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Technical Analysis and System Design

Integrated Orbital Servicing Study Follow-on

MARTIN MARIETTA

TRW
SYSTEMS GROUP

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INTEGRATED ORBITAL SERVICING STUDY FOLLOW-ON

Contributors:

R. O. Zermuehlen
K. R. Zant
A. E. Wudell
M. R. Snodgrass
R. A. Skidmore
A. W. Schlaht
J. Schlafer
J. Rombach
D. H. Mitchell (TRW)
W. M. Miller
H. Meissinger (TRW)
R. J. LaBaugh
G. M. Kyrias
C. Grant
W. L. DeRocher, Jr.

Approved:

W. L. DeRocher, Jr.

W. L. DeRocher, Jr,
Program Manager

MARTIN MARIETTA CORPORATION
P. O. Box 179
Denver, Colorado 80201

FOREWORD

This study was performed under Contract NAS8-30820 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. The final report consists of three volumes:

Volume I - Executive Summary,
Volume II - Technical Analyses and System Design,
Volume III - Engineering Test Unit and Controls.

CONTENTS

CONTENTS (Continued)

CONTENTS (Continued)

Page

FIGURE

Page

FIGURE (Continued)

Page

FIGURE (Continued)

Page

FIGURE (Continued)

Page

TABLE (Continued)

Page

TABLE (Continued)

I. INTRODUCTION AND CONCLUSIONS

Orbiting satellites continue to grow in capability, increase in cost and proliferate in numbers as the nation continues to struggle slowly, but consistently across the frontiers of space. The great exploitation of space has by no means begun; in fact, the exploration is far from complete. More and more frequently, however, glimpses of its great potential are offered the informed and aware. Unlimited sources of power, highly efficient communications and a continuously expanding application of specialized manufacturing processes are only a few opportunities. Man's imagination will undoubtedly multiply these manyfold. The challenge to the technological community today is not necessarily to find a use for space but rather to find the best ways to use it. The entrepreneurs will follow if space can be accessed quickly, safely and cost effectively. The Space Transportation System, particularly the Shuttle Orbiter is the first major step in that direction. It will eventually assure the necessary access. It is by no means the last step. A greater confidence in operating and working in space over long durations must be acquired. Space stations are obvious milestones towards this end. The sheer magnitude in both numbers and diversity of future orbiting satellites, facilities, and utilities necessary to encourage these opportunities will demand longer hardware life, more reliable operation and a continuously greater accountability to cost. This will eventually demand routine maintenance of the orbiting equipment much as electrical power facilities, communications transmitters, manufacturing tools, and computing systems are maintained on earth today.

It is this theme that has motivated the last several years of conceptual studies of on-orbit satellite maintenance under MSFC contract NAS8-30820.

Many alternatives for satellite maintenance have been identified--unmanned orbital servicing systems, manned extravehicular activities, highly reliable expendable designs, and retrieval and return for ground refurbishment. The first Integrated Orbital Servicing Study (IOSS) completed in September 1975 concluded that:

- On-orbit servicing is the most cost effective satellite maintenance approach.
- The development of an on-orbit servicer maintenance system is compatible with many spacecraft programs.

- Spacecraft can be designed to be serviceable with acceptable design, weight, volume and cost effects.

As satellite designs continue to evolve it becomes apparent that there is room for virtually all the alternatives of satellite maintenance at one point or other in the future. The question has become one of "How?", not "Which?" or "Why?". In a word, the "How?" sums up the thrust of this contract's activities. To that end the following major outputs were produced.

- An optimum configuration for an on-orbit satellite servicer system was selected.
- Preliminary design of a flight version satellite servicing mechanism was performed.
- A control system was configured for the arm, and control modes defined.
- Maintainable spacecraft designs were completed for typical high and low earth orbit applications.
- A simulation/demonstration was conducted that demonstrated feasibility and utility of the servicer concept and designs.
- An engineering test unit of the proposed servicing arm was designed and assembled for eventual use as an evaluation tool at MSFC.
- The optimum approach to repair of geosynchronous satellites was identified and the life cycle costs of on-orbit servicing were detailed.

Martin Marietta was aided in this follow-on activity by TRW, Inc. under the direction of David H. Mitchell who was responsible for the serviceable spacecraft design work. It should also be noted that the servicer configuration, mechanism and control system work has broader application to the whole field of teleoperator technology.

This volume documents the trade study work completed in the first year of the follow-on work. It will be complemented by an Executive Summary and an Engineering Test Unit and Control System technical volume.

A. STUDY OBJECTIVES

The overall objective of this study was to continue the development of the orbital maintenance concepts which emerged from the first Integrated Orbital Servicing Study. This current study was to further the design definition of an automated spacecraft servicing system supported by the space transportation system. The objective was to be attained by an evolutionary effort characterized by analytical study, simulation, analysis, and three-dimensional modeling and mockup activities in preparation for the design and fabrication of functional prototype subsystems and systems. These systems were to be evaluated for fit, function, interface, and adequacy with all other elements of the system to ensure the elements and objectives were in phase and represented the best interests of the NASA.

1. Background

The first Integrated Orbital Servicing Study was an 18-month, $264,000 effort completed in 1975. It was primarily concerned with developing a recommended approach to servicing/maintaining spacecraft on orbit as opposed to flying the mission with expendable spacecraft or with spacecraft returned to earth for refurbishment. In this regard, it was not necessary to define the various systems in any high degree of detail. Its objective was to provide the basis for the selection of a cost-effective orbital maintenance system supported by the space transportation system. The large number of prior studies for NASA and DOD were used as a basis.

Of the many approaches to providing servicing function, module exchange was selected for maintenance concept evaluation because it satisfies the majority of the servicing operations with a single technique. This selection is consistent with the findings of the majority of the prior studies. Module exchange can provide the servicing functions of (1) repair failed equipment; (2) repair degraded equipment; (3) overcome design failures; (4) replace/replenish wornout equipment; and (5) update equipment with new models. Equipment includes mission equipment as well as subsystem equipment.

At the outset of the first IOSS, the 1973 NASA payload model was reviewed and 47 spacecraft programs were selected as the maintenance applicable spacecraft set. Based on these spacecraft designs and from the alternative on-orbit maintenance concepts in the literature the pivoting arm mechanism, which exchanges modules in an axial direction, was selected. Figures I-1 and -2 illustrate serviceable configurations of the large X-ray telescope and the INTELSAT being serviced by an on-orbit servicer where the orbiter and tug are the respective carrier vehicles. These figures show two applications of the on-orbit servicer system, recommended by this first IOSS, that can also be applied to an earth-orbital teleoperator system, to a geosynchronous freeflyer, to the solar electric propulsion system, and to some forms of the interim upper stage. The pivoting arm was also found better than either EVA or the Shuttle remote manipulator system for maintenance at the Orbiter.

The extensive cost analysis showed that the savings, across the 47 spacecraft programs, were significantly greater than for ground refurbishment when compared with the expendable spacecraft mode. In addition to the on-orbit servicer preliminary design, one-tenth scale models of the servicer, stowage rack, and three spacecraft were delivered. Two versions of the significant structural interface between modules and spacecraft were designed, fabricated, and delivered. The first Integrated Orbital Servicing Study and several spacecraft studies have clearly shown that space hardware and operational economies may be obtained through the maintenance or service of certain spacecraft on orbit. These study conclusions were obtained even though the servicer system and spacecraft trade studies were not conducted at a level which could determine the degree of dependence or degree of autonomy which should be given to the servicer system, spacecraft, delivery vehicle, or other elements of the operational system. The wide range of spacecraft configurations and/or options which can efficiently utilize maintenance were not developed nor displayed. The current study showed that quite simple systems are obtained when the interdependence of the spacecraft and servicer complexity is considered and the two systems are designed with the other's capabilities in mind.

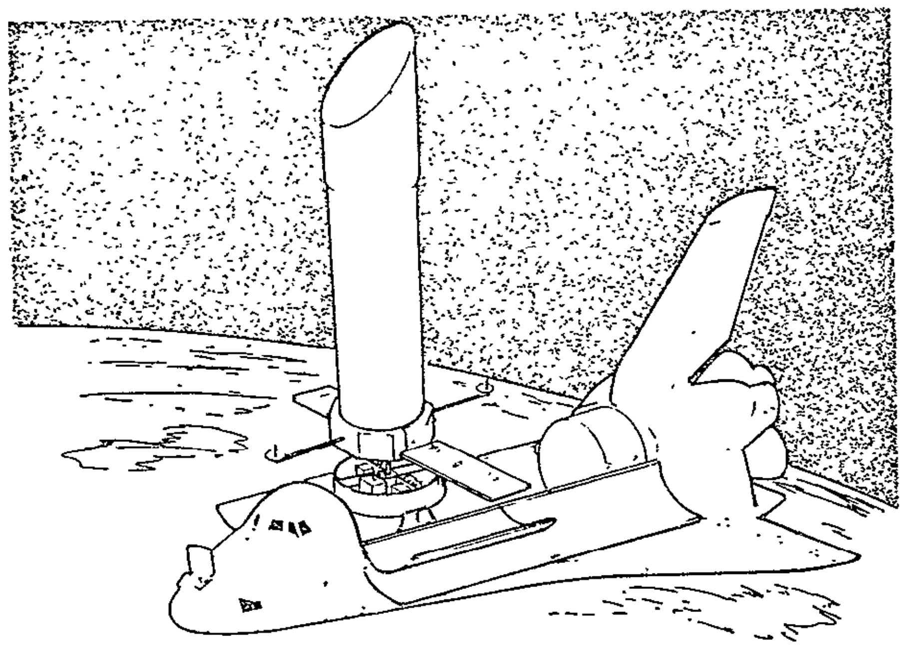

Figure I-1 Servicing the Large X-ray Telescope at the Orbiter

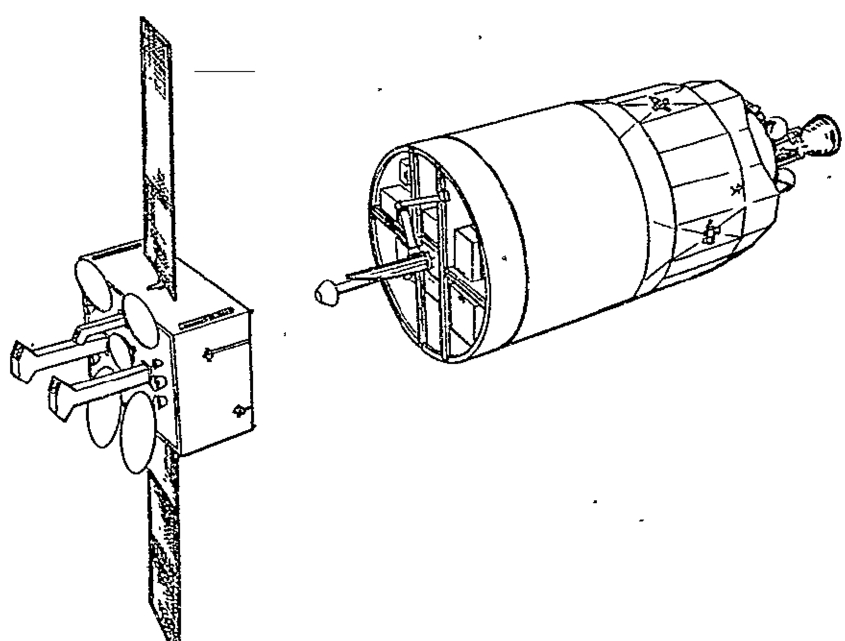

Figure I-2 Servicing the Intelsat via the Full-Capability Tug

2. Specific Objectives

This follow-on effort was intended to provide the definition of the functional and physical requirements of the system. Both low and high earth orbit servicing/maintenance operations were to be addressed. This effort was to continue the first IOSS preliminary design of the sytem in such detail as to support the fabrication of functional mockups combined with engineering trade studies leading to the definition of compatible, maintainable spacecraft and servicer system configurations.

The full range of variables were to be exercised to define the system elements through engineering trades of the servicer system interacted with a typical carrier spacecraft matched to the emerging upper stage designs and the STS system capabilities. The activity was to examine a range of serviceable spacecraft to determine the on-orbit servicer design requirements. It was to result in a preliminary design of the servicer system and of the interface of a serviceable spacecraft, with rationale for the design approach selected. In addition the effort was to provide for the design, fabrication and demonstration of hard functional mockups of the servicer system and the corresponding spacecraft interface. An operable physical representation of the on-orbit servicer preliminary design was to be delivered to NASA.

The specific objectives for this study then are directed toward optimizing those initial designs from the first IOSS and developing more detail in all hardware areas related to servicing. These objectives were:

- To define the on-orbit servicer functional and physical requirements to support both low and high earth orbit servicing/maintenance operations;
- To select an on-orbit servicer and interface mechanism concept that will maximize the utility of a single design approach;
- To provide a higher level of servicer system design detail than that of the first IOSS and to describe a preferred and highly integrated design which fulfills all established servicing requirements;
- To identify a refined and usable control system preliminary design which increases the operational utility of the servicer mechanism;

- To develop a detail characterization of all potentially maintainable spacecraft;
- To prepare a preliminary design of serviceable versions of three selected spacecraft;
- To conduct an analysis to develop an understanding of and approaches to the design of mission equipment for serviceability;
- To conduct a demonstration/simulation of a functional hard mockup of the on-orbit servicer system and associated portions of the selected spacecraft in order to validate the concepts and demonstrate feasibility;
- To prepare an evaluation of the relative utility/profitability of selected high earth orbit maintenance/servicer approaches;
- To prepare a comprehensive review of servicer system life cycle costs as derived from an analysis of system requirements;
- To deliver a full-scale, powered, counterbalanced engineering test unit of the servicer mechanism structure; and
- To provide an on-orbit servicer implementation plan involving an early ground demonstration and subsequent flight demonstration.

The significant issues to be addressed in the study included:

- Identification of criteria for selection of the on-orbit servicer concept;
- Selection of a representative set of mission equipment and identification of approaches to their design for serviceability;
- Identification of the significant serviceable spacecraft design issues;
- Identification of an effective approach to servicer mechanism operation in one-g;
- Selection of a best approach for conducting the demonstration/simulation;
- Identification of the servicer mechanism structure design approach that will maximize the return to NASA.

B. RELATIONSHIP TO OTHER NASA EFFORTS

After years of spacecraft evaluations, maintenance trade studies, and conceptual designs, the work on this contract can best be characterized by a focussing of all this earlier work toward selection of an optimum on-orbit servicing system configuration, and preliminary design of that selection. Mockups and early prototype hardware were actually fabricated and demonstrated to validate the concepts selected.

As a result of the more hardware-oriented nature of this contract the interaction with other NASA efforts was markedly reduced from the first IOSS which was used as the basic reference and the lead to many of the other NASA efforts.

Most related NASA studies on satellite maintenance were completed at the start of this contract. Some of the more significant of these are listed in Table I-1. Their conclusions and results were all available and used where applicable. They proved most useful in the analysis performed to characterize potentially maintainable satellites and the operations analyses. In this regard the DSCS-II study by TRW was based on existing TRW spacecraft and provided much detail data for the serviceable spacecraft preliminary design.

Table I-1 Significant Prior Studies

Payload Supporting Studies for Tug Assessment, MSFC In-house, 1973.
In-space Servicing of a DSP Satellite, SAMSO/TRW, March 1974.
Unmanned Orbital Platform, MSFC/RI, September 1973.
Payload Utilization of Tug, MSFC/MDAC, GE and Fairchild, May 1974.
Operations Analysis, NASA/Aerospace, July 1974.
Servicing the DSCS-II with the STS, SAMSO/TRW, March 1975.
Earth Observatory Satellite System, GSFC/In-house and Contracted, 1976.
Integrated Orbital Servicing and Payloads Study, MSFC/COMSAT, September 1975.
Multi-mission Support Equipment, MSFC/MMC, April 1975.
Orbital Assembly and Maintenance, JSC/MMC, August 1975.
Study to Evaluate the Effect of EVA on Payload Systems, AMES/RI, January 1976.
Multi-mission Support Equipment (Launch Site), MSFC/MMC, June 1975.
Earth Orbital Teleoperator Systems Concepts and Analysis, MSFC/MMC, April 1976.

Table I-2 Concurrent Studies

Proto-flight Manipulator Arm Assembly, MSFC/MMC, April 1977.
Analytical Study of Electrical Disconnect System for Use on Manned and Unmanned Missions, MSFC/MMC, January 1977.
Design, Development, Fabrication and Testing of a Fluid Disconnect for Space Operation Systems, MSFC/Fairchild Stratos, September 1976, 28 months.
High Energy Astronomy Observatory (HEAO) Block II Study, MSFC/Preliminary Design, December 1975.
PLUS, Payload Utilization of SEPS, MSFC/Boeing, July 1976.

Table I-2 lists five concurrent studies that provided helpful information to the IOSS follow-on. The servicer mechanism electromechanical drives for both the space version and the engineering test unit were adapted from the Proto-flight Manipulator Arm designs. The HEAO Block II study data were used as basic information for development of the serviceable Characteristic Large Observatory spacecraft preliminary design by TRW. The PLUS data, along with other Solar Electric Propulsion System (SEPS) data, were used in the geosynchronous spacecraft servicing operations analysis.

C. STUDY APPROACH

The many alternative forms of satellite maintenance are illustrated in Figure I-3. Each of these alternative forms is directed towards increasing spacecraft availability, which is a measure of the time that a spacecraft is ready to perform its intended mission. On-orbit maintenance, or servicing is one way to reduce the cost of spacecraft availability.

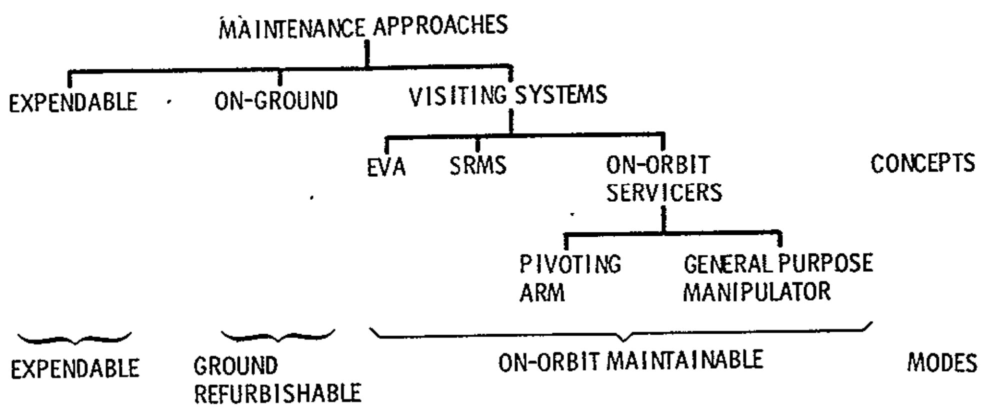

Figure I-3 Spacecraft Maintenance Approaches

The prior studies evaluated these varied concepts and modes and in the first IOSS concluded on-orbit spacecraft maintenance with a special purpose (pivoting arm) manipulator was the preferred approach. As stated in Section A, the objective of this study is to review the initial configuration and expand upon the hardware design and interface definition of that concept to the point of fabricating preprototype/mockup hardware for concept validation in an MSFC servicing demonstration facility. The study approach divides the effort into seven tasks. The tasks and their interrelationships are illustrated in Figure I-4.

Task 1 is a trade study to determine if there is a better alternative to the pivoting arm on-orbit servicer mechanism or to the side- and bottom-mounting space replaceable unit (SRU) interface mechanisms resulting from the first IOSS. It also includes the development of rationale for the selection of a spacecraft and set of mission equipment for use as the reference in the other study tasks.

Task 2 is a further level of design of the on-orbit servicer concept selected in Task 1. It provides a greater level of detail than the first IOSS

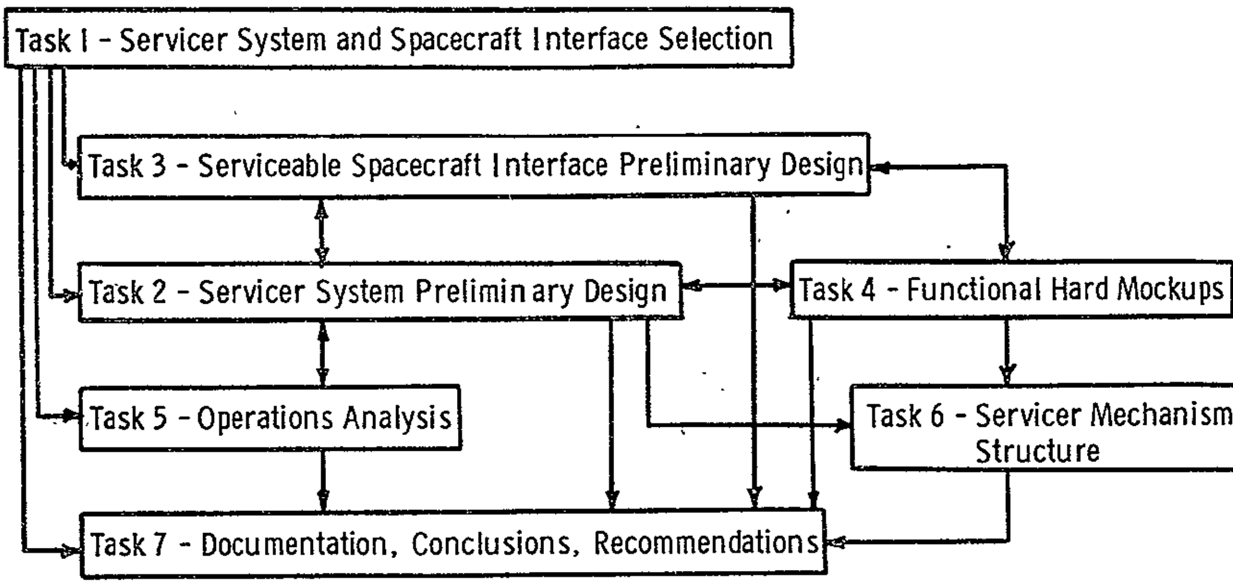

Figure I-4 Study Task Flow

and is directed to the space application. Coordination between the servicer and spacecraft interface design activity of Task 3 is maintained to ensure a highly integrated design.

The purpose of Task 3 is to develop a serviceable spacecraft design for space application of a selected spacecraft with emphasis on the interface aspects. Both spacecraft subsystems and mission equipment are addressed. The module mechanical interfaces are addressed as are the servicing operations aspects.

The mockups of Task 4 employ existing Martin Marietta motion generators, control stations, and control logic systems combined with the interface mechanisms fabricated under the first IOSS to obtain the hard functional aspects.

The first part of Task 5 further explores the benefits of on-orbit servicing in high earth orbits, while the second part addresses the major part of on-orbit servicer life-cycle costs, which occur in the operations phase.

