoulder and wrist roll joints.

The arms are of modular design. Each arm can be easily disassembled in three sections, using the other arm. Electrical disconnects for the control circuits are provided at the arm segment interface. A tool stowage rack is provided on the manipulator body for adapter tools compatible with the end effector interface.

Considerable development is required in order to use the ASMS mechanism as a ground servicer. If only one arm is used, new shoulder joints are needed for pivoting around the docking probe and for elevation.

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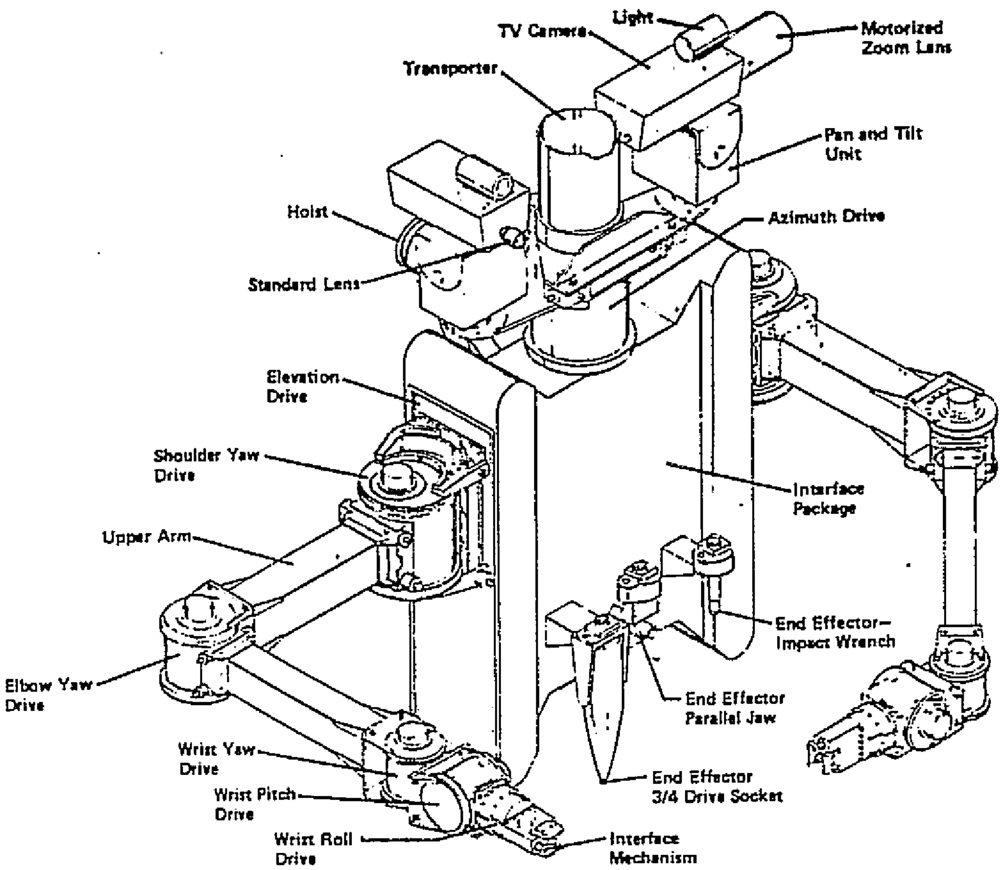

Figure 3.2.2-19 Advanced Servomaniplator System

If the dual arm configuration is retained, an extra joint is required for pivoting motion and all the counterbalancing system disadvantages of the ROSS system would apply.

Coordination of the two arms will require a complex control system.

The principal advantages and disadvantages of using the ASMS as the servicer mechanism for ground demonstrations are:

Advantages:

- Redundant design - two arms
- Capable of doing two operations at the same time
- Designed to be maintained in a closed environment by a second ASMS

Disadvantages:

- Unit has not been built
- Requires development work for integration into the servicer system
- Two arms - mechanical complexity
- Complex control system - coordination of two arms
- Lower fidelity, 1-g vs, flight

3.2.3 Servicer Mechanism Coarse Screening

The following servicer mechanism candidates for the ground demonstrations system were eliminated in a coarse screening process for not meeting the "Must" requirements, defined in Section 3.2.1:

The Remote Manipulator System

- Docking probe extension required (100%)
- Arm or docking probe must be offset

- Arm length must be reduced
- All joints must be reduced in size
- Hardware is expensive
- End effector adapter is required
- Stowage rack modifications required

<u>The Remote Orbital Servicing System</u>

- Hardware does not exist
- Complex control system
- Arm length must be increased
- Stowage rack modifications required
- Many single arm features incorporated in PFMA

<u>The Slave Manipulator Arm</u>

- Docking probe extension required (100%)
- Arm or docking probe must be offset
- Hardware does not belong to MSFC
- Wrist is not compact
- End effector adapter is required
- Stowage rack modifications required

<u>The Advanced Servomanipulator System</u>

- Hardware does not exist
- Difficult to adapt to servicing
- Mechanically complex
- Concepts are included in ROSS

The remaining candidates considered for the servicer mechanism selection for ground demonstrations are the Engineering Test Unit of the IOSS and the PFMA arm.

3.2.4. Comparison of ETU and PFMA

A comparison of the two servicer mechanisms was performed based on the requirements defined in Section 3.2.1. The results are shown in Table 3.2.4-1. Weighting factors were assigned to each requirement and they reflect our opinion regarding the importance of these criteria in the selection of the servicer mechanism. This was a subjective process and was based on our experience and best judgment in considering all the elements affecting the performance and the cost of the servicer.

For each requirement that is best met by the PFMA, a plus sign (+) was marked in the respective column of Table 3.2.4-1. Consequently a negative sign (-) was marked in the PFMA column for the requirements best met by ETU and a zero (0) when the two candidates are approximately the same. If both candidates meet a certain requirement but the use of PFMA involves a delta cost in order to be upgraded to the performance level of the ETU, a (C) was marked in the PFMA column. These costs were estimated and are given later. The comparison between ETU and PFMA assumes the use of the seventh PFMA joint to obtain the desired motions.

Table 3.2.4-1 Comparison of ETU and PFMA for 1-g Servicer

REQUIREMENT	WT	ETU	PFMA	
HIGH FIDELITY				
Efficient representative operations	10	- Can do all operations	- Can do all operations	(0)
			- Potential wrist counterbalance interference	(0)
Similarity to flight Configuration	8	- Software for coordinated motion is available	- Software for coordinated motion can be developed	(C)
		- Software for some module exchange trajectories is available	- Software for module exchange trajectories must be developed	(C)
			- Stowage rack must be modified	(C)
			- More complex equations for Coordinated joint control	(C)

Table 3.2.4-1 Continued

REQUIREMENT	WT	ETU	PFMA	
ACCURACY				
Number of joints	4	- 6 joints	- 7 joints	(-)
Length of arm segments	3	- 118 in. to end	- 137 in. to end effector (to obtain same reach)	(-)
Length of docking probe	5	- 60 in.	- 60 in. based on use of seventh joint	(0)
VERSATILITY				
Full reach envelope	8		- Arm length must be increased	(C)
			- Mechanism or docking probe must be offset	(-)
Compact wrist/end effector	5		- Larger wrist moment arm (21 vs. 15 in.)	(-)
Adapter compatibility	4	- Refueling and electrical connectors	- Refueling and electrical connectors can be added	(0)
			- Modify end effector to IOSS configuration	(C)
RELIABILITY				
Operational experience	5	- Unit operated at MSFC	- Unit operated at MSFC	(0)
		- Suppliers and expertise available	- Suppliers and expertise available	(0)
			- Space available joints	(+)
			- Longer operating experience	(0)
Performance margins	5		- Wrist counterbalance not effective for some arm configurations	(0)
			- Load and moment capability may be too low (13 vs. 20 lb)	(-)
			- Unbalanced gravity loads reacted more by drive motors	(0)
System Complexity	3		- Uses seventh joint	(-)
			- Control laws more complex	

Net of 23 Negatives (Weighted Score)

The activities required in order to bring the PFMA to the current status of the ETU and the estimated costs involved are shown in Table 3.2.4-2

Table 3.2.4-2 Costs to Bring the PFMA to Current Status of ETU

ACTIVITIES	COST (K$)
- Software for coordinated motion	30
- Software for module exchange trajectories	40
- Control system interfaces	20
- Stowage rack and docking probe modifications	40
- Increase in arm length (includes wiring and counterbalance changes)	30
- End effector modifications	20
TOTAL:	180

The PFMA was designed and built as space qualified hardware. The supplemental costs detailed in Table 3.2.4-2 must be weighted against the delta costs for designing, building and flight qualifying the ETU alternative mechanisms for use in space.

Before selecting the servicer mechanism a risk analysis was performed separately for the PFMA and ETU in order to assess the risk in terms of program schedule impact and probability of occurence of problems and to identify the necessary risk control actions to be implemented (see Table 3.2.4-3).

Table 3.2.4-3 Risk Considerations

Candi-date	Concern	Proba-bility	Program Impact	Risk Assessment
Both	CRITICAL FAILURE MODES - Not enough Redundancy (Failure analysis not started)	MED	HIGH - Becomes evident during tests or operation - Requires considerable effort to fix	UNACCEPTABLE Perform failure analysis and influence design
Both	REMOTE DOCKING - Not available for flight demonstration	MED	HIGH - Becomes evident late in the program - Considerable time and effort to correct	UNACCEPTABLE Coordinate docking development (OMV)
Both	FUNDING STREAM Not adequate	MED	HIGH - Schedule slips - Cost increases	UNACCEPTABLE Establish a budget line item
Both	FAILURE TO ACCEPTABLE PRO-MOTE THE SERVICING TECHNOLOGY (Assuming viable, flexible system)	MED	HIGH - Orbital servicing Not accepted	UNACCEPTABLE Establish industry standards for servicing inter-face

Table 3.2.4-3 Continued

Candi-date	Concern	Proba-bility	Program Impact	Risk Assessment
	<u>STATE-OF-THE-ART</u>			
	- Unable to flight qualify components or suppliers not available			
IOSS	All components	LOW	MED	LOW
PFMA	Electronics	LOW	LOW	LOW
Both	- Flight qualified refueling components not available	MED	LOW - Availability predictable - Ground demo. does not need qualification	LOW
	<u>MARGIN OF SAFETY</u>			
	- Not enough load capability for 1-g			
IOSS		LOW	LOW	LOW
PFMA		MED	LOW	LOW Controlled through design analysis
Both	- Not enough accuracy ground demo flight demo (docking)	 LOW MED	 LOW MED - Discovered during design - Large effort to correct	 LOW MED Controlled through automated target recognition

3.2.5 Servicer Mechanism Recommendation

In concluding this servicer mechanism selection trade study our recommendation is to continue to use the Engineering Test Unit for servicer ground demonstrations.

ETU was designed to conduct 1-g module exchange demonstrations. Its counterbalancing system is efficient, producing minimum load in the drive motors. It has a lifting force capability approximately twice the PFMA capability.

The ETU servicer mechanism is compact and efficiently performs module exchange and other servicing tasks, and requires only a 60 in. spacing between stowage rack and spacecraft. It has high quality joints of the PFMA type, which can be built and qualified for space applications with minimum expense.

The PFMA is not as desirable as ETU because it requires extensive development work in order to integrate it in a servicer ground demonstration system. The main drawbacks which make the PFMA less desirable for ground servicer demonstrations are:

- Offset of mechanism or docking probe
- Limited module lifting force
- Use of seventh joint and related control system complexity
- Larger wrist moment arm
- Longer arm affects stiffness and accuracy

3.3 GROUND DEMONSTRATION ACTIVITIES

The Engineering Test Unit of the IOSS was selected as the servicer mechanism for ground demonstrations based on the results of the tradeoff study presented in Section 3.2. The selection of the required hardware for ground demonstrations of MMS module exchange, refueling and other servicing tasks are documented in Section 3.1

In this Section, the ground servicer demonstration objectives were reviewed and several demonstration activities are recommended. A cost estimate of the hardware and software modifications of the ETU required for conducting the proposed ground demonstration activities, was performed.

3.3.1 Objectives of the Servicer Ground Demonstrations

The principal objectives of the servicer ground demonstrations, using a modified Engineering Test Unit, are:

1) To Demonstrate the Adaptability and Flexibility of the Module Exchange Concept. This can be best done by demonstrating the exchange of the MMS module, because it is the only on-orbit serviceable modular concept that is operational and because it was designed for a different servicing interface. Additional demonstrations should be conducted to prove that the IOSS is a flexible servicing system, without imposing important constraints on spacecraft design. Exchange of equipment at the individual component level, such as battery replacement, including the opening or removal of an access door/thermal protection cover can further demonstrate the versatility of this servicer system;

2) To Demonstrate the Use of the Ground Servicer as a Laboratory Tool for Development of new servicing concepts, new hardware and software, before further flight testing and operational implementation. A good example is the development of a satellite remote refueling capability. The ground servicer can also be used as an integration and checkout facility. Development of an automatic target recognition and error correction system, of new controls or of new tools and adapters can benefit from the use of the ground servicing demonstration system as a laboratory tool. New sensors, sophisticated end effectors and other elements of the next generation of servicing systems can be developed using the ground and the flight servicer demonstration units.

If problems arise during the flight tests or operational servicing, the ground demonstration unit could be used for finding and/or checking out solutions;

3) To Demonstrate the Use of the Ground Servicer as a Training Facility. Training of the operators for the flight demonstrations as well as for actual servicing operations can be done using the ground servicer system. For this reason, it is important that hardware and software commonality with the flight units is designed into the ground demonstration servicer. This will also make possible more convincing, high fidelity ground servicing demonstrations.

The main role of the servicing ground demonstrations is to support further flight demonstrations. The availability of on-orbit servicing capability can be convincingly demonstrated to the user community only through flight tests. The acceptance of on-orbit servicing methods by the spacecraft designer is also linked to the financial and programmatic commitment of NASA for timely development of the operational capability.

Not all concepts tested on the ground unit will develop into flight hardware. It is important that as much development as possible be performed using the ground demonstration unit before testing on the flight demonstration system. However, the flight demonstrations should not wait until all the development projects have been tested on the ground. Flight demonstrations and tests should be scheduled as soon as one particular technology (for instance module exchange) has been proven in ground demonstrations. This will improve the acceptance of on-orbit satellite servicing methods and help speed up their incorporation in new spacecraft designs.

3.3.2 Candidate Activities for the Servicer Ground Demonstrations

Several near-term activities were proposed and the costs involved were estimated (see Table 3.4.2-1).

1) Upgrading of the Control System of the ETU The refurbishment requirements of the ETU are discussed in Section 4.0. The need for upgrading the control software and hardware to provide smoother, more accurate operation and to add a manual-augmented mode was identified.

New control software should be developed based on a combination of the software being used by MSFC and that used during the Engineering Test Unit Design Acceptance Review, conducted at Martin Marietta Aerospace.

A new, simple control console should be built. It should incorporate two 3-DOF hand controllers, provided by MSFC for the manual-augmented mode, as well as the servicer control panel that is part of the existing servodrive console, a new television monitor and the existing computer terminal presently used with the MSFC PDP 11/34 computer.

Refurbishment of the electro mechanical systems of the ETU should include the repair of worn cable ties on the arm, check out of all cables for electrical continuity, replacement of the lamp support plate, repair or replacement of optical targets on spacecraft mockup and on stowage rack, and paint touch-up where necessary.

The servicer control software and hardware operation should be checked out at MSFC in three modes: 1) supervisory (with and without operator action between steps), 2) manual-augmented and 3) manual joint by joint. The ability to control the system in each of the three modes in performing module exchange between the mock-ups of the spacecraft and the stowage rack should be demonstrated. A servicer software user's manual should be prepared with sufficient information to permit MSFC personnel, familiar with the operation of the PDP 11/34 computer, to use the new software.

2) <u>Multimission Satellite Module Exchange</u> A light weight mockup of the MMS module should be designed and two units should be built. The design goal is a maximum weight of 20 lb for the module, including the electrical connector(s) and the module retention hardware. The module mockup should be a full size representation of the outside shape and dimensions of the MMS module, should have the same attachment interface and provide adequate structural support for the two attaching fasteners and for the two latch receiving brackets. As much as possible of the actual module attachment hardware and connector mounting hardware should be used in the mockup. The fastener operating torque should be the same or as close as possible to the nominal value for the actual flight unit.

An adapter tool interfacing with the ETU end effector at one end and with the MMS module servicing tool (MST) interface at the other should be designed and built. It could be a standard MST without batteries, controls and EVA handles, and provided with a standard ETU end effector interface and an electrical connector.

Preliminary contacts were made with the MMS Project Office at NASA - Goddard Space Flight Center and they are willing to cooperate with MSFC and the contractor in defining the spacecraft to servicer arm interfaces. The GSFC Satellite Servicing Project has recently issued a Research and Technology Objectives Plan (RTOP) to perform a study for defining the interface requirements for the remote servicing of the MMS spacecraft and for adaptation of the standard MST for remote orbital servicing. Arrangements should be made for obtaining as GFE from Goddard Space Flight Center a standard MST to be modified for use as an adapter for the ETU and also for obtaining the necessary MMS module retention hardware for two module mockups and for three attachment locations on the stowage rack and spacecraft mockups. Close cooperation between MSFC, GSFC and the contractor should be developed for designing and building the MMS servicing adapter tool, the MMS module mockups and other elements of the MMS module servicing demonstration.

Modification of the present spacecraft mockup is necessary in order to incorporate support structure and compatible attachment interfaces, connectors and sensors for one MMS module (see Figure 3.1.1-23).

Modifications of the stowage rack mockup are necessary for receiving the MMS module in two locations.

The modified spacecraft and stowage rack mockups, the module and the adapter tool should be integrated with the ETU at MSFC and exchange of the MMS module mockup should be demonstrated.

The increased end effector load due to the MMS module mockup, tool adapter and other servicer modifications should not exceed the servicer lifting capability. An engineering analysis of all affected components should be conducted to ensure their safety and integrity. Some redesign and modification of the ETU is anticipated.

3) <u>Refueling Interconnection Equipment</u> For initial ground demonstrations, transfer of water between the storage rack and spacecraft mockup, using air as pressurant gas, should be demonstrated, using a special refueling resupply interface unit and a cable/hose management system.

A servicer refueling module mockup should be built, comprised of water tank, air tank, piping and valves, a hose/cable management system, a refueling interface unit, instrumentation, controls and support structure. A conceptual design of such a module is shown in Figure 3.1.2-15. The refueling interface unit should carry disconnects for water, air and electric cables, should have a translation mechanism, attachment alignment mechanism and a dust cover removal mechanism. A conceptual design of such a refueling interface unit, prepared by Martin Marietta Aerospace, is shown in Figure 3.1.2-8. Simplified functional mockups should be built for disconnect valves with leak test and purge capability.

Modifications of the spacecraft mockup are necessary in order to accommodate the water and pressurized air tanks, piping, valves instrumentation and controls and the refueling interface.

4) AXAF Module Exchange Demonstrations A demonstration of focal plane instrument module exchange requires building two large volume, light weight module mockups (shown in Figure 3.1.3-9). The module retention system could be a light weight version of the base mounting interface mechanism (see Figure 3.1.3-2). The design should also include electrical and fluid disconnects.

Modification of the spacecraft mockup is required to accommodate radial removal of the AXAF module mockup including a hinged thermal cover with an unlatching/opening mechanism, actuated by the power takeoff of the servicer.

Modification of the stowage rack to accommodate the AXAF module in two locations should provide structural support and latch interfaces. A hinged thermal cover, similar to the one on the spacecraft mockup should be fitted on one of the two stowage rack locations. The other location is for temporary storage.

5) Battery Exchange Demonstration Demonstration of battery or other such individual component level exchange is necessary in order to prove the operational flexibility of the ETU servicer.

A representative battery mockup should be designed and two units should be built. The mockup should have an electrical disconnect and a light weight latch mechanism capable of mating/demating the disconnect. As an alternative, the battery mockup could be attached to the base structure using captive fasteners. An adapter tool should then be built to actuate the fasteners and mate or demate the disconnect. MMS type fasteners and the MMS adapter tool could be used instead of standard captive fasteners.

A removable thermal/access cover should be designed and built. It can be removed like a module, in a separate servicing sequence, and placed on the stowage rack. It should be provided with a light weight latch and attachment mechanism. Another option is to use a hinged cover, similar to that to be used for AXAF module exchange.

The stowage rack and the spacecraft should be modified to receive the batteries: one compartment with cover on the spacecraft mockup, one compartment with cover and one temporary attachment location on the stowage rack.

6) Automatic Target Recognition and Error Correction A previous study* of the expected error of mechanical arms, conducted by Martin Marietta Aerospace with internal funding, shows that accuracy approaching ± 0.80 in. (3σ) is achievable for a system like the IOSS. This number should not be compared with the ETU repeatability of 1/8 in. which is only one small component of the overall error. Among the dominant sources of error considered, is the vehicle docking misalignment. Without special provisions, docking misalignment can be on the order of degrees. Docking misalignment not exceeding 0.3° in any of the three axes were considered in the above-mentioned study. However, if the standard RMS end effector is used as a docking probe, post rigidization accuracy of $\pm 0.4°$ is expected in the roll direction and $\pm 0.15°$ in the pitch and yaw direction**. Roll is the most critical and is unfortunately the most difficult to align accurately.

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* Orbital Inflight Maintenance (Project 27D) Vol. 2 - Accuracy Capability of Mechanical Arms, Martin Marietta Aerospace, Report No D76-48727-002, Dec. 1976.

** R.G. Daniell et. al. "The Design and Development of an End Effector for the Shuttle Remote Manipulator System" 16th Aerospace Mechanisms Symposium (J.F. Kennedy Space Center, Florida) May 13-14, 1982 NASA Conference Publication 2221.

The IOSS end effector capture envelope is ± 0.75 in. and the guide capture capability of the side mounting interface mechanism is ± 0.50 in. These capture capabilities are marginal, when using the RMS end effector as a docking probe. The use of adapters for the ETU end effector and/or docking probe enhances the system operational flexibility but, at the same time, may appreciably decrease its accuracy below the minimum acceptable level. For radial module removal up to 63% larger errors are expected.

In manual control modes the operator can make the required corrections before engaging the module or the end effector, by using the video capability.

In the supervisory mode, however, an equivalent capability needs to be developed, in the form of an automatic target recognition and error correction system. The system can use the existing video equipment and the on-board computing capability, to scan and interpret the TV image, prior to engagement detect the error, issue the required commands for correction to the servicer arm control system, verify the results and then command the final engagement. An autonomous video rendezvous and docking system is being developed by Martin Marietta Aerospace under a contract from MSFC***. It requires a modified optical target with three reflective spots, special software and a special computer interface box to handle the data processing. The system has been proved in the Space Simulation Laboratory of Martin Marietta Aerospace and the technology is readily applicable to the servicer.

7) Engineering Test Unit Electronics Update Improvements in the reliability of the Engineering Test Unit can be obtained by updating some of its controls electronics, such as replacement of relays with solid state switches, replacement of wire wrapped boards with printed circuit boards and by improving some other

*** Development of an Autonomous Video Rendezvous and Docking System, Martin Marietta Aerospace MCR-83-584, Phase 2, June 1983, MSFC Contract NAS8-34679 circuit elements. The ETU has had few failures compared to the expected level for equipment of its complexity (as shown in Section 4.0). However, most of the recorded problems are linked to failures of electronic components. The changes proposed have a potential for improving the overall reliability of the ETU if it is to be used extensively in the future.

8) Convert ETU Control System from Analog to Digital The modifications list includes digital sensors (like optical encoders inside the joints), digital inputs and displays, microprocessor computations and a new control panel. These modifications will improve the accuracy and the stability of the controls system. Process controllers are available off-the-shelf, for use as microprocessors.

9) Other Ground Demonstration Activities Which will prove the system flexibility by performing potentially useful satellite servicing tasks are listed below.

- Tank exchange (or other propulsion system modular components exchange). Development of an in-line coupling meeting the requirements of Section 3.1.2 is necessary.
- Light weight side and bottom attachment, as needed. Current technology can be used.
- Other interface mechanisms - as needed.
- Other adapter tools for specialized tasks. May include adapter tools for removing special fasteners, or a PFMA type end effector (see Figure 3.1.4-3) for gripping and deploying an antenna or performing other contingency tasks.

3.3.3 Recommendation for the Servicer Ground Demonstrations

Before more ground demonstrations are performed, upgrading of the servicer controls as outlined in Section 3.3.2 is recommended, to ensure high fidelity, convincing demonstrations.

MMS module exchange is recommended to be the next demonstration activity. The required technology is available and the potential space applications are immediate.

Other generic modules, like the AXAF or communications satellite module exchange demonstrations are recommended next. These activities must be coordinated with the respective project offices in order to help incorporate remote servicing capability in the spacecraft design.

In parallel with the activity described above, the development of refueling hardware and its ground demonstrations are recommended. This activity should use the results of other development efforts, described in Section 3.1.2, and integrate them in a remote refueling system for ground tests and demonstrations.

Development and ground demonstration of an automatic target recognition and error correction system is recommended as the next activity. The purpose is to assure the required accuracy for the success of the flight servicing demonstrations as discussed in Section 3.3.2.

Flight demonstration simulations, training and problem solving using the ground servicer as a laboratory development tool and training facility are the next recommended activities, in support of flight servicer demonstrations and actual remote servicing operations.

3.4 GROUND DEMONSTRATION PLAN

A recommended schedule for performing the servicer ground demonstrations and the estimated cost of these activities are shown in this section.

3.4.1 Schedule

The ground demonstration schedule is shown in Figure 3.4.1-1. The first five items are ground demonstrations and are arranged in a waterfall pattern with significant overlapping during procurement and preparation for the demonstrations. However, the actual demonstrations (tests) are conducted one at a time. The last three activities shown in the figure are flight support activities. Generally, the demonstrations themselves are a month or less with most of the time being spent in preparation and checkout. The ground demonstration activities were described in Section 3.3.2.

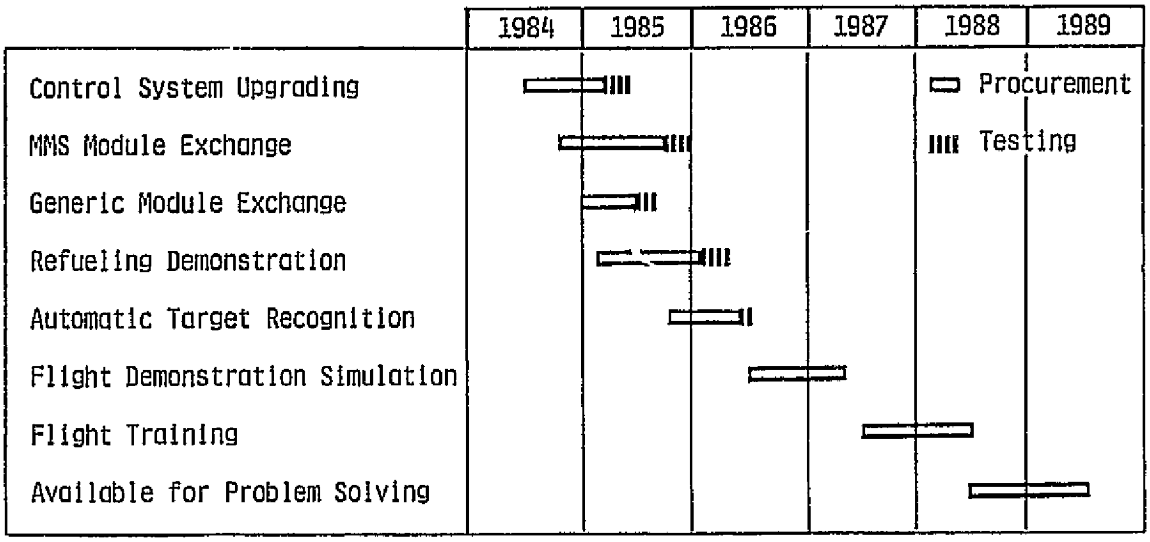

Figure 3.4.1-1 Ground Demonstrations Program Plan

Other activities that should be supported by the ground servicer demonstrations during breaks in the flight support activities are:

- Development of special refueling and electrical disconnects such as cryogenic or high pressure disconnects, self aligning conical electrical connectors, etc.
- Development of in-line fluid couplings for tank and other propulsion system components replacement
- Demonstration of other servicings tasks specific to the space station operations such as resupply of other fluids, space maintenance and assembly tasks, etc.

3.4.2 Cost Estimate

Cost estimates were performed for the activities recommended for the servicer ground demonstrations and the results are given in Table 3.4.2-1. The effort required for each activity was described in Section 3.3.2. Included in the cost were the design, procurement, fabrication, delivery and checkout at MSFC. Not included in the cost estimates are the test planning, test activities, data collection and analysis, and test report preparation. Costs are in 1984 dollars.

Table 3.4.2-1 Ground Demonstrations Cost Estimate (Thousands of 1984 dollars)

Item	Total
1. Control System Upgrading	100
2. MMS Module Exchange	450
3. Generic Module Exchange (Three Types)	600
4. Refueling Demonstration	250
5. Automatic Target Recognition	100
Total	1,500

4.0 ENGINEERING TEST UNIT REFURBISHMENT REQUIREMENTS

The Engineering Test Unit (ETU) of the IOSS was inspected at Marshall Space Flight Center, Information and Electronics Laboratory, during a four day trip on October 10-13, 1983. The purpose of the trip was to determine the refurbishment requirements for using the ETU in the ground demonstration program.

A review of the ETU failure history was performed and its condition was assessed and compared with the needs and requirements of the demonstration program. Electromechanically, the system is in very good condition. The necessary improvements in the software and controls as well as the analysis and design effort required in order to increase the ETU capability were identified.

4.1 MSFC OPERATING EXPERIENCE

We reviewed the ETU operational records since it was installed in the Information and Electronics Laboratory in April, 1978. A complete log of run-time was not available. However, the existing records cover most of the activities between April 1978 and December 1980 and all the failures and repairs to date.

There were no failures in the arm itself (mechanical or electrical). A few minor failures were experienced in the control system. They were due to dirty card contacts, failed electronic components or overloads through faulty grounding when the new PDP 11/34 microcomputer was installed. Also, the TV camera that is attached to the arm had several failures and had to be removed twice for repair. The cause of the failure was traced to foreign matter inside the electronics compartment of the camera. The rate of failures was, however, lower than normally expected for a control system of this type and complexity. The repairs were performed by MSFC personnel. The Information and Electronics Laboratory personnel are satisfied with the overall performance of the ETU. The documentation delivered with the equipment was considered complete and very useful for quick and easy failure isolation and repair.

A review of the operational history showed that the activities of the ETU consisted of initial development tests and demonstrations at Martin Marietta in Denver, and then, after the delivery to Marshall Space Flight Center, demonstrations to various groups, including interested technical personnel, news media and tour groups, practice and tests of the manual mode controls, software development, module exchange activities and other studies.

The total (continuous) operating time per joint was estimated at 85 hours, as detailed in the following.

Prior to delivery to MSFC, the total arm operation time was approximately 200 hrs, and because of sequential joint actuation, about one-fourth of this time represents continuous joint operation. At MSFC, the arm was actuated approximately ten minutes for each hour of demonstrations, practice, tests or software development and one-fourth of this time represents continuous joint operation.

<u>Prior to Delivery to MSFC:</u>
Approx. 200 hr of arm operation /4--------------------50 hr per joint

<u>After Delivery to MSFC:</u>
200 one hr demonstrations
200 hr of practice
120 hr of software development
300 hr of module exchange and other studies

Total 820 hr x 10 min arm operation/hr /4----------------<u>34 hr per joint</u>
Total, approx. 85 hr per joint

Based on Martin Marietta experience, the remaining operating life of the joints before brush change and other maintenance is required should be more than 100 hours. Assuming the same level of use as in the past, the engineering test unit can be operated for five additional years before joint refurbishment.

4.2 SERVICER MECHANISM TESTS

A series of accuracy tests in the Manual-Direct (joint by joint) mode were run. The starting position was established by lining up vertically two sharp pencils, one attached to the end effector and the other to the stowage rack structure. The position of each arm joint was established by bringing the meters to zero position with the corresponding potentiometer on the control console (all indicator lights out). The arm was then moved in all axes, through large amplitudes at various rates and returned to the reference point within 1/16 in. or better. This test was repeated twice, and after that, a 10 lb weight was attached to the end effector. The same test was repeated twice and the accuracy was 1/8 in. or better.

The last accuracy test performed during this visit was in the Manual-Direct mode with a module attached to the end effector in a horizontal position for maximum wrist moment load. Accuracy was within 1/8 in. or better.

These tests produced the same results as previous tests made when the arm was initially assembled.

After completing these accuracy tests, a test of the internal noise generated by each joint, when operated one at a time at low and high speed was conducted. An extension bar was used between the test person's ear and various points of the gearbox housing. No periodic or random distinct noise was detected which normally would indicate broken teeth, defective bearings or excessive wear. The test results are tabulated below:

Joint	Speed		Remarks
	High	Low	
T	Very quiet	Very quiet	
U	Audible noise w/slight periodic hammering	Audible noise w/slight periodic hammering	"Normal" condition was same all the time
V	Very quiet	Very quiet	
W	Motor reducer noisy, uniform sound. Worm gear quiet.	Noisy reducer-uniform sound. Worm gear quiet.	"Normal" condition
Y	Uniform, slight noise	Quiet	"Normal" condition
Z	Uniform noise	Low level, uniform noise	"Normal"
End Effector jaw closing	Noisy, uniform sound, less than W	Noisy, uniform sound, less than W	"Normal"

The free play of the Z joint was measured using an indicator with a magnetic base. Total free play was 0.004 in. at 2.75 in. radius, with the brake applied. This value corresponds to 5 arc min. and is considered to be within the normal, original tolerance.

Axial free play of approximately 0.030 in. was found in the ball screw of the jaw mechanism at mid-stroke. It appears to be the normal, original free play. The jaws have some free play in the pins and wear/impact marks at the tips. Also the chrome finish of the guiding cone has rather unsightly nicks and dents shallow enough to not affect function.

The cable harness on the arm was visually inspected and no worn or broken insulation was found. Some of the cable ties have wear marks and need replacement. No binding points were detected in any position.

An interference condition exists between the lamp support plate and the arm tube between joints Y and Z in some extreme positions and was present all the time. Metal chips from the rubbing, contaminated the ball screw and other moving parts of the end effector. This problem can be corrected by bending the lamp support plate or by replacing it with one of different design.

The results of the inspection and of the tests were reviewed, as well as the ETU operations history. In conclusion, the mechanical arm was found to be in very good condition and no dismantling was necessary for further inspection.

Higher moment capability of the wrist joints will be required for demonstrating MMS type module exchange using an adapter tool. Replacement of the Globe motor of the W (wrist yaw) joint was investigated. The results of the analysis showed that replacement was not necessary. The W joint actually has a larger margin of safety than the Y joint (wrist pitch), considering the motor capability and the worst load, using either the side or base mount interface mechanisms. The analysis of the V joint (shoulder pitch) linear actuator has also shown an adequate margin of safety and that no replacement is necessary.

4.3 CONTROL SYSTEM TESTS

The arm was successfully used to exchange a module in radial position, with side interface mechanism, using the Manual-Direct mode.

Computer controlled operation (Supervisory mode) of the ETU arm was demonstrated. The accuracy was rather poor and the module needed hand pushing, or manual control corrections, close to the engagement position. The end effector and latching controls were manually actuated. A "drift" in the trajectory was, presumably, due to faulty A/D converters associated with the PDP 11/34 microcomputer. Other possible causes of the problem could be the fact that the computer and the control console have separate grounds and there is no slaving of the reference voltages between the A/D converters and the servo drive console. The arm trajectory in the Supervisory mode of control followed wavy, irregular paths in some places. Software modifications are needed for smoother operation.

The module attachment guides were held in position within the spacecraft or stowage rack mockups only by friction, in order to prevent mechanism damage. However, because of inaccurate positioning of the arm in the Supervisory mode of control, shifting of these module attachment guides occurred frequently and manual repositioning was necessary. In addition to improvements in the accuracy of the arm controls, a compliant, easy to reposition, hold-down system should be added to the attachment guides.

During one of the accuracy tests, while a module was attached to the end effector in a horizontal position, the W (wrist yaw) joint started to slip slowly down (approximately 1/8 in. per min, measured at the tip of the module). We could hear the gear reducer moving. The rate did not increase when applying extra load. When the test was repeated we could not reproduce the condition. It was assumed to be caused by a stray current reaching the W joint motor. Such stray currents were noticed on several other occasions, in different joints, producing erratic movement or "drift" of the arm while in the Manual-Direct mode of control. The cause of the problem is yet to be identified. It could be a failed component of the servo power amplifiers, or an effect due to interconnections between the servo system and the computer system.

Three of the six meters of the control console had internal friction that prevented free movement of the needle and they need repair or replacement.

The torque sensitivity constant (K_T) was measured for several of the DC torque motors. The results are summarized in the following table.

AXIS	Applied Torque (ft-lb)	Measured Current (Amps)	Gear Ratio	Measured K_T (ft-lb/Amp)	Vendor Specified K_T (ft-lb/Amp)
T	80	2.6	113	0.27	0.33
V	69	2.3	127	0.24	0.33
Y	53	2.2	110	0.22	0.31
Z	21	1.9	50	0.22	0.25

The above test was performed with a spring balance. It is estimated that the accuracy of test was $\pm$ 30%. All of the measured values are sufficient to produce smooth reliable operation of the system.

4.4 RECOMMENDATIONS

The changes recommended here are designed to eliminate the existing minor problems of the Engineering Test Unit and upgrade its control system by adding a Manual-Augmented control mode. Smoother operation in performing the present module exchange demonstrations will be achieved, as well as increased system flexibility in preparation for further demonstrations such as refueling or exchange of different types of modules, as outlined in Section 3.3.

4.4.1 Mechanical Arm

The following refurbishment is recommended for the mechanical arm:

1) Replacement of the lamp support plate. Modify design to eliminate interference;

2) Replacement of the worn cable ties. Check all cables for electrical continuity;

3) Paint touch-up where necessary.

4.4.2 Stowage Rack and Spacecraft Mock-Up

Following are the recommended changes in order to properly perform module exchange demonstrations using the side or the base interface mechanisms:

1) Design, fabricate, install and check-out a method for rigidizing the location of the module attachment guides in the stowage rack and in the spacecraft mockup. Two levels of rigidization are required: (1) soft, to be used for those tests where there is a potential for damage to the Engineering Test Unit and (2) firm, to be used for tests where the chance for damage is low. Limiting force devices, such as shear pins should be provided. Means should be provided to indicate when a module is not in the desired soft or firm rigidization location. Four sets of equipment are required for rigidization of the module attachment guides - two for modules in the spacecraft mockup and two for modules on the stowage rack mockup;

2) Repair or replacement of the optical targets on spacecraft mockup and stowage rack used in connection with the Manual-Direct and Manual-Augmented control modes;

3) Paint touch-up where necessary.

4.4.3 Controls Hardware

The recommended changes shown below are needed in order to eliminate a few problems found during the inspection and to add a Manual-Augmented mode of control:

1) Design, fabricate, assemble and checkout a simple control console for use in conjunction with the existing Engineering Test Unit. The console should accommodate: (1) the servicer control panel that is part of the servicer servo drive console, (2) a new television monitor, (3) the existing alpha-numeric display and keyboard (computer terminal) used with the MSFC PDP 11/34 computer and (4) a set of two hand controllers, each having three degrees of freedom, to be provided by MSFC. The hand controllers are used for the Manual-Augmented mode and need only generate on/off signals (switch closures). The control console, with the appropriate equipment installed should be compatible with the demonstration of the Supervisory, Manual-Augmented and Manual-Direct control modes;

2) Replacement of three failed meters on the control panel of the servicer servo drive console;

3) Checkout of the power supply unit and of the D/A and A/D converters to determine the cause of stray currents to the arm joints or "drift" and to correct the problem.

4.4.4 Controls Software

During the inspection of the Engineering Test Unit, the need for improved control software was identified, to provide smoother, more accurate operation and to add a Manual-Augmented mode of control. In order to upgrade the existing software, the following actions are recommended:

1) Establish the requirements for the upgraded control software. These requirements for the Supervisory mode should include the ability to go through a complete replacement of a "failed" module

with a "good" module and the storing of the "failed" module in the initial storage rack location of the "good" module. The operator should be able to initiate the exchange so that it proceeds from the beginning to the end without further operator actions or so that the computer waits at the end of each step for operator directions to continue. Provisions should be made for operation with the initial "failed" module location being in the: (1) radial, (2) near-axial, or (3) far-axial locations in the spacecraft mockup. Two stowage rack mockup module locations should be accommodated: (1) "good" module and (2) temporary stowage. The control software design should be such that it would be possible to easily change the stowage rack mockup module locations. The interface mechanism latch and the end effector attach operations should be controlled by the computer when in the Supervisory mode. A safety override control should be provided the operator that would inhibit the computer from opening the end effector jaws. The software data file should be large enough to include the data for demonstration of connection and disconnection of a refueling probe or electrical umbilical. The software should include equations and instructions for the Manual-Augmented mode as well as the Supervisory mode.

The software should be compatible with an existing MSFC PDP 11/34 computer and the electronic interfacing equipment;

2) Review the capabilities and operations of the existing MSFC electronic interface equipment (analog to digital and digital to analog converters). Include any necessary instructions for control of the electronic interfacing equipment in the software;

3) Modify the existing software to satisfy the requirements that are identified and check out the new program;

4) Prepare a Servicer Control Software User's Manual with sufficient information to permit MSFC personnel familiar with operation of the PDP 11/34 computer to use the new software. This user's manual should contain a full program listing with appropriate comments;

5) Check out the operation of the Engineering Test Unit using the new software. The goals of this checkout should be: (1) to demonstrate the capability to exchange mockup modules between the mockups of the spacecraft and the stowage rack, and (2) to demonstrate the ability to control the system in each of three ways--Supervisory with operator action between steps, Supervisory without operator action between steps, and Manual-Augmented with coordinated joint control. The Supervisory mode tests should include the demonstration of a complete replacement of a "failed" module with a "good" module and the storing of the "failed" module in the initial storage rack location of the "good" module.

4.5 OTHER GROUND SERVICER IMPROVEMENTS

Several improvements of the Engineering Test Unit that were considered after reviewing the results of the inspection have not been included in the recommendations of paragraph 4.4, because their contribution to the improvement of the ground demonstration program was less cost-effective or of a lower priority. However, some or all of these ideas for improvement could be incorporated in the servicer design in the future should the ground demonstration requirements change. Following is a list of these improvements:

1) Add automatic calibration and electronic circuit trimming to simplify arm operation;

2) Add automatic target recognition using TV scanning to improve electronically the arm accuracy;

3) Redesign the servo control boards as printed-circuits. Add input differential protection to the operational amplifiers and replace the relays with electronic switches;

4) Provide the capability for computer control of servo disable by axis. Also provide programmable dynamic current limit by axis. This would permit the servos to be mechanically backdriven;

5) Design, fabricate, deliver, and check out at MSFC two devices that will indicate, in digital form, the specific location in which a module is latched. These module location indicators shall be suitable for installation on the existing interface mechanism baseplates, one indicator to a baseplate.

5.0 FLIGHT DEMONSTRATION PLAN

The objective of this phase of the work activity was to identify and define the major elements of an on-orbit servicing demonstration in the orbiter cargo bay. The objective of the cargo-bay demonstration is to help convince satellite designers that on-orbit servicing in the form of remote module exchange can be done on orbit and that the major elements of the system can be designed, built, and operated. The cargo-bay demonstrations are considered to be a significant step on the path to obtaining user acceptance of on-orbit servicing.

The approach to this task was to review prior work on the subject to identify elements of the operational system, requirements, constraints, and alternative concepts. The desirable characteristics of a cargo-bay experiment were then identified and the rationale was stated. This was followed by an identification of candidate flight demonstration activities. These are discussed only in a general way as the specifics are expected to evolve as new spacecraft are designed and new functional equipment, such as for refueling, becomes available. Several arrangements of equipment in the orbiter cargo bay are then described, evaluated, and a recommended arrangement is selected. This is followed by a discussion of a free-flight demonstration and a summary of the flight demonstration plan. The elements that were considered in preparing the flight demonstration plan are shown in Table 5-1. All of the ground demonstration activities, such as exchange of an MMS module and refueling probe connection, were addressed. The orbiter cargo-bay size effects are discussed in Section 5.3 in terms of three alternative arrangements of the servicing equipment. Three alternatives for location of the control station--aft flight deck, spacelab module, or on ground--are also addressed in Section 5.3.

Table 5-1 Orbiter Cargo-Bay Demonstration Considerations

Activities to be demonstrated
Orbiter cargo bay size constraints
Orbiter impacts
Control station approach
Flight crew requirements

The source and eventual utilization of the cargo-bay demonstration hardware is a major concern. One approach is to upgrade the 1-g demonstration equipment, but this approach means that only the flight hardware will be available for procedure development and operator training. If a new set of hardware is to be built for the cargo-bay demonstration, then the question arises as to whether it should be designed to do the operational servicing activities as well. If these additional requirements are placed on the demonstration equipment then its costs will increase.

5.1 DESIRABLE CHARACTERISTICS

This section of the report reviews candidate scenarios for on-orbit servicing operations as an approach to establishing desirable characteristics for the fight demonstration plan activities. However, the specifics of the plan are not important in that different sets of specifics can satisfy the goal. What is important is that the plan leads to a commitment to perform a flight demonstration. The existence of a plan, combined with a committed funding stream, will help to convince the user community that on-orbit servicing in the form of module exchange can become a reality. The specifics of the flight demonstration plan can be defined based on where the support comes from and the interests of the supporting groups. Some candidate specifics are identified in the following so that the plan can be expressed in more detail.

5.1.1 Operational Scenarios

Three representative on-orbit servicing scenarios are postulated and examined to identify candidate characteristics of the flight demonstration plan. These scenarios are:

1) Low earth orbit (LEO) using a free flyer such as the Orbital Maneuvering Vehicle (OMV) operating from the orbiter or from the space station;

2) Low earth orbit in the orbiter cargo bay;

3) Geosynchronous earth orbit (GEO) using a carrier vehicle.

Use of an OMV in LEO for servicing is a good way to overcome the limited orbit transfer capability of the orbiter and to enhance the space station's capabilities. Use of a servicer mechanism in the orbiter cargo bay is a technique that is an alternative to using astronauts on extra-vehicular activity (EVA) for module exchange. It is a way of reducing EVA burdens (costs, safety considerations, limits to length of EVA, and lost time during preparations) on some missions. An illustration of how the Integrated Orbital Servicer System (IOSS) could be used to service a characteristic large observatory at the orbiter is shown in Figure 5.1-1. The IOSS final report of April 1978 addressed 12 different geosynchronous satellite servicing scenarios and concluded that the differences in costs were not a driver. The concept of using a chemically propelled carrier vehicle with a representative set of spare modules has the advantage of lower first costs while being above average in cost savings. The chemically propelled carrier vehicle could be the OMV. The OMV and servicer can be transported to GEO by the Centaur or Transfer Orbit Stage (TOS) on a one-way basis or by the Orbital Transfer Vehicle (OTV) from the space station on a round trip basis. The OMV and servicer would be separated from the transport vehicle once they reached GEO. The servicer and OMV would rendezvous and dock with the OTV after completing the servicing missions for return to the space station.

A representative flight profile for a LEO servicing mission using the OMV as the carrier vehicle is shown in Figure 5.1-2. The OMV, with the servicer equipment and replacement modules, is launched in the orbiter. At the appropriate time, the servicer and OMV are deployed from the orbiter and initiate a transfer trajectory to the failed satellite. Then the OMV would dock the servicer to the payload. The servicer would exchange modules or perform propellant resupply while under control from the ground via the Tracking and Data Relay Satellite System (TDRSS). After the servicer has completed servicing the payload, the OMV returns the servicer to the vicinity of the orbiter.