Task 6 addresses the design, fabrication, assembly, and delivery of a full-scale, counterbalanced, powered engineering test unit of the on-orbit servicer mechanism.

Task 7 develops an on-orbit servicing implementation plan with emphasis on identification of ground and on-orbit demonstrations that will lead to early user acceptance. The documentation/coordination activites have been included in Task 7 for simplicity.

The study task and subtask relationships are straightforward with a few interactions. The major interactions are between Tasks 2 and 3 which are the on-orbit servicer design and the spacecraft interface design respectively. Task 5, operations analysis, results can have an effect on Task 2, the on-orbit servicer design. The control system and interface evaluation of Task 4, functional hard mockups, affect both the on-orbit servicer and the spacecraft interface designs. Task 6 draws on the results of Tasks 2 and 4 as a basis for the servicer mechanism structure design.

The Systems Group of TRW, Inc. was subcontracted for Task 3, the serviceable spacecraft interface preliminary design effort.

The trade studies of Task 1 addressed the important question of system complexity as opposed to system capability. This question is difficult when applied to on-orbit servicing as the level of capability required in terms of module removal directions and mechanism reach are not known at this time, nor can they be known. Our first iteration through this question was based on a detailed analysis of 28 serviceable spacecraft designs from the literature. The alternative servicer configurations were evaluated in three dimensional one-tenth scale mockups. The preliminary considerations led to a trade study of five different servicer configurations that represented five combinations of complexity and capability. Each configuration's capability was optimized for its level of complexity. The NASA was thus able to select the combination of complexity and capability that was most suitable at that stage of on-orbit servicer development. These trade studies involved all six considerations used in the first IOSS, namely: spacecraft design aspects, space transportation system impacts, technical feasibility, operational areas, programmatics, and cost.

The analyses of Task 1 also led to a firm recognition of a number of factors that became the very basis of our approach to servicer system design. One is a realization of the very simple nature of the tasks, or actions, involved in module exchange. They are: remove, flip, relocate, and insert. These four actions are all that are involved in replacing a whole set of modules in a failed spacecraft. There are no other tasks. Another factor is that the module locations both in the spacecraft and in the stowage rack are known well before launch of a servicing mission. There is no need to search for the failed module location. Thus the module locations can be stored in the onboard computer and all of the module exchange trajectories (Figure I-5) can be preprogrammed as well. A third factor is that the working volume for the on-orbit servicer mechanism is a solid of revolution with its axis coincident with the docking axis. The last major point is that the control system design and the mechanism design should be developed together so that they complement each other and so that system operability is enhanced. These factors became the basis of our approach to servicer system design.

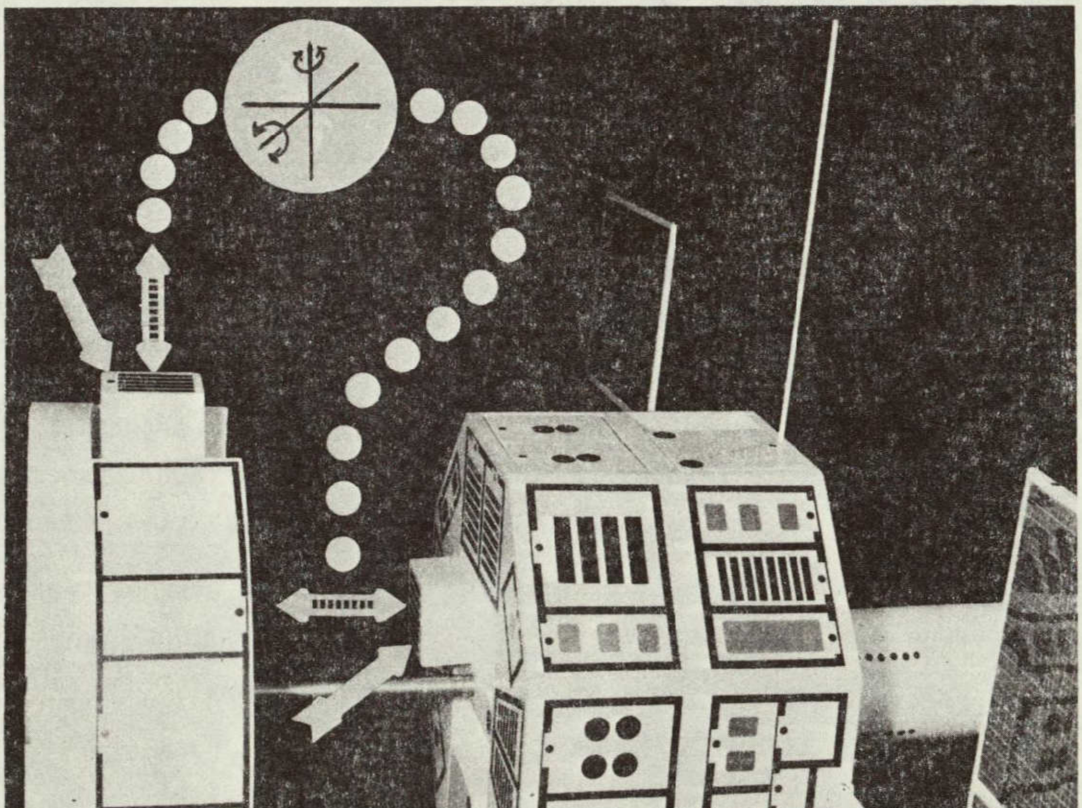

Figure I-5 Preprogrammable Module Trajectory

D. RESULTS AND CONCLUSIONS

The results and conclusions from this study are provided below by chapter. The top level conclusions are presented first under 1. Under each set of results the more significant conclusions are shown in italics.

1. Top Level Conclusions

a) *On-orbit servicing is a feasible and useful method of significantly reducing spacecraft program costs.* (All chapters)

b) *A single on-orbit servicer development can satisfy serviceable spacecraft requirements.* (Chapter II)

c) A continuing servicer system development program is necessary for user acceptance. (Chapters II, IV, and IX)

d) The space transportation system is designed to accept on-orbit servicing and no serious potential impacts were identified. (Chapter IX)

e) Serviceable spacecraft designs by TRW in this study and by COMSAT Labs in the prior study have validated the servicer design and demonstrated the value of an understanding of servicer system capabilities by the spacecraft designer. (Chapters IV and VI)

f) An on-orbit servicer implementation plan involving an early ground demonstration and subsequent flight demonstration has been prepared. (Implementation Plan)

g) Servicer system life cycle costs are small compared to potential cost savings and are greatest when spacecraft and servicer systems are designed with consideration of the other's capabilities. (Chapter IX)

h) A central docking system is more useful to servicing systems than other concepts. (Chapter III)

i) Development of geosynchronous upper stages with a rendezvous and docking capability should be accelerated to maximize potential savings from on-orbit servicing. (Chapters IV and IX)

2. Satellite Servicing Requirements (Chapter II)

a) *A valid set of servicer system requirements has been developed from an analysis of 28 serviceable spacecraft designs.*

b) A one-tier module exchange capability is preferred over a two-tier capability.

c) Both radial and axial module exchange directions are required for some large spacecraft.

d) The servicer mechanism reach need not exceed two tiers of modules per docking.

e) The servicer mechanism working volume should be a solid of revolution.

3. Servicer Mechanism Configuration Selection (Chapter III)

a) *An integrated set of five modular servicer mechanisms was found to span all servicer requirements.*

b) *The axial/near radial servicer mechanism configuration which has the best balance between capability and complexity was selected for preliminary design.*

c) The modular set, with its capability for growth and adaptability, was recommended for long term development.

- Axial;
- Axial/Near Radial;
- Near Radial;
- Two-Tier Radial;
- Axial/Two-Tier Radial.

d) An effective set of criteria was developed for selection of servicer mechanism configurations.

e) A complete range of servicer mechanism configurations was systematically and iteratively reduced to a few for detail consideration.

f) One-tenth scale models were found to be cost-effective during detail selection between servicer configurations.

4. Serviceable Spacecraft Preliminary Design (Chapter IV)

a) *Design of spacecraft for serviceability is straightforward with acceptable weight and cost effects.*

b) *The significant attributes and design characteristics of a serviceable spacecraft which can enhance user acceptance have been identified.*

c) The TRW preliminary layouts of three serviceable spacecraft were iterated with the MMC servicer configurations to easily reach compatible designs for:

- DSCS-II--Defense Satellite Communications System II;
- SEOS--Synchronous Earth Observatory Satellite;
- CLO--Characteristic Large Observatory.

d) The fundamental characteristics of mission equipment repair in orbit have been identified.

e) The CLO replaceable modules are arranged around two separate docking ports.

5. On-Orbit Servicer System (Chapter V)

a) *A servicer system preliminary design has been prepared that exploits the simple nature of the module exchange task.*

b) Functions have been allocated between the man, control system, servicer mechanism, interface mechanism, spacecraft and carrier vehicle.

c) The control system design goal of simplifying servicer system operations was accomplished.

d) The Orbiter Payload Specialists Station is recommended for control of servicer operations in or near the Orbiter.

e) The cylindrical geometry of the servicing task lends itself to simple, easily implemented module exchange trajectories.

6. Servicer Mechanism Preliminary Design (Chapter VI)

a) *A servicer mechanism preliminary design has been prepared which satisfies all the established requirements.*

b) The mechanism can replace modules in both the axial and radial directions on a single mission.

c) All aspects of the mechanism preliminary design are well within today's state of the art.

d) Three standard interface mechanism sizes can handle 90 percent of the anticipated modules (These are applicable to either the side or base mounting interface mechanisms):

- 17 inch cube, less than 75 lb modules;
- 26 inch cube, less than 200 lb modules;
- 40 inch cube, less than 400 lb modules.

e) The spacecraft designer may design his own interface mechanisms as long as they are compatible with the servicer mechanism and module stowage rack. However, two general purpose interface mechanisms have been designed.

f) A truss type module stowage rack is more weight efficient than a monocoque structure.

g) The selected stowage rack configuration can stow sufficient modules for servicing:

- Two DSCS-II spacecraft;
- Two SEOS spacecraft; or
- One CLO spacecraft.

7. Servicer Control System (Chapter VII)

a) *A control system approach and implementation have been developed which involve three modes:*

- *Supervisory as the primary mode;*
- *Manual direct as the backup mode;*
- *Manual augmented to represent conventional teleoperator control.*

b) All three control modes were found to be essential and their development should be continued.

c) Use of the emerging microprocessor technology can permit a dedicated servicer computer featuring low weight, flexibility, capability, and operational simplicity, and will lead to expedited servicer system development.

d) The same basic trajectory sequence can be used for all three control modes and for every module--only the trajectory segment end conditions will vary.

e) The cylindrical coordinate system is most optimum for servicing.

f) Hand controller correlation with the TV screen is highly desirable for the manual augmented mode.

g) A single TV camera mounted on the end effector is effective.

8. Simulation/Demonstration (Chapter VIII)

a) *A simulation/demonstration using existing Martin Marietta motion generator, computer, and control station was conducted.*

b) The selected control system was verified to be useful and effective.

c) The supervisory mode is effective and easy to learn.

d) The manual direct mode is feasible, but visual aids are highly desirable.

e) The capture volume of the interface mechanism should be reassessed.

9. Operations Analysis (Chapter IX)

a) *On-orbit maintenance is the most cost-effective mode for maintenance of geosynchronous spacecraft.*

b) All twelve servicing methods considered were contained within a ± 3 percent cost spread.

c) The Tug-demand launch servicing method resulted in minimum total program cost.

d) The fixed location warehouse with incremental satellite servicing by the SEPS resulted in minimum servicing response time.

e) Use of expendable upper stages is useful but not as cost-effective as reusable systems.

f) The current NASA launch cost reimbursement policy favors expendable servicer systems on geosynchronous missions.

g) *Operations costs are the largest part of servicer system life cycle costs for a range of mission model sizes.*

h) The current NASA launch cost reimbursement policy enhances STS near term objectives, but puts many on-orbit maintenance requirements into the extra-cost category. However, these extra costs are not expected to overcome the advantages of on-orbit maintenance.

i) The current NASA launch cost reimbursement policy results in higher launch costs for all maintenance modes with an increase in savings for the on-orbit maintenance mode.

10. Engineering Test Unit (Volume III)

a) *An engineering test unit of the servicer mechanism structure has been designed, fabricated, and delivered to MSFC.*

b) The engineering test unit can become the basic building block of a NASA on-orbit servicing demonstration facility.

II. SATELLITE SERVICING REQUIREMENTS

This chapter collects and compiles the multitude of assorted requirements that interact with and influence the overall servicing scenario. The satellite servicing requirements are used as a basis for the servicer mechanism configuration of Chapter III, and the servicer system design of chapters V, VI, and VII, as well as supporting the serviceable spacecraft design of Chapter IV. The approach to developing these requirements is depicted in Figure II-1.

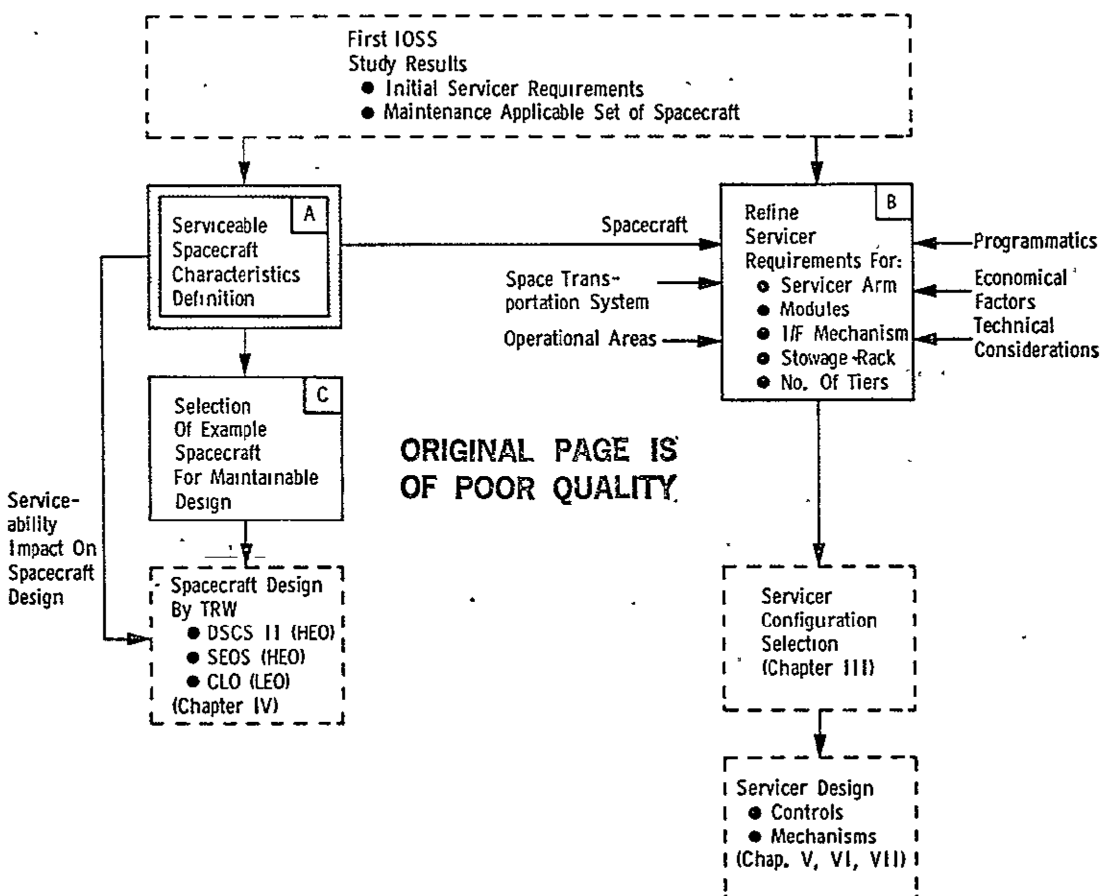

Figure II-1 Servicing System Requirements Definition

There are three requirements-related discussion areas included in this chapter. They are blocks A, B, and C on the figure. These letters also correspond to the section headings in this chapter.

Some preliminary data from the first IOSS study were available as a starting point at the beginning of this study. As noted in the block on the top of Figure II-1, this took the form of a preliminary definition of servicer requirements. Other useful data and analyses results regarding the maintainability of satellites were also available from that prior study.

The real objective of this chapter is to accomplish block B, which is a refinement and expansion of the preliminary servicer requirements. The arrows showing inputs to block B represent consideration areas used in developing the servicer system requirements. The emphasis was to identify servicer mechanism configuration drivers. In order to provide more detail and arrive at the desired credibility, it was found necessary to first perform an in-depth analysis of the most current data on future serviceable spacecraft. The purpose of this analysis, shown as block A on the figure, was to review, then classify and characterize in detail all the potentially maintainable spacecraft identified during the first IOSS literature search. Each of these spacecraft represent different individual's concepts of how spacecraft should be designed for serviceability. This task was a principal effort in the requirements area. It became the cornerstone, not only for the servicer requirements of block B, but also resulted in a set of requirements, or implications if you will, on the design of spacecraft for maintainability. It should be noted these implications were most useful as an independent cross check against maintainable spacecraft design requirements generated on a subcontract with TRW (see Chapter IV)..

The analysis in block A also provided the necessary data for selection of the spacecraft to be designed for maintainability by TRW. That selection process is shown in the figure as block C.

The relationships of the requirements task to the other activities conducted on the contract such as the TRW spacecraft designs, are shown as dotted boxes on the figure for reference.

A. SERVICEABLE SPACECRAFT CHARACTERISTICS

A detailed analysis regarding this subject was conducted and documented in a memorandum, "The Effect of Serviceable Spacecraft Design on Servicer System Requirements," dated May, 1976. This section will summarize that document which reviewed and classified 28 serviceable spacecraft designs from the literature.

The approach used in this analysis to arrive at a credible characterization of maintainable spacecraft focused initially on the Maintenance Applicable Set (MAS) of the previous IOSS study. The MAS was used as a reference in the spacecraft classification task described in 1. below. The cost savings data from the previous IOSS for on-orbit servicing, as opposed to expendable spacecraft or ground refurbishment programs, was then used to determine the relative importance of each category from a cost savings standpoint. This is discussed in 2. To identify the requirements imposed by each category a study of 28 serviceable spacecraft designs was performed and presented in 3. The servicer requirements to be implemented in this Integrated Orbital Servicing Study Follow-On were identified in 4., based on the potential cost savings recoverable through providing the servicing capability. The implications of serviceability on spacecraft design were abstracted and presented in 5.

1. Spacecraft Classification

The maintenance applicable set of the first IOSS was derived from the 1973 NASA payload model and the 1974 Space Shuttle Payload Description (SSPD) The applicability of the MAS to the present study was evaluated with respect to present payload planning as presented in the 1975 SSPD. Of the 47 spacecraft in the MAS, six programs were deleted in the 1975 SSPD. These were all from the astronomy payloads. Six programs were added in the SSPD, however, which offset the deletions. The MAS was determined to be a representative sample group of spacecraft to be flown in the Shuttle era. A summary of those 47 spacecraft in the MAS is provided in Table II-1.

Categories or classifications of spacecraft according to imposed servicer requirements were determined. Classification according to mission objective yielded a first cut, as configuration is often driven by program

Table II-1 Summary of Maintenance Applicable Set

Payload Number	Payload Model Code No.	Spacecraft Name
AS-01-A	AST-6	Large Space Telescope
AS-03-A	AST-1B	Cosmic Background Explorer
AS-05-A	AST-1C	Advanced Radio Astronomy Explorer
AS-07-A	AST-N1	3M Ambient Temperature IR Telescope
AS-11-A	AST-N2	1.5M IR Telescope
AS-13-A	AST-N3	UV Survey Telescope
AS-14-A	AST-N4	1M UV Optical Telescope
AS-16-A	AST-8	Large Radio Observatory Array
AS-17-A	AST-N5	30M IR Interferometer
HE-01-A	AST-9B	Large X-Ray Telescope Facility
HE-03-A	AST-5A	Extended X-Ray Survey
HE-05-A	AST-5D	High Latitude Cosmic Ray Survey
HE-07-A	PHY-1A	Small High Energy Satellite
HE-08-A	AST-5B	Large High Energy Observatory A
HE-09-A	AST-4	Large High Energy Observatory B
HE-10-A	AST-5C	Large High Energy Observatory C
HE-11-A	AST-9A	Large High Energy Observatory D
HE-12-A	PHY-5	Cosmic Ray Laboratory
SO-02-A	AST-7	Large Solar Observatory
SO-03-A	AST-3	Solar Maximum Mission
AP-01-A	PHY-1B	Upper Atmosphere Explorer
AP-02-A	PHY-1C	Explorer-Medium Altitude
AP-04-A	PHY-2A	Gravitational and Relativity Satellite - LEO
AP-05-A	PHY-3A	Environmental Perturbation Satellite - A
AP-07-A	PHY-3B	Environmental Perturbation Satellite - B
EO-08-A	EO-3	Earth Observatory Satellite
EO-09-A	EO-4	Synchronous Earth Observatory Satellite
EO-10-A	EO-5	Applications Explorer (Special Purpose Satellite)
EO-12-A	EO-6	TIROS
EO-56-A	NN/D-8	Environmental Monitoring Satellite
EO-57-A	NN/D-9	Foreign Synchronous Meteorological Satellite
EO-58-A	NN/D-10	Geosynchronous Operational Meteorological Satellite
EO-59-A	NN/D-12	Geosynchronous Earth Resources Satellite
EO-61-A	NN/D-11	Earth Resources Survey Operational Satellite
EO-62-A	NN/D-13	Foreign Synchronous Earth Observation Satellite
OP-02-A	EOP-5	Gravity Gradiometer
OP-04-A	EOP-7	GRAVSAT
OP-05-A	EOP-8	Vector Magnetometer Satellite
OP-51-A	NN/D-14	Global Earth and Ocean Monitoring System
LS-02-A	LS-1	Biomedical Experiment Scientific Satellite
CN-51-A	NN/D-1	INTELSAT
CN-52-A	NN/D-2A*	DOMSAT A
CN-53-A	NN/D-2B	DOMSAT B
CN-54-A	NN/D-3	Disaster Warning Satellite
CN-55-A	NN/D-4	Traffic Management Satellite
CN-56-A	NN/D-5A	Foreign Communication Satellite - A
CN-58-A	NN/D-2C	DOMSAT C
CN-59-A	NN/D-6	Communications R&D Prototype
CN-60-A	NN/D-5B*	Foreign Communication Satellite - B

* Dropped from maintenance applicable set in economic analysis.

objectives. The final classification is summarized in Table II-2. It is based on configuration and mission orbit or servicer carrier vehicle. The orbiter is used for low earth orbit and an upper stage or tug for high earth orbit. The criteria used was whether the subsystem and mission equipment could be packaged into one tier or required more than one tier. A tier was defined as a cylinder of 4.57 m diameter and approximately 1 m deep. This is compatible with the orbiter cargo bay diameter. The classification according to configuration was complemented by whether the spacecraft was delivered to orbit by the Shuttle or some upper stage.