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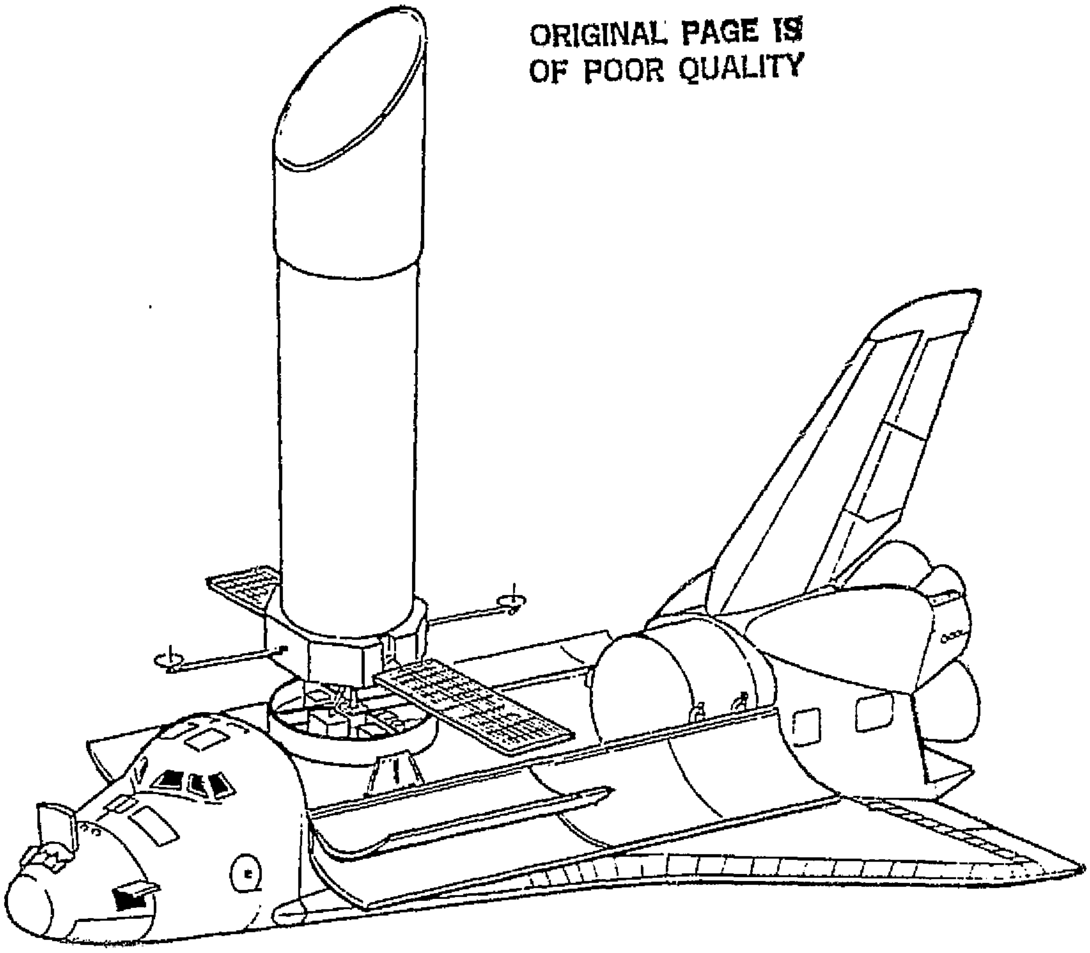

Figure 5.1-1 Servicing a Characteristic Large Observatory

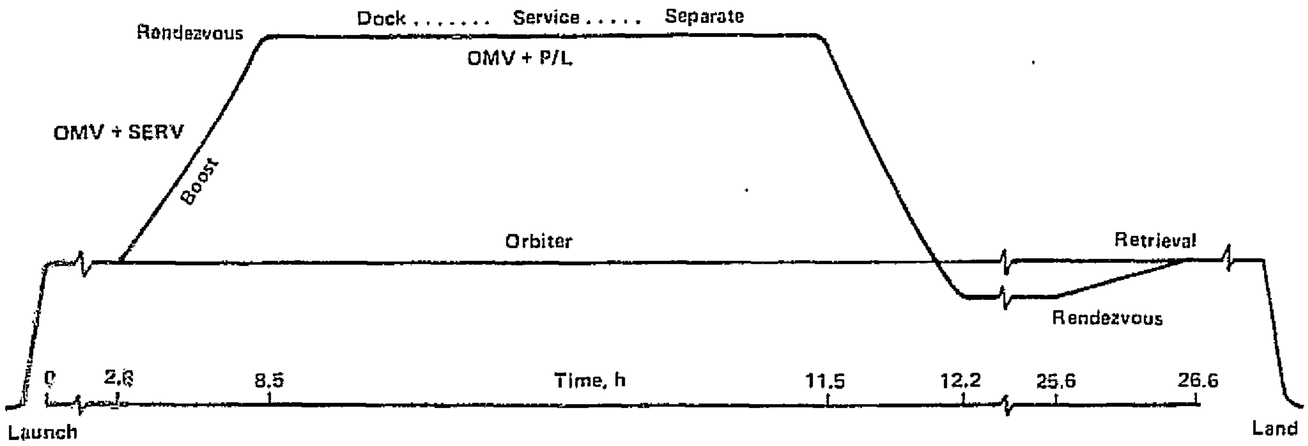

Figure 5.1-2 Servicing Profile

The OMV and servicer would then be retrieved by the Remote Manipulator System (RMS) and stowed in the orbiter cargo bay for return to earth. Alternatively, the servicer and failed modules could be retrieved from the OMV and returned to earth while the OMV remains on orbit for later missions. In addition to the one hour period required for docking, this mission assumes that it takes two hours to complete module exchange servicing for a total satellite orbit time of three hours. It is anticipated that -opellant resupply will take longer than module exchange when a fluid probe is used because of small lines and low flow rates. Multiple servicing missions could be designed so that the OMV and servicer would service more than one satellite before returning to the orbiter. The mission time required for an OMV servicing mission is highly dependent upon the time necessary to complete the satellite servicing and the number of phasing orbits required. For servicing times of two hours, mission times are less than 52 hours.

When the servicer is operated from the space station, the stowage rack will be loaded with the modules and refueling equipment specific to the servicing mission. The servicer is then attached to the OMV using the space station's Remote Manipulator Arm. The OMV transports the servicer to the spacecraft to be serviced in LEO. After completion of the servicing mission, the OMV will return the servicer to the space station.

A mission in LEO involving repair in the orbiter cargo bay involves a combination of the retrieval and delivery mission profiles shown in Figures 5.1-3 and -4, along with the servicing activity at the orbiter. The payload retrieval profile for the OMV involves rendezvous and docking with the payload. The profile of Figure 5.1-3 assumes that it takes one hour to dock with the payload. The time to complete the delivery profile of Figure 5.1-4 depends on several parameters including delivery altitude, time to orient and separate, and phasing orbit altitude. When returning from the delivery mission, the OMV can use extended phasing orbits or it can go into a long-term orbital storage mode. When the OMV has brought the failed satellite close to the orbiter the RMS can be used to retrieve the satellite from the OMV

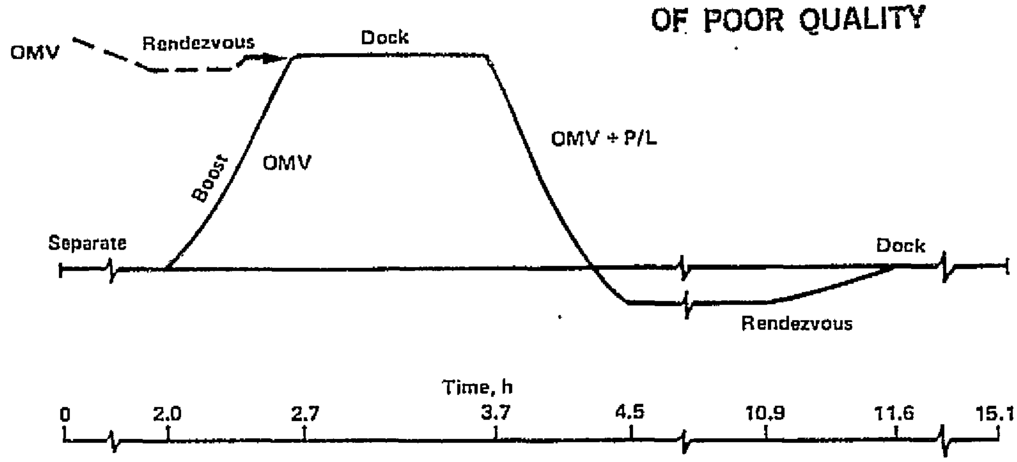

Figure 5.1-3 Retrieval Profile

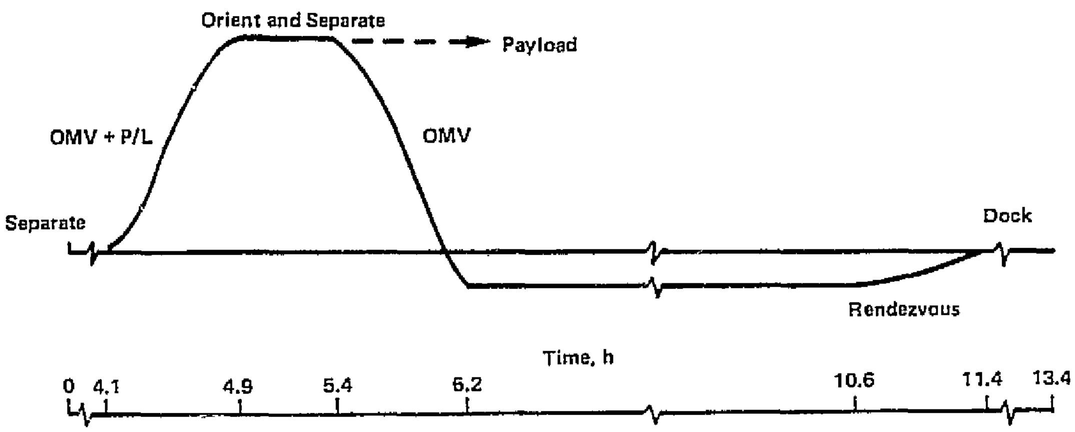

Figure 5.1-4 Delivery Profile

and position the satellite in the cargo bay for servicing. Servicing can be accomplished by astronauts on EVA, by the RMS, or by a servicing mechanism if the satellite can be repaired by module exchange. After repair and checkout, the satellite is positioned for pickup by the OMV, which then delivers the repaired satellite to the desired orbit.

Servicing in geosynchronous earth orbit is not likely to include EVA for some time and thus servicing is limited to remote activities, such as module exchange or propellant transfer. In this example case, the servicer mechanism and a stowage rack with a selected set of modules is mated to an OMV in LEO at the orbiter or at the space station. The OMV is then mated, for example, to an Orbit Transfer Vehicle for transfer to geosynchronous orbit. Upon reaching GEO, the OMV and servicer separate from the OTV. The OMV, under control directly from the ground, transfers to and services those vehicles that required near-term refueling or other servicing. The OMV then either goes into a long-term orbital storage mode until subsequent refueling or other servicing needs are identified or it rendezvous and docks with the OTV for return to the space station.

These three representative servicing scenarios are used to identify the general characteristics of the servicing missions, support systems that are required, and the areas that could be demonstrated in an orbiter cargo-bay servicing experiment.

5.1.2 Support Systems

The on-orbit servicer requires a number of support systems for it to be useful. There must be a way to get the servicer into space, to rendezvous and dock with the failed spacecraft, and to control the module exchange operations. Fortunately, the majority of the required functions are being developed as characteristics of the various parts of the Space Transportation System. Table 5.1-1 lists the functions that are required and indicates their applicability to the three operational scenarios.

The satellite deployment and retrieval function is a capability of the orbiter's Remote Manipulator System. This same capability is applicable to deployment/retrieval of the OMV, servicer, and both together. The OMV development has been started and the on-orbit servicing capability is considered as an add-on kit to the OMV. Thus, the OMV and on-orbit servicer should be compatible. The extended OMV mission scenarios include a long-term orbital storage (LTOS) mode that indicates a need to be able to mate the servicer and OMV on-orbit at the orbiter or at the space station. The servicer is not self-contained in that it needs certain functions from the carrier

Table 5.1-1 On-Orbit Servicing Support Systems

Function/Equipment	Operational Scenarios		
	LEO-OMV	LEO-Orbiter	GEO
Satellite Deployment (RMS)	X	X	X
Satellite Retrieval (RMS)	X	X	—
Orbital Maneuvering Vehicle	X	X	X
Mating/Demating of Servicer & OMV	X	—	—
Provision of Power, Attitude Control, and Thermal Control	OMV	Orb	OMV
Two-Way Communications Links to Ground	X	—	X
Servicer Control Station	Gnd	Orb	Gnd
Rendezvous Capability	OMV	OMV	OMV
Docking Capability	OMV	OMV	OMV
Orbit Transfer Stage	—	—	OTV

vehicle. These include: electrical power, attitude control, thermal control, video data compression when necessary, and two-way communication links. The downlink includes a small amount of data and a video signal that can have a lower than normal refresh rate. The up communications link only involves a few command signals. The carrier vehicles have or are planned to have these capabilities. The LEO mission using the OMV can communicate through the TDRSS to a ground control station or to the space station. The LEO-orbiter mission does not require an RF communications capability as the control station will likely be on the orbiter, and communications from GEO can be directly to the ground.

In addition to the three communications networks (Space Tracking and Data Network - STDN, Tracking and Data Relay Satellite System - TDRSS, and NASA Communications System - NASCOM), the various ground control centers (Mission Operations Control Center - MOCC and Project Operations Control Centers - POCC) can also be used. A servicing ground control station is visualized for the LEO-OMV and the GEO scenarios. This station could be part of the MOCC, a separate POCC or the POCC for the OMV could be used. The servicer control station for the LEO-orbiter scenario would likely be at the orbiter aft flight deck or it could be on the ground. In which case, the orbiter communications links and the TDRSS would be used.

A rendezvous and docking capability is being developed for the OMV and thus should be directly transferable to the OMV-servicer combination. The payload to OMV rigidization function can be used for the OMV to servicer docking, but is not required for the servicer to satellite docking. If the servicer-OMV combination is to be used at GEO then an acceptable orbit transfer stage must be developed or evolved from one of the existing upper stages.

The next step is to identify those support functions from Table 5.1-1 that should be considered as part of a servicer experiment in the orbiter cargo bay. A candidate list is given in Table 5.1-2. Most of the scenarios of Table 5.1-1 involve the satellite deployment and retrieval functions of the Remote Manipulator System and thus, they should be considered. Especially as the RMS equipment is planned to be available for use on every orbiter flight and on the space station. The Orbital Maneuvering Vehicle will have its own development schedule and need not be involved in the servicer cargo-bay tests. The mating/demating of the OMV and servicer at the orbiter is being recommended for a different orbital flight test because the functional equipment involved is different. The mating test involves functional OMV docking and docking rigidization equipment. The servicer tests do not require functional docking rigidization equipment. Power, attitude control, and thermal control should be provided by the orbiter. It is recommended that the servicer control station be on the ground to simplify the system and reduce costs. This approach clearly includes bit rate limits and communication system delays. The servicer control station would be part of the MOCC or a POCC and involve the two-way communications system. However, if desired, the servicer control station could be on the orbiter.

Table 5.1-2 On-Orbit Servicing Support Functions for Consideration in Cargo-Bay Experiment

Satellite Deployment/Retrieval by RMS
Provision of Power, Attitude Control, and Thermal Control
Servicer Control Station Location
Docking Capability

The rendezvous capability of the OMV will be checked as part of the OMV flight test program and need not be repeated as part of the servicer cargo-bay test. While the OMV docking capability will be checked as a part of the OMV flight test program the adequacy of rigidizing the connection between the satellite and servicer (servicer docking probe) should be checked as part of the servicer cargo-bay test. Test of the orbit transfer stage will be part of the OMV test program.

5.1.3 Servicer Verification Areas

Most functional aspects of the servicing operation are the same for all three scenarios. However, some are different. Candidate servicing verification areas are listed in Table 5.1-3. The table is divided to show which areas are affected by whether or not the OMV is the carrier vehicle. The areas to be investigated will have already been investigated in the 1-g servicing facility. However, the evaluations will have been performed in a 1-g environment under simulated space lighting. The 1-g environment necessitates the use of counterbalance systems. Also, lightweight module mockups will have been used to minimize tip loading of the servicer arm.

The on-orbit evaluations will permit the servicer performance to be verified in an actual space environment using full mass modules. Performance will be tested without counterbalance system effects and with actual space lighting. The absence of the gravity force could affect the end effector and interface mechanism capture volumes. The majority of the items in the top part of Table 5.1-3 can be readily incorporated into a servicer cargo-bay test.

Table 5.1-3 Servicer Verification Areas

Areas the Same for the Three Scenarios
Deployment of Servicer Docking Probe
Servicer Mechanism Performance
Interface Mechanism Performance
Connector Performance including Mate and Demate - Electrical, Waveguide, Thermal, and Fluids
Methods of Accommodating (compliance) Attach Errors
End Effector Capture
Interface Mechanism Capability for Capture, Latch, Unlatch and Release
System Force and Torque Levels
Repeatability Accuracy (Electro/Mechanical)
Spacecraft to Servicer Alignment
Spacecraft Module Removal and Replacement Trajectories
Control System Modes Validation
Man Machine Interaction
Lighting
Malfunction Mode/Backup System
Mission/Man/STS System Safety
Pre and Post Module Exchange Condition Analysis
Areas Different when OMV is Not Used as the Carrier Vehicle
Launch and Boost Support Structure
Deployment of Stowage Rack (applicable LEO-orbiter only)
Communications Links
Control Station Location
Supplementary Visual Aids (applicable to LEO-orbiter only)
Supplementary TV Cameras (applicable to LEO-orbiter only)
Direct Viewing (applicable to LEO-orbiter only)

The performance verification areas that are different when the OMV is used as the carrier vehicle are listed in the lower part of Table 5.1-3. The launch and boost phase support structure for the servicer and stowage rack will be subject to different loads depending on its configuration during these phases. The loading conditions will also be different when it is being transferred to GEO by an OTV. Deployment of the stowage rack is unique to the LEO-orbiter scenario. Verification of the ability of the RMS to be used for positioning the stowage rack at the desired location and rotating it 90 degrees can be included in the cargo-bay experiment. Docking of the spacecraft to the central docking mechanism of the servicer will be performed by the RMS in LEO-orbiter and by the OMV for the other two scenarios. Communications will be hard-wired for LEO-orbiter with the servicer operator located in the orbiter aft flight deck. For the other two scenarios, the communications will consist of the the satellite tracking net with a ground or space station based control station. Supplementary visual aids in the background will exist for the LEO-orbiter application. This is particularly true if cargo-bay cameras are used for added information. The benefits to be gained from direct viewing of the cargo-bay servicing operation for LEO is also a consideration item. For some locations in the cargo bay, the aft flight deck windows do provide a direct view.

Several of the areas in the lower part of Table 5.1-3 can be readily included in a servicer experiment in the orbiter cargo bay. These include: (1) deployment of the stowage rack, (2) communications links, (3) control station location, (4) supplementary visual aids, (5) supplementary TV cameras, and (6) direct viewing from the orbiter aft flight deck.

5.1.4 Programmatic Considerations

In addition to the support system considerations of Section 5.1.2, the servicer verification areas of Section 5.1.3, and the flight demonstration activities of Section 5.2, certain programmatic aspects also need to be considered. These are listed in Table 5.1-4. The first item - enhance user acceptance of on-orbit servicing - is a statement of the objective of the planned activity. If acceptance is to be enhanced, then the probability of failure must be reduced. A module exchange demonstration before the TV world that fails will not enhance user acceptance. The demonstration must be planned and accomplished successfully.

Table 5.1-4 Desirable Characteristics of Flight Demonstration Plan

Enhance user acceptance of on-orbit servicing
Incorporate representative servicing operational equipment
Include verification of procedures, analysis techniques, and 1-g simulations
Adaptability to changes in knowledge level
Compatible with OMV development schedule
Costs that are phased to user acceptability

The degree to which the experiment equipment represents the operational equipment must be addressed. One approach is to use the existing Engineering Test Unit, with some modifications, for the cargo-bay experiment equipment. While this approach may reduce costs, it has many drawbacks. The ETU equipment will be more than ten years old by the flight date, it was never designed for flight use, gravity is used to remove backlash in several drives, and the potentiometers and electronics are not flight qualified. It is expected that there will be some evolution in the design, especially the interface mechanism design, before operational use. Additionally, there will be a need for a training and procedures development unit that can be well satisfied by the existing Engineering Test Unit. The potential for expansion of the knowledge base, expression of new requirements by candidate users, and potential design changes due to the orbiter cargo-bay experiment results all argue that the experiment equipment should not become the first operational unit. There is also a possibility that the need for an operational unit might be delayed, in which case, it would be better to spend less money on the experiment unit. It is thus recommended that the plan include three sets of servicer equipment:

1) Existing Engineering Test Unit for 1-g demonstrations, procedures development and training;

2) Cargo-bay experiment unit designed to technology development mission requirements;

3) Operational unit(s) designed for free-flight test and use with the OMV.

An important part of the recommended plan is to work with candidate users to increase their awareness of the values of on-orbit servicing. It is expected that this information interchange will result in new ideas and new uses for the servicer. Thus, the development plan must be flexible enough in the early stages to be able to accommodate these new ideas. Similarly, the expenditure plan must be phased to the level of acceptance being obtained. It is expected that funding levels will follow growth in user acceptance and growth in user acceptance will follow funding levels.

5.2 CANDIDATE FLIGHT DEMONSTRATION ACTIVITIES

The objective of this discussion is to identify candidate activities for the cargo-bay flight demonstration and then to select a representative set for planning purposes. The interests of potential users of on-orbit servicing, new ideas, and the results of the 1-g demonstrations may modify the list, but that is not important as the cost of a specific activity is expected to be low. So any change in activities more than about 18 mos before flight should not seriously impact costs. It is important that the final selected set of activities be well checked out in the 1-g demonstrations. It is expected that the 1-g demonstration activity list will be longer than the list of those tested in the orbiter cargo-bay.

The list of candidate activities started with the 1-g candidate list and is given in Table 5.2-1. Descriptions of the activities and sketches of some equipment can be found in Section 3.3.2. The list of candidate activities has been separated into four groups, the first of which is module type. The battery module is used to represent a small heavy module. Batteries will need to be replaced because of their limited and somewhat unpredictable lifetimes.

Table 5.2-1 Candidate Flight Demonstration Activity Considerations

Module Type - Battery module - Multimission Modular Spacecraft type modules - Propellant tank module - Electrical connection interface unit - Propellant resupply module with interface unit - Access door - Electrical connector - Fluid in-line coupling - Wave guide connector - Thermal connector
Interface Mechanism Type - Lightweight side interface mechanism - Alternative interface mechanism concepts - Hinged access cover drive
Special Tools - MMS module servicing tool - Other interchangeable adapter tools - Refueling/resupply interface unit - Hose or cable management device - Propellant in-line coupling drive
Direction of Module Motion - Near axial - Far axial - Near radial - Compound motions

The Multimission Modular Spacecraft (MMS) module is representative of large modules with dual latches for securing the module to the satellite. In addition to MM spacecraft, this type of module is also being considered for use on the Advanced X-ray Astrophysics Facility (AXAF) and the Space Based Laser. As these modules are available commercially, it is expected that they may be used on additional satellites as well. The propellant tank module is included as an alternative to the propellant resupply module with probe. While both are designated as propellant resupply they can also be thought of as a fluid, gas or liquid resupply unit.

Two types of interface units are listed - electrical connection and refueling/resupply. They are similar and they both require connections (cables or hoses) back to the stowage rack that must be managed. Combinations of electrical connection and refueling/resupply have also been proposed. While small electrical connectors may be mated using a simple interface mechanism, large electrical connectors and the fluid disconnects will likely require a translation device to provide the high mating and demating forces required. Dust covers with their removal mechanisms may be required on both the spacecraft and servicer sides of the fluid disconnects and electrical connectors.

The access door is listed as a module type to show that access covers or doors can be treated as a module where the interface mechanism is a special configuration to properly secure the door. Four connector types are listed to indicate that they could be part of a module that is being exchanged.

The lightweight side interface mechanism is a redesigned version of the side interface mechanism that is used with the Engineering Test Unit at MSFC. As noted above, it is expected that new interface mechanism concepts will evolve as potential servicing users start to accept the concept of module exchange. The hinged access cover drive is another approach to using covers over modules to provide thermal protection. In this case, the cover is hinged to the satellite and latched down. The servicer end effector attaches to a fitting on the satellite near the door. The interface mechanism drive, or end effector power takeoff, is used to power a mechanism that frees the access cover latches and drives the cover to an open position. The end effector jaws are then opened and the servicer can be used to remove the uncovered module in the normal way. After the module has been replaced, the access door can be driven closed and latched by using the servicer end effector and interface mechanism drive.

The first special tool is an adaptation of the tool designed for use by the astronauts to replace MMS type modules. As power can be obtained from the servicer, the batteries on the astronaut tool are not required. Other specialized adapter tools may be developed for specialized tasks such as deploying an antenna. A hose or cable management device is required to demonstrate the use of the propellant resupply or electrical interface units. It is important that the hoses be kept out of the way of the servicer arm, or any modules, if the servicer is scheduled to do other things while the probes are attached to the satellite. The propellant in-line coupling drive is conceptualized as an attachment fitting on a propellant tank module and a set of linkages. The end effector attaches to the fitting, similar to the hinged access cover drive, and the interface mechanism drive (power takeoff) is then used, with proper linkages and mechanisms, to rotate a nut that connects or disconnects the propellant lines. A separate propellant in-line coupling drive could be used for each propellant line if desired to be sure of providing the proper torque level.

Four satellite module motion directions are listed to cover the full range of anticipated uses including compound motions such as were considered for early versions of the OMV.

The recommended activities for the first test flight are:

1) A Multimission Modular Spacecraft type module using a modified MMS module servicing tool, incorporating an electrical connector, and mounted so that the module moves axially;

2) Battery module on a lightweight side interface mechanism using an electrical connector and with a near-radial module motion direction;

3) Hinged access door mounted so that the servicer end effector is attached in a radial direction.

These three activities incorporate nine, or 41%, of the items from Table 5.2-1. If desired, a fourth activity involving the electrical connection interface unit with a cable management device and an alternative interface mechanism type could be added for the first flight or considered for the second flight.

The recommended activities for the second test flight are:

1) A multiple line propellant resupply module including a refueling interface unit and a hose and cable management device mounted in a far axial direction;

2) A propellant tank module on a lightweight side interface mechanism using a propellant in-line coupling drive and mounted in a near radial direction;

3) An access door treated as a module on a lightweight side interface mechanism and mounted in the near axial position.

The second flight has been dedicated to propellant transfer so that all the safety considerations relative to the handling of hydrazine can be addressed on one flight. If desired the multiple line propellant resupply module can include gas resupply and electrical connections. It would also be possible to retest any anomalies that occur on the first flight.

It is suggested that the hardware for the refueling demonstrations be obtained, if possible, from an ongoing Johnson Space Center refueling demonstration flight program. The JSC equipment may be designed for astronaut use, but it could be reconfigured for use as part of the servicer orbiter cargo-bay demonstration.

The candidates remaining on the Table 5.2-1 list, but not assigned to a flight, could be considered for either flight if the concepts are supported by potential users. These include:

1) Wave guide connector;

2) Thermal connector;

3) Other interchangable adapter tools;

4) Compound motions.

It is recommended that the servicer be exercised in all three control modes. If each activity were conducted in each of the three modes, and each took 45 min, then the total experiment time would be seven hours per flight. This seems acceptable unless the setup and stow for reentry activities become too long. In that case, the supervisory mode should be used for all three activities and the two manual modes used for one activity each.

5.3 CANDIDATE CARGO-BAY ARRANGEMENTS

The experiment equipment to be arranged in the orbiter cargo-bay consists of a spare module stowage rack, the servicer mechanism, docking system, servicer electronics, spacecraft mockup and any support equipment required. The servicer electronics equipment should be packaged quite small and can be ignored at this level of discussion. Connections to the servicer will be data, commands, video, electrical power, ground connections, and some separately wired emergency control functions. The data and command functions can be digitized and put on data buses. Thus, only a few connections between the experiment equipment and the orbiter are required. The small number of connections allows this function to be ignored in these early arrangement considerations.

In the sketches that follow, the experiment equipment has been located near the aft end of the orbiter cargo-bay. This location was selected to have a large field of view from the orbiter aft flight deck windows and to avoid RMS arm reach problems. In most cases, it might be better to locate the equipment where module motions could be more easily seen from the orbiter aft flight deck windows. Other considerations include requirements for other equipment and experiments on a specific flight, center of gravity control for launch and landing, field of view of the orbiter cargo-bay cameras, and location of the keel and trunnion fittings, especially the active keel fittings. These considerations can be worked into the arrangement when specific flight opportunities are identified.

The three arrangements discussed are:

1) Fixed tandem arrangement;

2) Use of RMS for docking;

3) Use of an orbital flight test pallet.

5.3.1 Fixed Tandem Arrangement

The fixed tandem arrangement of the experiment equipment is shown in Figure 5.3-1. This arrangement was selected for its simplicity with the stowage rack and spacecraft mockup mounted rigidly to each other in the proper orientation for module exchange. All module flips are done outside the spacecraft/stowage rack envelope to avoid potential interferences. All three module removal directions are easily accommodated as are the range of activities selected for flights one and two. Should the arm fail and be locked in a position that prohibits closing the cargo-bay doors then three options are available: (1) use of pyrotechnics to separate the interfering parts of the arm, (2) use of the RMS to fold the arm (and module) to an acceptable position, or (3) use of EVA to fold the arm (and module) to an acceptable position. These options are applicable for all three experiment equipment arrangements. The stowage rack and spacecraft mockups can be made strong enough so that any landing loads can be handled and any loose parts can be contained.

Figure 5.3-1

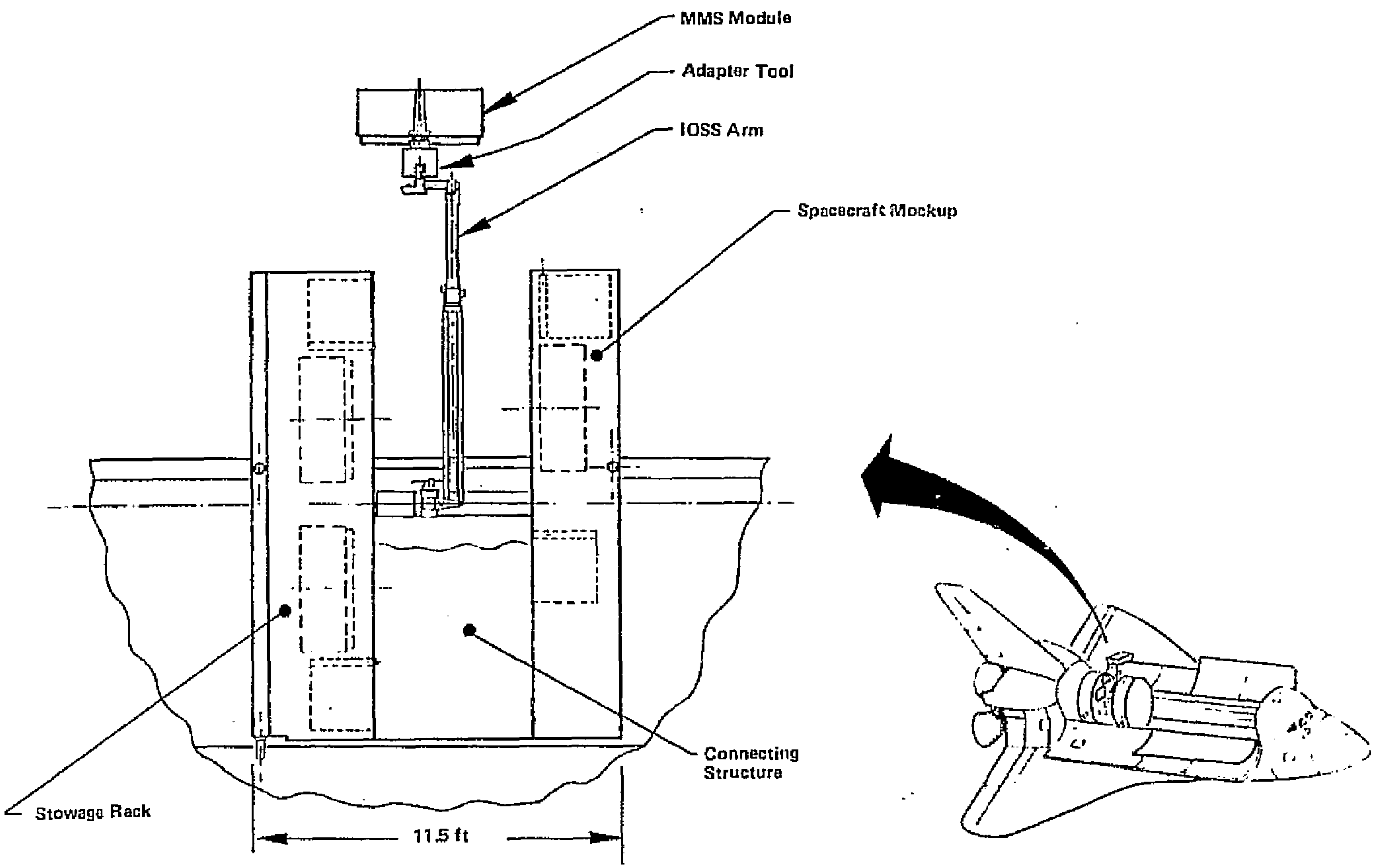

Figure 5.3-1 Fixed Tandem Arrangement of Experiment Equipment

The effects of docking misalignments are not included in this arrangement as the mockups are pre-aligned before launch. The servicer arm cannot be exercised over its full range of travel and the servicer docking probe stowage and deployment system cannot be evaluated. Direct viewing of module exchange from the aft flight deck is not possible. However, the cameras on the RMS could be used to supplement the IOSS and cargo-bay cameras. The amount of cargo-bay space devoted to the experiment could be reduced slightly by deleting the lower part of the spacecraft mockup.

5.3.2 Use of RMS for Docking

This approach is different from the other two candidates in that one arrangement is used for launch and reentry and a different arrangement is used for the module exchange demonstrations. The launch and reentry configuration is similar to that of Figure 5.3-1 except that the two mockups are not rigidly attached to each other except through the orbiter's structure. The module exchange configuration is shown in Figure 5.3-2.

Two approaches to handling the servicer docking probe were investigated. The fixed, or non-stowable, configuration is shown in Figure 5.3-2. The non-stowable docking probe of the servicer protrudes inside an opening in the back of the spacecraft mockup to reduce the launch length of the experiment equipment. When the spacecraft mockup is deployed by the RMS, it is turned so that the servicer docking probe can match with the grappler on the front side of the spacecraft. The second approach is to incorporate a foldable docking probe into the IOSS design as shown in Figure 5.3-3. This approach reduces the IOSS length by about two feet when it is stowed in the orbiter cargo bay. The intent is to reduce launch costs that are based on length in the cargo bay.

The RMS and MMS flight support system (FSS) cradle A prime are used to change from one configuration to the other. The stowage rack and servicer are attached to the FSS for launch with the stowage rack centerline parallel to the orbiter centerline. The spacecraft mockup

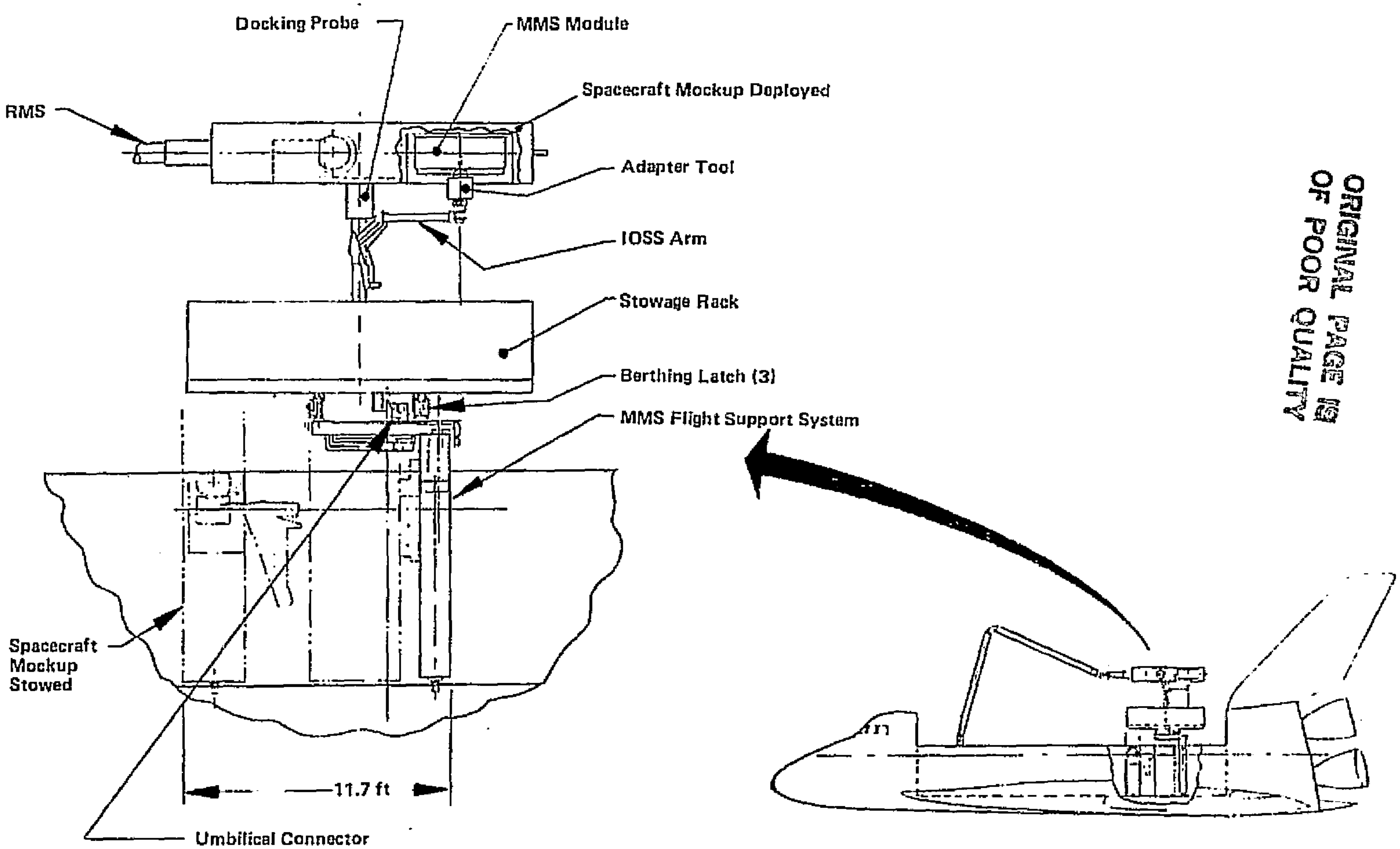

Figure 5.3-2 Use of RMS for Docking Arrangement

Figure 5.3-2

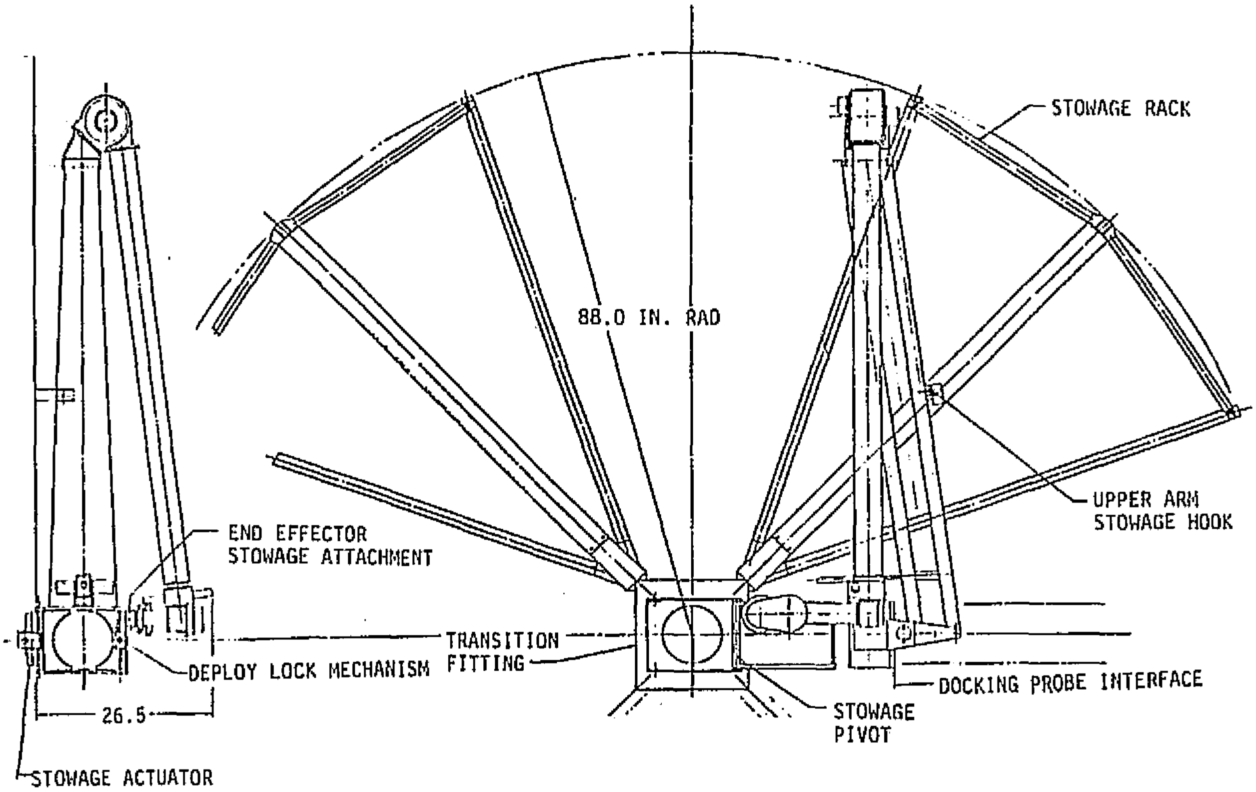

Figure 5.3-3 IOSS with Foldable Docking Probe

is launched with its centerline parallel to the orbiter centerline. The spacecraft is mounted in a set of deployable trunnions and uses a keel fitting. The RMS is used to lift the spacecraft mockup up and out of the trunnion and keel fitting. The RMS holds the spacecraft to one side while the FSS is used to rotate the stowage rack and servicer to the position shown (heavy lines) on Figure 5.3-2. The RMS is then used to dock the spacecraft mockup to the servicer. The RMS arm is released and the servicer docking system is used to rigidize the alignment between the stowage rack and the spacecraft.

Modules can be moved axially or radially and flipped outside the spacecraft and stowage rack envelope. The range of activities selected for flights one and two can be readily accommodated. In addition to the range of emergency separation techniques outlined in Section 5.3.1, the FSS can also be used for emergency separation. Reentry and landing loads can be accommodated when the spacecraft is secured by the trunnions and keel fitting and the stowage rack and servicer arm are secured by the MMS flight support system.

The effects of docking misalignments are included explicitly in this arrangement. The servicer arm can be exercised over its full range of travel. The servicer docking probe stowage and deployment mechanism should be included in this arrangement and properly exercised before, and after, the spacecraft is docked to the servicer. As the servicer docking probe stowage and deployment system folds the docking probe against the front of the stowage rack, its use removes the need to notch the spacecraft mockup for docking probe clearance. Use of the stowage and deployment system results in a more realistic representation.

Direct viewing of module exchange from the orbiter aft flight deck windows is possible for all practical FSS locations. After the RMS is detached from the spacecraft, the FSS rotational capability can turn the stowage rack and spacecraft mockups for even better direct viewing of the module exchange process from the orbiter aft flight deck windows. Again, the RMS arm cameras, elbow and wrist, could be used to supplement the IOSS and cargo-bay cameras.

Use of the FSS and the RMS complicates the experiment, but their use also significantly increases the number of investigation areas that can be verified.

5.3.3 Use of An Orbital Flight Test Pallet

There are two significant considerations in this arrangement-use of a short pallet to mount the experiment equipment, and use of the Spacelab itself for the control station. The major advantage is the ability to package things and check interconnections out well before the equipment is assembled into the orbiter. A disadvantage might be the availability of Spacelab equipment. One arrangement of servicing equipment on an orbiter flight test (OFT) pallet is shown in Figure 5.3-4. The equipment was arranged with the docking probe in a transverse direction to improve the direct viewing of the module exchange operations and it incidentally completed the range of possible cardinal directions. If the docking probe had been arranged parallel to the orbiter's centerline, then the arrangement would have been similar to that in Figure 5.3-1.

The arrangement shown in Figure 5.3-4 is somewhat crowded because of the space taken up by the pallet and because of the need to stay within the nominal 15 ft clearance diameter of the orbiter cargo-bay. The spacecraft mockup has one corner clipped so that one of the 24 in. modules can be removed in the near-radial direction. However, the full complement of modules and module motion directions can be accommodated. The advantages and disadvantages of the spacelab integrated arrangement are similar to those of the fixed tandem arrangement except that direct viewing from the Spacelab module might be easier and there may be more room available in the spacelab module for the servicer control station. The advantages of this arrangement are:

1) All four module motion directions are accommodated;

2) The range of activities selected for flights one and two can be accommodated;

3) There are acceptable methods for overcoming a frozen arm/module location that inhibits closing the cargo-bay door;

4) Direct viewing of module exchange can be incorporated into the experiment.

Disadvantages of the arrangement are:

1) Need for more care in structural design to contain "loose" parts during a maximum load landing;

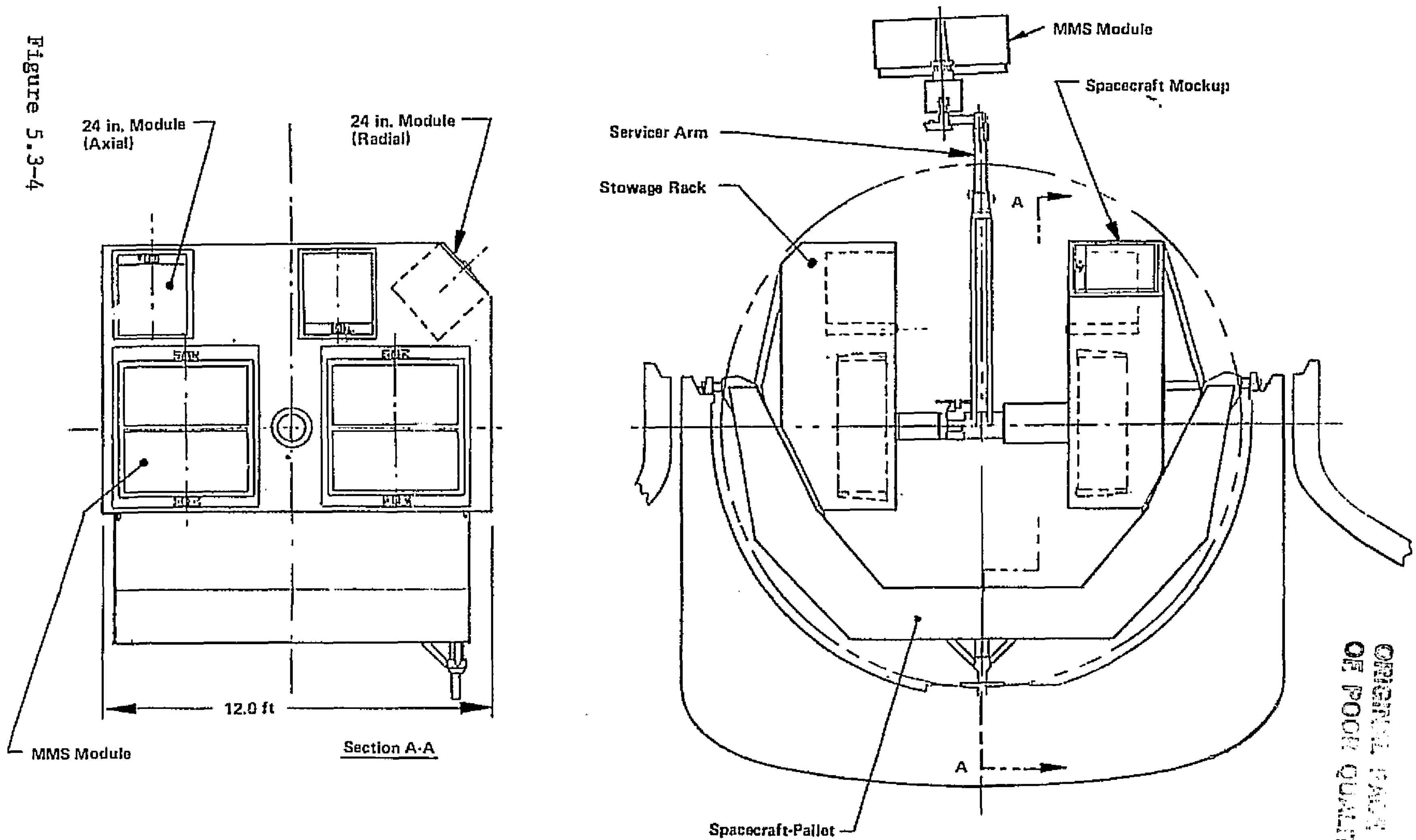

Figure 5.3-4 Experiment Equipment Arranged on OFT Pallet

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2) Effects of docking misalignments are not included;

3) The servicer cannot be exercised over its full range of travel;

4) The servicer mechanism stowage and deployment system cannot be evaluated;

5) The room available for module stowage is very limited.

Incorporation of the servicer control station into the Spacelab module appears to have some advantages in that Spacelab is designed to accept a variety of experiments and to provide the necessary support services, such as electrical power, communications, and data storage. This arrangement of the servicer control station can be used in connection with any of the three candidate arrangements.

5.3.4 Recommended Arrangement

The objective of this section is to present the rationale leading to a recommendation for a selection of one of the three candidate arrangements of servicer experimental equipment in the orbiter cargo bay. All these arrangements can satisfy many of the basic requirements. These requirements are:

1) Module motion direction (Table 5.2-1) accommodation;

2) Flight one and two activities (Section 5.2) accommodation;

3) Ability to handle a stuck arm.

Table 5.3-1 lists the requirements that are satisfied by only some of the arrangements. The arrangement involving use of the RMS for docking satisfies all of the requirements while the other two arrangements are deficient in at least four areas each. In particular, they are deficient in two major areas: (1) inclusion of docking misalignment effects, and (2) ability to evaluate the servicer mechanism and docking probe stowage and deployment system. Even though the RMS docking

Table 5.3-1 Alternative Arrangement Evaluation

Item	Arrangement: Tandem	Arrangement: Docking	Arrangement: OFT Pallet
1. Direct viewing of module exchange	No	Yes	Yes
2. Containment of loose parts during hard landing	Yes	Yes	No
3. Inclusion of docking misalignment effects	No	Yes	No
4. Ability to fully exercise servicer mechanism	No	Yes	No
5. Ability to evaluate servicer docking probe stowage and deployment system	No	Yes	No
6. Adequate room for modules and other experiment equipment	Yes	Yes	No

arrangement will be a more expensive experiment, it is recommended that the RMS docking arrangement (Figure 5.3-2) be used because of its ability to involve the docking misalignment effects and to demonstrate stowage and deployment of the servicer mechanism. The RMS docking arrangement also provides for good direct viewing of module exchange, containment of loose parts during a hard landing, an ability to fully exercise the servicer mechanism, and provides adequate room for modules and other experiment equipment.

This recommendation leaves a number of options open that are better decided as more information becomes available. These include:

1) Specific experiment activities;

2) Sequence of on-orbit activities;

3) Backup modes;

4) Safety considerations.

None of these options are expected to seriously impact the cost estimates. However, it is recommended that the servicer control station be located on the ground because it should result in a lower overall cost. By the time of the proposed experiment, all of the communications links and the POCC/MOCC protocols should be well developed. Thus, these functions can be easily included in the experiments along with the bit rate limits and time delays expected for an operational system.

5.4 CARGO BAY DEMONSTRATIONS

The objective of the cargo-bay demonstrations of the orbital servicer system is to help gain easier acceptance of module exchange as a viable technique for spacecraft maintenance. The recommended cargo-bay demonstrations have been separated into two flights - one involving module exchange and access door operation, and the other involving refueling operations. This discussion combines the analyses and conclusions of Sections 5.1 - 5.3 into a consistent plan.