Table II-2 Spacecraft Classification Categories

	CATEGORY			
	Low Earth Orbit		Medium or High Earth Orbit	
	One Tier	Two or More Tiers	One Tier	Two or More Tiers
Number of MAS Programs	11	15	18	3

It is apparent that the MAS contains representative spacecraft for each category. The small number of High Earth Orbit, two tier configurations is to be expected due to weight constraints on delivery to high earth orbit.

2. Cost Savings Per Category

Having classified the maintenance applicable set according to the configuration categories above, the cost savings recoverable through on-orbit maintenance as opposed to expendable or ground refurbishment programs were calculated for each category. The data was taken from the extensive cost analysis performed during the first IOSS. The results of this analysis is shown in Figure II-2.

Approximately 50% of the savings are in the LEO, two tier spacecraft category. This result places an emphasis on servicer capability to service more than one tier spacecraft. This ratio is consistent within the sample group. Of the ten programs with the potential for the highest cost savings, six are two tier or greater. These ten programs make up 50% of the cost

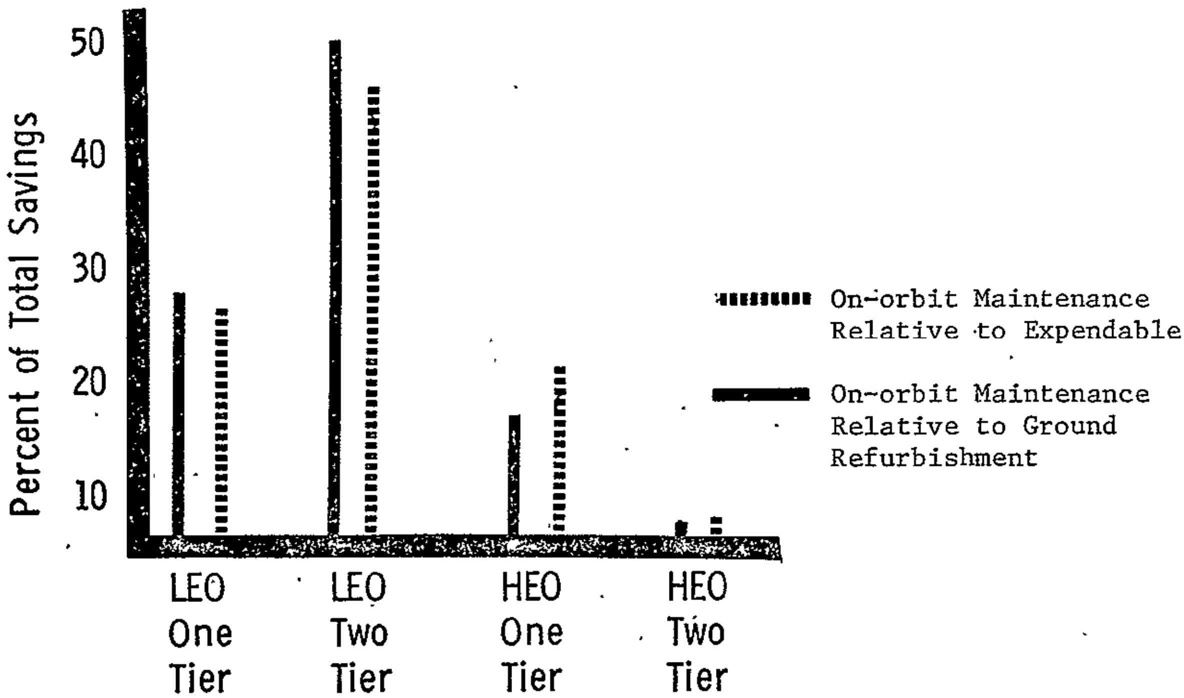

Figure II-2 MAS Cost Savings Per Category

savings. It may be noted that programs representing 74% of the potential cost savings were scheduled for initial launch prior to 1984.

Highest cost savings programs:

- EO-08-A	Earth Observatory Satellite	LEO - One
- AS-01-A	Space Telescope	LEO - Two
- EO-61-A	Earth Resources Survey Operational Satellite	HEO - One
- AS-07-A	3m Ambient Temperature IR Telescope	LEO - Two
- AS-11-A	1.5m IR Telescope	LEO - Two
- LS-02-A	Biomedical Experiments Science Satellite	LEO - One
- SO-02-A	Large Solar Observatory	LEO - Two
- HE-12-A	Cosmic Ray Laboratory	LEO - Two
- AS-14-A	1m UV Optical Telescope	LEO - Two
- HE-07-A	Small High Energy Satellite	LEO - One

3. Serviceable Spacecraft Design Evaluation

An extensive review and evaluation of the serviceable spacecraft designs found in the literature was conducted. The variety (28) of serviceable spacecraft designs considered provides trend data over a wide range of spacecraft configuration variables. The serviceable spacecraft designs were obtained from the extensive literature file collected during the first IOSS. These were complemented with a few additional concepts from the more recent literature. The primary emphasis of this review was on identifying parameters that would aid in selecting the best servicer mechanism. As part of identifying servicer configuration drivers, it was necessary to find a way of determining the relative importance of conflicting information. One of the techniques used was to identify the most significant spacecraft program class in terms of potential savings and then use it to help resolve the differences. The method of collecting and analyzing the serviceable spacecraft descriptive data provided important information on serviceable spacecraft design approaches. This information was also collated, analyzed, evaluated, and made available to TRW for use in their serviceable spacecraft design work.

The approach used was to develop a classification scheme based on the first IOSS maintenance applicable set and then to identify the important class on the basis of potential savings. The low earth orbit, more than one tier spacecraft programs represent the greatest savings. The serviceable spacecraft designs were then separated into the classifications. Extensive descriptions of each of the 28 serviceable spacecraft were prepared. Summaries of characteristics, at the spacecraft and module levels, were made both at the class level and then across classes. Conclusions and implications on serviceable spacecraft design were drawn and servicer system requirements were selected. The results were cross checked against the SSPD spacecraft definitions, but no deficiencies were uncovered. Table II-3 presents the serviceable spacecraft and the company that performed each study in the four categories introduced earlier. Several of the spacecraft (DSP, EOS, SEOS, and AGOES) were studied by more than one company

Table II-3 Serviceable Spacecraft Designs

- HIGH EARTH ORBIT - ONE TIER SPACECRAFT (11)
  - DSCS-II, DSP - TRW
  - COMR&DSAT, CSCSAT - MDAC/Fairchild
  - SEOS, AGOES - MDAC/GE
  - SEOS - GE
  - UOP - RI
  - INTELSAT - COMSAT
  - DOMSAT B - MSFC
  - DSP - Aerospace
- HIGH EARTH ORBIT - TWO TIER SPACECRAFT (8)
  - Geosynchronous Platforms, 7 Versions - RI
  - AGOES - GE
- LOW EARTH ORBIT - ONE TIER SPACECRAFT (2)
  - EOS - Aerospace
  - EOS - Lockheed
- LOW EARTH ORBIT - TWO TIER SPACECRAFT (7)
  - MMS EOS - GSFC
  - ST - MMC
  - HEAO 1.2m TELESCOPE - MSFC
  - Solar Astronomy Platform, Stellar Astronomy Platform, HE Phy. Platform, Plasma Phy. Platform - RI

To better illustrate the design type of spacecraft in these four categories a typical design has been selected from each of the four and is described in more detail.

a) Serviceable High Earth Orbit - One Tier Design - The TRW serviceable design of the DSCS II is presented in Figure II-3 as an example of a high-earth-orbit, one tier design. It is a web structure, of box shape 128 in. x 99 in. x 40 in. It is designed for central docking and axial removal of replaceable modules. Three versions of this spacecraft were studied by TRW. The versions vary in number and size of modules. The configuration shown has the least number of modules (eight). Each module contains an entire subsystem and is approximately 32 in. x 40 in. x 40 in. The heaviest module weighs 444 lbs. The maximum number of modules considered was 30. In that configuration, subsystems were broken down into functions.

This spacecraft requires minimum complexity in servicer design. There is no axial reach requirement and the maximum radial reach is 78 in.

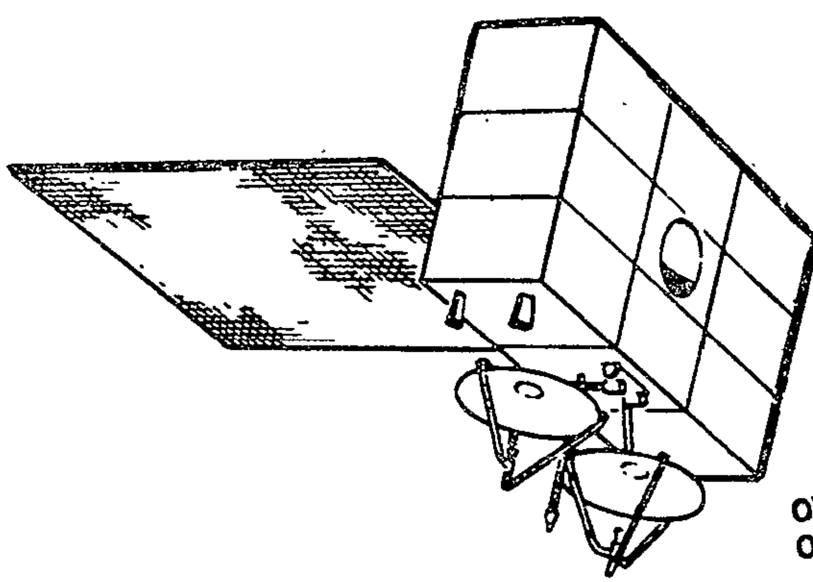

Figure II-3 Serviceable High Earth Orbit - One Tier Design (DSCS-II)

b) Serviceable High Earth Orbit - Two Tier Design - The Geosynchronous Earth Observation Platform is presented in Figure II-4 as representative of the HEO, two or more tier category. This Rockwell International configuration has separated the mission modules into two separate tiers. One tier is at the focal plane of the telescope. The other tier is at the entrance aperture of the telescope. The subsystem modules are located in a separate tier also at the entrance aperture. The overall size of the spacecraft is 144 in. diameter x 309 in. long. In the Rockwell studies, servicing was considered by an automated servicer as well as by crewmen under both EVA suited and shirtsleeve conditions. The latter drove the design to the inward radial removal configuration. To maintain the central region clear the docking must be peripheral. The basic structure for mounting the modules or space replaceable units (SRUs) is web type. There are locations for 36 replaceable units which range in weight from 13 to 200 lbs. The maximum servicer reach is 42 in. radial and 54 in. axial.

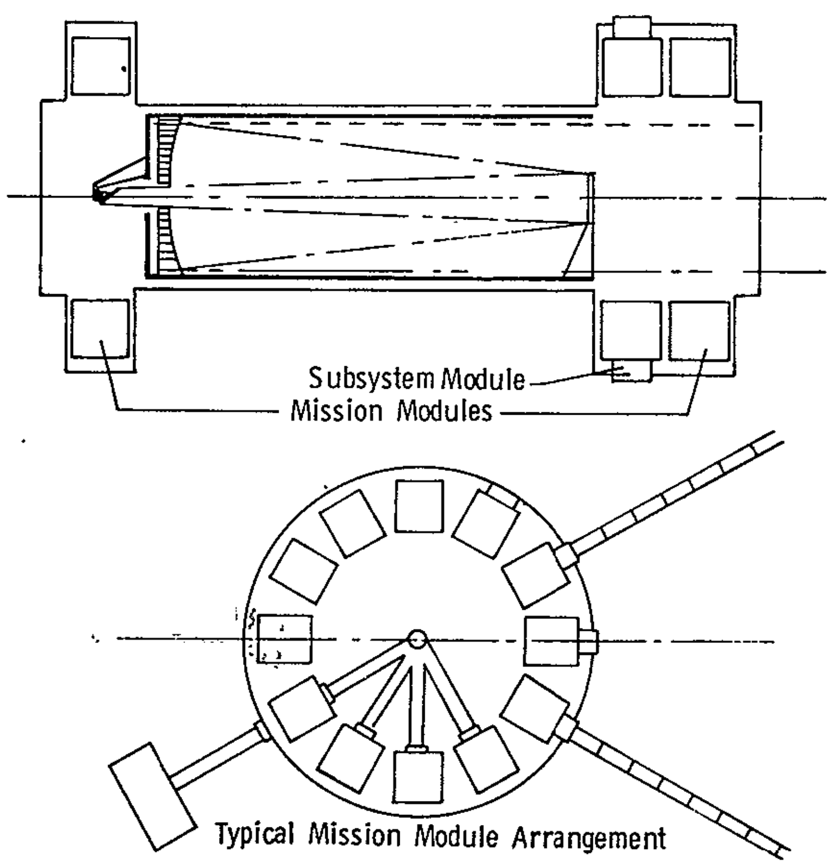

Figure II-4 Serviceable High Earth Orbit - Two Tier Design (SEOS)

This class of spacecraft represents the more complex servicer requirements. The potential for redesign exists if servicing by shirtsleeve crewmen is eliminated. Module removal could then be axial and radial outward. Locating all three tiers at the base of the telescope would eliminate multiple docking, peripheral docking and possibly save on spacecraft design in structure and cable run lengths.

c) Serviceable Low Earth Orbit - One Tier Design - The Earth Observation Satellite serviceable design by the Aerospace Corporation is presented as a representative of the LEO one tier spacecraft. The configuration, shown in Figure II-5, is also representative of 26 additional spacecraft configured by the Aerospace Corporation in their Operations Analysis Study. The spacecraft is designed consistent with the one tier definition and is compatible with the Orbiter cargo bay. Its overall dimensions are 180 in. diameter and 60 in. deep. The configuration shown here has equipment mounted to individual pallets which are removed axially. Only one face of the cylinder is used for mounting here. More recent data from Aerospace shows modules mounted to the interior face of the cylinder as well.

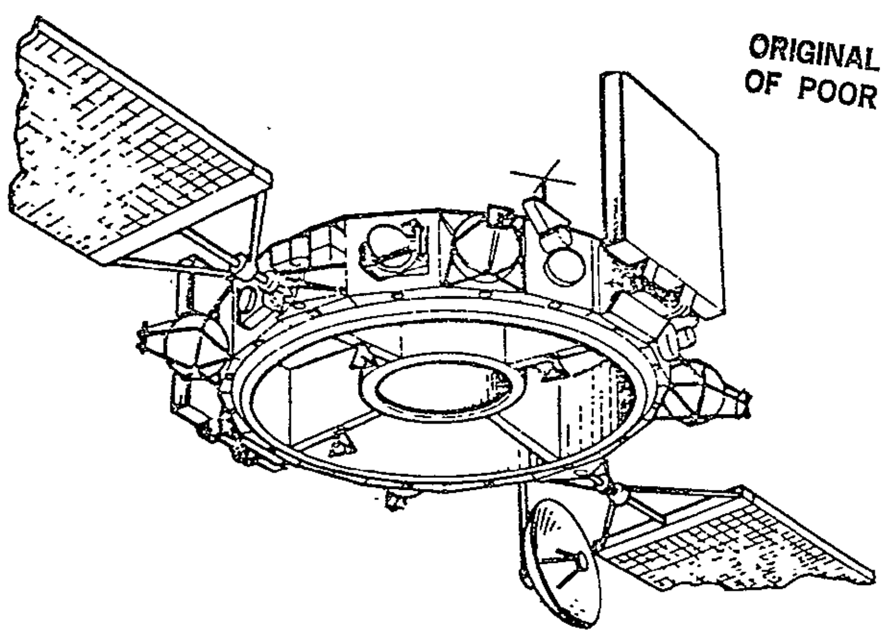

Figure II-5 Serviceable Low Earth Orbit - One Tier Design (EOS)

This category of spacecraft does not generate any critical requirements which drive the servicer design. Docking is performed axially with a central type docking. Only one docking is required. All modules are removed axially requiring only radial servicer reaches. The majority of the modules are less than a 40 in. cube in size; however, mission modules may reach 40 in. x 40 in. x 80 in. The maximum module weight is 719 lbs.

d) <u>Serviceable Low Earth Orbit - Two Tier Design</u> - It was desired to include a representative of the GSFC work on serviceable spacecraft. The recent documentation on the Multi-Mission Spacecraft does not discuss servicing, nor does it include any set of mission equipment. It was decided to use the older GSFC information on the Earth Observatory Satellite (EOS) that did provide the servicing and mission equipment data required. It is shown in Figure II-6.

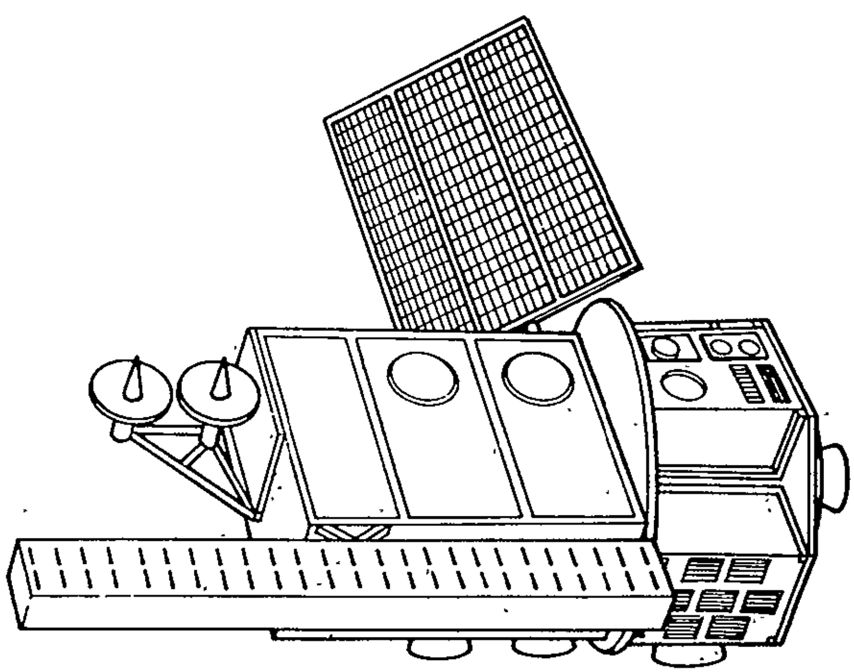

Figure II-6 Serviceable Low Earth Orbit - Two Tier Design (EOS)

The EOS was designed for launch on Shuttle, Delta or Titan launch vehicles. This limits the design in the radial dimension and forces the two tier length with the tiers essentially perpendicular to each other. In order to service this spacecraft with the single, axial dock, excessive reach requirements of up to 230 in. axial would be required. Therefore, two dockings should be considered, an axial one for the two tiers of subsystem equipment and a radial one for mission equipment. Subsystem module removal is radial and mission equipment module removal is axial. All modules exceed the 40 in. cube module size. The subsystem modules are 48 in. x 48 in. x 20 in. Each module contains a complete subsystem or mission equipment system.

4. Space Serviceability Impact on Servicer Design

The results from 3. above were carefully tabulated in the previously referenced memo. The conclusions from the analysis and resulting data are reflected in servicer system requirements as follows:

a) All Replaceable Components Can Be Modularized - No evidence was found in any of the serviceable spacecraft descriptions that there was any servicing activity required which could not be put in the form of module exchange. A possible exception is the extension/retraction of appendages by an astronaut on EVA as discussed for the space telescope. Note that the mechanical drive system of the servicer end effector (normally used for interface mechanism operation) could be used to power extension/retraction systems in an emergency.

Similarly, all equipment that appeared to need replacement was modularized by one design team or another. In some cases, the resulting modules were large and heavy, but the replaceable components were modularized. This was true for both subsystem equipment and mission equipment.

b) Maximum of Two Tiers per Docking - Twenty-one of the twenty-eight serviceable designs reviewed incorporated one or two tiers per docking. Five were three tier designs. The other two spacecraft were the AGOES GE design (5 tiers) and the RI High Energy Physics Platform. The AGOES was redesigned to a one tier spacecraft in a MDAC/GE design. The RI spacecraft could be serviced using two dockings. In general, two tiers per docking was the maximum servicing requirement.

c) <u>Module Size - 15 in. Cube Minimum and 40 in. Cube Maximum</u> - The majority of modules, both subsystem and mission equipment, used in the serviceable spacecraft evaluated are within the size bracketed by a 15 inch cube and a 40 inch cube. Approximately half of the spacecraft used some modules smaller than a 15 inch cube and approximately 30% used some modules larger than a 40 inch cube.

The optimization of module size is an extensive trade study which must consider failure rates, the number of spacecraft which can be serviced on a flight, the weight and structural effects, and module interfaces among other parameters. In the spacecraft reviewed, the module philosophies generally ranged from packaging complete subsystems in a module to packaging individual functions in a module. In their DSCS-II study, TRW evaluated three separate configurations which varied mainly in the size and number of modules. They tended to favor the larger or complete subsystem modules.

Several of the programs evaluated used both packaging philosophies. In the Space Telescope design, subsystem equipment was packaged according to the functional philosophy resulting in many small replaceable modules. The mission equipment was packaged according to the complete subsystem philosophy. This resulted in a few large modules.

The modules smaller than a 15 inch cube can be repackaged into larger functionally complete modules. The modules larger than a 40 inch cube generally are mission equipment modules such as the Thematic Mapper used on EOS satellites and the scientific instruments on Space Telescope. Repackaging instruments into smaller modules requires additional study. In the DSP Satellite, however, a large instrument was redesigned so that the detector set became a replaceable module as opposed to replacing the complete instrument.

d) Module Weight - 10 to 700 lbs. - Most of the modules fell within the weight bounds of 10 to 700 lbs. Three spacecraft with modules weighing less than 10 lbs. were identified. They were Space Telescope, Plasma Physics Platform, and Unmanned Orbital Platform. In all cases, the items could be grouped with other components to result in larger modules. There are nine spacecraft with modules in the 11 to 20 lb. increment. Thus the lower bound of 10 lbs. is appropriate.

Only two spacecraft programs were identified with modules heavier than 700 lbs. These are the Aerospace Corporation design of EOS and the Space Telescope. The serviceable spacecraft data indicated an upper bound of 500 lbs. would be appropriate as there were no maximum weight modules in the 500 to 700 lb. interval. However, the SSPD data has a number of modules falling between 500 and 700 lbs. Thus the 700 lb. value is used.

e) Servicer Mechanism Reach From Docking Port - It is recommended that the minimum reach be 0 in., axial and 20 in. radial and that the maximum reach be 100 in. axial and 90 in. radial. The maximum radial reach is bounded by the radius of a tier which equals half of the Orbiter cargo bay diameter. The minimum radial reach occurs on the end of the spacecraft near the docking drogue. Some spacecraft designs, using peripheral docking, have a module directly on the spacecraft centerline. With central docking, modules cannot interfere with the docking drogue. Additionally, the reach is to the interface mechanism attach point which can be on the outboard side of the module. The data collected is for the innermost part of the module. It is recommended that a minimum radius of 20 in. be used.