In addition to design and development of the experiment equipment, the major precursor activities are accomplishment of the ground demonstration plan and the JSC refueling demonstration program. It is recommended that the JSC cargo-bay experiment equipment be obtained and reworked for the second servicer cargo-bay demonstration. The recommended characteristics of the servicer cargo-bay demonstration were developed in Sections 5.1, 5.2, and 5.3. They are summarized in Table 5.4-1. More detailed specifics of servicer performance areas are provided in Table 5.1-3.

5.4.1 Flight Plans

Example flight plans for the two cargo-bay demonstration flights are discussed to provide a better understanding of the equipment and flight time required. Should the specific activities and equipments of Section 5.2 be changed at a later date, then the flight plans will also change. The flight plan starts out after the flight equipment,

Table 5.4-1 Servicer Cargo-Bay Demonstration Characteristics

- Satellite mockup unstow and stow by RMS
- Supply of power, attitude control, and thermal control by orbiter
- Two-way communications links to ground through orbiter and TDRSS
- Servicer control station at OMV ground control station
- Docking rigidization by servicer docking probe
- Deployment of servicer docking probe
- Module exchange demonstration
- Refueling demonstration
- Servicing equipment performance demonstration
- Control modes evaluation
- Man-machine interactions evaluations
- Compliance with orbiter system safety requirements
- Deployment of stowage rack in orbiter by MMS flight support system
- Use of representative servicing operational equipment
- Operator training
- Servicer docking probe normal to orbiter wing plane

communications connections and the control station have been operated together on the ground. This test could use land lines instead of the radio links planned for flight. The servicer, stowage rack, spacecraft mockup, and servicer electronics are launched in the orbiter in the configuration shown dashed in Figure 5.3-2. It is suggested that the servicing demonstration be scheduled for later in the orbiter flight plan so that the spacecraft to be deployed will be out of the cargo bay and direct viewing of the module exchanges from the aft flight deck will be better. Before the experiment equipment is deployed, the ground control station should be activated and continuity of communication links should be verified.

A sequence of activities for the first servicer cargo-bay demonstration is given in Table 5.4-2. Before the sequence is started, two-way communications between the servicer control center and the orbiter must be established. The sequence starts with activation of the orbiter RMS, goes through three forms of module exchange in each of the three control modes and ends with the RMS and all demonstration equipment being secured. The three specific module exchange activities are:

1) A Multimission Modular Spacecraft type module using a modified MMS module servicing tool, incorporating an electrical connector, and mounted in the spacecraft so that the module moves axially;

Table 5.4-2 First Servicer Cargo-Bay Demonstration Sequence

Step No.	Activity	Step Time (min)	Cum Time (min)
1.	Activate and check out RMS	30	30
2.	Remove spacecraft from cargo-bay	20	50
3.	Hold spacecraft clear of servicer	5	55
4.	Activate and check out FSS	10	65
5.	Operate FSS to bring stowage rack to vertical position	15	80
6.	Rotate FSS for best direct viewing from aft flight deck	10	90
7.	Establish that ground control station can transmit to and receive from servicer	20	110
8.	Unlatch and unfold servicer docking probe	10	120
9.	Unstow servicer	10	130
10.	Exercise servicer mechanism in manual control and return to rest position	30	160
11.	Dock spacecraft mockup to servicer using RMS	20	180
12.	Release RMS and position for best use of its cameras and clear of direct view from aft flight deck	10	190
13.	Rigidize servicer to spacecraft docking attachment	5	195
14.	Put servicer in Supervisory control mode	5	200
15.	Exchange battery module	40	240
16.	Exchange MMS type module	70	310
17.	Open and close access door	30	340
18.	Put servicer into Manual-Direct control mode	5	345
19.	Exchange battery module	90	435
20.	Exchange MMS type module	150	585
21.	Open and close access door	70	655
22.	Put servicer into Manual-Augmented control mode	5	660
23.	Exchange battery module	60	720
24.	Exchange MMS type module	110	830
25.	Open and close access door	50	880
26.	Put servicer in Manual-Direct control mode	5	885
27.	Soften servicer to spacecraft docking attachment	5	890
28.	Attach RMS to spacecraft	15	905
29.	Remove spacecraft from servicer and move clear of servicer	10	915
30.	Stow servicer	10	925
31.	Fold and latch docking probe	10	935
32.	Turn servicer off	5	940
33.	Rotate FSS to position for stow initiation	10	950
34.	Operate FSS to put stowage rack into reentry position	15	965
35.	Secure FSS	10	975
36.	Operate RMS to stow spacecraft for reentry and release RMS	25	1000
37.	Secure RMS	10	1010

2) Battery module on a lightweight side interface mechanism using an electrical connector and with a near-radial module motion direction;

3) Hinged access door mounted so that the servicer end effector is attached in a radial direction.

The first series of activities in Table 5.4-2 involves repositioning the experimental equipment to the selected arrangement for the module exchange activities. The RMS and FSS are used for this activity. While only indicated once (as Step 6), it is intended that the FSS rotational capability be used to keep the spacecraft and stowage rack positioned for best viewing of each step in the module exchange sequences. Communication between the ground control station and the servicer is established next. This is followed by unstowing the servicer mechanism and docking probe and then exercising the servicer to verify that it is ready for the demonstration.

The handoff of the spacecraft mockup between the RMS and the servicer docking probe must be done carefully so that nothing is damaged. This operation takes advantage of the fact that the RMS end effector and the servicer docking probe each have two operating modes - capture and rigidize. The handoff will go smoothly as long as the two mechanisms are not in the rigidize mode at the same time.

The Table 5.4-2 sequence involves all three control modes. These are:

1) Supervisory control mode where a microprocessor commands the servicer mechanism to go through a preprogrammed set of motions with the operator only being involved to start the sequence and if there is a problem;

2) Manual-Direct control mode where the operator controls the servicer one joint at a time following a written sequence;

3) Manual-Augmented mode where the operator uses hand controller motions coordinated with a TV picture of the scene to move the modules. The hand-controller signals are converted to mechanism joint angle rate commands by a microprocessor.

After the demonstrations have been completed (Step 26), the experiment is stowed for reentry and landing using the self-stow feature of the servicer, the RMS and the FSS. The servicer control station can be shut down at this time. The 37 steps of the table are estimated to take almost 17 hours, which is composed of:

1)	Setup		200 min
2)	Supervisory control mode		140 min
3)	Manual-Direct control mode		315 min
4)	Manual-Augmented control mode		230 min
5)	Stow and secure		125 min
		Total	1010 min

This total time could be separated into a number of phases with no phase longer than the 200 min to set things up. Conducting the demonstration in phases will add more total time (1 hr per phase) to secure and setup between phases. If the two manual control modes are not exercised, then the demonstration would take just under eight hours. The order of doing the specific module exchanges and the control modes used can be switched around to suit other experiment or operational constraints. Of the 17 hrs of experiment time, the ground station crew must be involved for the total time. The orbiter crew will also need to be involved for most of the demonstration to assure that good photographic and video data are obtained.

A sequence of activities for the second servicer cargo-bay demonstration, after establishing communications between the servicer control station and the orbiter, is given in Table 5.4-3. The first thirteen steps involve setting the experiment equipment up and checking it out and are the same as for the first flight. The specific demonstration activities are:

1) A multiple line propellant resupply module with a refueling interface unit and a hose management device mounted in a far axial direction;

Table 5.4-3 Second Servicer Cargo-Bay Demonstration Sequence

Step No.	Activity	Step Time (min)	Cum Time (min)
1-13.	As per Steps 1-13 of Table 5.4-2	195	195
14.	Put servicer in Supervisory control mode	5	200
15.	Connect refueling interface unit	20	220
16.	Initiate propellant transfer	10	230
17.	Exchange propellant tank module	50	280
18.	Remove and temporarily stow access door	15	295
19.	Put servicer into Manual-Direct mode	5	300
20.	Exchange propellant tank module	100	400
21.	Pick-up access door from temporary stow location and reinstall	35	435
22.	Put servicer into Manual-Augmented mode	5	440
23.	Exchange propellant tank module	75	515
24.	Remove and replace access door	25	540
25.	Put servicer in Supervisory control mode	5	545
26.	Stop propellant transfer	10	555
27.	Vent and secure propellant supply lines	20	575
28.	Disconnect refueling interface unit and move to stow location	20	595
29.	Put servicer into Manual-Direct control mode	5	600
30-40.	As per Steps 27-37 of Table 5.4-2	125	725

2) A propellant tank module on a lightweight side interface mechanism using a propellant in-line coupling drive and mounted in a near radial direction;

3) An access door treated as a module on a lightweight side interface mechanism and mounted in the near axial position.

As for the first flight, the FSS rotational capability can be used to position the stowage rack and spacecraft for best viewing from the aft flight deck windows. The anticipated long propellant transfer time (several hours) suggested that propellant transfer only be demonstrated once and that refueling interface unit connection only be demonstrated using the Supervisory control mode. The propellant tank module exchange and the access door repositioning are demonstrated with all three control modes while propellant is being transferred. At the conclusion of the steps in Table 5.4-3, the servicer control station could be shut down.

The 40 steps of Table 5.4-3 were estimated to take just over 12 hrs, which is composed of:

1) Setup		200 min
2) Supervisory control mode		150 min
3) Manual-direct control mode		145 min
4) Manual-augmented control mode		100 min
5) Stow and secure		130 min
	Total	725 min

This total time can be separated into a number of phases as for the first flight with a one hour penalty added for each phase. The propellant transfer activity has an elapsed time of almost seven hours in Table 5.4-3, but it could be shortened to 1.5 hrs plus the propellant transfer time. All other phases can be kept under 200 min. Both the ground and orbiter crews will be involved for the whole demonstration period. In addition to photographic and video data coverage, the orbiter crew will need to monitor for propellant spills.

5.4.2 Equipment Required

Certain of the equipment required for the servicer cargo-bay demonstration is auxiliary equipment available for use on the orbiter as part of the Space Transportation System. This equipment is listed in Table 5.4-4. Its provision, control, and use should present no difficulties except for the communications links. Control of the servicer is nearly continuous and involves use of the TDRSS. The cargo-bay demonstrations should not be scheduled until all three TDRSS satellites are in orbit and operating. If all three TDRSS satellites are not available then the time spans of Section 5.4-1 will have to be lengthened.

Table 5.4-4 Orbiter Related Equipment

- Remote Manipulator System
- Attitude control, electrical power, thermal control
- MMS flight support system
- Orbiter communications equipment
- Cargo-bay cameras
- RMS cameras

The equipment specific to the cargo-bay demonstration is listed in Table 5.4-5. With a few exceptions, this equipment will have to be procured specifically for the demonstrations. The exceptions include the servicer 1-g trainer that is assumed to be available from the servicer ground demonstration program and the propellant resupply equipment that is assumed to come partially from the JSC orbital refueling program. Two sets of replacement modules and doors are required (Section 5.4.1) - one set for each flight. The propellant resupply equipment is only required for the second flight.

Table 5.4-5 Cargo-Bay Demonstration Equipment

- Integrated Orbital Servicing System
- Servicer control station in OMV ground control station
- Replacement modules and doors
- Propellant resupply equipment
- Servicer to orbiter interface equipment
- Servicer ground checkout equipment
- Servicer 1-g trainer
- Spare module stowage rack
- Stowage rack to orbiter interface equipment
- Stowage rack ground checkout equipment
- Spacecraft mockup
- Spacecraft to orbiter interface equipment
- Spacecraft ground checkout equipment

The three major pieces of equipment are the Integrated Orbital Servicing System (IOSS), the spare module stowage rack, and the spacecraft mockup. The latter two pieces of equipment must be reworked between flights to accommodate the propellant resupply demonstration. Each of these major pieces of equipment requires interface equipment with the orbiter and ground checkout equipment. It is recommended that the IOSS be built to experiment equipment standards to reduce costs and to allow for design differences in the operational units. The spacecraft mockup will require data transfer to and from the orbiter. This could be provided by RF links, by a direct cable connection to the orbiter, or by a connector that is mated after the spacecraft is docked to the servicer. Data transfer between the stowage rack and the orbiter can be by direct cable connection.

If the OMV ground control station (GCS) is not far enough along in its development process for use at the desired time, then the control station equipment could be installed at the MOCC or at a convenient location that has been used for other orbiter flight test or operations activities. While Table 5.4-5 shows the servicer control station on the ground, it may be advisable to have it on the orbiter aft flight deck.

5.4.3 Schedule

The servicer cargo-bay demonstration schedule was developed from an OMV development schedule provided by MSFC. The key points from the OMV schedule were an OMV authority to proceed (ATP) for Phases C and D on Jan 1, 1986 and the end of supporting development of a servicer kit in July of 1988. These dates are shown on Figure 5.4-1 along with other key OMV dates. The July 1988 date was selected as the beginning of Phase B for the free-flight verification or operational servicer development. This approach integrated well with the use of representative time spans for the various demonstrations and verification activities. It was decided that the results of the servicer cargo-bay demonstration would be most useful if the cargo-bay demonstration period started at the same time as the operational servicer development in July 1988. The two cargo-bay flight dates then became September 1988 and June 1989. As is shown in Section 5.5.3, the cargo-bay flights are completed well before the operational servicer preliminary design review (PDR) so that the cargo-bay demonstration results can be factored into the operational servicer design and development.

The schedule for the demonstration servicer was separated from the schedule for the spacecraft mockup and the stowage rack because these latter two equipments must be reworked between the first and second flights to integrate the refueling demonstration equipment while the servicer need only have its software changed and be checked out before the second flight. Phase B, for both the servicer and the mockups, are

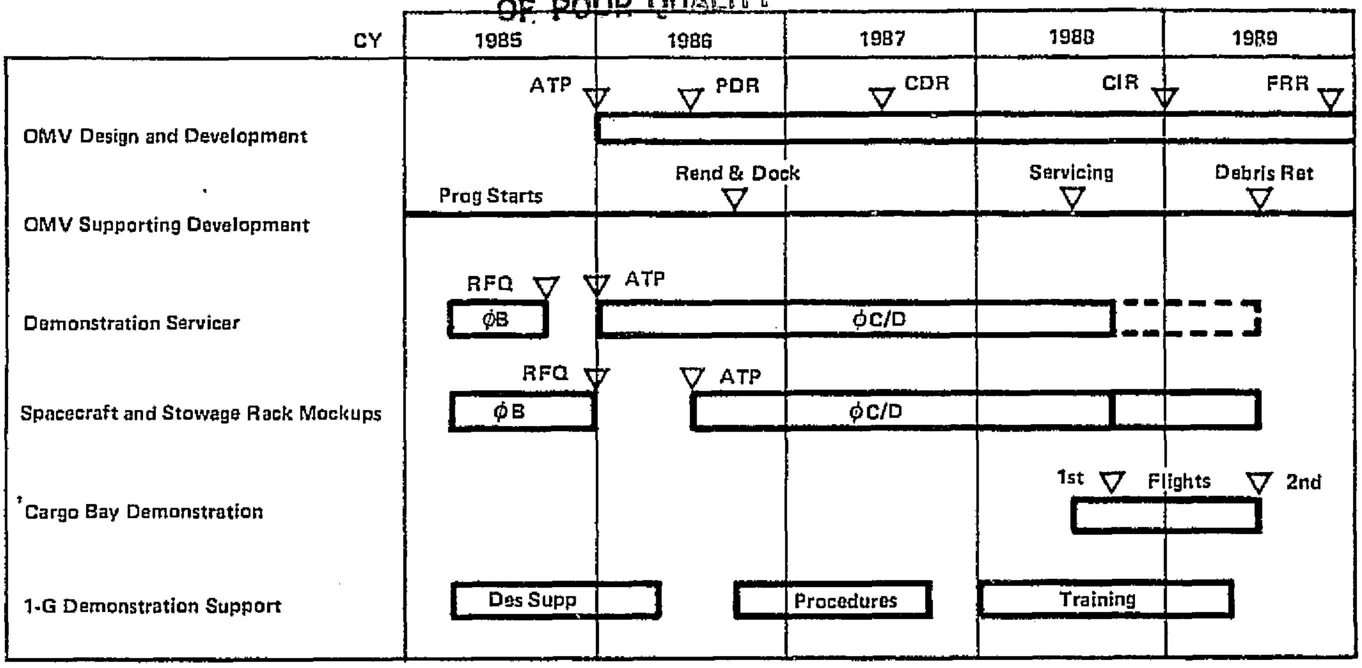

Figure 5.4-1 Servicer Cargo-Bay Demonstration Schedule

recommended to start together in April of 1985. However, the spacecraft and stowage rack mockups have been allowed 9 mos for Phase B as compared to 6 mos for Phase B for the servicer as less is known about the mockups. A longer time is allowed for the servicer Phase C/D (32 vs 26 mos) because of its greater complexity. A dashed extension is shown on the servicer Phase C/D to indicate the need to check out the servicer for the second flight. Similarly, an extension period is indicated for the mockup Phase C/D to allow for the rework of the spacecraft and stowage rack mockups for the second demonstration flight. Three periods of support from the 1-g demonstration equipment are shown. The first is for design support and parallels the Phase B activity and the servicer design before PDR. The second period is for procedures development and the third is for operator training. As the control station is on the ground the demonstration operators need not be astronauts. However, some astronaut training will be involved in terms of RMS and FSS operations, data collection, and assistance in overcoming anomalies.

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Figure 5.4-2 provides the next level of detail for the demonstration servicer schedule. A three month break has been allowed between the end of Phase B and the start of Phase C/D to allow for bid and evaluation. The schedule items are representative of this type of project. The system test equipment activity is scheduled to start after PDR and to be complete before system tests start. The system test equipment block also includes any component testing that is required. The fabrication and assembly of the flight unit and airborne support equipment (ASE) are estimated to take 13 months. Software development and documentation for the ASE is included in the design and development block, while that to be used at the ground station is included in the control station block. The control station design, fabrication, assembly, software, and checkout has been scheduled to start after the preliminary design review and to be complete by the start of system qualification tests.

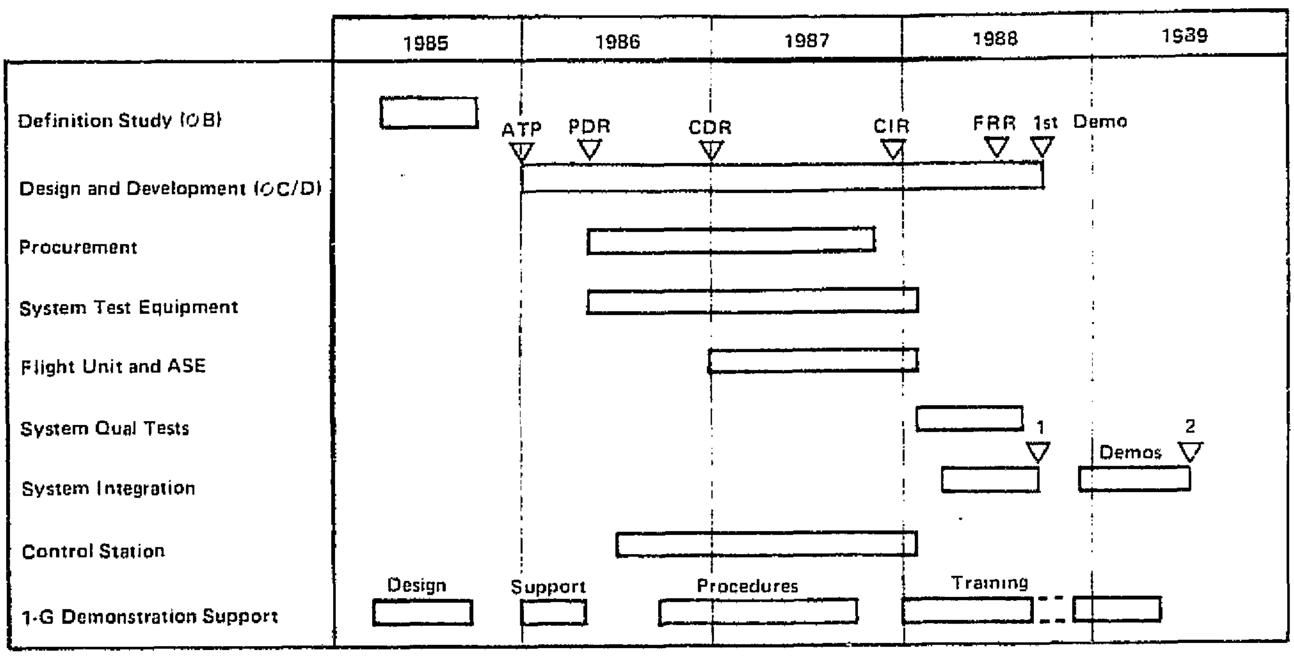

Figure 5.4-2 Demonstration Servicer Schedule

The servicer system integration tests have been divided into separate blocks for demonstrations 1 and 2 as shown on Figure 5.4-2. Support from the 1-g demonstration equipment is shown on the last line of the figure. Design support is required in parallel with Phase B and with the pre-PDR work of Phase C/D. Detailed procedures development work including failures, back-outs, and work-arounds is shown to start before CDR and to end well before the start of the system qualification tests. The operator and astronaut training phases are shown in two segments - one for each demonstration. On the basis of this schedule, the first servicer cargo-bay demonstration could occur in September of 1988 and the second in June of 1989.

A similar schedule for the spacecraft mockup and the stowage rack is given in Figure 5.4-3. This schedule allows for a longer Phase B because of the need to evaluate availability of parts from other programs, the lower level of definition of the spacecraft mockup, and the need to consider module arrangements for two separate demonstrations. A six month period has been allowed between Phase B and Phase C/D to allow for competitive procurement. Each of the schedule line items have been divided into parts associated with demonstrations 1 and 2. No production tooling has been indicated because of the one unit build. The intent is to overdesign the equipment so qualification testing can be limited.

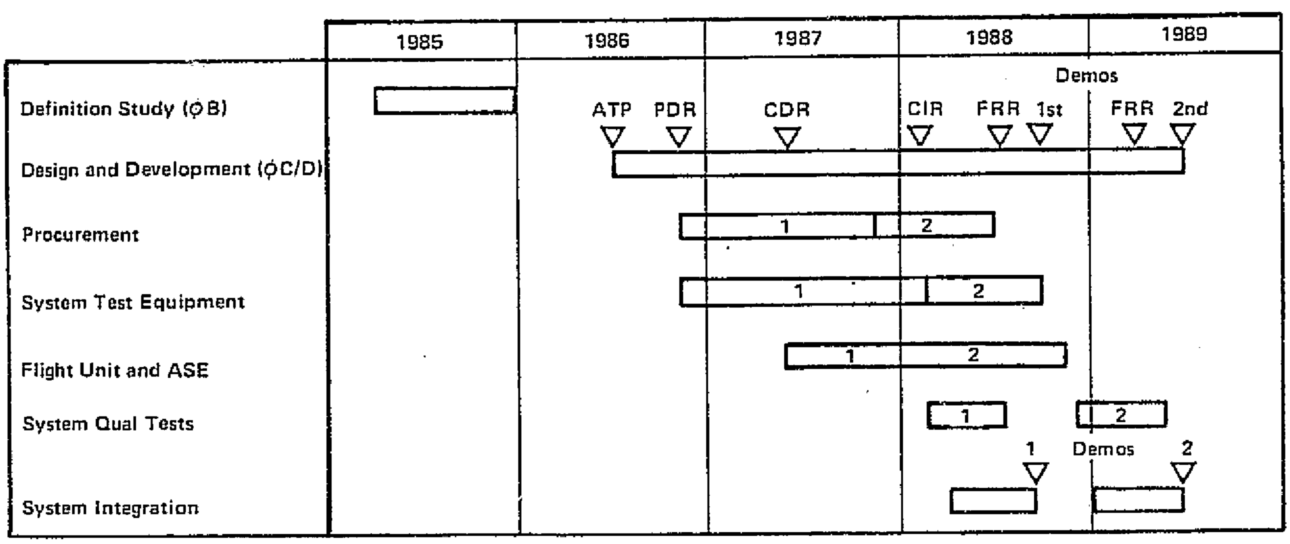

Figure 5.4-3 Spacecraft Mockup and Stowage Rack Schedule

5.4.4 Cost Estimate

A preliminary cost estimate was prepared for the proposed cargo-bay servicer system demonstrations. A total cost is given as well as separate estimates for the two flight demonstrations - one involving module exchange, battery exchange, and access door operation and the other involving refueling operations, tank changeout and access cover replacement.

The costing was based on estimated weights of the cargo-bay demonstration equipment listed in Table 5.4-5. The cost estimate was developed using cost estimating relationships (CER) contained in the Martin-Marietta Aerospace Cost Data Base and in several NASA pricing models. The various cost elements and the basis for cost estimates are shown in Tables 5.4-6. and 5.4-7.

The total estimated cost of the servicer cargo-bay demonstrations, including a contingency of $2.0 million, will be approximately $20 million. In estimating the costs of the cargo-bay servicer demonstrations, the following assumptions were made:

1) All costs are in 1984 dollars;
2) Costs include design/development and fabrication of the experimental unit;
3) Not included are the launch costs for shuttle, the cost of using the MMS flight support system, the cost of an RMS standard end effector and grapple fixture for docking, and the part of the propellant resupply equipment from the NASA/JSC orbital refueling program;
4) Test data reduction and analysis and report preparation are not included;
5) The cost of cargo bay demonstration equipment is assumed to be one half of the equivalent operational equipment, since qualification will be for cargo-bay experiments rather than for fully operational equipment;

Table 5.4-6 Cost Estimate For Cargo-Bay Servicer Demonstrations-First Flight

ELEMENT	COST, $M (FY 84 $)	BASIS
IOSS System		
Servicer Mechanism	2.0	Airborne Structures and Mechanisms CER NASA REDSTAR CER
Airborne Support Equipment	3.0	Airborne Avionics CER
Docking Probe	0.5	Airborne Structures and Mechanisms CER
Replacement Modules & Doors	2.5	Structural CER
Servicer to Orbiter Interface Equipment	1.5	Airborne Avionics CER
Spare Module Stowage Rack	3.5	Airborne Structures and Mechanisms CER NASA REDSTAR CER
Stowage Rack to Orbiter Interface Equipment	0.5	Airborne Structures and Mechanisms CER Airborne Avionics CER NASA REDSTAR CER
Subtotal	13.5	
Servicer Control Station in OMV GCS		
Control Consoles & Software	1.0	Analogous to Viking Control Console Analogous to Peacekeeper Monitor and Control Console.
Servicer Ground Checkout Equipment		
Mechanical C/O Equipment	0.1	Mechanisms and Structures GSE CER
Electrical C/O Equipment	0.2	Electrical GSE CER
Stowage Rack Ground Checkout Equipment		
Mechanical C/O Equipment	0.1	Mechanisms and Structures GSE CER
Electrical C/O Equipment	0.2	Electrical GSE CER
Spacecraft Mockup		
Structure	0.3	Structural CER
Spacecraft Ground Checkout Equipment		
Mechanical C/O Equipment	0.2	Mechanisms and Structures GSE CER
Electrical C/O Equipment	0.2	Electrical GSE CER
Personnel		
Engineers and Technicians (Including Training)	0.2	Development Schedule
Total	16.0	

Table 5.4-7 Cost Estimate For Cargo-Bay Servicer Demonstrations-Second Flight

ELEMENT	COST, $M (FY 84 $)	BASIS
Propellant Resupply Equipment		
Propellant Module	0.5	Propulsion CER Airborne Structures and Mechanisms CER
Propellant Resupply Probe	0.5	Propulsion CER Airborne Structures and Mechanisms CER
Hose Management System	0.5	Propulsion CER Airborne Structures and Mechanisms CER
Subtotal	1.5	
Rework Spacecraft Mockup for Propellant Tank Exchange		
Propellant Tank	0.5	Propulsion CER Airborne Structures and Mechanisms CER
Personnel		
Engineers and Technicians (Including Training)		Included in First Flight
Total	2.0	
Total Cargo Bay Demo:	18.0	

6) There will be minimal rework of the servicer and stowage rack between the two cargo-bay demonstration flights.

The cost of design, development and manufacturing the servicer system for the first cargo-bay demonstration flight is estimated at $13.5M.

The servicer mechanism will be similar to the Engineering Test Unit (ETU) used in the ground demonstrations with minor design modifications for 0-g operation and will use the protoflight design approach. It will be based on the preliminary design for the on-orbit servicer developed during the IOSS study effort. The cost of the design, development and manufacturing will be reduced as much as possible by minimizing the traceability and configuration control requirements. However, the orbiter safety requirements will be fully satisfied. A removable counterbalance system will be provided for ground checkout.

Thermal coatings and heaters for thermal control in earth orbital operation will be provided. Low outgassing, flat viscosity index wet lubricant compatible with earth orbital environment will be used. All materials and processes used will be space-compatible. Seals will be provided outside of all lubricated bearings. Each drive will be tested under thermal vacuum conditions simulating the cargo-bay working environment. The plastic film potentiometers on the shoulder and elbow joints will be replaced by resolvers for improved accuracy. Other joint modifications such as a more accurate ball screw drive for the shoulder pitch joint and a belleville spring arrangement for the worm of the wrist yaw drive will be incorporated. Full length arm segments (79 in.) will be used instead of the reduced length version of the ETU. A space qualified TV camera will be used instead of the one presently installed on the ETU.

The docking probe will be fully functional, will incorporate a folding and latching mechanism for stowage and a RMS type standard end effector, modified for docking operations, based on the results of the OMV docking probe development program. Grapple fixtures and optical targets will be installed on the spacecraft mockup for RMS and docking probe interface and TV camera and lights will be provided on the stowage rack for docking operations. Costs were not included for the RMS standard end effector and grapple fixture. These units will also be used during the free-flight verification.

Two sets of MMS module mockups, battery module mockups, and access doors with actuation/latch mechanisms will be built. This hardware is intended to be reused in the free flight demonstrations but will be built to experiment requirements as it is not part of the operational equipment.

Included in the servicer to orbiter interface equipment will be the orbiter communication system as well as the power interface. The control and power cables will be routed across the moving interface of the deployable MMS flight support system. Other sensor and power circuits will be provided for the stowage rack to orbiter interface.

A spare module stowage rack will be designed and built to satisfy requirements similar to those applicable to the servicer mechanism. Built-in attachment points for all the modules, tanks, refueling modules, cables, and equipment used in the two cargo-bay demonstrations will be provided on the stowage rack for easy change-over.

The cost of design and fabrication of the servicer control station within the OMV ground control station includes the control console and the related software required for controlling the servicer from the ground and for test data acquisition and storage. The traceability and configuration control requirements will be minimized, consistent with the orbiter safety requirements, in order to reduce cost.

The spacecraft mockup will be designed as a cargo bay experiment rather than fully flight qualified hardware. Traceability and configuration control requirements will be minimized and structural testing will be reduced by increasing the design margins. Special emphasis will be placed on the orbiter safety requirements. All the attachment points for the modules, battery, access cover, replaceable tank and for other components, cable and piping for both flight demonstrations will be provided for easy changeover. Two grappler fixtures with optical targets will be provided, one for the docking probe and the other for the RMS. Two trunnions, a keel fixture, and their support structure will be provided for attachment to the orbiter during launch and return.

All the experimental equipment will be checked out on the ground prior to launch. The necessary mechanical and electrical checkout equipment will be designed and built.

The personnel required to support both flights of the servicer system cargo bay demonstrations was estimated as follows:

1) Test procedures writing will require two persons for six months:
2) Training of two operators will span a four month period;
3) Ground operational support for both flights will require 15 persons for 48 hours.

For the second flight, a propellant resupply module will be designed and built. It will include support structure, stowage rack attachments, tanks for hydrazine and pressurant gas, thermal protection, plumbing and other components, electrical connectors, sensors and cabling, flexible fluid lines and their management system including the latches for securing it in a stowed position during launch and the refueling interface unit modified for interfacing with the servicer end effector. The design will be based on the refueling unit demonstrated in the ground tests with the required modifications for reducing the weight and changing from water/air to hydrazine/nitrogen gas. The experience accumulated during the NASA-JSC propellant transfer tests will be fully utilized to minimize the design and qualification tests. Orbiter safety and contamination requirements will be major design drivers.

There will be a configuration change between the first and second cargo-bay demonstrations, affecting mostly the stowage rack, the spacecraft mockup and the orbiter interfaces. Retesting of elements from the first flight and modifications are possible and a cost allowance was provided.

As mentioned, the personnel requirements were estimated once for both flights. It was assumed that the same individuals will support both flights to minimize the training costs.

5.5 FREE-FLIGHT VERIFICATION

While the objective of the orbiter cargo-bay experiment was to encourage potential users of on-orbit servicing in the form of module exchange and show that the major elements of the system can be designed, built and operated, the objective of a free-flight verification is to verify that the equipment is operational and ready for use. Thus, the verification equipment must be designed to satisfy the operational needs and to operational requirements and processes.

5.5.1 Flight Plan

An example flight plan is introduced and explained so that the equipment involved and time phasing can be better understood. The example flight plan is representative and alternative plans, with different initial assumptions, can also be prepared and evaluated. One of the precursors to a free-flight verification is the need to demonstrate mating of the servicer stowage rack and the OMV. This demonstration has been suggested as part of the space station technology development mission (TDM). It is also assumed that the OMV has progressed through its development program to where an OMV is available and can be launched with adequate propellant for the free-flight verification mission onboard. The serviceable spacecraft is assumed to be a special spacecraft for the verification. It might be that there is a failed spacecraft requiring servicing of the kinds to be demonstrated when it is needed, but it is very unlikely. So the plan is to obtain a special serviceable spacecraft. In addition to the full size operable modules to be exchanged, the spacecraft would require an attitude control system, two-way communications to the ground through the TDRSS, a docking receptacle, a translational thrust capability to put it on a drift orbit with respect to the orbiter, and the usual structure, power supplies, and thermal control. This plan shows the serviceable spacecraft being returned to earth (to avoid more space debris), but it may be possible to use it for some other mission after servicing has been verified. An alternative to a special design and build of a serviceable spacecraft is to use the Shuttle Pallet Satellite (SPAS-01) built by Messerschmitt-Boelkow-Blohm (MBB). The SPAS-01 would have to be reconfigured for this special use including the addition of replaceable modules and a propellant resupply system, and the upgrading of its communications system to work with the TDRSS. However, the SPAS-01 is an interesting alternative that should be considered.

The flight plan starts out with the servicer, with replacement modules, the OMV and the serviceable spacecraft being launched together in the orbiter. The servicer would be fastened to the OMV before launch and would be returned to earth with the OMV. At the appropriate time in the mission, the sequence of activities listed in Table 5.5-1 would be started. All activities, other than those involving the orbiter Remote Manipulator System (RMS), are controlled from the ground. Because of the close relationship between the servicer and the OMV, ground control of the servicer could be from the OMV ground control station (GCS).

At step 9, the spacecraft attitude control system must be shut off so that it does not fight the OMV attitude control system. Two types of servicing are recommended - module exchange and propellant transfer. Example modules are battery, electrical power conditioner, attitude control electronics, or communications. Propellant transfer will likely be a mono-propellant for the spacecraft attitude control system. High pressure gas could also be transferred. An electrical connector could be mated to aid in control of the propellant transfer. Alternatively, control of propellant transfer could be from the ground to the servicer and spacecraft through separate communications links.

Alternatives to the OMV for boosting the spacecraft back towards the orbiter would be to use the spacecraft attitude control system thrusters to initiate the transfer or for the OMV and spacecraft to return to near the orbiter in the docked configuration.

The OMV is put into a quiescent mode when it is near the orbiter. The orbiter will then do whatever maneuvering is necessary for the RMS to be able to reach out and retrieve the OMV and servicer. When both the servicer and spacecraft have been stowed in the orbiter, the orbiter crew can continue with their other mission tasks or initiate reentry and landing.

The mission duration time has been estimated at 21 hrs. This time period can be modified extensively depending on the desired separation between the orbiter and spacecraft and on whether time or propellant is used to achieve and remove the separation distance. Of the total mission time of 21 hrs, the orbiter crew need only be involved for eight hrs. The ground operations crew will need to be involved for the total mission time.

Table 5.5-1 Free-Flight Verification Sequence

Step No.	Activity	Step Time (hr)	Cum Time (hr)
1.	Spacecraft deployed by RMS	2	2
2.	Spacecraft checked out from ground	1	3
3.	Spacecraft put into drift orbit ($\Delta V \approx 5$fps)	1	4
4.	Servicer and OMV are unstowed and deployed by RMS	1	5
5.	OMV initiates transfer trajectory towards spacecraft	--	5
6.	Servicer mechanism is unstowed and docking probe is deployed	--	5
7.	Servicer is checked out	1	6
8.	OMV rendezvous with spacecraft	2	8
9.	OMV docks with spacecraft using servicer docking mechanism	1	9
10.	Servicer exchanges modules	2	11
11.	Servicer transfers propellants	2	13
12.	OMV boosts spacecraft onto trajectory to near orbiter	--	13
13.	Servicer undocks	1	14
14.	Servicer folds and stows servicer docking probe	--	14
15.	OMV and servicer transfer to orbiter	3	17
16.	OMV rendezvous with and station keeps close to orbiter	1	18
17.	OMV is put in quiescent mode	--	18
18.	RMS attaches to OMV and servicer	--	18
19.	RMS stows OMV and servicer in orbiter	1	19
20.	Orbiter rendezvous with and station keeps close to spacecraft	.1	20
21.	RMS attaches to spacecraft	--	20
22.	RMS stows spacecraft in orbiter	1	21

5.5.2 Equipment Required

The equipment required, in addition to that which is part of the Space Transportation System (STS), can be separated into two groups - "existing" equipment and verification specific equipment. By "existing" is meant equipment that is planned to be in existence as part of other programs at the time of planned use.

Table 5.5-2 lists existing equipment that is required. The orbiter and other parts of the STS, such as the TDRSS, will be used, but were not included in the list. The list is relatively short and involves mostly items related to the OMV project. It is also required that the OMV be available and have sufficient propellant on board to perform the free-flight demonstration. It is assumed that the servicer 1-g trainer is available from the servicer cargo-bay demonstration program.

Table 5.5-2 Equipment Available From Other Programs

Orbital Maneuvering Vehicle
Ground Control Station for OMV
OMV Docking and Rigidization System
Servicer 1-g Trainer

The free-flight verification equipment that must be procured specifically for the free-flight verification project is listed in Table 5.5-3. The two major pieces are the Integrated Orbital Servicing System and the serviceable spacecraft. Each of these requires a ground-based control station, interface equipment with the orbiter, and ground checkout equipment. It is recommended that the IOSS be built to operational equipment standards and that planning include the delivery of two operational units and the ability to produce additional units for use with the space station. The servicer support equipment should also be designed to operational standards and for repeated use. The serviceable spacecraft and its support equipment could be designed on a one-time use basis and might even incorporate equipment from other programs into its design or the SPAS-01 spacecraft built by MBB might be used. The cost estimates are based on rental of a spacecraft bus and an allowance for refurbishment of the rented spacecraft.

Table 5.5-3 Free-Flight Verification Equipment

Integrated Orbital Servicing System
Servicer Control System in OMV Ground Control Station
Replacement Modules
Propellant Resupply Equipment
Servicer to Orbiter Interface Equipment
Servicer Ground Checkout Equipment
Serviceable Spacecraft
Spacecraft Control System in OMV Ground Control Station
Spacecraft to Orbiter Interface Equipment
Spacecraft Ground Checkout Equipment

5.5.3 Schedule

The free-flight verification schedule was developed from an OMV development schedule provided by MSFC. The key points from the OMV schedule are an OMV authority to proceed (ATP) of Jan 1, 1986 and a first flight on Jan 31, 1990. These dates are shown on Figure 5.5-1, along with other key OMV dates, such as the preliminary design review (PDR) and critical design review (CDR). OMV operations were assumed to start immediately after the first flight although a transition flight test period is likely to occur. The OMV schedule also showed a series of end dates for OMV supporting development. Rendezvous and docking development could start on Nov 1, 1986 and servicing development could start on July 1, 1988. It was assumed that a first rendezvous and docking flight could occur on July 1, 1990. This represents a 44 mo development span for rendezvous and docking. A slightly longer span of 49 mos was selected for the servicer and serviceable spacecraft development. The result is a verification flight in July of 1992.

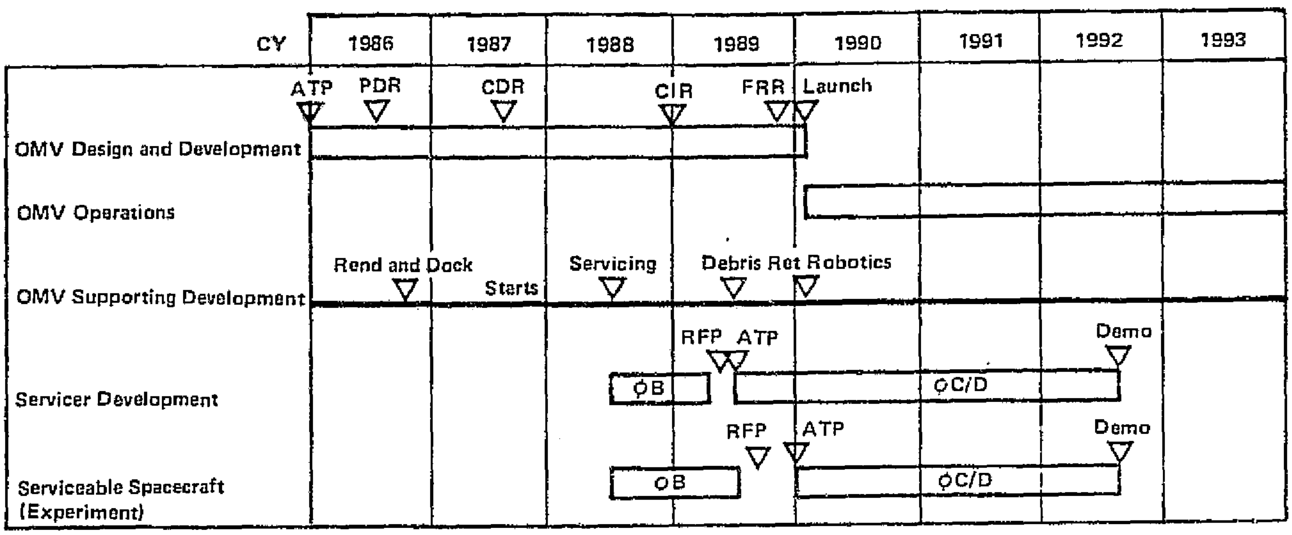

Figure 5.5-1 Free-Flight Verification Schedule

Phase B of both the servicer and serviceable spacecraft start together in July 1988. However, the spacecraft Phase B is estimated as 12 mos with 9 mos for the servicer because less is known about the serviceable spacecraft. A longer time is allowed for the servicer Phases C and D (37 vs 31 mos) because the servicer is considered to be operational equipment and thus, will require a longer design and development period.

Figure 5.5-2 provides the next level of detail for the servicer development schedule. A three month span has been allowed between the end of Phase B and the start of Phases C and D to allow for bid and evaluation. The schedule items are representative of this type of project. Tooling has been included so that multiple flight units may be manufactured in subsequent contract phases. A structures/propulsion test article was not included because there is no propulsion equipment and a structural test article should not be required except for the stowage rack. The servicer will be designed for stiffness to keep the mechanism natural frequencies high and thus will have high strength margins. The system test equipment is scheduled so that component tests can be conducted early and so that a full set of equipment is available for the system qualification tests. The control station design, fabrication, assembly, and checkout has been scheduled to start after the preliminary design review and to be complete by the start of system qualification tests. On the basis of this schedule, the flight verification could occur in July of 1992.

A similar schedule for the serviceable spacecraft is given in Figure 5.5-3. This schedule allows for a longer Phase B because of the need to evaluate availability of parts from other programs and because of the lower level of definition of the serviceable spacecraft. A six months period has been allocated between Phase B and Phases C and D to allow for a competitive procurement. The Phase C/D span for the serviceable spacecraft is shorter than for the servicer because the spacecraft is being treated as experiment equipment rather than as operational equipment. As only one spacecraft would be built there is no need for production tooling. Otherwise, the serviceable spacecraft schedule is similar to the servicer development schedule. The schedule is long enough that it may well be compatible with the use of the SPAS-01.

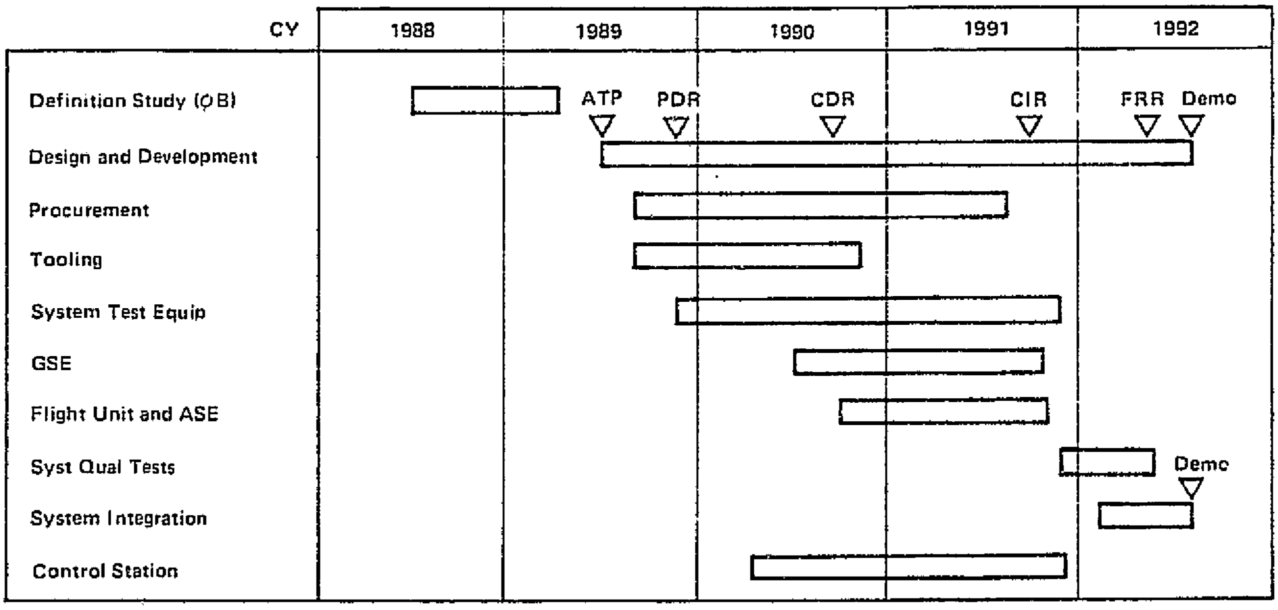

Figure 5.5-2 Servicer Development Schedule

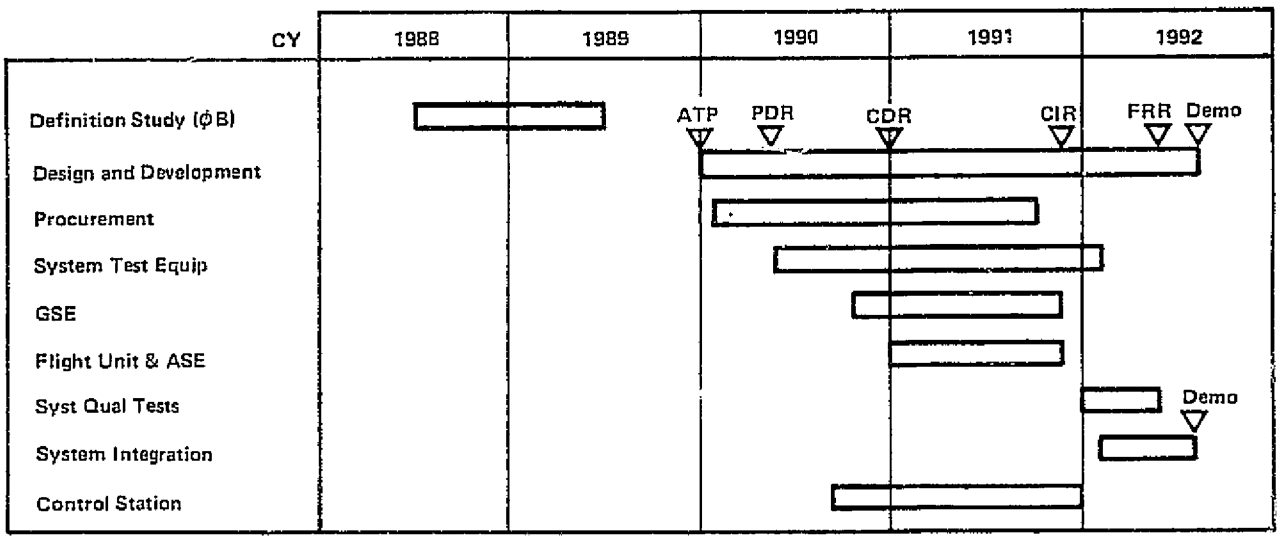

Figure 5.5-3 Serviceable Spacecraft Development Schedule

5.5.4 Cost Estimate

A preliminary cost estimate was prepared for the free-flight servicer system verification. Module exchange and monopropellant refueling operations were included.

The costing was based on estimated weights of the equipment for the free-flight servicer verification as listed in Table 5.5-3. The cost estimate was developed using cost estimating relationships (CER) contained in the Martin Marietta Aerospace Cost Data Base and in several NASA pricing models.

The various cost elements and the basis for cost estimates are shown in Table 5.5-4. The total estimated cost of the engineering effort and building the servicer system for free-flight verification will be approximately $35 million, including a contingency allowance of $3.2 million.

In estimating the cost of the free-flight servicer verification the following assumptions were made:

1) All costs are in 1984 dollars;
2) Costs include design, development, and fabrication of the operational units;
3) Not included are shuttle launch costs for the servicer, OMV and the serviced satellite;
4) Test data reduction and analysis and the report preparation are not included;
4) The cost of the OMV ground control station was not included except for the modifications required for the operation of the servicer attached to OMV;
5) The design and build of two units of the servicer for the free-flight verification will use the traceability, configuration control and qualification requirements of fully operational equipment;
6) Leasing of the SPAS-01 satellite and its modification for the module exchange and refueling functions as the serviced satellite was assumed for minimum cost;
7) Replacement modules and doors, refueling modular system and the stowage rack will be reused and reworked as necessary from the cargo-bay servicer demonstration tests.