A maximum axial reach of 100 inches also fulfills the reach requirements of most spacecraft. The two spacecraft with greater than 200 inch reach requirements are the GSFC MMS/EOS and the Space Telescope. These reach requirements are based on single axial dockings as the spacecraft are designed. The MMS is designed for maintenance using a Shuttle Orbiter fixture. The ST is designed for servicing by a suited crewman. Both excessively long reach requirements could be decreased by multiple docking and redesign of the spacecraft with automated servicing in mind. The 100 in. reach permits reaching slightly past two tiers to solar arrays, antenna drives, or specialized mission equipment.

f) Both Axial and Radial Outward Module Removal are Required on the Same Spacecraft - The axial module removal direction is preferred by the high earth orbit, one tier designs. Radial inward removal is indicated on eight designs, but these are all RI geosynchronous platform variants. As was shown by the UOP Alternative Configuration, inward radial can be readily replaced by outward radial when the need for shirtsleeve operation is deleted. Both axial and radial module removal are shown for five spacecraft. This dual capability is also a significant encouragement to spacecraft designers. If the dual capability is not retained as a requirement, then the individual needs for axial and radial could result in different concepts for development and higher total life cycle costs.

g) Provide For Off-Axis Radial Module Removal - Four serviceable spacecraft designs involved off-axis radial module removal. By itself, this data might not be adequate justification. However, this is another area where it is desired to minimize restrictions on the spacecraft designer. The additional complexity in the servicer mechanism appears minimal and the possible gain to the spacecraft designer could be significant.

5. Implications of Serviceability on Spacecraft Design

While the analysis described in 1., 2., and 3. was directed towards deriving the servicer requirements of 4. above, another very pertinent and useful output was obtained regarding the design of spacecraft themselves for serviceability. Some of the more significant conclusions derived from the data are provided below. They were provided to TRW to support their typical serviceable spacecraft design (see Chapter IV). These requirements have also been useful as an independent check on the designs performed by TRW.

a) Use Central Docking System - Both peripheral and central docking were used by the spacecraft evaluated. Where peripheral docking was used it was normally due to having presupposed a peripheral type servicer or to accommodate shirtsleeve crew servicing and the associated radial inward removal. The majority of the designs could easily accommodate central docking. The use of a peripheral docking system establishes a "fence" between the areas for axial module removal and the areas for radial module removal. The "fence" seriously inhibits the design of a servicer configuration capable of both axial and radial module removal.

b) Minimize the Number of Dockings Per Spacecraft Service - Multiple docking was only required for four of the twenty-eight spacecraft. A maximum of two tiers per docking was required. Multiple docking should be permitted, but at the same time it should be minimized.

c) Docking Direction Should be Normal to the Solar Array Drive Axis Direction - In almost all cases, the docking direction was normal to the solar array drive axis and component placement was such that there was minimum likelihood of interference. One spacecraft design used an unbalanced solar array configuration to avoid interference between the solar array and the docking operation.

d) Solar Arrays and Other Appendages Need Not Be Retractable - Slightly more than half of the designs did not provide for retracting appendages. The proportion of non-retracting designs was higher for the low earth orbit spacecraft. This is the opposite of what might be expected. Extended appendages could cause damage to the Orbiter wings or tail. The possible interferences are less for Tug based servicing. However, the main rationale for an appendage retraction capability is to be able to return the spacecraft to earth. This capability can also be obtained by incorporating a method of severing the appendages in an emergency.

e) Consider Use of Replaceable Solar Array and Antenna Drives - Most solar arrays were driven to face the sun. Only four cases of replaceable drives were identified. Serviceable designs of the solar array drives were investigated by the Aerospace Corporation. They planned to replace the drives as a module while leaving the solar array and the axis support bearings in place.

More fixed than driveable antennas were reported. This probably reflects the number of communications satellites in the group of satellites evaluated. Only a few cases of replaceable antenna drives were noted. This could be because the driveable antennas were often located at the ends of deployable booms which made access difficult.

It is recommended that additional consideration be given to techniques for replacing solar array and antenna drives. As noted above, Aerospace Corporation as well as TRW have looked at the replaceability of solar array drives. Also, TRW has considered placing antenna drives at the base of the extension booms instead of out on the booms. This could make the replacement of antenna drives more feasible.

f) <u>Use Most of the Orbiter Cargo Bay Diameter</u> - All but two of the spacecraft had diameters, or diagonal measurements, greater than twelve feet. These were the GSFC Multi-Mission Spacecraft which is designed for use with expendable launch vehicles and the other was an early design of AGOES by General Electric. However, many of the designs had dimensions close to twelve feet and others needed most of the 15 foot diameter, but did not use the volume effectively.

g) <u>Dominant Structural Type is Web</u> - Two types of structure were found. Plate structure was the name given to large flat sections made up of honeycomb panels. These tended to be four to six inches thick and were the primary structure for mounting the modules. Web structure was the name given to the egg-crate or pigeon hole arrangement. The "holes" were generally outlined by some form of beams and then thin plates were used to connect and stiffen the beams in the planes of the thin plates. This name was also assigned to the open trusswork type of construction.

Plate construction was found on four spacecraft. On two of these, the larger part of the structure was web.

h) <u>Select Spacecraft Shape to Suit Other Design Requirements</u> - The structural shapes of serviceable spacecraft identified were cylinder, toroid, disk, and rectangular box. Combinations were also found. Approximately one-third of the spacecraft were toroids, one-third were rectangular boxes, and the remaining third were cylinders, disks, or combinations. The spacecraft shape does not have much effect on servicer requirements other than reach distance, which is discussed above.

j) Use Between Ten and Thirty Modules - The number of modules includes both subsystem and mission equipment modules. The fewest modules on a spacecraft were eight and this occurred for one version of the DSCS-II by TRW, the DSP by Aerospace Corporation and the SEOS by General Electric. Forty-two modules were used on the UOP by Rockwell International and 90 were used on the ST by Martin Marietta. Three spacecraft used fewer than ten modules and seven used more than thirty modules. The ratio between average number of modules for one tier and more than one tier spacecraft was not as large as anticipated. The averages being 21 and 28 modules for one and more than one tier respectively. The data did show a larger difference between low and high earth orbit spacecraft where the average number of modules were 33 and 20 respectively. The LEO, more than one tier numbers tend to be distorted by the Space Telescope data.

k) Module Size - 15 in. Cube to 40 in. Cube - Refer to servicer requirements section A.4.c) for rationale.

l) Module Weight - 10 to 700 lb. - Refer to servicer requirements section A.4.d) for rationale.

m) Servicer Reach 20 in. Minimum to 100 in. Maximum - Refer to servicer requirements section A.4.e) for rationale.

n) Electrical, Waveguide, and Fluid Connectors Are Acceptable - All modules required electrical connectors. Twenty-four of the spacecraft required waveguides, or at least high frequency, connectors. Only three spacecraft required fluid connectors. However, this may have been a desire to avoid the development problems of a fluid connector. Fluid connectors should be permitted. They can provide useful functions in terms of cross-strapping propellant supplies and permitting replacement of instrument fluids.

o) Avoid Conductive Type Thermal Connectors - Thermal connectors were used on 14 spacecraft designs. These were predominantly the Rockwell International Geosynchronous Platform spacecraft. They proposed using a crushable packet of material to obtain good thermal contact. However, the problems associated with obtaining good thermal conductance in a separable joint mitigate against its use. Radiation type thermal connectors could be useful.

B. SERVICER REQUIREMENTS

The requirements for the serving system are derived from the following considerations:

- serviceable spacecraft;
- space transportation system
- operational areas
- programmatics
- economic considerations
- technical considerations

These considerations were also used in each of the trade studies of the first IOSS and thus help provide continuity between this analysis and the prior work.

The following discussion will elaborate on each in the order shown.

1. <u>Servicer Requirements - Serviceable Spacecraft</u>

This is by far the major contributor to servicer system requirements. The serviceable spacecraft analysis summarized in the previous section, A., was initiated primarily to derive servicer requirements. The resulting requirements were provided in Section A.4. For completeness, they are repeated here in Figure II-7 in summary form.

MODULE REMOVAL DIRECTION

- Axial and Radial
- Off-Axis Radial

MODULE END EFFECTOR ATTACH POINT LOCATION

- Anywhere on outer surface of a two-tier spacecraft
- 100 in. axial; 180 in. diameter.
- Anywhere on end surface outside 20 in. of center

MODULE SIZE

- Maximum, 40 in. x 40 in. x 40 in.
- Minimum, 15 in. x 15 in. x 15 in.

MODULE WEIGHT RANGE

- 10 to 700 lbs

Figure II-7 Servicer Requirements - Spacecraft Defined

The combined capacity for axial, radial, and off-axis radial module removal on any spacecraft with a single docking will provide the spacecraft designer with maximum design freedom. Off-axis module removal occurs when a spacecraft is configured with an array of modules in a plane not perpendicular to the docking axis. Some of these modules could be removed in a plane containing the docking axis (radial plane), but others cannot. The others would be removed in planes parallel to the radial plane, thus the off-axis radial designation. The end effector attach point locations have been selected to cover one end of a spacecraft (less the docking probe area) plus enough of the spacecraft outside surface to cover two tiers plus a small margin for servicing solar array or antenna drives.

The module sizes and weights which were arrived at in the first IOSS were verified by the serviceable spacecraft analysis. The vast majority of module sizes lie within these limits or can be reconfigured to be within these limits.

The above requirements are primarily module related. Another area not discussed thus far is the interface mechanism. The interface mechanism provides 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 which is fastened to the module and a baseplate receptacle which is fastened to the spacecraft or to the stowage rack. The baseplate receptacle is passive. The baseplate has the linkages, cams, and rollers which latch the baseplate into the receptacle. The baseplate mechanism is mechanically driven from the servicer end effector. The interfaces of the interface mechanism are thus with the modules, the servicer end effector, the spacecraft, and the stowage rack.

The first IOSS study conducted an interface mechanism analysis. It was reviewed, along with the serviceable spacecraft configurations analysed in A of this chapter, to identify a logic for selecting a single interface mechanism as a standard. The data did not lead to such a logic, rather it indicated that a variety of interface mechanisms are possible and could be useful.

The first IOSS resulted in the design and fabrication of two interface mechanisms -- one for bottom mounting, and one for side mounting. These plus some of the other possibilities are shown in Figure II-8. Various non-redundant attachment configurations can be used with each of the location alternatives.

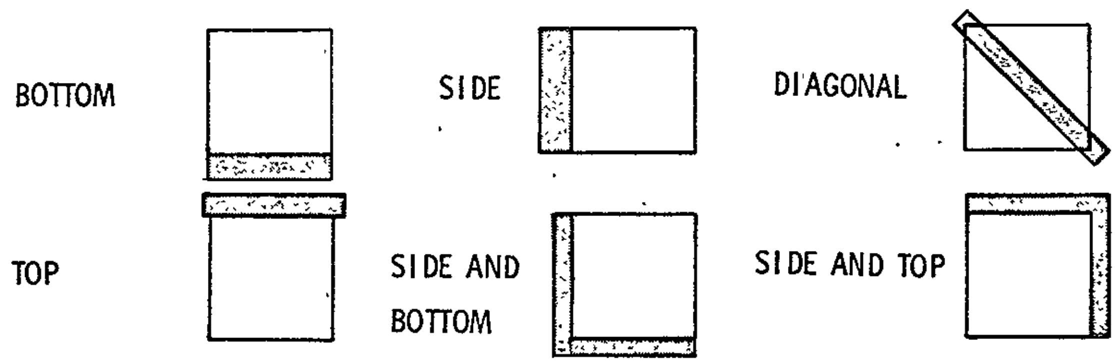

Figure II-8 Alternative Interface Mechanism Locations

The disadvantage of multiple interface mechanism alternatives is, of course, probable higher cost. The servicer mechanism end effector and its mechanical drive system should, ideally, be standardized across all interface mechanisms. Similarly, the method for attaching the interface mechanism baseplate receptacle alternatives into the stowage rack should also be standardized. In this way, a single, or few, stowage rack designs could be used for all missions.

The first IOSS suggested the development of an interface mechanism as a two-part kit in perhaps three sizes. These standard interface mechanisms could be made available to spacecraft designers. Each designer could then make his choice within his own set of design and economic constraints. The initial guidelines and conclusions from the first IOSS are, in general, still valid for this report.

One other component of the servicer system merits some discussion at this time. That is the stowage rack design. Its requirements are not derived directly from spacecraft design but are instead derived from the servicer arm, from the module sizes and weight and from the interface mechanisms. In other words, it is the last element to be defined. Based on the requirement of the primary elements the following stowage rack requirements in Figure II-9 have been derived.

DESIGN TO BE COMPATIBLE WITH SERVICER MECHANISM

- Module replacement direction
- Alternative forms of modularized servicer mechanism
- Provide for servicer mechanism stowage
- Provide for docking probe stowage

PROVIDE 40-in. AXIAL LENGTH TO ACCOMMODATE MODULES

USE MAXIMUM ALLOWABLE DIAMETER = 14.7 ft

PROVIDE ONE TIER OF MODULE STOWAGE

- Average volume margin = 6.5

SELECT STRUCTURAL TYPE FOR MINIMUM WEIGHT

DESIGN TO BE COMPATIBLE WITH VARIETY OF INTERFACE MECHANISM CONCEPTS

IT MAY BE DESIRABLE TO HAVE MORE THAN ONE STOWAGE RACK DESIGN TO MINIMIZE TUG MISSION WEIGHT

Figure II-9 Servicer Requirements - Stowage Rack Configuration

2. Servicer Requirements - Space Transportation System

The Space Transportation System places requirements on the servicer system mainly in the areas of compatibility with and safety of the Orbiter and Upper Stage as illustrated in the summary in Figure II-10. The Upper

Stage considered can be any version that has a capability for rendezvous and docking with a free-flying spacecraft. If on-orbit servicing becomes a significant docking mission for an upper stage, then the servicing needs should be strongly considered. All the servicer mechanism configurations work well with a central docking system and nearly all have complications with a peripheral docking system. If all the geosynchronous servicing missions were to single-tier, axial module removal spacecraft, then peripheral docking could be accommodated. For Orbiter operations, it is assumed that the SRMS is used to place the spacecraft on a central docking probe which extends from the stowage rack.

MUST BE COMPATIBLE WITH ORBITER AND UPPER STAGE

USE EXISTING DOCKING APPROACHES

- Orbiter - Shuttle Remote Manipulator System
- Upper Stage - Servicing is a significant requirement

DO NOT REQUIRE SPACECRAFT APPENDAGE RETRACTION BEFORE DOCKING

- Interference with Orbiter wings and tail
- Interference with SRMS
- Implies that spacecraft should not be rotated during servicing

MUST BE ABLE TO SEPARATE FROM SPACECRAFT IF SERVICING FAILS

MUST BE ABLE TO CLOSE CARGO BAY DOORS

SRMS IS AVAILABLE FOR REPOSITIONING EQUIPMENT BEFORE AND AFTER SERVICING SPACECRAFT

VALUE OF OPERATING WITHIN CARGO BAY ENVELOPE IS NOT KNOWN

Figure II-10 Servicer Requirements - Space Transportation System

The shuttle remote manipulator system will also be used to position the servicer, stowage rack, and adapters in the best locations for servicing. It may also be used for replacement of outsize mission equipment.

3. Servicer Requirements - Operational Areas

The first IOSS concluded that a single servicer development would satisfy both the Low Earth Orbit (LEO) based and High Earth Orbit (HEO) based requirements. This approach still appears valid. While the Orbiter volume capability is large enough for multiple servicing, the number of spacecraft in very similar orbits that will need servicing at the same time will probably be low. Thus multiple servicing on the same Orbiter mission is not likely. The possible need for servicing different classes of spacecraft on a single HEO mission should be evaluated if the different spacecraft classes require more complex servicer configurations than would be required for servicing individual spacecraft on a mission. Servicing multiple spacecraft of a single class on one mission seems straight forward as regards servicer mechanism configurations. Key requirements are summarized in Figure II-11.

SINGLE APPROACH FOR LEO AND HEO BASED SERVICING

LEO SPACECRAFT TEND TO BE LARGER

MULTIPLE SPACECRAFT SERVICING IN LEO IS POSSIBLE BUT THERE ARE SERIOUS PROPULSIVE CONSTRAINTS

HEO SPACECRAFT GENERALLY COULD BE MADE ONE TIER AXIAL

- Could use simpler servicer configuration
- Implies lower cost to HEO spacecraft programs

MULTIPLE SPACECRAFT SERVICING IN HEO IS DESIRABLE

- Same spacecraft class
- Different spacecraft classes

Figure II-11 Servicer Requirements - Operational Areas

4. Servicer Requirements - Programmatic Factors

The programmatic related requirements are summarized in Figure II-12. The first three points on the figure show an undesirable feature in that the more complex LEO missions will probably occur before the simpler (from a servicing mechanism point of view) HEO missions. It was preferred to develop and verify the simpler servicer first and, in fact, was the selected approach in the configuration selection of Chapter III.

MAJOR PART OF POTENTIAL SAVINGS ARE FOR LEO SPACECRAFT
EARLY MISSIONS WILL BE LEO
IMPLIES FIRST SERVICER USE COULD BE MOST COMPLEX FORM
SIMPLER FORMS OF SERVICER CAN LEAD TO AN EARLIER ACCEPTANCE BY USERS
APPROACH SHOULD BE CONSISTENT WITH USE ON EARTH ORBITAL TELEOPERATOR SYSTEM, AND SOLAR ELECTRIC PROPULSION SYSTEM
IN-FLIGHT DEMONSTRATION/VERIFICATIONS ARE REQUIRED
MINIMIZE LIFE CYCLE COSTS

Figure II-12 Servicer Requirements - Programmatic Factors

Concern regarding this early complex servicer prompted an analysis to determine feasibility of using a simple, single-tier servicing capability for the more complex two tier LEO missions. That analysis was documented in a memo titled, "Applicability of the Simple Version of the On-orbit Servicer to the Low Earth Orbit Mission area," dated August 13,

1976. The results from that study will be summarized here.

The analysis focused on the Maintenance Applicable Set (MAS) of the first IOSS as a representative sample of the satellite programs in the shuttle era. There are twenty-six LEO programs in the MAS. Each of these was evaluated as shown on the flow chart of Figure II-13. For a program to go directly through the logic flow, data had to be available on module sizes, the modules had to weigh less than 300 kg and be smaller than a cube,one meter on a side, and the total of the modules had to weigh less than 2,100 kg. Where these criteria were not met the module sizes were evaluated to determine if redesign was feasible, or an evaluation of reliability of modules was performed to determine the feasibility and benefits of locating low reliability modules within access of a single tier servicer, or data was extrapolated.

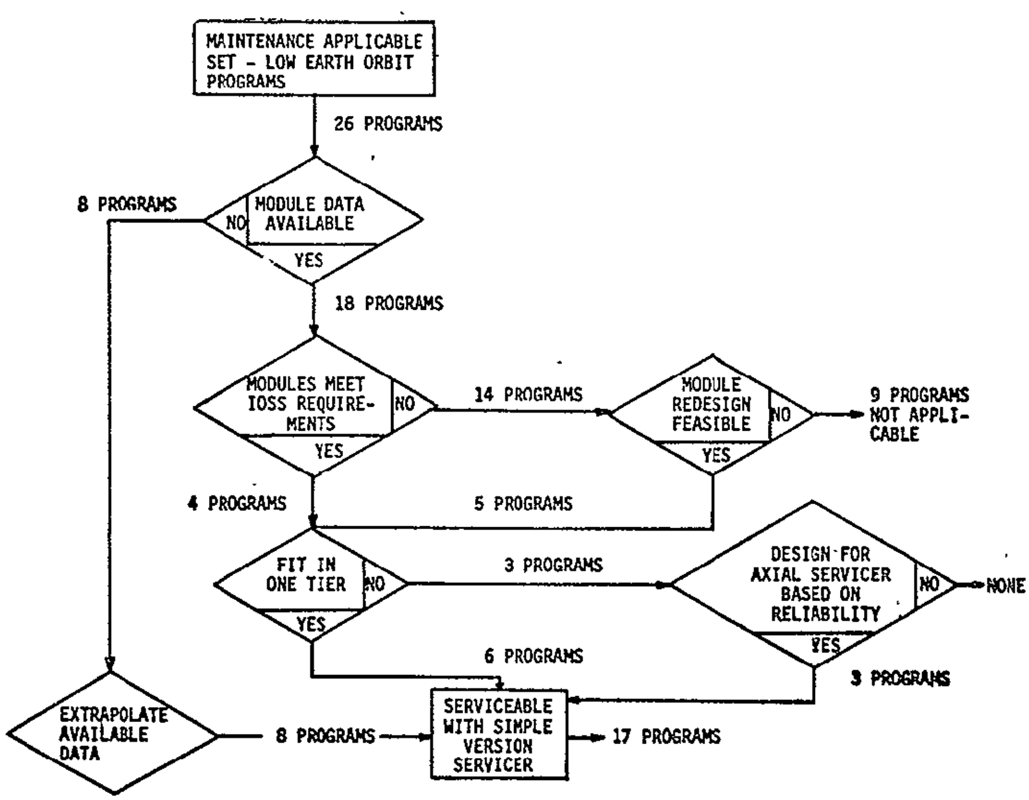

Figure II-13 Evaluation Logic Flow

Specific module data required to perform this analysis was obtained from the Level A and B, SSPD's, July 1975. Of the twenty-six programs in the MAS, LEO category, this data was available for only 18. The evaluation was based on these programs and the results were extrapolated to the whole set at the conclusion of the analysis.

Fourteen of the eighteen programs contained at least one module that exceeded the IOSS requirements. Five of these 14 programs could be redesigned for serviceability. These programs normally contained mission equipment modules for which an entire instrument was packaged into a single module. In these cases the instruments could be separated into two or more modules which are replaceable. For example, a complete instrument could be separated into the optics and detector in one module and the electronics in another module.

The programs for which module redesign was not considered feasible nominally contained very large mission equipment modules. Examples are: the 2700 kg Nuclear Calorimeter in HE-12-A and the 6000 kg Large Gamma Ray Survey Instrument in HE-08-A. High energy particle detection necessitates shielding and heavy modules for which redesign for replacement does not appear feasible at this level of evaluation.

The five redesignable spacecraft plus the earlier four that met initial weight and size requirements were examined for location within one tier of the subject spacecraft. A tier is defined for a Shuttle era designed spacecraft as a cylinder 4.57 meters diameter by 1.2 meters long. This is a volume of 16.3 cubic meters. Six of the programs permitted one tier and three required more than one. To evaluate the potential benefit of servicing just the first tier of these spacecraft, a study of module reliability based on data from the Aerospace Corporation was performed. The highly reliable modules were classed as being non-serviceable and were placed in the second tier.

The final analysis showed that seventy-five percent of the potential cost savings accruable through on-orbit maintenance in the LEO case can be obtained by servicing only one tier of the spacecraft with a simple version of the servicer. This is primarily possible through the significant result that for spacecraft of more than one tier, the additional cost of considering the second tier a nonreplaceable unit is minimal when high reliability components are located there.

5. Servicer Requirements - Economic Factors

The impact of economic factors on servicing are summarized in Figure II-14. The potential spacecraft program cost savings were evaluated under a number of variations in the first IOSS cost sensitivity analysis. In all cases, the cost savings were always significantly larger than the servicer life cycle costs. This point strengthens the development of a more versatile servicer -- more versatile both in maximum capability and existence

SPACECRAFT PROGRAM SAVINGS ARE SIGNIFICANTLY GREATER THAN SERVICER LIFE CYCLE COSTS

LAUNCH COSTS ARE MORE SIGNIFICANT FOR HEO THAN LEO

- Minimize Weight
- Minimize Stowed Length

SERVICER MODULARIZATION

A modularized servicer will cost more to develop and operate

A modular development program could imply lower peak funding requirements

Lighter versions of servicer will be an incentive to spacecraft designer for simpler spacecraft configurations

DEEMPHASIZE ACCOMMODATING SPACECRAFT PROGRAMS WHERE POTENTIAL SAVINGS ARE LOW

Figure II-14 Servicer Requirements - Economic Factors of different versions (servicer modularization). The additional costs associated with modularization could be offset by low launch costs for the HEO missions. Note that if modularized forms are not needed for the early missions, their development could be delayed. This would spread out the development costs and thus lower the peak funding requirement.

If the servicer development is not initiated soon, and user acceptance obtained, then the potential benefits can be sharply reduced. Once a spacecraft is developed in an expendable mode, then the costs of redevelopment to a serviceable version can significantly reduce the potential benefits.

The uncertainty in the anticipated traffic model is another point in favor of development of a versatile servicer system.

6. Technical Considerations

There are several other sources of requirements for the servicer that don't really fall in any of the above categories. For lack of a better classification they have been called technical considerations. One of these is the significance of minimizing servicer arm length and another is to minimize complexity.

The rationale for minimizing servicer arm length is evident from the following relationships.

- Joint torque and weight increase with arm length for constant end effector force.
- Joint weight increases with arm length for constant arm stiffness.
- Arm tube weight increases as fourth power of arm length for constant arm stiffness.
- Joint angular accuracy required increases as inverse of arm length for constant end effector position accuracy.

Mechanism, or arm, length should also be minimized of itself because of its effect on arm stowage problems. However, arm length is also a measure of concern with respect to the other system parameters shown. End effector force was selected as being greater than 20 lb. in the first IOSS. The desired arm stiffness is related to control system stability and to the operator's ability to control the system. End effector position accuracy is related to the capture volume of the end effector to interface mechanism and capture volume of the interface mechanism baseplate to the baseplate receptacle.

In order to amplify on the minimized complexity requirement a means of measuring complexity must be established.

The factors listed on Figure II-15 were selected to represent servicer mechanism complexity. It is desirable to minimize complexity while retaining the ability to satisfy the performance reuqirements. All configurations have three degrees of freedom to provide for relative attitude. This requirement arises because of the combined axial, radial, and offset radial module removal directions along with the greater reach distances and desire to give the spacecraft designer freedom as to how he orients his interface mechanisms.

NUMBER OF DEGREES OF FREEDOM

- Three required for relative attitude
- Three is minimum required for relative translation

TYPE OF TRANSLATIONAL DEGREE OF FREEDOM

- Rotational drives are simplest
- One stage translational drive is more complex
- Telescoping translational drives are even more complex

Figure II-15 Servicer Reqnirements - Complexity Factors

Three degrees of freedom are the minimum required for the three components of relative linear position or translation. However, a fourth degree of freedom is used, in some configurations, to provide a reach-around capability or so that the elbow joints, or arm segments, can be positioned away from spacecraft structure.

Translational drives are considered to be more complex than rotational drives because of the difficulty in controlling stiffness and backlash. The one stage translational drives, such as was used on the pivoting arm servicer mechanism, are less complex than the multiple-stage translational drives. These multiple-stage, or telescoping, drives have definite limits on the extension per stage that is possible while still retaining reasonable stiffness and backlash properties. The low stiffness arises because of circumferential bending in the tube cross sections.

C. SELECTION OF EXAMPLE SATELLITES FOR MAINTAINABILITY DESIGN

Chapter IV documents the design of three typical maintainable spacecraft performed by TRW under subcontract. Before this preliminary design subcontract was initiated, it was necessary to select representative spacecraft to be designed for maintainability. This selection was documented in a memo, "Geosynchronous Maintenance Spacecraft Selection", dated June 23, 1976. That selection process is summarized here.

In order to obtain the maximum depth of analysis in the serviceable spacecraft design work of Chapter IV, it was decided to concentrate on three spacecraft. Two of these to be geosynchronous and one to represent all large, low earth orbit observatories. The geosynchronous spacecraft are smaller, tend to be one tier with axial module removal, and thus emphasize the utility of the simpler of the servicer mechanism modular configurations. The large, low earth orbit observatories represent a significant part of the potential savings from on-orbit servicing. The desire was to address all LEO observatory classes by using a single representative. The differences in observatory mission objectives to be handled by a consideration of mission equipment alternatives.

As a matter of interest, it was found in the analysis summarized in Section B.4 that a simpler, single-tier servicer design could be very effective even for a two tier low earth orbit spacecraft, merely by placing the highly reliable modules in the second tier.

It was agreed to select two high earth orbit spacecraft. The criteria initially used in the selection process are:

a) Be contained in the maintenance applicable set of the first IOSS;

b) Be representative of the high earth orbit one-tier spacecraft for which significant savings can accrue when orbital servicing is used;

c) Be able to span the "design for serviceability" aspects for spacecraft subsystems and for mission equipment; and

d) Adequate spacecraft definition should be available -- it is desirable to have a spacecraft for which a study of serviceable design has been performed.

There were initially 101 spacecraft programs from which selection could have been made. These were evaluated during the first IOSS with respect to orbit, schedule, necessity for servicing, data availability, and probable savings from servicing. The result was that 49 spacecraft programs were selected for maintenance consideration. The 49 were later reduced to 47 for the economic analysis. These 47 formed the maintenance applicable set of spacecraft introduced earlier in Table II-1.

The objective for this selection has been established as the group of spacecraft whose orbits require the high earth orbit upper stages as opposed to merely the shuttle orbiter as the servicer carrier vehicle. This further reduces the spacecraft programs to the 21 listed in Table II-4. Each of these spacecraft could be designed for servicing as one or one-plus tier configurations requiring a simple servicer. One or one-plus tier refers to a spacecraft configuration which is ~ 180 inches in diameter and ~ 40 inches deep. Module removal is in either the axial or radial direction. The programs listed in Table II-4 are in order of cost savings for on-orbit servicing beginning with the highest cost savings.

The cost savings recoverable through on-orbit maintenance are closely related to the number of maintenance cycles. Each maintenance cycle represents either an additional expendable spacecraft that would have to be flown or a ground refurbishment that would be performed if on-orbit servicing were not available. As such, the spacecraft selection can be pared down further by eliminating those programs with less than 3 maintenance cycles. This results in 13 programs.

Table II-4 MEO and HEO One or One-Plus Tier Spacecraft Programs

Payload Number	Payload Model Number	Program Name	Weight (kg)	Initial Launch Date	Maint. Cycles
EO-61-A	NN/D-11	E. Res. Sur. Oper. Sat.	733	07/81	9
EO-59-A	NN/D-12	Geosync. E. Res. Sat.	1475	06/88	8
EO-62-A	NN/D-13	For. Sync. E. Obs. Sat.	3300	06/88	8
EO-09-A	EO-4	Sync. Earth Obs. Sat.	3300	09/83	6
CN-51-A	NN/D-1	INTELSAT	1472	06/83	9
CN-53-A	NN/D-2B	DOMSAT B	1472	08/82	7
EO-56-A	NN/D-8	Environ. Monitor. Sat.	2204	06/82	6
CN-56-A	NN/D-5A	Foreign Comm. Sat. A	308	09/82	12
AS-16-A	AST-8	Lg. Rad. Obs. Array	1300	06/85	3
CN-59-A	NN/D-6	Comm. R&D Proto.	1438	04/83	2
CN-55-A	NN/D-4	Traffic Management Sat.	299	08/82	7
EO-58-A	NN/D-10	Geosync. Oper. Met. Sat.	286	07/82	6
AP-07-A	Phy-3B	Env. Perturb. Sat. B	3946	07/87	1
AP-02-A	Phy-1C	Expl. Med. Alt.	272	04/83	2
CN-54-A	NN/D-3	Disaster Warning Sat.	583	07/82	2
EO-57-A	NN/D-9	For. Sync. Met. Sat.	286	07/82	4
CN-58-A	NN/D-2C	DOMSAT C	2100	07/83	3
EO-12-A	EO-6	TIROS	1636	06/82	1
AS-05-A	AST-1C	Adv. Radio Ast. Exp.	596	07/83	2
AP-01-A	Phy-1B	Upper Atmos. Exp.	909	06/85	1
AP-05-A	Phy-3A	Env. Perturb. Sat. A	1488	07/84	0

EO-61-A	EO-56-A	EO-58-A	EO-09-A
EO-59-A	CN-56-A	EO-57-A	
CN-51-A	AS-16-A	CN-58-A	
CN-53-A	CN-55-A	EO-62-A	

These 13 programs can readily be identified to consist of seven earth observations programs, five communications satellites, and a physics program. Similarity of mission normally leads to similarity in subsystem and mission equipment, spacecraft design, and therefore servicing requirements. The communication satellite group similarities are: containing antennas, both transmitting and receiving, traveling wave tube amplifiers, a basic subsystem group, and a structure encompassing the equipment. The earth observing programs contain mission equipment that is similar from the standpoint of servicing. The only difference comes in whether the instruments house the required optics or whether they are located at the focal point of a telescope. This does not affect servicing except as the size of the mission equipment containing the optics is larger. The selection of a communications satellite will not adequately represent the mission equipment peculiarities of earth observation programs. It was therefore recommended that two spacecraft be selected, one for each category -- communications and earth observations. These programs are listed by category in Table II-5.

Table II-5 High Earth Orbit Program Mission Categories

EARTH OBSERVATIONS
EO-61-A, Earth Resources Survey Operational Satellite
EO-59-A, Geosynchronous Earth Resources Satellite
EO-56-A, Environmental Monitoring Satellite
EO-58-A, Geosynchronous Operational Meteorology Satellite
EO-57-A, Foreign Synchronous Meteorology Satellite
EO-62-A, Foreign Synchronous Earth Observation Satellite
EO-09-A, Synchronous Earth Observation Satellite
COMMUNICATIONS
CN-51-A, INTELSAT
CN-53-A, DOMSAT B
CN-56-A, Foreign Communications Satellite A
CN-55-A, Traffic Management Satellite
CN-58-A, DOMSAT C

The large Radio Observatory Array can be thought of as a communication satellite with a large antenna. Its servicing requirements are represented by the communications satellite category.

In order to select a spacecraft, the serviceable designs for high earth orbit one-tier spacecraft were reviewed. The similarity in serviceable design is evident in Table II-6. Eleven applicable serviceable designs were presented in the memorandum, "The Effect of Serviceable Spacecraft Design on Servicer System Requirements", May, 1976. These eleven are listed in Table II-6 according to serviceable design configuration. Seven of the spacecraft fall in the design category of a basic box shape, using web-type structure and axial removal. Figure II-3 earlier presented the TRW DSCS-II design. This is representative of the classification mentioned above and also of all communications spacecraft. Since TRW, our subcontractor, has performed the very recent and extensive study, "Servicing the DSCS-II with the STS", we have selected this as one of the spacecraft for study.

Table II-6 High Earth Orbit Single-Tier Serviceable Design Study Configurations

- Box Shape, Axial Removal, Web --
  - DSCS-II - Defense Support Communications Satellite, TRW Design
  - INTELSAT - COMSAT
  - DSP, Defense Support Program, TRW
  - COMR&DSAT - Communications R&D Satellite, MDAC/Fairchild Design
  - CSCSAT - Commercial Synchronous Communications Satellite, MDAC/Fairchild
  - SEOS - Synchronous Earth Observation Satellite, MDAC/GE
  - AGOES - Advanced Geosynchronous Observation Environmental Platform, MDAC/GE
- Plate
  - UOP - Unmanned Orbital Platform, Rockwell International
  - DOMSAT, MSFC
- Concentric Ring
  - DSP - Defense Support Program, Aerospace Corporation
- Radial Removal
  - SEOS - Synchronous Earth Observations Satellite, General Electric

As indicated in Table II-6, two of the serviceable spacecraft studies on earth observations programs were performed on the Synchronous Earth Observation satellite (SEOS). One by General Electric configured this spacecraft for radial module removal. A subsequent study by McDonnell Douglas and GE configured it as a box structure using axial module removal. The SEOS is also one of the programs showing the most potential for cost savings to accrue through the use of on-orbit servicing. Together the SEOS and the Foreign SEOS require 14 maintenance cycles. The SEOS was selected as the second high earth orbit spacecraft.

For completeness, it was found necessary that the maintainability design effort by TRW should also include a representative low earth orbit spacecraft, though preliminary servicer design would not necessarily be directed toward this spacecraft. The low earth spacecraft in the MAS in Table II-1 were evaluated in a manner similar to that above for high earth orbit. Three large observatory types were identified: high energy astrophysics, solar and stellar. Each of these classes represent large potential savings. It was decided to address the mission equipment from a representative of each class. The selected representatives were:

HE-11-A, Large High Energy Observatory D (1.2m x-ray telescope)
AS-01-A, Large Space Telescope
SO-02-A, Large Solar Observatory

The mission equipment from each of these is addressed in Chapter IV. It was decided to look at the total servicing requirements by looking at one of the above three which was to be called a Characteristic Large Observatory (CLO). The Space Telescope could not be used as a basis for the CLO as it was in a pre-RFP stage and thus any serviceable design would have been sensitive and controversial. It was found that little hard data existed on the Large Solar Observatory. A good expendable spacecraft design was essential for the CLO so that effort could be expended on serviceability aspects as opposed to expanding general mission requirements into a basic spacecraft design.

Fortunately an adequate amount of data existed for the 1.2 m x-ray telescope. It had been studied in some detail by MSFC in 1976 and their data was available. Additionally, TRW had significant experience on high energy observatory spacecraft and mission equipment that could be directly applied. Thus, the 1.2 m x-ray telescope was selected as the subject for the Characteristic Large Observatory design for serviceability.

The resulting maintainable designs of the DSCS II, SEOS and CLO are presented in Chapter IV.

III. SERVICER MECHANISM CONFIGURATION SELECTION

This chapter will discuss a relatively unique aspect of the servicer design. In the normal design evolution there is an orderly process that starts with a concept and a set of requirements and grows into a system design and finally the detailed hardware design. In the evolution of the servicing system there was another step that was not only a critical point in the design process but also one about which there were many uncertainties and questions that had to be carefully resolved before a true design phase could be initiated. That step was the selection of a configuration for the servicer. Explicitly, the task entailed selection of the most optimum arrangement of servicer arm segments (quantity and length), joint orientation and joint order. The configuration directly affects almost all aspects of the subsequent design, particularly the mechanism and control system. The challenge in this task arises from the large number of variables in the problem which result in, theoretically, a seemingly almost infinite number of potential configurations.

This chapter will document the analysis conducted to reduce this myriad of configurations down to the optimum servicer.

The servicer requirements definition, provided in Chapter II, encompasses all the prerequisites as can be seen in the flow diagram of the approach, depicted in Figure III-1. Also aiding the analysis was the configuration work done on the previous IOS study.

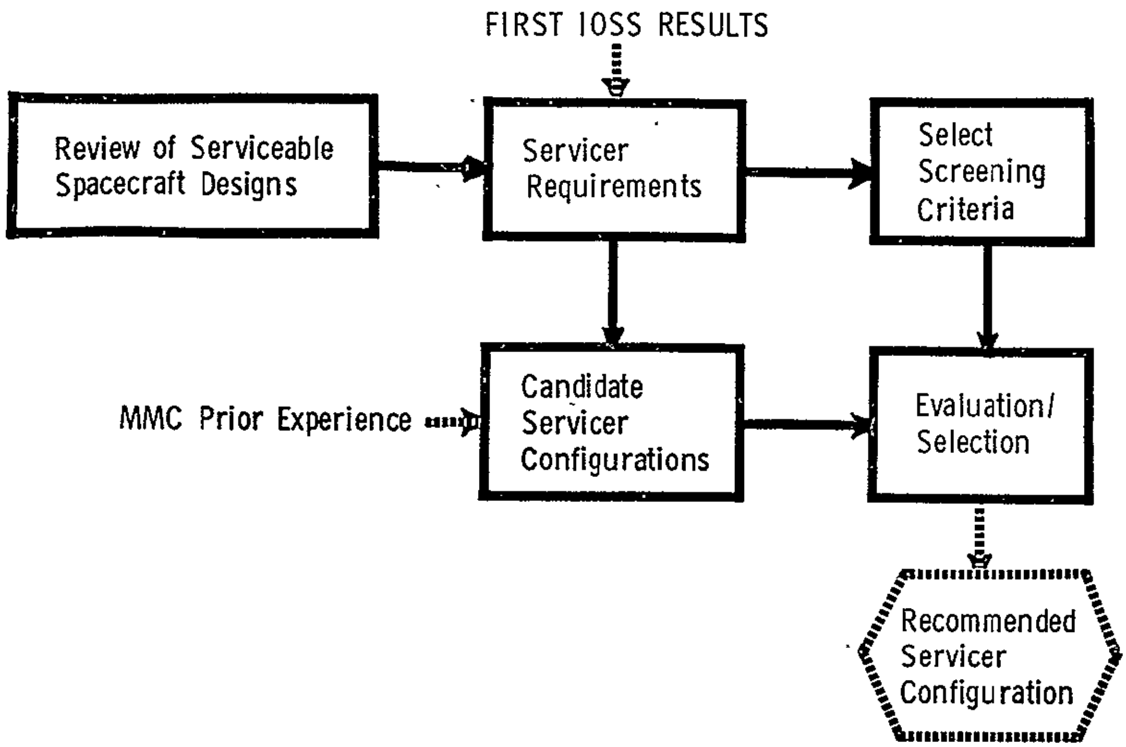

Figure III-1 On-orbit Servicer Configuration Selection Approach

A. CANONICAL FORMS ANALYSIS

The first IOSS required use, or modification, of a servicer design that existed in the literature. This follow-on study has not been constrained in that manner. It was considered extremely important to make an unbiased, thorough, bottom up evaluation of all conceivable configurations of the arm. An analysis was conducted and documented in a memo, "Canonical Forms of Two Segment Manipulator Arms," dated 26 March 1976. Note that even this analysis was limited to no more than two arm segments. These two-segment arms are connected to joints and oriented so as to provide up to three degrees of translation motion of the end of the last segment. Three and four segment arms are of course feasible, but the number of possible configurations was considered totally unmanageable within the study resources. It will be seen that two segments, properly connected and oriented, can satisfy many servicing activities. However, three segments are needed to reach around multiple tier spacecraft. It should be noted that the desired motion of concern which the configurations must supply is translational movement of the tip. Rotation of an end effector about the end of the last arm's segment is called attitude and does not really enter into the configuration selection.

The analysis identified four classes of configurations:

- Three translational directions of motion
- Two translational directions of motion and one rotational
- One translational direction of motion and two rotational, allowing one of the segments to be a fixed length
- Three rotational degrees of freedom and fixed lengths for both arm segments

These four general configurations can be formulated into 31,185 specific configurations. The canonical forms analysis reduced these 31,185 possibilities down to 12 canonical forms. By way of explanation, a canonical, or normal, form is the simplest and most significant form to which the configurations can be reduced without loss of generality.

They thus permit reducing the number of cases to a few that are representative of all possibilities. The results of the analysis are depicted in Table III-1.

Table III-1 Canonical Forms Analysis Results

DOF and Segments	Possible Configurations	Canonical Forms	Carried to Evaluation as Two Segment Arms	Extrapolated to Three Segment Arms
2T, 1R	81	1	0	0
1T, 2R, 1F	1,944	7	3	1
1T, 2R		1	0	0
3R, 2F	29,160	3	0	3
TOTAL	31,185	12	3	4

F - fixed length segment, T - variable length segment
R - rotational degree of freedom

Of the 12 canonical forms the two-translational configuration was not used because of its complexity. Note that no candidates are even identified for the three translational motion grouping because of its undesirable complexity features. Four of the 12 forms were not used because of limited working volume. Of the seven remaining, three were carried to evaluation as two segment arms and four were extended into three-segment configurations by adding another degree of freedom (joint) and another fixed segment. The latter was done so that in the next section an evaluation of as broad spectrum of representative configurations as possible could be provided. It was preferred to include representative three segment and, if possible, four segment configurations. It was stated earlier that a bottoms up derivation of these many segment configurations was impractical, therefore a reasonable solution was to extrapolate the most logical of the 12 canonical forms in Table III-1 into three segment versions. All seven arms are included in the evaluation in the next section.

In the process of reducing the large number of possible configurations to the canonical forms, a number of configuration equivalency rules for identifying invalid configurations were identified. These are presented in Figure III-2. They are either intuitively obvious or can be easily proven. These equivalency rules were used in evolving to the ten configurations and in reviewing the set of configurations for completeness.

THE DIRECTION OF THE FIRST DEGREE OF FREEDOM AXIS CAN ALWAYS BE ARBITRARILY DESIGNATED (e.g. X)

A ROTATIONAL DOF MUST ALWAYS BE FOLLOWED BY A LINEAR SEGMENT (FIXED OR VARIABLE LENGTH) WITH ITS AXIS PERPENDICULAR TO THE AXIS OF THE ROTATIONAL DOF

THE ORDER OF TWO ADJACENT LINEAR SEGMENTS CAN BE INTERCHANGED

THE ORDER OF A ROTATION AND A LINEAR SEGMENT ABOUT AND ALONG THE SAME AXIS CAN BE INTERCHANGED

THE ORDER OF TWO ADJACENT ROTATIONS ABOUT DIFFERENT AXES CANNOT BE INTERCHANGED

THE ZERO DEFINITION OF A ROTATION AXIS DOES NOT MATTER

WHEN MORE THAN ONE ADJACENT ROTATION IS ABOUT THE SAME AXIS, EITHER DEGREES OF FREEDOM ARE LOST OR THERE ARE REDUNDANT DEGREES OF FREEDOM

Figure III-2 Configuration Equivalency Rules

B. EVALUATION OF FULL-CAPABILITY CONFIGURATIONS

Having reduced the initial myriad of configurations down to seven canonical forms that credibly represent all reasonable two and three segment configurations, the next step was to define each of these more explicitly so that a quantitative analysis could be performed. In the process it was found that two "special-case" configurations should be added to the seven because they were the subject of previous studies. One was the general purpose manipulator from the first IOSS with its circular track. The other was the pivoting arm from the first IOSS, however another rotational degree of freedom was added along with another fixed segment to give it the necessary capability for both axial and radial exchange. This addition makes it a four segment configuration; a desirable feature from the standpoint of broadening the spectrum of configurations evaluated. This makes nine candidates. One final configuration was added in the process of extending the four two-segment canonical forms to three segment configurations to more completely represent the category of three segment arms. The resulting total of ten configurations evaluated is listed in Table III-2 along with the basic source of the configuration.