Table 5.5-4 Cost Estimate for Free-Flight Servicer Verification

ELEMENT	COST, $M (FY 84 $)	BASIS
IOSS System		
Servicer Mechanism	8.0	Airborne Structures and Mechanisms CER NASA Space Division CER
Airborne Support Equipment and Software	8.0	Airborne Avionics CER NASA Langley CER
Docking Probe	3.0	NASA REDSTAR CER NASA Space Division CER
Stowage Rack Rework	2.0	Adjustment of Cargo-Bay Estimate
Replacement Modules and Doors	0.5	Rework from Cargo-Bay Demo.
Servicer to OMV Interface	3.5	Airborne Avionics CER Airborne Structures and Mechanisms CER
Stowage Rack to OMV Interface	0.5	Airborne Avionics CER Airborne Structures and Mechanisms CER
Subtotal	25.5	
Servicer Control Station in OMV GCS		
Control Consoles and Software	3.0	Analogous to Viking Control Console Analogous to Peacekeeper Monitor and Control Console
Servicer Ground Checkout Equipment		
Mechanical C/O Equipment	0.2	Mechanisms and Structures GSE CER
Electrical C/O Equipment	0.3	Electrical GSE CER
Stowage Rack Ground Checkout Equipment		
Mechanical C/O Equipment	0.2	Mechanisms and Structures GSE CER
Electrical C/O Equipment	0.1	Electrical GSE CER
SPAS-01 Satellite		
Leasing Fee	1.0	10% of Original Cost
Experiment Package	1.0	Previous Experiment Packages
Spacecraft Ground Checkout Equipment		
Mechanical C/O Equipment	0.2	Mechanisms and Structures GSE CER
Electrical C/O Equipment	0.2	Electrical GSE CER
Personnel		
Engineers and Technicians (Includes Training)	0.1	Development Schedules
Total Free-Flight Demo:	31.8	

The servicer mechanism design will be similar to the cargo-bay demonstration servicer except for the traceability, configuration control and qualification tests suitable for fully operational space flight equipment. Modifications resulting from the cargo-bay demonstrations will be incorporated in the design. The modules to be exchanged, the access covers and doors and the refueling hardware will be the ones used in the cargo-bay demonstrations but reworked and updated for this one-time use. The stowage rack cost is based on reworking the stowage rack from the cargo-bay tests and building a new second unit.

The servicer control station will be redesigned and integrated with the OMV ground control station. The servicer control software will be updated.

The mechanical and electrical equipment for the servicer ground checkout will be designed and built.

It is assumed that the SPAS-01 satellite built by MBB will be available for lease when needed for use as a serviceable spacecraft. It will be modified to add an experiment package, comprised of the modules to be exchanged, tanks, piping and other components for the refueling verification. The spacecraft to orbiter interface will be provided by SPAS-01. Ground checkout equipment for this serviceable spacecraft will be designed and built.

The personnel costs include test procedure writing, operator training and manning of the ground control station during the 21 hours of free-flight servicer operations.

The engineering effort and hardware build for the overall servicer system development is estimated to be $56.5 million. This is a preliminary estimate based on the conceptual design of the Integrated Orbital Servicer System and the proposed cargo-bay demonstration and free-flight servicer verification plans.

6.0 SERVICING DEVELOPMENT PLAN

The objective of this activity is to integrate the results of Sections 3 and 5 of this report into an orderly development plan leading to a fully verified operational on-orbit servicing system based on the module exchange and refueling/resupply technologies. The key servicing development plan issue is the need to balance the number and complexity of development activities against available funds. The proposed approach is to lay out a program with most of the desired features, that overlaps the 0-g, 1-g, and operational servicer demonstrations, and attempts to get an early operational capability. This approach minimizes costs by taking advantage of parallel activities such as the JSC refueling program, and advocates renting a spacecraft bus rather than buying a new one. The program is also scoped large enough to become a recognized part of NASA's long-range plans. The promise of a clear plan by NASA to develop and use module exchange for many years will encourage the user, or spacecraft designer, to incorporate module exchange in his plans.

In evolving this approach, a range of alternatives were considered. At the high end of the spectrum was a servicer development program to demonstrate several forms of module exchange, several cover door opening or removal approaches, three or four approaches to refueling (propellant resupply), and several approaches to cryogenic resupply in each of three areas--1-g, cargo bay, and free flight. The three phases were put in series so full advantage of prior work could be incorporated in subsequent activities; this resulted in a long and expensive program. Additionally, on-orbit servicing opportunities would be lost with a concurrent loss of potential savings.

If NASA is unable to fully fund the development of the module exchange form of orbital servicing at this time, one approach is to take advantage of the opportunities that arise, such as experiment opportunity announcements. In this way it is possible to maintain the momentum that now exists. As a living document, the servicer development program plan can be adapted to opportunities as they arise.

The approach to presenting the servicing plan is to take the information in Sections 3 and 5 of this report, to abstract the results and conclusions, and to put them in a different order. The selected order is to first discuss the plan elements in terms of objectives, issues to be addressed, approach, and results expected. This is followed by discussions of schedules and estimated funding needs. Each of these areas--plan elements, schedule, and costs--are also addressed in terms of (1) total plan, (2) ground demonstrations, (3) cargo-bay demonstrations, and (4) free-flight verification.

The major characteristics of the spacecraft servicing demonstration plan are listed in Table 6-1. The plan leads to the existence of two units of an operational servicer system that has been verified by free-flight testing and is suitable for use with the space station. The plan can be completed by mid 1992 at a cost of 56.5M in 1984 dollars. Deletion of the second operational unit could save $3M.

Table 6-1 Major Characteristics of Plan

- Three integrated activities
  - Ground demonstrations
  - Cargo-bay demonstrations
  - Free-flight verification
- Based on proven IOSS design and test hardware
- Emphasis on exchange of MMS modules
- Inclusion of other representative servicing tasks
- Inclusion of refueling
- Two cargo-bay flights
- Free-flight verification of an operational servicer suitable for use with the space station
- Activity completion schedule

Ground demonstrations	Mid 1986
Cargo-bay demonstrations	1988/89
Free-flight verification	Mid 1992

- Funding Estimate

Ground demonstrations	$ 1.5M
Cargo-bay demonstrations	$20.0M
Free-flight verification	$35.0M
Total	$56.5M

Three servicer mechanism configurations are involved:

1) The Engineering Test Unit currently in use at MSFC would be used for ground demonstrations;
2) A protoflight quality unit that would be built for the two demonstration flights in the orbiter cargo bay;
3) Two fully operational units that have been qualified and documented for use in the free-flight verification tests.

The Engineering Test Unit (ETU) is retained on the ground as it would require extensive rework before space flight and it is needed for training, procedures development, and troubleshooting. The protoflight unit is recommended for the cargo-bay tests to minimize costs, and a pair of fully operational units is necessary if an operational capability is to exist.

6.1 PLAN ELEMENTS

The plan elements are those major characteristics of the plan that are necessary to establish a basis for cost estimation. Plan elements are identified by starting with a statement of objectives of the activity and an identification of the issues to be addressed. Next is the approach to performing the activity. The recommended approach resulted from consideration of a variety of alternatives and includes a definition of the plan elements. This part concludes with discussions of expected results.

6.1.1 Servicing Development

The overall servicing development plan is discussed first so that its major components--ground demonstrations, cargo-bay demonstrations, and free-flight verification--can be better understood.

6.1.1.1 Objective--The objective of the spacecraft servicing demonstration activity is to develop an on-orbit servicing capability that is ready for use by others, is integrated into the Space Transportation System, can perform module exchange and refueling/resupply, and can operate at, or away from, the orbiter and the space station.

6.1.1.2 Issues to be Addressed--The following servicing development issues were identified and discussed in Sections 3 and 5:

1) Enhancement of user acceptance;
2) Incorporation of representative servicing equipment at each stage;
3) Completion of the program within three years after first OMV flight;
4) Inclusion of verification procedures, analysis techniques, and 1-g simulations in plan;
5) A funding stream that is phased to user acceptance;
6) Minimization of overall costs while constraining risks;
7) Performance of tests in lowest cost environment;
8) Separation of development into viable segments;
9) Maximization of knowledge transfer to potential users;
10) Adaptability to changes in knowledge level.

6.1.1.3 Approach--The basic approach to the servicing development program is to establish a continuing program that includes three interrelated activities--ground demonstrations, orbiter cargo-bay demonstrations, and free-flight verification. The ETU is used for the ground demonstrations, a protoflight servicer and stowage rack along with a spacecraft mockup are used for the cargo-bay demonstrations, and an operational servicer mechanism, the protoflight stowage rack, a rented spacecraft bus, and mocked up modules are used for the free-flight verification. Representative servicing equipment modules, refueling equipment and control systems are used and functionally upgraded throughout the program.

Risks are constrained and costs are minimized as concepts, methods, and techniques are investigated and checked out in the easiest environments first before progressing to the more demanding situations. Also ground test hardware is available to help analyze any anomalies that may occur in space. The existence of a continuing ground test program means that potential users can become involved at any time, even to the extent of having their specific spacecraft situations demonstrated.

Separation of the program into the three interrelated activities provides a number of advantages:

1) Users can get involved at any step and influence what is done in subsequent steps;
2) The early year funding requirements are low yet the users can be made aware of the specific end goal of the program;
3) Each segment of the program is viable once its precursors are well under way;
4) The program can be modified as better knowledge concerning technology, user acceptance, competitive technologies, and available funds become known.

6.1.1.4 Results Expected--The primary result of the servicing development program is the existence of an operational on-orbit servicing system that is available for use. Secondary results include:

1) Methods and equipment for module exchange and on-orbit refueling and resupply that are applicable to the space station;
2) Better control system approaches;
3) Data on how to configure spacecraft for servicing;
4) More useful orbital maneuvering and transfer vehicles;
5) The potential for saving hundreds of millions of dollars on future spacecraft programs.

6.1.2 Ground Demonstrations

The ground demonstrations are conducted first as they are less expensive, the equipment is more accessible and is easier to reconfigure, a wider range of tests can be conducted, and the data is easier to collect.

6.1.2.1 Objective--The objectives of the ground demonstrations are to obtain a better understanding of on-orbit servicing so that the cargo-bay demonstrations may be better focussed and to increase user confidence in the technology and in the program. These objectives can be expanded as:

1) To demonstrate the adaptability and flexibility of the module exchange concept;
2) To demonstrate the use of the equipment as a laboratory tool for development of servicer equipment and procedures;
3) To demonstrate the use of the ground servicer as a training facility.

6.1.2.2 Issues to be Addressed--The following ground demonstration related issues were identified and discussed in Sections 3 and 5 of this report:

1) Control system upgrading and refinements;
2) Adaptability of the module servicing tool to the Engineering Test Unit;
3) Methods for handling the MMS modules;
4) Development of operating procedures;
5) Operator training;
6) Astronaut training;
7) Identification of refueling methods and fuel line management techniques;
8) Demonstration of battery and other module exchanges and access door removal activities;
9) Evaluation of alternative interface mechanisms;
10) Demonstration of axial, radial and compound motions;
11) Tank and other refueling system components exchange;
12) Automatic target recognition and error correction;
13) Evaluation of new equipment, tools, end effectors, and adapters;
14) Evaluation of new sensors;
15) Evaluation of alternative electrical and waveguide disconnects;
16) Demonstration of space station specific tasks.

It has been recommended that the Engineering Test Unit mechanism and end effector be used for the ground tests. Thus these items are no longer considered to be issues.

6.1.2.3 Approach--The basic approach to the ground demonstration is to use the existing servicing demonstration facility at MSFC with the Engineering Test Unit, mockups, electronics, and computer. The existing end effector and interface mechanism would also be used as a starting point. The MSFC staff and their contract help would be used to operate the facility, run tests, collect data, and publish results. New and modified equipment and software could be obtained in-house or from outside contractors.

A series of overlapping tests and demonstrations would be conducted. Each test or demonstration would involve:

1) Planning;
2) Equipment procurement, installation, and checkout;
3) Software definition, modifications, and checkout;
4) Procedures development;
5) Test or demonstration operations;
6) Data reduction and analysis;
7) Report preparation and distribution.

The recommended tests and demonstrations include:

1) Control system upgrading;
2) MMS module exchange;
3) Generic module exchange;
4) Refueling demonstrations;
5) Automatic target recognition tests;
6) Tests suggested by users or spacecraft designers.

It was also suggested that the ETU and its electronics be upgraded by conversion from an analog to a digital system and by going from wire wrapped boards to printed circuit boards. It is recommended that this type of improvement be delayed until it is clearly shown to be necessary.

Upon completion of the above list of tests and demonstrations, the servicing demonstration facility would be used to support the flight activity in terms of (1) flight demonstration simulations, (2) flight training, and (3) problem solving.

In parallel with the flight support work, the servicing demonstration facility could be used for:

1) Demonstration of propellant tank and other system components exchange;
2) Development of light weight side and bottom mounting interface mechanisms;
3) Development of new interface mechanism concepts;
4) Development of other adapter tools;
5) Development of new end effectors;
6) Development of special refueling and electrical disconnects;
7) Development of in-line fluid couplings;
8) Development of space station specific servicing tasks.

6.1.2.4 Results Expected--The primary result from the ground demonstrations is the knowledge and confidence to continue to the orbiter cargo-bay demonstrations. Secondary results include:

1) Existence of an operating ground test facility for the evaluation of on-orbit servicing systems;
2) An increased level of user acceptance;
3) A better understanding of the adaptability and utility of the module exchange concept;
4) An operable training facility.

6.1.3 Cargo-Bay Demonstrations

The cargo-bay demonstrations are conducted after the ground demonstrations so they can benefit from the results of the ground demonstrations. A smaller number of demonstrations will be required and a set of equipment that satisfies the requirements of the experiments to be conducted in the orbiter is appropriate. Two flights are recommended. A first flight that only involves module exchange and a second flight that involves refueling. In this way, the control, mechanism, and communications aspects can be settled before the fluid flow aspects.

6.1.3.1 Objective--The objectives of the cargo-bay demonstrations are to confirm the ground tests, show that there are no anomalies, to demonstrate that module exchange and on-orbit refueling can be done, and to increase user confidence in the technology and the program. It is recommended that the servicer control station be on the ground to bring communications link aspects into the situation and to place fewer demands on the flight crew.

6.1.3.2 Issues to be Addressed--The following cargo-bay demonstration related issues were identified and discussed in Sections 3 and 5 of this report:

1) Demonstration of MMS module exchange;
2) Demonstration of other module exchange activities;
3) Demonstration of refueling;
4) Demonstration of tank and other refueling system components exchange;
5) Communications links;
6) Control station location;
7) Supplementary visual aids;
8) Supplementary TV cameras;
9) Direct viewing;
10) Deployment of servicer docking probe;
11) Servicer mechanism performance;
12) Interface mechanism performance;
13) Connector performance including mate and demate--electrical and fluids;
14) Methods of accommodating attach errors;
15) End effector capture;
16) Interface mechanism capability for capture, latch, unlatch, and release;
17) System force and torque levels;
18) Repeatability accuracy (electro/mechanical);
19) Spacecraft to servicer alignment;
20) Spacecraft module removal and replacement trajectories;
21) Control system modes validation;
22) Man machine interaction;

23) Lighting;
24) Malfunction mode/backup systems;
25) Mission/man/STS system safety;
26) Pre and post module exchange condition analysis.

6.1.3.3 Approach--The basic approach to the orbiter cargo-bay demonstrations is to fabricate a new servicer mechanism to protoflight, or experiment, standards and to use it for two demonstration flights. The first demonstration flight would only involve module exchange, primarily MMS modules, and the second flight would involve refueling. There is a sufficiently large number of functions to be verified (see Section 6.1.3.2) that it is felt to be a better approach to leave the complexities of a refueling demonstration to a second flight.

The recommended configuration arrangement involves use of the orbiter RMS for docking the spacecraft mockup to the deployed servicer. The servicer and spare module stowage rack would be supported by and deployed by the flight support system cradle A prime of the MMS. Specific cargo-bay demonstration characteristics are:

1) Satellite mockup unstow and stow by RMS;
2) Supply of power, attitude control, communication link access and thermal control by orbiter;
3) Two-way communications links to ground through orbiter and TDRSS;
4) Servicer control station at OMV ground control station;
5) Docking rigidization by servicer docking probe;
6) Deployment of servicer docking probe;
7) MMS module exchange demonstration;
8) Refueling demonstration;
9) Servicing equipment performance demonstration;
10) Control modes evaluation;
11) Man-machine interactions evaluations;
12) Compliance with orbiter system safety requirements;
13) Deployment of stowage rack in orbiter by MMS flight support system;
14) Use of representative servicing operational equipment;
15) Operator training.

The recommended activities for the first test flight are:

1) A Multimission Modular Spacecraft type module using an MMS module servicing tool, incorporating an electrical connector, and mounted so that the module moves axially;
2) Battery module on a lightweight side interface mechanism using an electrical connector and with a near-radial module motion direction;
3) Hinged access door mounted so that the servicer end effector is attached in a radial direction.

The recommended activities for second test flight are:

1) A multiple line propellant resupply module with a refueling interface unit and a hose management device mounted in a far axial direction;
2) A propellant tank module on a lightweight side interface mechanism using a propellant in-line coupling drive and mounted in a near radial direction;
3) An access door treated as a module on a lightweight side interface mechanism and mounted in the near axial position.

Certain equipment required for the servicer cargo-bay demonstration is part of the auxiliary equipment available for use on the orbiter as part of the Space Transportation System. This equipment includes:

1) Remote Manipulator System;
2) Attitude control, electrical power, thermal control;
3) MMS flight support system;
4) Orbiter communications equipment;
5) Cargo-bay cameras;
6) RMS cameras.

The cargo-bay demonstration equipment that must be procured new, or adapted from another use, includes:

1) Integrated Orbital Servicing System;
2) Servicer control station in OMV ground control station or on orbiter aft flight deck;
3) Replacement modules and doors;
4) Propellant resupply equipment;

5) Servicer to orbiter interface equipment;
6) Servicer ground checkout equipment;
7) Servicer 1-g trainer;
8) Spare module stowage rack;
9) Stowage rack to orbiter interface equipment;
10) Stowage rack ground checkout equipment;
11) Spacecraft mockup;
12) Spacecraft to orbiter interface equipment;
13) Spacecraft ground checkout equipment.

The servicer 1-g trainer is assumed to be available from the servicer ground demonstration program and the propellant resupply equipment is assumed to come partially from the JSC orbital refueling program. Two sets of replacement modules and doors are required - one set for each flight. The propellant resupply equipment is only required for the second flight. Communication from the orbiter to the ground control station, if used, would go through the TDRSS.

6.1.3.4 Results Expected--The primary result from the demonstration is the confidence that modules can be exchanged and propellants can be successfully transferred in zero-g by remote control. Secondary results include:

1) Confirmation of the ground test results;
2) Validation of the ground test equipment;
3) An increase in user acceptance;
4) An understanding of communication link effects.

6.1.4 Free-flight Verification

The free-flight verification tests are considered to be the final proof that establishes an orbital servicing capability. Thus the design, development, and test process must be suitable for operational equipment. Similarly all the appropriate documentation must be prepared so that the capability can be used by others. It is recommended that at least two production units and adequate spares be procured so there will be a higher availability of servicing equipment for operational flights.

6.1.4.1 Objective--The objective of the free-flight verification tests is to verify that an operational servicer capability exists and is available for use by the user community. These verification tests should also increase confidence that the servicer can be used at the orbiter, at, or near, the space station, and in geosynchronous orbit.

6.1.4.2 Issues to be Addressed--The following free-flight verification related issues were identified and discussed in Sections 3 and 5 of this report:

1) Demonstration of MMS module exchange;
2) Demonstration of other module exchange activities;
3) Demonstration of refueling;
4) Demonstration of rendezvous and docking;
5) Communications links;
6) Control station location;
7) Deployment of servicer docking probe;
8) Servicer mechanism performance;
9) Interface mechanism performance;
10) Connector performance including mate and demate - electrical and fluids;
11) Methods of accommodating (compliance) attach errors;
12) End effector capture;
13) Interface mechanism capability for capture, latch, unlatch and release;
14) Repeatability accuracy (electro/mechanical);
15) Spacecraft to servicer alignment;
16) Control system modes validation;
17) Man machine interaction;
18) Malfunction mode/backup systems;
19) Mission/STS system safety;
20) Pre and post module exchange condition analysis.

6.1.4.3 Approach--The basic approach to the free-flight verification tests is based on the desire to have a fully operational on-orbit servicer system at the end of the program. This means that the servicer system must go through the full design and development process including the production of two flight units so that a backup unit is available for each operational flight. A single verification flight is recommended. It is possible to use less fully qualified components for the serviceable spacecraft and the modules to be exchanged. As the serviceable spacecraft is a one-time use, it may be possible to rent a spacecraft bus and to fit it with an experiment package, where the experiment package is the serviceable spacecraft.

Specific free-flight verification characteristics are:

1) Serviceable satellite mockup supported by a rented spacecraft bus;
2) Supply of power, attitude control, thermal protection, communication link access and control of the servicer from the OMV;
3) Use of OMV for rendezvous and docking of servicer to the serviceable spacecraft mockup;
4) Two way communication links to ground through TDRSS;
5) Servicer control station at OMV ground control station;
6) Docking rigidization by servicer docking probe;
7) Deployment of servicer docking probe;
8) MMS module exchange demonstration;
9) Refueling demonstration;
10) Servicing equipment performance verification;
11) Control mode verification;
12) Operator training.

Certain existing equipment will be required. It includes:

1) Orbital Maneuvering Vehicle;
2) Ground control station for OMV;
3) OMV docking and rigidization system;
4) Servicer 1-g trainer.

The orbiter and other parts of the STS, such as the TDRSS, will be used, but were not included in the list. The list is relatively short and involves mostly items related to the OMV project. It is assumed that the servicer 1-g trainer is available from the ground demonstration program. The free-flight demonstration equipment that must be procured specifically for the free-flight demonstration project includes:

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13) Interface mechanism capability for capture, latch, unlatch and release;
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1) Integrated Orbital Servicing System;
2) Servicer control system in OMV ground control station;
3) Replacement modules;
4) Propellant resupply equipment;
5) Servicer to orbiter interface equipment;
6) Servicer ground checkout equipment;
7) OMV servicer electronics control interface equipment;
8) Serviceable spacecraft mockup;
9) Spacecraft bus (rented);
10) Spacecraft control system in OMV ground control station;
11) Spacecraft to orbiter interface equipment;
12) Spacecraft ground checkout equipment.

The two major pieces are the Integrated Orbital Servicing System (IOSS) and the serviceable spacecraft equivalent. Each of these requires a ground-based control station, interface equipment with the orbiter, and ground checkout equipment. It is recommended that the IOSS be built to operational equipment standards and that planning include the delivery of two operational units. The servicer support equipment should also be designed to operational standards and for repeated use. The serviceable spacecraft mockup and its support equipment could be designed on a one-time use basis. The spacecraft bus might be the SPAS-01 spacecraft built by MBB.

6.1.4.4 Results Expected - The primary result of the free-flight verification activity is the existence of an operational servicer capability ready for use by specific spacecraft programs or by the space station. Secondary results include:
1) Increased user confidence;
2) Potential for significant savings in future spacecraft programs.

6.2 SCHEDULES

Schedules were developed in Sections 3 and 5 for each of the three on-orbit servicing development phases. As each schedule was based on the OMV development schedule, they integrate easily with each other.

6.2.1 Servicing Development Schedule

The schedule for the overall servicing development plan was based on an OMV first flight date of Jan 1990. The OMV supporting development schedule also provides a servicing supporting development end date of July 1988. The start of Phase B development for the operational servicer configuration was also assumed to be July 1988. Figure 6.2-1 shows the timing of the major steps in developing the on-orbit servicer.

The ground demonstrations overlap the cargo bay demonstrations at the beginning of the cargo-bay demonstration activity and then the ground demonstration equipment is used to support the flight activities. The free-flight verification activity overlaps the cargo-bay demonstrations and leads to verification of the servicer 30 months after the first flight of the OMV.