Table III-2 Candidate Servicer Configurations

No.	Configuration Description	Source
1	Extended Pivoting Arm--Axial Translational Drive	First IOSS version extended to four segments
2	Extended Pivoting Arm--Shoulder Pitch	One of four canonical forms extended to three segments
3	Extended Pivoting Arm - Shoulder Pitch and Second Elbow Pitch	One of four canonical forms extended to three segments
4	Modified General Purpose Manipulator	MDAC circular track version from first IOSS
5	General Purpose Manipulator - Shoulder Yaw	One of four canonical forms extended to three segments
6	3 DOF Extendable Drive (second segment) - Elbow Pitch	One of three 2-segment canonical forms carried forward
7	3 DOF Extendable Drive (first segment) - Shoulder Yaw	One of three 2-segment canonical forms carried forward
8	3 DOF Extendable Drive (first segment) - Elbow Pitch	One of three 2-segment canonical forms carried forward
9	4 DOF Extendable Drive (first segment)--Shoulder and Elbow Pitch	One of four canonical forms extended to three segments
10	Three Segment Manipulator--Shoulder, First Elbow, and Second Elbow Pitch	One of four canonical forms extended to three segments

A three-dimensional 1/10 scale mockup of the servicer elements was constructed to aid in the evaluation. It proved to be invaluable. Photos of each of mocked up ten configurations is provided in Figure III-3. One point of clarification should be made regarding No. 4, which is the MDAC circular track version from the previous study. The circular drive track is not shown on the photo. It has been replaced by an equivalent mechanization for ease of mockup.

This study's progress reviews have described these configurations in much more detail than the format of this report will permit. The First

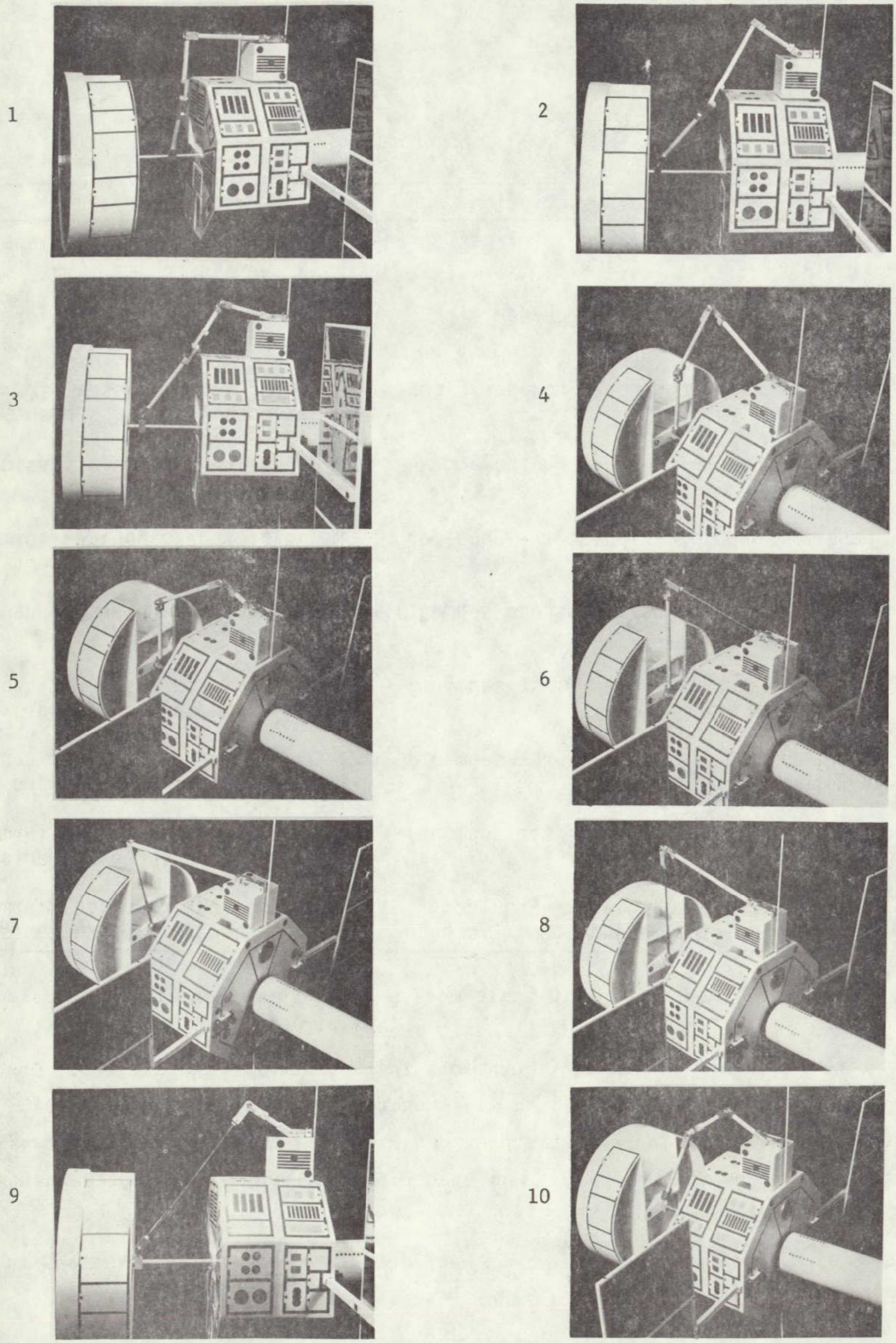

Figure III-3 Configuration Pictorial Description

Quarterly Progress Review, dated May 1976, should be referred to for more detail. Table III-3 provides a condensed tabulation of some of the pertinent characteristics of each of the ten configurations.

The evaluation of the ten servicer configurations was performed on three levels. The evaluation results from each level were used to determine which configurations should be carried forward to the next evaluation level. Also, more details were introduced in each subsequent level of evaluation.

In Evaluation Level 1 the servicer mechanism length for each configuration was optimized using as variables: relative length of arm segments, mechanism base location, and separation distance. In Evaluation Level 2 the remaining configurations were mocked up and investigated in a 3-D soft mockup. The capability of the configurations to accommodate reference module transfer trajectories was studied. In Evaluation Level 3 the remaining configurations were investigated for the capability to accommodate stowage of the servicer mechanism, modularization and counterbalancing. Stowage approaches allowed the servicer mechanism to be hinged, and thus stowed up against the face of the stowage rack. Modularization approaches for axial only, radial two-tier, radial minimum, and axial and radial minimum were studied. Counterbalancing approaches for the launch site and lab testing were investigated.

The screening criteria used to select between the ten servicer configurations at each of the three evaluation levels are listed in Figure III-4. A check means that a given criteria was used in the evaluation level. The first three criteria are spacecraft servicing requirements which every configuration should meet in order to be considered further. Servicer mechanism length is significant in that it affects weight, accuracy and stiffness. Spacecraft to stowage rack separation distance is important in that it affects servicer mechanism length, accuracy and structural weight. Mechanization complexity involves the number and type of degrees of freedom and the number of arm segments. Control complexity involves singular control axis and control equation complexity. The last three criteria are

Table III-3 Configuration Reference Data

Configuration Characteristic	CONFIGURATION 1	2	3	4	5	6	7	8	9	10
• Degrees of Freedom										
Arm										
Rotational	3	4	4	2	4	2	2	2	3	4
Translational	1	0	0	1	0	1*	1*	1*	1*	0
End Effector										
Rotational	3	3	3	3	3	3	3	3	3	3
Total	7	7	7	6	7	6	6	6	7	7
• Number of Arm Segments	4	3	3	3	3	2	2	2	2	3
• Base Offset (inches)	8	3	10	0	0	0	0	0	0	0
• Segment No. 1 (inches)	45	97	107	84**	84	140	20 to 140	255	86 to 179	113
• Segment No. 2 (inches)	70	97	53	101	108	30 to 158	150	140	44	65
• Segment No. 3 (inches)	70	44	56	101	43	-	-	-	-	45
• Segment No. 4 (inches)	88	-	-		-	-	-	-	-	-
• Total Arm Length (inches)	281	241	226	286	235	298	290	395	223	223
• Separation Distance (inches)	60	60	75	75	75	75	75	75	75	75

* Extendable

** Radius of Circular Track

measures of how effectively a configuration can accommodate stowage, modularization and counterbalancing.

Screening Criteria	Evaluation Levels		
	Level 1	Level 2	Level 3
Axial and Radial Module Removal	X	X	X
Reach-End Effector/Module Attach Locations	X	X	X
Module sizes			
Maximum, 40 in. x 40 in. x 40 in.	X	X	X
Minimum, 15 in. x 15 in. x 15 in.	X	X	X
Servicer Mechanism Length	X	X	X
Spacecraft to Stowage Rack Separation Distance	X	X	X
Mechanization Complexity	X	X	X
Control Complexity		X	X
Stowage			X
Modularization			X
Counterbalancing			X

Figure III-4 Servicer Configuration Screening Criteria

The results of the level one evaluation are summarized on Figure III-5.

The servicer mechanism length was optimized for each of the ten configurations considering as variables: relative length of arm segments, mechanism base location, and separation distance between the spacecraft and stowage rack. Data were tabulated on all ten configurations for the Level 1 Screening Criteria listed in the left hand column of Figure III-4. A comparison of these results indicated that Configurations 4, 6, 7, and 8 should be eliminated and not studied further. The driving factors for eliminating each configuration are indicated on Figure III-5 by a box.

SCREENING CRITERIA	CONFIGURATION				
	IDEAL	4	6	7	8
Axial and Radial Module Removal	Both	Both	Both	Both	Both
Reach End Effector/Module Attach Locations	OK	OK	OK	Can't reach central region of end	OK
Servicer Mechanism Length (Inches)	198	286	298	290	395
Separation Distance (Inches)	60 - 75	75	75	75	60
Complexity					
Degrees of Freedom					
Rotational	3	2	2	2	2
Translational		1*	1	1	1
Total	3	3	3**	3	3
Number of Segments	2	3	2	2	2

* Circular Track Drive

** First Arm Segment must be folded for stowage

☐ Indicates driving factors contributing to the elimination of a configuration

Figure III-5 Level One Evaluation Results

The data in the left hand column labeled "Ideal" represent what the screening criteria values would be for an "ideal" (most desirable) configuration. The other configurations can be measured relative to the "ideal" configuration values. It should be noted that the remaining systems which passed this gate compare much more favorably to the ideal. This data will be seen in the subsequent evaluation results.

Each of the four configurations (4, 6, 7, and 8) can accommodate both axial and radial module removal/insertion at the spacecraft. This was a fundamental requirement. Each configuration can reach the end effector/module attach locations with one exception. Configuration 7 cannot reach the end central region of the spacecraft. This factor along with the fact that its total arm length is 46 percent greater than the "ideal" configuration resulted in Configuration 7 being eliminated. Configuration 4 was eliminated based on a 44 percent increase in arm length and the complexity of the peripheral circular track. The circular track results in a considerable weight penalty which was seen in the previous IOSS study. Two factors resulted in the elimination of Configuration 6. The first factor was a 50 percent increase in arm length. The second factor was the need for folding the first arm segment during stowage which results in a considerable complexity penalty. Configuration 8 was eliminated based on an extremely long arm; 100 percent greater than the "ideal" configuration.

After the Level One Evaluation and subsequent elimination of configurations 4, 6, 7, and 8, the remaining configurations were mocked up and evaluated further in a three-dimensional mockup of the spacecraft and stowage rack. The segment lengths specified previously in the description of each configuration were used. The configurations were evaluated for their capability to perform the three module exchange transfer trajectories. Two factors were observed which resulted in the elimination of configurations 1, 2, and 5. Factor one was interference of the servicer arm when trying to reach the attach point location and then again during removal/insertion of the modules. Factor two was control complexity in terms of singular axes and non-planar control. The results for these three eliminated configurations is provided in Figure III-6.

Configuration 1 has a singular control axis problem when removing modules radially near the front edge of the spacecraft.
Also, for radial module removal the arm motion is non-planar which results in more complex control than for planar motion. These factors along with a fairly long arm length (281 inches) resulted in the elimination of configuration 1.

Configuration 2 has a servicer arm interference problem for radial module removal/insertion. The arm interferes with the corner of the spacecraft and stowage rack. Also, it has a non-planar arm motion for radial module removal.

SCREENING CRITERIA	CONFIGURATIONS			
	IDEAL	1	2	5
Axial and Radial Module Removal	Both	Both	Both	Both
Reach End Effector/ Module Attach Locations	Axial OK	OK	OK	Restricted to outer 10 in. on axial removal
	Radial OK	Near Radial Control Problem	Interference Problem with S/C & S/R corners	OK
Servicer Mechanism Length (inches)	198	281	241	235
Separation Distance (inches)	60 - 75	60	60	75
Mechanization Complexity				
Degrees of Freedom				
Rotational	3	3	4	4
Translational		1	0	0
Total	3	4	4	4
Number of Segments	2	4	3	3
Control Complexity				
Motion				
Axial	Planar	Planar	Planar	Non-Planar
Radial	Planar	Non-Planar	Non-Planar	Planar
Singular Axis				
Axial	None	None	None	None
Radial	None	Exists for Near Radial	None	None

☐ Indicates driving factors contributing to the elimination of a configuration

Figure III-6 Level Two Evaluation Results

Configuration 5 cannot reach axial modules which are more than ten inches in from the periphery of the spacecraft. Also, it has a non-planar arm motion for axial module removal.

Configurations 3, 9, and 10 were investigated in more depth on the 3-D mockup. The results are provided in Figure III-7. The areas of stowage, counterbalancing and modularization were studied. The three configurations are comparable for many of the screening criteria. Each configuration accommodates radial and axial removal over the spacecraft surface areas designated. Also, the servicer mechanism length and separation distance is essentially the same for each configuration. Configuration 3 has two major limiting factors. The first factor is control complexity. Its non-planar motion results in more complex control laws than for configurations

9 and 10. Counterbalancing for testing configuration 3 is more complicated than for configurations 9 and 10 because of the non-planar motion of the arm. These two factors were the major reasons for elimination of configuration 3. In addition configuration 3 will require a greater stowage depth because the shoulder and first elbow rotational axes are not parallel.

SCREENING CRITERIA		CONFIGURATIONS			
		IDEAL	3	9	10
Axial and Radial Module Removal		Both	Both	Both	Both
Reach End Effector/ Module Attach Locations	Axial	OK	OK	OK	OK
	Radial	OK	OK	OK	OK
Servicer Mechanism Length (inches)		198	226	223	223
Separation Distance (inches)		60 - 75	75	75	75
Mechanization Complexity - DOF Rotational Translational Total Number of Segments		 3 0 3 2	 4 0 4 3	 3 $\boxed{1}$ 4 2	 4 0 4 3
Control Complexity Motion Axial Radial		 Planar Planar	 $\boxed{\text{Non-Planar}}$ $\boxed{\text{Non-Planar}}$	 Planar Planar	 Planar Planar
Singular Axes		None	None*	None	None
Stowage		See Discussion	Fair	Good	Good
Modularization		See Discussion	---	Good	Good
Counterbalancing		See Discussion	$\boxed{\text{Poor}}$	Good	Good

* There is a singular axis for stowage rack radial far.

$\boxed{}$ Indicates driving factors contributing to the elimination of a configuration.

Figure III-7 Level Three Evaluation Results

Configurations 9 and 10 are comparable in the complexity required for stowage and counterbalancing. Also, both appear to have acceptable approaches for modularization. The major deciding factor between configurations 9 and 10 is the type of drives. Configuration 9 uses an extendable drive with a maximum to minimum extension ratio of 2.3. This drive is ranked as being considerably more complex than the rotational drive of configuration 10. A telescoping (extendable) translational drive is more complex than a non-extendable translational drive. The 2.3 ratio will require three or four segments which also adds to the complexity. These complexity factors resulted in the selection of configuration 10 as the recommended configuration. As the study progressed, however, it became apparent that it was still not possible or practical to launch into a design phase for this configuration. For one thing, it was a relatively complex mechanism and its development was not necessarily straightforward. It became apparent that simplified versions of configuration 10, which may not satisfy all current and future requirements, could be much more confidently developed. Consequently, at this point the requirements were reevaluated to see if a more phased development approach was feasible. The result was adoption of a modular arm configuration approach that had the ability of eventually growing toward the full capability of configuration 10 while permitting initial development of a simpler version that meets most of the requirements of the early years of servicing. The next sections will present that modular approach, rationale for it, and, in section E, make a final selection that will serve as a basis for the design phases described in the remaining chapters of this report

C. MODULAR FORMS DESCRIPTION

This section presents the more detailed level of analysis that was found necessary in order to finally arrive at an optimum servicer configuration. As stated earlier, the following discussion will show how the preferred configuration 10 was modified and compromised in order to provide a sensible development approach. Risks were reduced through simplicity of initial designs, yet nearly all important requirements were met. The capability for straightforward development to fuller capabilities in the future was preserved.

The first step in this phase of selection was the reevaluation of spacecraft requirements. It was determined that a number of these requirements are not really dominant in selection of a servicer configuration. Some typical requirements in this category are:

• Spacecraft Diameter	12 to 15 feet
• Docking Type	Central
• Number of Modules	10 to 30
• Module Size	Minimum -- 15 in. cube Maximum -- 40 in. cube
• Module Weight	10 to 700 lbs.

It was found that just two requirements really influenced the servicer configuration. They are:

- Number of tiers of modules on spacecraft
- Module removal directions

There are five logical combinations of tiers and module removal direction. They are listed in Table III-4.

Table III-4 Logical Spacecraft Servicing Requirements Groupings

Removal Direction	Number of Tiers	Designation
Axial	1	Axial
Axial and Radial	1	Axial/Near-Radial
Radial	1	Near-Radial
Radial	2	Two-Tier Radial
Axial and Radial	2	Axial/Two-Tier Radial

Obviously, a servicer mechanism design is matched to a set of spacecraft servicing requirements. Therefore, the question is: Which one or more of the five spacecraft servicing requirements groupings in Table III-4 should the servicer mechanism be designed to? The difficulty in resolving the question lies in the fact that the designs of future serviceable spacecraft are in the early stages. Thus, it is not known explicitly what the spacecraft community will need. It is also realized as the spacecraft servicing requirements are broadened to gain spacecraft designer flexibility, the servicer mechanism complexity increases. In light of these uncertainties and conflicts, it was decided the best approach was selection of a modular configuration that would permit accommodating all of the above groups by adding or subtracting segments or joints as necessary from a relatively simple basic configuration.

To verify the validity of this approach, a modular version of configuration 10 was devised and each of the five configurations necessary to accommodate the five groups in Table III-4 was mocked up with the 1/10 scale servicer elements. Each of these configurations will be described in detail in the following paragraphs. Section D will describe the evaluation and E will select the preferred of these five configurations for initial design and development.

1. Axial Servicer Configuration

The axial servicer is designed to accommodate servicing of a one tier spacecraft. In the prime operational mode modules are removed in an axial direction. However, the servicer mechanism can remove modules in an off-axis direction also. The off-axis removal is a significant feature for centrally located sensor packages on large telescope type spacecraft.

The series of photographs of the servicer sytem mockup in Figure III-8 demonstrate how a module would be exchanged. Modules can be located anywhere on the end surface of the spacecraft. However, structural and thermal requirements will dictate where and how the modules should be

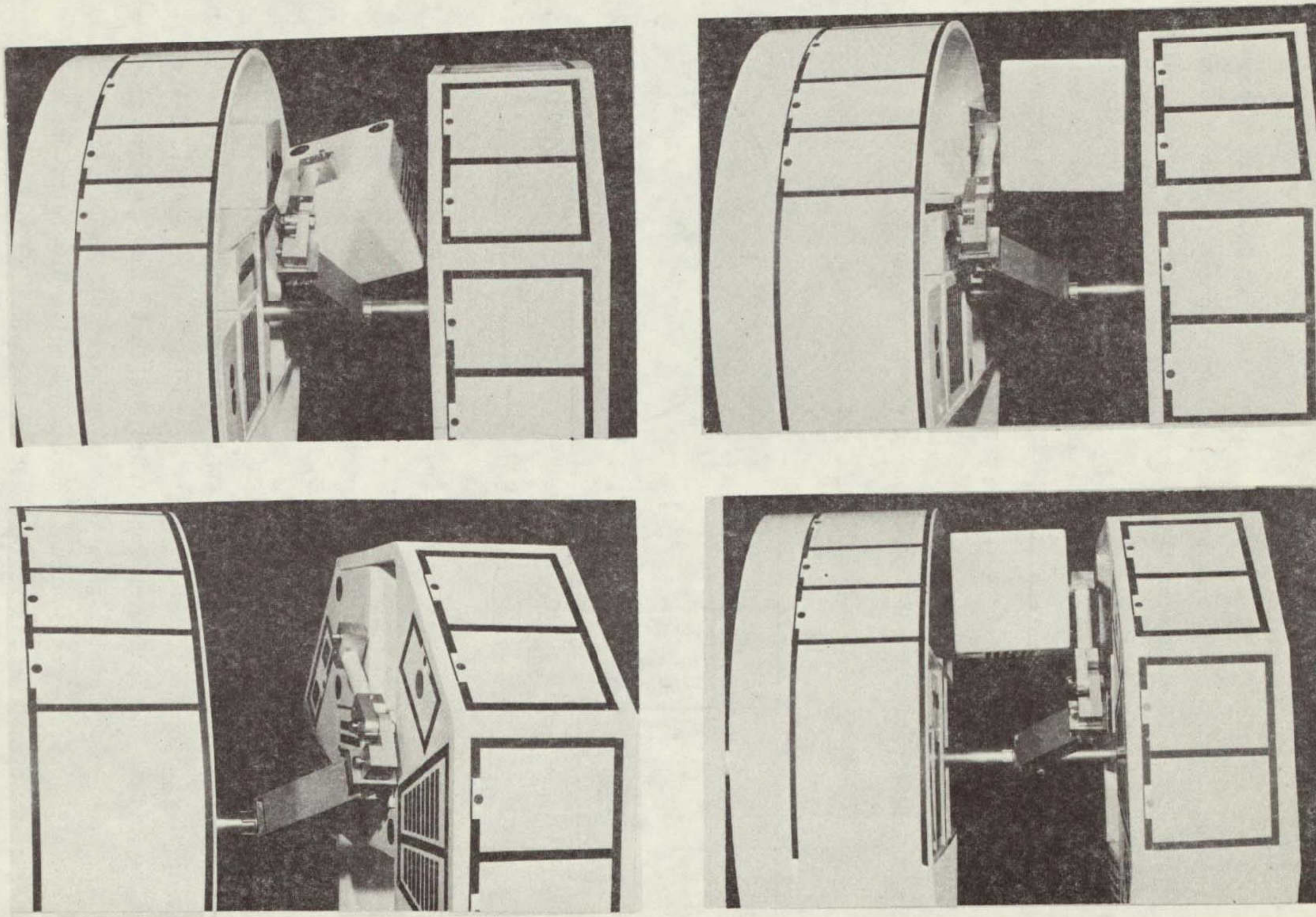

Figure III-8 Axial Servicer Configuration attached to the spacecraft. The servicer mechanism will accommodate both side and bottom mount interface mechanisms. The interface mechanisms can be located to within 20 inches of the central axis. Obviously, the module itself can extend inward from this point.

Operationally, this servicer has the advantage of being able to perform a module exchange within the 15 foot diameter of the spacecraft and stowage rack envelope. This feature has application when servicing within the Orbiter cargo bay.

A layout sketch of the configuration is shown in Figure III-9.

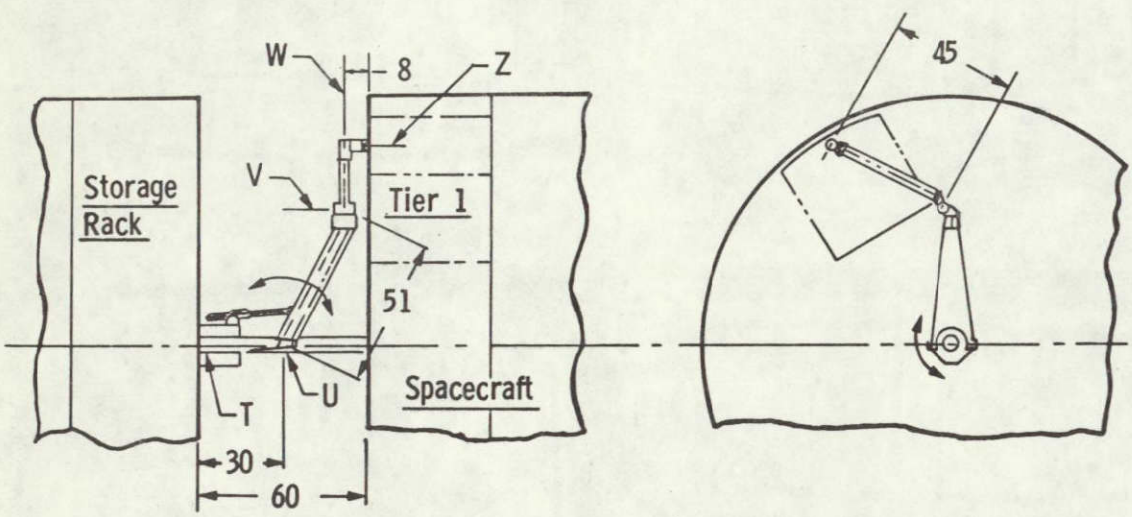

Figure III-9 Axial Servicer Configuration Layout

Note that a considerable number of design refinements and simplifications have been incorporated in the original configuration 10 design. In fact, the desired simplicity and the modularity features lead to a basic design, for this as well as the following four configurations, that is more closely aligned with the design of the pivoting arm servicer of the previous IOSS. One major design improvement, which is directly applicable to two of the modular configurations, is the replacement of the pivoting arm servicer's translational drive with a much simpler mechanization. This has been accomplished by attaching a four bar linkage (as shown in the layout) to the rotary portion of the shoulder roll drive (T). The four bar linkage (U) can be driven (as shown) in a push/pull manner by a ball screw, or a rotary drive can be installed at one of the lower pivot points of the linkage. The four bar linkage translates the elbow roll joint (V) between the spacecraft and stowage rack. The linkage is such that it maintains the elbow to wrist arm segment in planes parallel to the front of the spacecraft as the module is moved in the region between the spacecraft and stowage rack.

The wrist has two degrees of rotational freedom resulting in a total of five degrees of freedom for this configuration. It has an indexing drive (W) which is required for flipping the module. This drive does not have to be a continuous, accurate type of drive. Only the end points have to be accurate. A wrist roll drive (Z) is required to align the end effector to the interface mechanism for attachment.

Two types of module flip are available. One is inside the spacecraft and stowage rack envelope, and the other is outside of the spacecraft and stowage rack envelope. The flip inside the envelope is more complicated and requires a coordinated motion between the four bar linkage drive and the wrist indexing drive. The outside flip merely puts the arm in its fully extended position, and the flip then performed with the index drive.

The growth alternatives for this configuration, shown in Figure III-10, are: 1) Tandem module locations for growth to two tiers and 2) off-axis axial module removal for mission equipment access to central region.

A two-tier spacecraft can be serviced with the axial servicer using the tandem module locations approach. An end effector adapter is required. Also, there are certain restrictions placed on the spacecraft structural layout. The first tier modules must be larger than the second tier modules immediately behind, and an opening in the structure between the first and second tier must be provided.

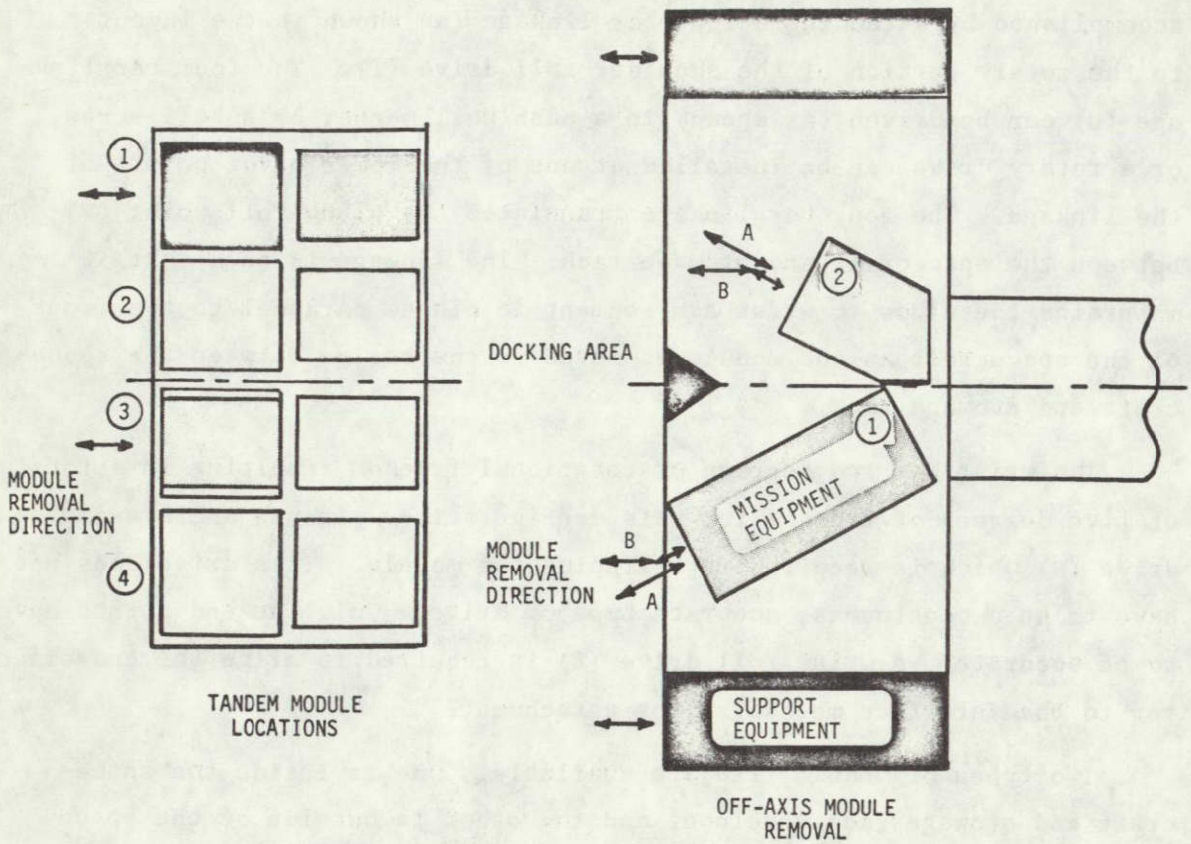

Figure III-10 Axial Module Removal Growth Alternatives

The sequence of steps to replace a second tier module starts with removing the first tier module (1) and stowing it in the stowage rack. The second tier module (2) is now accessible through an opening in the structure between the tiers. The servicer mechanism picks up an end effector adapter from the stowage rack. The adapter allows the interface mechanism of the second tier module to be reached within the translational travel of the servicer mechanism. By means of the adapter the end effector unlatches the second tier module. The servicer mechanism then translates the module to a temporary stowage location in the first tier (3). The module is latched in place. The end effector releases and stows the adapter. Now the second tier module is removed from its temporary stowage location in the first tier (4) in the normal manner. A reversal of the steps allows a second tier module to be inserted into the spacecraft. The current end effector design would require an additional drive to operate the jaws on the adapter.

The off-axis axial module removal approach on the right of Figure III-10 allows mission equipment and docking structure to be located in the central region. The docking structure is located near the front surface of the spacecraft. Behind the docking structure,mission equipment modules can extend to the centerline of the spacecraft. This is a desired location for sensors on some spacecraft. The module can be removed in an off-axis axial direction if space permits. If space is limited, the module is removed part way in an off-axis axial direction and the remainder in the axial direction B. The latter approach would require a different guide configuration and possibly have additional impacts on the interface mechanism. Module 2 location would require mounting the module on the interface mechanism so the end effector attach point is in the spacecraft frontal plane, or it would require use of the end effector adapter previously discussed.

2. Axial/Near-Radial Servicer Configuration

The axial/near-radial servicer is designed to accommodate a one tier spacecraft. Operationally, modules can be removed in both axial and radial directions from the spacecraft. This allows the spacecraft designer greater flexibility in structural and thermal design. However, some of this spacecraft design flexibility could also be gained through side and bottom mount interface mechanisms. Both off-axis axial and radial module removal can also be accomplished with this servicer mechanism.

The series of photographs in Figure III-11 demonstrate how a module would be exchanged. The servicer mechanism is very similar to the axial servicer previously discussed. The same types of rotary and four bar linkage drives are incorporated. The servicer mechanism accommodates both side and bottom mount interface mechanisms. Modules can be located anywhere on the end of the spacecraft. However, the end effector interface mechanism attach points for axial removal must be located outside of a 20 inch radius. For radial module removal the diameter of the spacecraft can vary from 80 inches to 174 inches. The radial attach points lie in a plane which can be thought of as a spacecraft frontal plane. Selecting this plane for attachment minimizes the segment length between the index drive and the

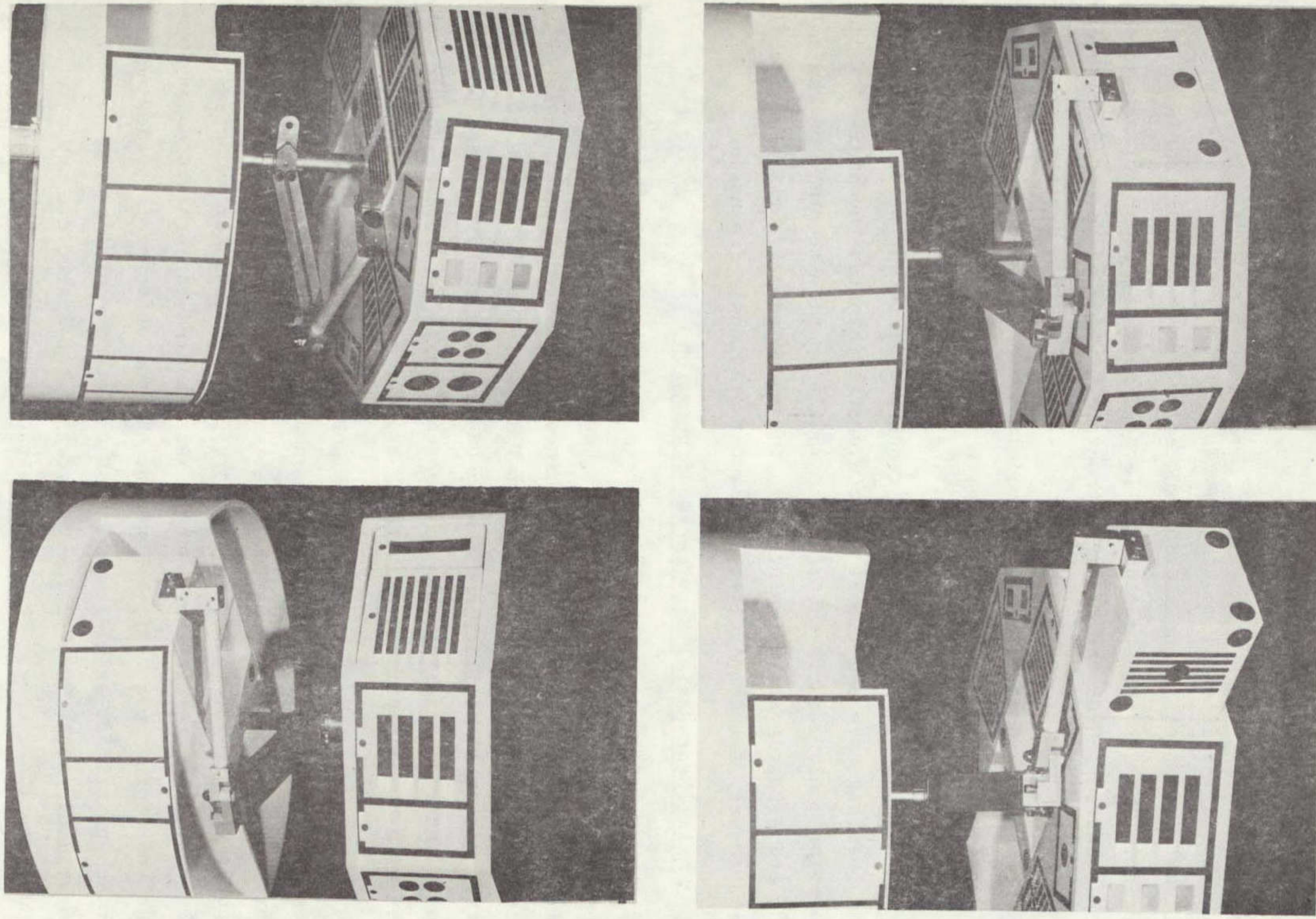

Figure III-11 Axial/Near-Radial Servicer Configuration end effector. However, this imposes certain restrictions on spacecraft volume and structural design. Alternatives are dicussed below.

A layout of the axial/near radial configuration is provided in Figure III-12.

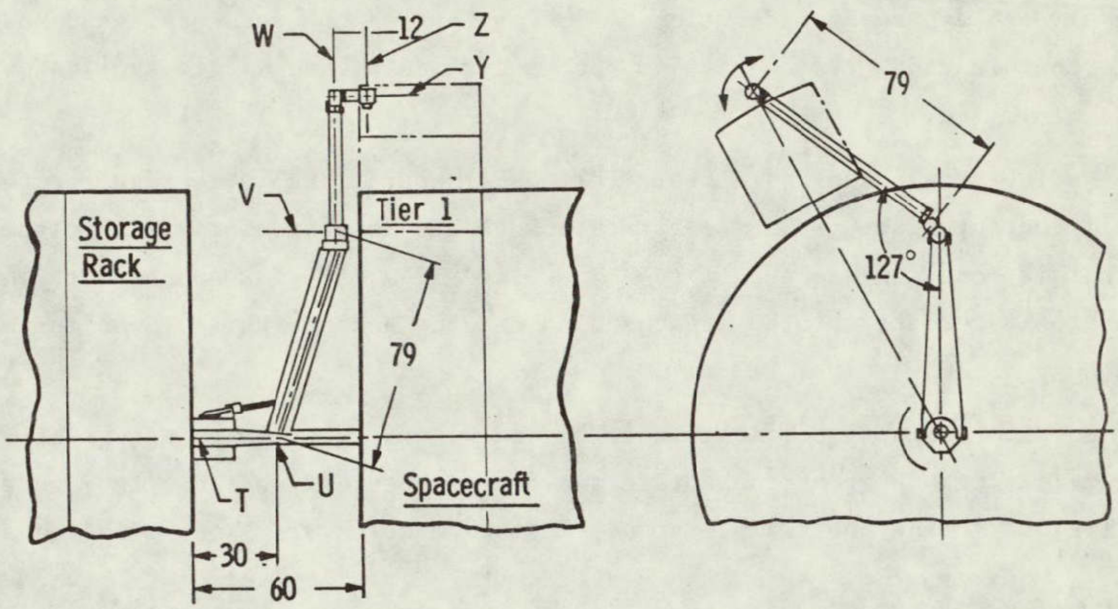

Figure III-12 Axial/Near-Radial Servicer Configuration Layout

The design is very similar to the axial servicer design with the exception of the following factors which have resulted from the addition of the radial module removal requirement:

- addition of one degree of rotational freedom in the wrist
- addition of a third arm segment
- longer first and second arm segments
- higher torques and greater accuracy in the drives

The added wrist pitch (Y) is required for orienting the wrist roll (Z) properly for the radial or axial module removal. It is also required for the radial module extract/insert motion. The third arm segment is required to allow the end effector to reach the radial attach points. The first and second arm segments were increased to 79 inches.

The total operating arm length has gone from 134 inches to 208 inches. This increase is arm length results in higher torque and greater accuracy requirements for the shoulder and elbow drives.

Two types of module flip are available: inside and outside the spacecraft and stowage rack envelope. The simpler mode is outside of the spacecraft and stowage rack envelope. The motions can be sequenced as opposed to coordinated. TV coverage is not as difficult, and hazard avoidance is minimized.

The axial/near-radial servicer normal operational mode requires that all the interface mechanism attach points lie in a frontal attach plane. In this mode the spacecraft to be serviced is a one-tier spacecraft. Also, there is symmetry of the attach points in the spacecraft to those in the stowage rack.

With a minor modification to the servicer mechanism this type of servicer can be grown to capture two-tier spacecraft as shown in Figure III-13. In the one example standard sized modules are shown with the attach

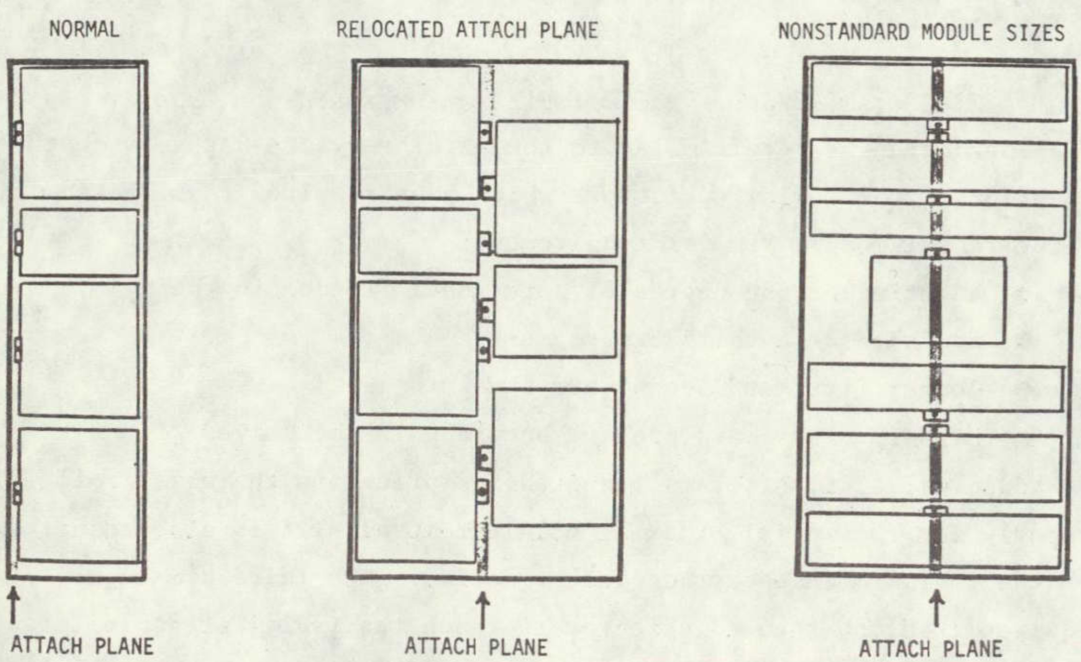

Figure III-13 Near-Radial Module Removal Growth Alternatives

C-2

plane in the center of the two tiers. The interface mechanism attach points lie in a plane because the module attachment points are interdigitated as shown. By using outsize modules as shown in the other example a two-tier spacecraft volume can also be captured.

The condition that must exist in the spacecraft is that all the attach points lie in a plane normal to the docking axis. The plane can be located anywhere in an axial direction from the front to the back. The modification to the servicer mechanism is affected by the distance the plane is from the front. The arm segment length between the index drive and the end effector must be changed so the end effector can reach the spacecraft attach plane. The condition that must exist in the stowage rack is that its attach plane is symmetrical with respect to the spacecraft attach plane. Thus, choosing the spacecraft attach plane location determines the impact on the stowage rack.

The modifications are minor compared to the increased spacecraft volume gained and could readily be incorporated on those servicing missions requiring them. Other servicing missions would have the nominal servicer mechanism. This would result in an effective match of spacecraft servicer requirements to servicer capability.

3. Near-Radial Servicer Configuration

The near-radial servicer, shown in Figure III-14, is designed to accommodate servicing of a one tier spacecraft. In the prime operational mode modules are removed in a radial direction. However, the servicer mechanism can remove modules in an off-axis radial direction also. The end-effector/interface mechanism attach points are located in a frontal plane normal to the longitudinal axes. There is symmetry between the spacecraft and stowage rack with respect to the attach points, and the attach points lie in a plane in both the spacecraft and the stowage rack.