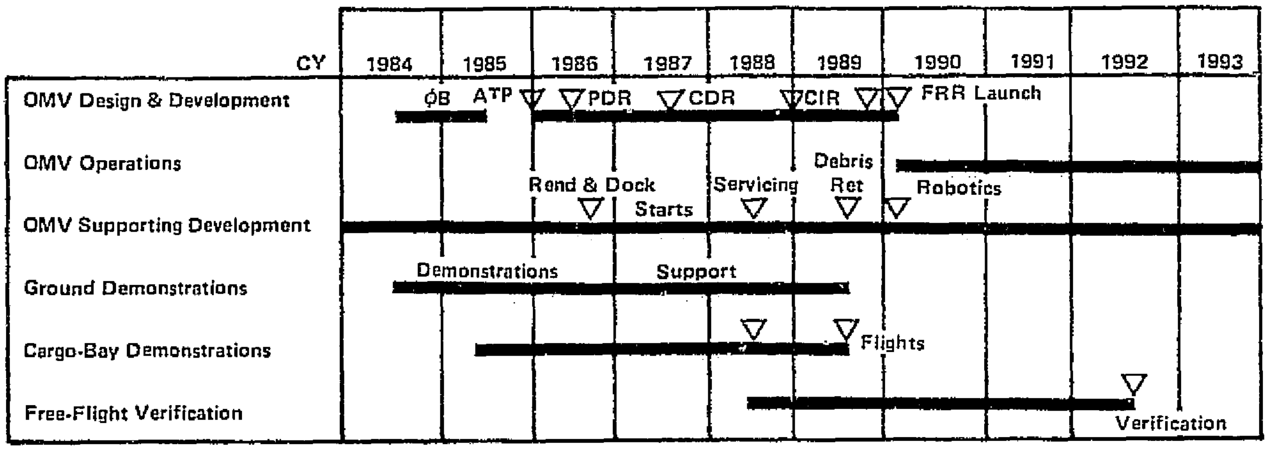

Figure 6.2-1 On-Orbit Servicing Development Schedule

6.2.2 Ground Demonstrations Schedule

The ground demonstration schedule is shown in Figure 6.2-2. The first five items are ground demonstrations and are arranged in a waterfall pattern with significant overlapping during procurement and preparation for the demonstrations. However, the actual demonstrations (tests) are conducted one at a time. The last three activities shown in the figure are flight support activities. Other activities discussed in Section

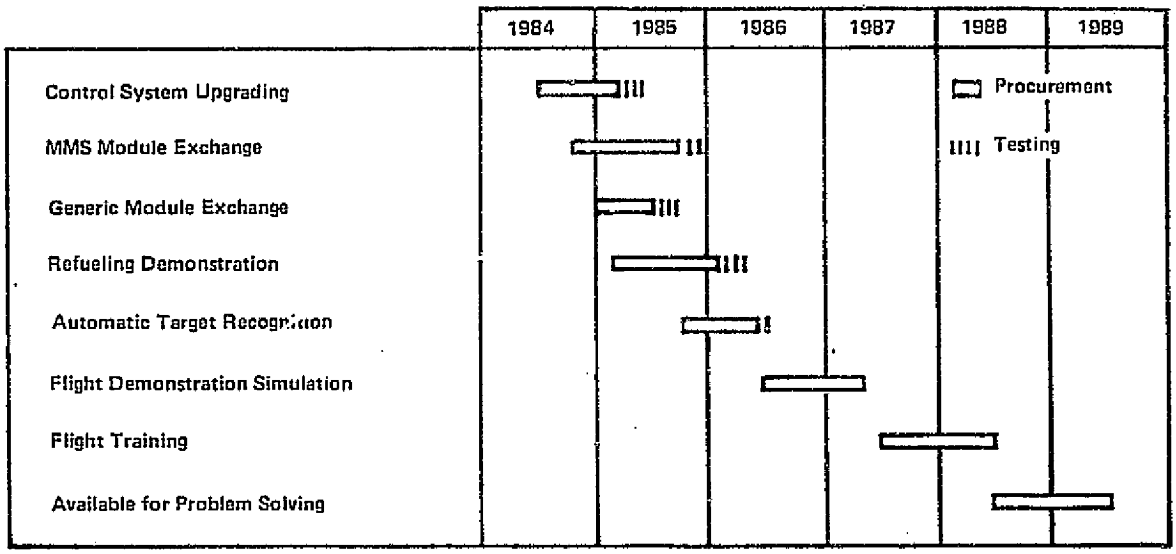

Figure 6.2-2 Ground Demonstration Program Plan

6.1.2 can be conducted during breaks in the flight support activities. Generally, the demonstrations themselves are a month or less with most of the time being spent in preparation and checkout.

6.2.3 Cargo-Bay Demonstrations Schedule

The top-level cargo-bay demonstration schedule is shown in Figure 6.2-3. The first two lines again show the OMV design and development schedule and the OMV supporting development schedule for reference. A short Phase B is shown for the cargo-bay demonstration servicer as sufficient work has been done to quickly arrive at a preliminary design. However, a thirty-two month period has been allowed for Phases C and D because of the design complexity and the need to integrate the servicer and stowage rack with the MMS flight support system and with the orbiter. Phase B for the spacecraft and stowage rack mockups are shown at nine months because less is known with regard to the desired configuration and the need to develop requirements and concepts for both flights into a single set of equipment. Phases C and D for the mockups are a little shorter because the equipment is not as complex as the servicer mechanism with its control electronics and software.

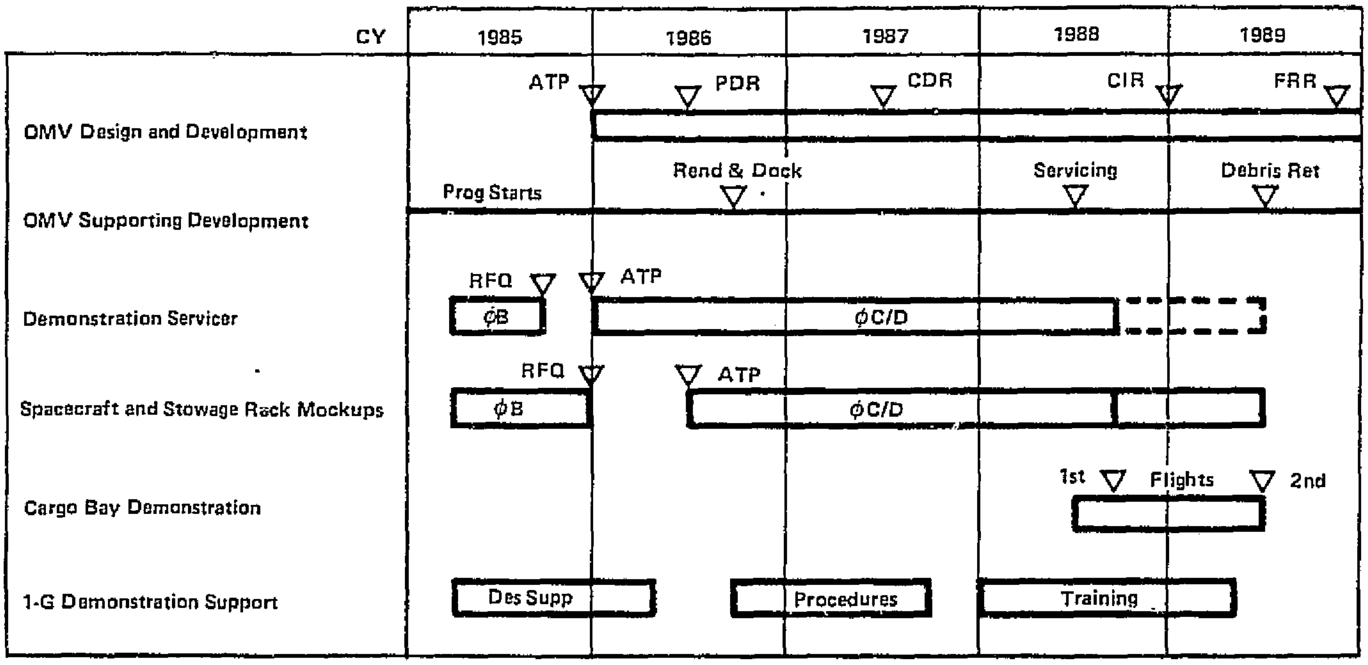

Figure 6.2-3 Servicer Cargo-Bay Demonstration Schedule

Two demonstrations are shown. The first flight--module exchange--is scheduled for September 1988 and the second flight--refueling demonstration--is scheduled for ten months later to allow for rework of the mockups and revisions to the servicer software. The three support phases for the 1-g demonstration equipment are also shown on Figure 6.2-3. The design support activity parallels the B Phases and up to PDR of the servicer Phase C. The procedures support activity includes CDR of the servicer phase C. The training activity is mostly for the operators, although some astronaut failure mode and backup procedure training will probably be required. The training period covers both flights. More detailed cargo-bay demonstration schedules are given in Section 5.4.3.

6.2.4 Free-Flight Verification Schedule

The top-level free-flight verification schedule is shown in Figure 6.2-4. The first and third lines respectively again show the OMV design and development schedule and the OMV supporting development schedule for reference. A bar indicating on-going OMV operations

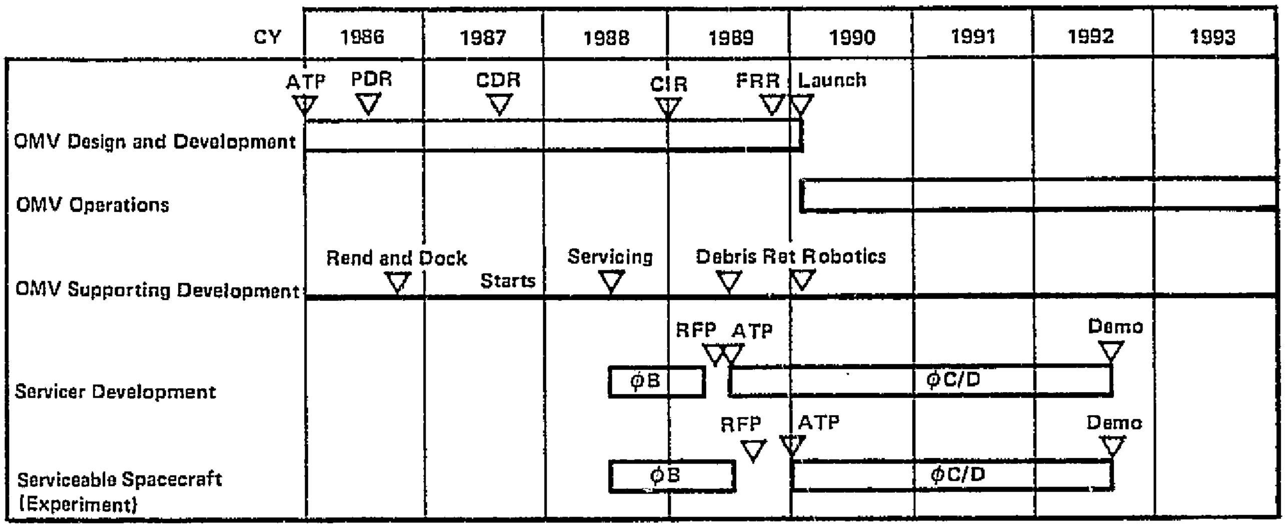

Figure 6.2-4 Free-Flight Verification Schedule

starting in 1990 is also shown. A nine month period is shown for the revised Phase B because of the high level of definition available. Three years have been allocated for the servicer Phases C and D as being representative for operational equipment of this complexity. One year has been allowed for the serviceable spacecraft Phase B because of the need to identify and verify the availability of a low cost approach to obtaining a spacecraft bus. The spacecraft Phase C/D span was selected to end at the same time as the servicer. The result is an on-orbit servicer verification flight in July of 1992. More detailed free-flight verification schedules are included in Section 5.5.3.

6.3 FUNDING ESTIMATES

Funding estimates were developed for each of the three on-orbit servicing development phases in Sections 3 and 5. The funding estimates have been combined with the schedules of Section 6.2 to arrive at yearly funding requirements.

An estimate of the funding required for servicing development is shown in Table 6.3-1 by development phase and by year. The total development

Table 6.3-1 Servicing Development Funding Estimate
(Millions of 1984 dollars)

Item	Total	1984	1985	1986	1987	1988	1989	1990	1991	1992
1. Ground Demonstrations	1.5	0.16	0.72	0.52	0.10					
2. Cargo-Bay Demonstrations	20.0		1.87	3.17	9.28	5.17	0.51			
3. Free-Flight Verification	35.0					3.50	3.80	10.54	13.43	3.73
Totals	56.5	0.16	2.59	3.69	9.38	8.67	4.31	10.54	13.43	3.73

cost is $56.5M. All funding requirements are shown in 1984 dollars. Inflation can be accounted for in later versions of the plan if desired. The plan has two funding peaks. The first peak is in 1987 and corresponds to preparation for the cargo-bay demonstrations. The second peak occurs in 1991 and corresponds to preparations for the free-flight verification tests.

An estimate of the funding required for the ground demonstrations is shown in Table 6.3-2. The funding requirements are based on the data derived in Section 3.0 and have been spread according to the schedule of Figure 6.2-2. Most of the activities involve two year spans. The peak funding requirement is in 1985. Funding requirements for operating the servicing demonstration facility including data collection, analysis, and reporting have not been included. Additional funds will be required for activities other than those listed in Table 6.3-2 that are discussed in Section 3.3.

Funding estimates for the cargo-bay demonstration activities are given in Table 6.3-3 and have been spread according to the schedule of Figure 6.2-3. The funding estimates are based on the cost data presented in Section 5.4.4. The most significant cost element is the servicer mechanism, its electronics, and ground checkout equipment. The peak funding requirement is in 1987 and involves procurement and fabrication

Table 6.3-2 Ground Demonstrations Funding Estimate
(Thousands of 1984 dollars)

Item	Total	1984	1985	1986	1987
1. Control System Upgrading	100	60	40		
2. MMS Module Exchange	450	100	250	100	
3. Generic Module Exchange	600		200	300	100
4. Refueling Demonstration	250		200	50	
5. Automatic Target Recognition	100		30	70	
Totals	1,500	160	720	520	100

Table 6.3-3 Cargo-Bay Demonstrations Funding Estimate
(Millions of 1984 dollars)

Item	Total	1985	1986	1987	1988	1989
1. Servicer Mechanism	7.3	0.73	1.31	3.29	1.82	0.15
2. Module Stowage Rack	4.3	0.43	0.77	1.94	1.16	
3. Spacecraft Mockup	0.7	0.07	0.13	0.28	0.16	0.06
4. Replacement Modules	2.0	0.20	0.40	1.20	0.20	
5. Refueling Equipment	2.5	0.25		0.75	1.25	0.25
6. Ground Control Station	1.2		0.24	0.90	0.06	
7. Contingency	2.0	0.19	0.32	0.92	0.52	0.05
Totals	20.0	1.87	3.17	9.28	5.17	0.51

of equipment. The data of Section 5.4.4 have been regrouped to the titles of Table 6.3-3 for convenience and a 2.0 million 1984 dollar contingency is included.

Certain important costs have not been included as discussed in Section 5.4.4. These include:

1) Launch costs;

2) Servicer development facility related costs;

3) Data collection, analysis, and reporting;

4) MMS equipment use costs;

5) Development of refueling equipment from JSC program;

6) Data collection, analysis, and reporting.

Funding estimates for the free-flight verification activities are given in Table 6.3-4 and have been spread according to the schedule of Figure 6.2-4. The funding estimates are based on the cost data of Section 5.5.4. The most significant cost element is the servicer mechanism, its electronics, and ground support equipment. The peak funding requirements are in 1990 and 1991 and involve procurement and fabrication of the equipment. The data of Section 5.5.4 have been regrouped to the titles of Table 6.3-4 for convenience and a 3.2 million 1984 dollar contingency has been included.

Table 6.3-4 Free-Flight Verification Funding Estimate (Millions of 1984 dollars)

Item	Total	1988	1989	1990	1991	1992
1. Servicer Mechanism	23.0	2.30	3.45	6.90	8.05	2.30
2. Module Stowage Rack	2.8	0.28		0.84	1.40	0.28
3. Spacecraft Bus and Mockup	2.4	0.24		0.56	1.10	0.50
4. Modules and Refueling Equipment	0.5	0.05		0.20	0.25	
5.Ground Control Station	3.1	0.31		1.08	1.40	0.31
6. Contingency	3.2	0.32	0.35	0.96	1.23	0.34
Totals	35.0	3.50	3.80	10.54	13.43	3.73

Certain important costs have not been included as discussed in Section 5.5.4. These include:

1) Launch costs;

2) OMV related costs;

3) Servicer development facility related costs;

4) MMS equipment use costs;

5) Equipment from cargo-bay demonstrations;

6) Data collection, analysis, and reporting.

The module stowage rack costs are based on reworking the stowage rack from the cargo-bay tests and building a new second unit. The costs related to replaceable modules and refueling equipment only include the refurbishment of the units used in the cargo-bay demonstrations.

7.0 CONCLUSIONS AND RECOMMENDATIONS

The significant conclusions and recommendations from this Spacecraft Servicing Demonstration Plan study are presented below. Many secondary conclusions and recommendations are given in Sections 3 through 6. The conclusions and recommendations are presented in order from the bottom up except that those conclusions spanning the study are given first.

7.1 ON-ORBIT SERVICING DEVELOPMENT

The following conclusions and recommendations apply to the overall on-orbit servicing development:

1) The recommended plan leads to the free-flight verification of an operational servicer suitable for use with the OMV and the space station;
2) The plan has three phases
   - Ground demonstrations,
   - Cargo-bay demonstrations,
   - Free-flight verification;
3) The free-flight verification can be completed by mid 1992;
4) The total estimated cost is 56.5 million 1984 dollars;
5) The plan includes three servicer mechanism configurations:
   - The Engineering Test Unit currently in use at MSFC would be used for ground demonstrations, procedures development, and training,
   - A protoflight quality unit would be used for the two demonstration flights in the orbiter cargo bay,
   - Two fully operational units that have been qualified and documented for use in the free-flight verification activity;
6) The plan is based on use of proven IOSS designs and test hardware;
7) A user's council should be formed to direct the implementation of an on-orbit servicing capability.

7.2 MULTIMISSION MODULAR SPACECRAFT

The following conclusions and recommendations apply to the involvement of MMS equipment in the demonstration plan:

1) Primary emphasis would be on demonstrating the exchange of MMS modules;
2) The module servicing tool and the ETU end effector should be adapted to work together for the exchange of MMS modules;
3) Lightweight MMS module mockups with standard MMS attachment fixtures and connectors should be used for ground demonstration;
4) On-orbit servicing of MMS modules should be effected by use of lateral docking with a straight docking probe adapter, tool adapter and modified stowage rack;
5) The MMS flight support system should be used to support the stowage rack and servicer during the cargo-bay demonstrations.

7.3 REFUELING DEMONSTRATIONS

The following conclusions and recommendations were made with respect to refueling demonstrations:

1) Refueling should be demonstrated;
2) Refueling/resupply modular units should be mounted on the stowage rack and connecting hoses should be positioned and connected by the servicer arm;
3) The refueling interface should be standardized;
4) The refueling demonstration equipment should be based on the NASA-JSC standardization effort;
5) Development work is necessary for in-line coupling and mate/demate mechanisms;
6) A Martin Marietta conceptualized refueling/resupply interconnection method looks promising.

7.4 REPRESENTATIVE SATELLITE MODULES

The following conclusions and recommendations were made with regard to selection of representative or generic module exchange:

1) A variety of modules other then MMS modules should be involved in the demonstrations;
2) Thermal cover removal/replace mechanisms and sensors for fastener and attach interface status need to be developed;
3) Changeout of modules representative of the AXAF and communications satellites should be included in the program;
4) Axial, near-radial, and off-axis module removal directions for spacecraft modules should be included;
5) Changeout of modules on the stowage rack need be in the axial direction only;
6) A variety of interface mechanisms are possible and could be useful;
7) A small, light interface mechanism or a tool adapter to remove conventional captive fasteners should be developed;
8) The interface between the servicer end effector and the interface mechanism, tools, and adapters should be standarized;
9) Deutsch or MMS electrical connectors may be used for representative satellite modules.

7.5 END EFFECTOR SELECTION

The following conclusions and recommendations were developed as part of the end effector configuration selection process:

1) The IOSS end effector is recommended for the ground and flight servicing demonstrations;
2) The IOSS end effector meets the end effector requirements and when complemented by a series of adapters can perform the servicing tasks considered;
3) An electrical disconnect should be added to the IOSS end effector;
4) Special adapters should be developed as required for other types of modules or servicing tasks;
5) Developments in the fields of robotics, telepresence, and artificial intelligence should be monitored for their applicability to on-orbit servicing;

6) The Engineering Test Unit and the cargo-bay demonstration equipment can be used as tools for the evaluation and development of servicer-applicable robotics, telepresence, and automation equipment.

7.6 SERVICER MECHANISM SELECTION

The following conclusions and recommendations were developed as part of the servicer mechanism selection process:

1) The Engineering Test Unit should be used for ground demonstrations;
2) The servicer mechanism selection was based on high fidelity, accuracy, versatility, reliability, cost, and risk avoidance;
3) The ETU servicer mechanism is compact and performs module exchange and other tasks efficiently. It was designed to conduct 1-g module exchange demonstrations and it has an effective counterbalance system;
4) The Proto-Flight Manipulator Arm is not as desirable as the ETU because it requires important development work in order to integrate it in a servicer ground demonstration system.

7.7 ENGINEERING TEST UNIT CONDITION

The following conclusions and recommendations were developed as part of the review and evaluation of the condition of the Engineering Test Unit at MSFC:

1) The Engineering Test Unit is in very good electromechanical condition and no dismantling was necessary;
2) The ETU operations history showed only minor easily resolved anomalies;
3) Recent ETU accuracy test data is similar to that taken when the unit was built;
4) Software modifications are needed for smoother operation and to obtain complete module trajectories;
5) The wrist yaw (Globe motor) drive was found to have a larger performance margin than the wrist pitch drive based on the side and base interface mechanism requirements;

6) Specific detail recommendations for upgrading the ETU are provided in Sections 4.4 and 4.5.

7.8 GROUND DEMONSTRATIONS

The following conclusions and recommendations were developed during the ground demonstration analyses:

1) The control system software of the MSFC servicing demonstration facility should be upgraded;
2) MMS module exchange should be the first ground demonstration activity;
3) The exchange of other generic modules—AXAF and communications satellite—should be coordinated with the respective project offices and then demonstrated;
4) Refueling/resupply hardware should be developed and the process demonstrated;
5) The exchange of batteries or other individual components should be demonstrated along with thermal blanket/access cover removal and replacement;
6) An automatic target recognition and error correction system should be developed and demonstrated;
7) The MSFC servicing demonstration facility should be made available for support of flight operations in terms of simulations, procedures development, training, and problem solving. The facility should also be made available as a laboratory development tool;
8) Additional development areas include:
   - Special refueling disconnects for cryogenics or high pressures, and self aligning conical electrical connectors,
   - Development of in-line fluid couplings for replacement of tanks and other propulsion system components,
   - Demonstration of other servicing tasks specific to space station operations.

7.9 CARGO-BAY DEMONSTRATIONS

The following conclusions and recommendations were developed during the cargo-bay demonstration analyses:

1) A protoflight quality servicer mechanism should be built for use in the two cargo-bay demonstration flights;
2) The orbiter Remote Manipulator System docking arrangement should be used;
3) The servicer should be exercised in all three control modes;
4) The servicer control station costs were based on a ground location. However, use of the orbiter aft flight deck should be investigated further;
5) The characteristics of the recommended servicer cargo-bay demonstration are:
   - Satellite mockup unstow and stow by RMS,
   - Supply of power, attitude control, and thermal control by orbiter,
   - Two-way communications links to ground through orbiter and TDRSS if ground location of service control station is used,
   - Servicer control station at OMV ground control station if appropriate,
   - Docking rigidization by servicer docking probe,
   - Module exchange demonstration,
   - Refueling demonstration,
   - Servicing equipment performance demonstration,
   - Control modes evaluation,
   - Man-machine interaction evaluations,
   - Compliance with orbiter system safety requirements,
   - Deployment of stowage rack in orbiter by MMS flight support system,
   - Use of representative servicing operational equipment,
   - Operator training;
6) The hardware for the refueling demonstrations should be obtained from the ongoing Johnson Space Center refueling demonstration flight program;

7) The recommended activities for the first test flight are:
   - A Multimission Modular Satellite type module using an MMS module servicing tool, incorporating an electrical connector, and mounted so that the module moves axially,
   - Battery module on a lightweight side interface mechanism using an electrical connector and with a near-radial module motion direction,
   - Hinged access door mounted so that the servicer end effector is attached in a radial direction;
8) The recommended activities for the second test flight are:
   - A multiple line propellant resupply module including a refueling interface unit and a hose and cable management device mounted in a far-axial direction,
   - A propellant tank module on a lightweight side interface mechanism using a propellant in-line coupling drive and mounted in a near-radial direction,
   - An access door treated as a module on a lightweight side interface mechanism and mounted in the near-axial position.

7.10 FREE-FLIGHT VERIFICATION

The following conclusions and recommendations were developed during the free-flight verification analyses:

1) A fully operational servicer system that has been qualified and documented should be built for use in the free-flight verification activity;
2) The Orbital Maneuvering Vehicle should be the servicer carrier vehicle;
3) The servicer control modes should be selected based on the cargo-bay demonstration results;
4) The servicer control station should be located on the ground;
5) A spacecraft bus, such as the SPAS-01, should be rented rather than a new spacecraft being built for this one-time application;

6) The characteristics of the recommended servicer free-flight verification are:
   - Serviceable satellite mockup supported by a rented spacecraft bus,
   - Supply of power, attitude control, thermal protection and control of the servicer from the OMV,
   - Use of OMV for rendezvous and docking of servicer to the serviceable spacecraft mockup,
   - Two way communication links to ground through TDRSS,
   - Servicer control station at OMV ground control station,
   - Docking rigidization by servicer docking probe,
   - Deployment of stowed servicer mechanism and docking probe,
   - MMS module exchange demonstration,
   - Refueling demonstration,
   - Servicing equipment performance verification,
   - Control mode verification,
   - Operator training;
7) Demonstration of the mating of the servicer stowage rack to the OMV should be a part of the space station technology development missions;
8) The recommended flight verification activities are:
   - Exchange of MMS module,
   - Exchange of representative modules,
   - Propellant transfer.