More than one tier spacecraft can be serviced with the type of approaches described under the axial/near-radial servicer. The end

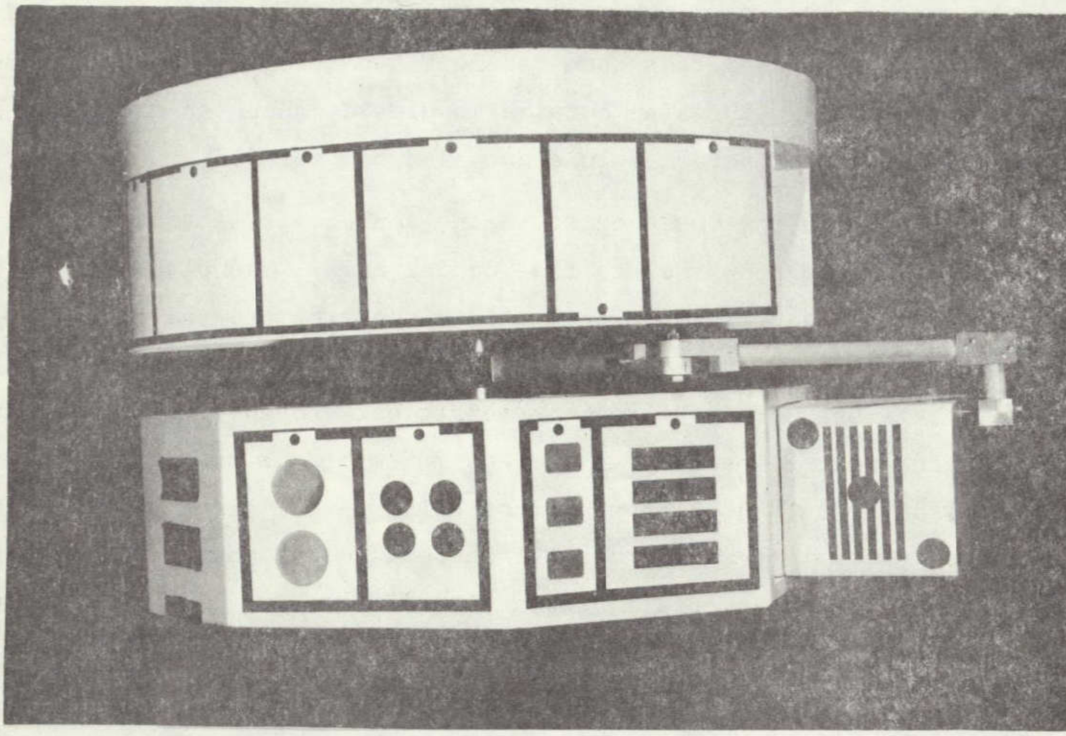

Figure III-14 Near-Radial Servicer Configuration

effector attach plane can be anywhere along the spacecraft. However, there must be symmetry of the attach plane in the stowage rack layout. The interdigitating of the attach points in the spacecraft allows a 40 inch depth stowage rack to accommodate a two tier spacecraft (40 inches per tier).

Considerable spacecraft structural design flexibility is gained through the use of side and bottom mount interface mechanisms. Spacecraft ranging in size from 80 to 174 inches in diameter can be serviced. With the off-axis radial module removal capability the spacecraft do not have to be cylindrical in shape. The spacecraft can have flat sides which are desirable for antenna farms and radiating surfaces.

The near-radial servicer shown in the configuration layout in Figure III-15 represents a simple design. It has the same number of degrees

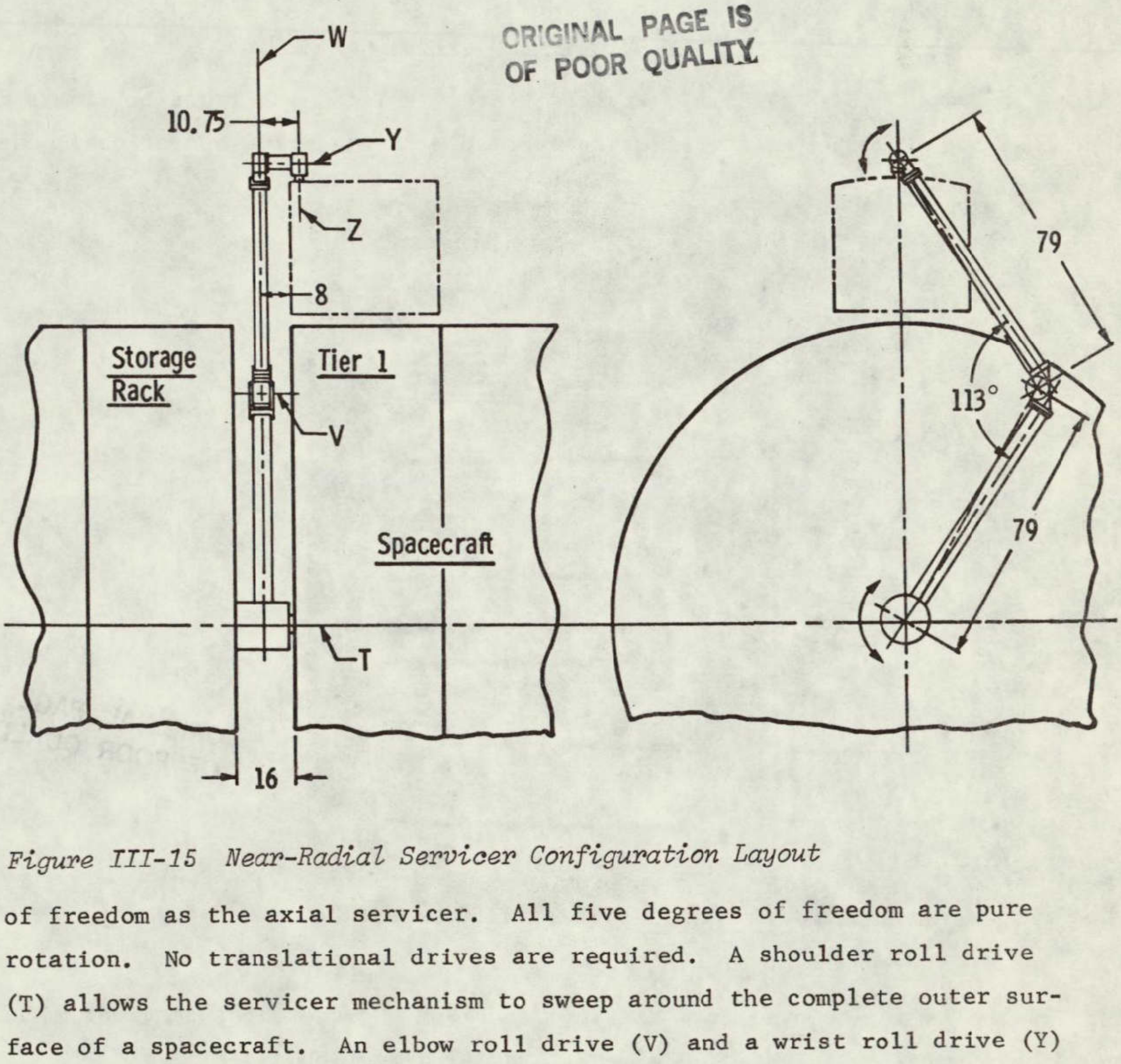

Figure III-15 Near-Radial Servicer Configuration Layout

of freedom as the axial servicer. All five degrees of freedom are pure rotation. No translational drives are required. A shoulder roll drive (T) allows the servicer mechanism to sweep around the complete outer surface of a spacecraft. An elbow roll drive (V) and a wrist roll drive (Y) along with the shoulder roll provide the extract/insert force. An indexing drive (W) is required to flip the modules between the spacecraft and stowage rack. This is a simple rotary drive requiring accuracy only at the end points. The wrist yaw (Z) aligns the end effector about the radial direction.

The arm segment lengths are 79 inches each with an overall operating length of 185 inches. The spacecraft to stowage rack separation distance is 16 inches. If the interface mechanism orientation in the spacecraft and stowage rack can be fixed, then the wrist yaw drive (Z) can be deleted and a four degree of freedom servicer results. This is the simplest of the servicer configurations that have been suggested.

4. Two-Tier Radial Servicer Configuration

The two-tier radial servicer system depicted in Figure III-16 is designed to service a two-tier spacecraft. In the prime operational mode

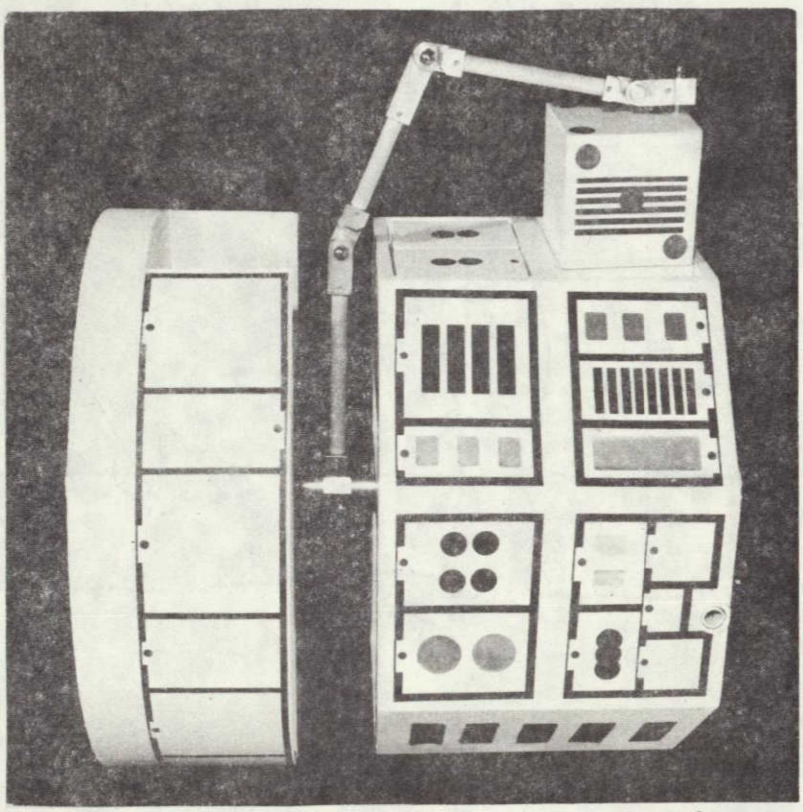

Figure III-16 Two-Tier Radial Servicer Configuration

modules are removed in a radial direction. However, the servicer mechanism can remove modules in an off-axis radial direction also.

It is of interest to compare the difference in spacecraft design between this servicer and the near radial. For this servicer the end effector/interface mechanism attach points do not have to lie in one plane normal to the longitudinal axis as they did for the near-radial servicer. Also, there does not have to be reflected symmetry in the stowage rack design. The attach points can be located anywhere on the outer surface of the spacecraft. This results in greater spacecraft designer flexibility. The significance of the greater design flexibility is difficult to assess. It should be noted that the near-radial servicer could service a two tier spacecraft using an adapter on the length of the third arm segment.

The two-tier radial servicer configuration layout is shown in Figure III-17. It has six degrees of rotational freedom. The three-segment arm forms a plane which always contains the longitudinal axis. The shoulder roll (T) allows the plane to rotate about the longitudinal axis. The two elbow rolls (V&W) along with the wrist roll (X) allow the top of the arm to be placed anywhere on the outer surface of the spacecraft and provide the extract/insert force for module replacement. This aspect of the configuration allows the end effector/interface mechanism attach point to be located anywhere on the outer surface of the spacecraft.

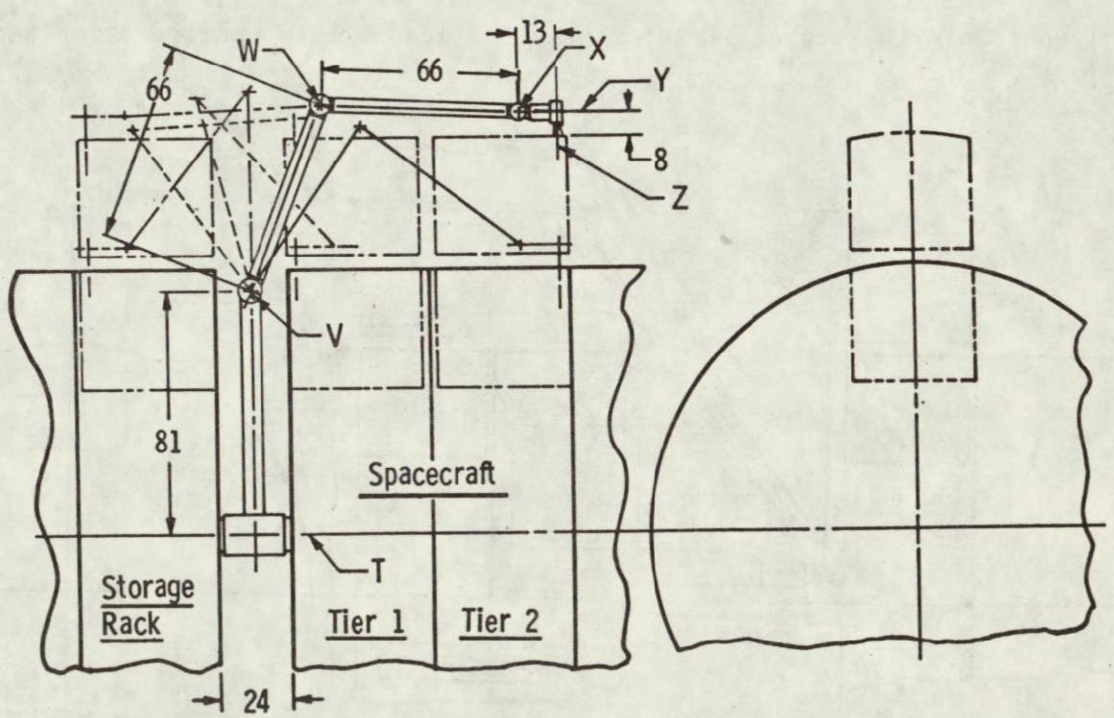

Figure III-17 Two-Tier Radial Servicer Configuration Layout

5. Axial/Two-Tier Radial Servicer Configuration

The axial/two-tier radial servicer is designed to service a two-tier spacecraft, with maximum flexibility for the spacecraft designer. A typical exchange is pictorialized in Figure III-18. Modules can be removed axially and off-axis axially from the first tier of the spacecraft. Modules can be removed radially and off-axis radially from the first and second tiers of the spacecraft. Maximum interface mechanism location and orientation flexibility are provided. On the outside of the cylindrical surface the interface mechanisms can be located anywhere, and the interface mechanisms do not have to be oriented to lie in a plane normal to the longitudinal axis. On the end surface the interface mechanisms must be located outside of a 40 inch diameter. There is complete orientation freedom within the end surface plane.

Two tier spacecraft varying in diameter from 80 to 174 inches can be serviced. The spacecraft can also have flat sides which require off-axis radial removal of modules. This axial/two-tier radial configuration, shown in layout form in Figure III-19, is the full-capability configuration ten of Section B.

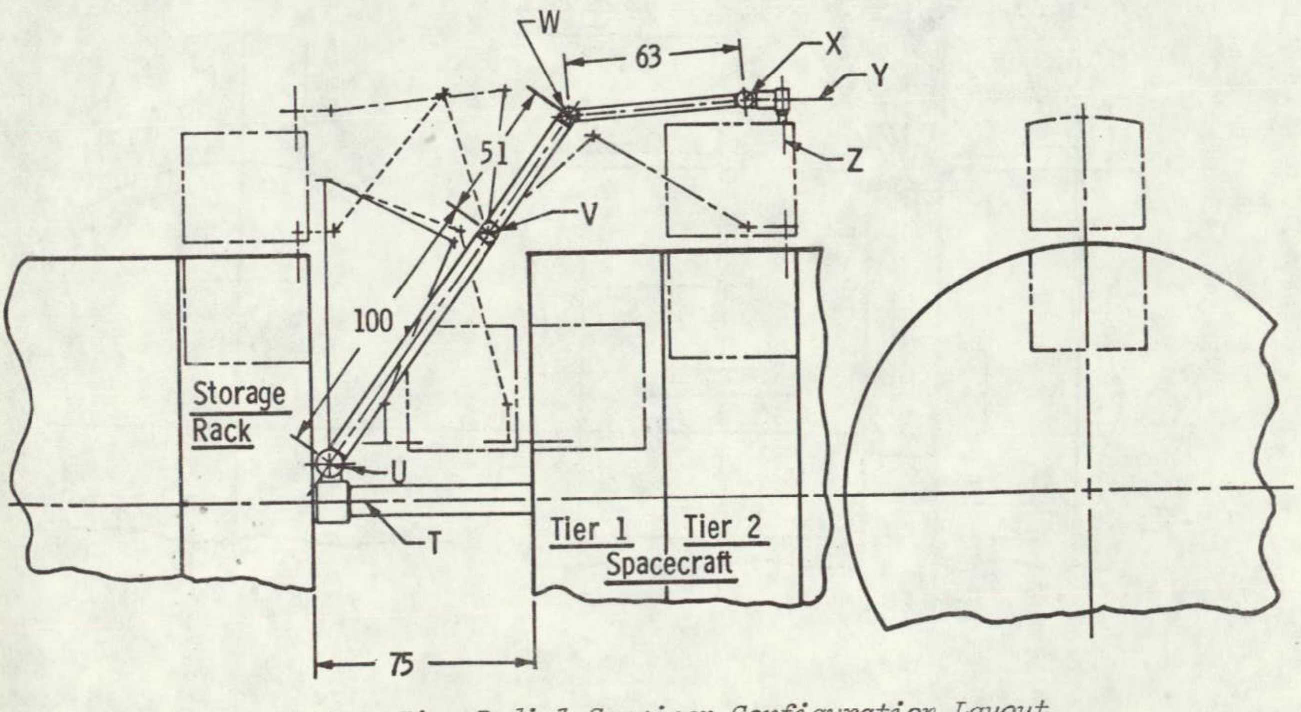

Figure III-19 Axial/Two-Tier Radial Servicer Configuration Layout

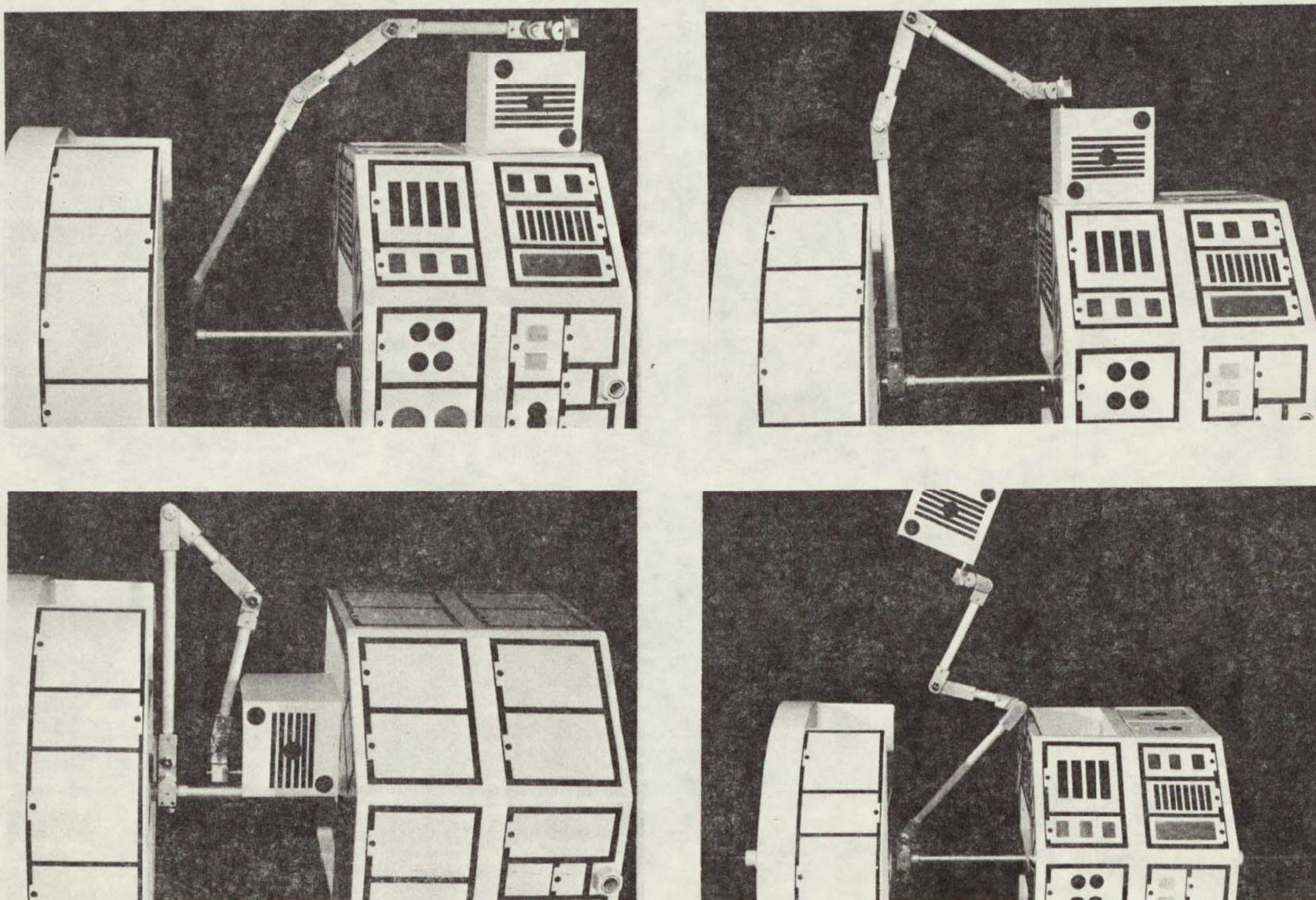

Figure III-18 Axial/Two-Tier Radial Servicer Configuration

The axial/two-tier radial servicer has 7 degrees of rotational freedom. The three segment arm forms a plane which always contains the longitudinal axis. The shoulder roll (T) allows the plane to rotate about the longitudinal axis. For radial and axial module replacement the shoulder pitch (U), the first elbow pitch (V), and the second elbow pitch (W) allow the tip of the arm to reach the end effector attach points on the outer and end surfaces of the spacecraft.

The three rotations of the wrist allow the end effector to be oriented properly for attachment. As shown in the layout the end effector is oriented for radial module removal. The wrist pitch (X) provides for attitude differences resulting from (1) different axial locations of the end effector attach points and (2) geometric changes during the module extract/insert motion. The wrist yaw (Y) is required for off-axis radial module removal. The wrist roll (Z) allows the interface mechanism orientation to vary (i.e., the interface mechanisms do not have to lie in planes normal to the longitudinal axis).

For axial module removal the end effector must be oriented 90 degrees out from the radial module removal direction. This is provided by the wrist pitch (X). Off-axis axial module removal attitude changes are provided by the wrist yaw (Y) and pitch (X). The wrist roll (Z) allows the interface mechanism to have any orientation in the spacecraft frontal plane.

The module flip is performed outside of the spacecraft and stowage rack envelope for both axial and radial module removal. The stowage rack need only be configured for radial module replacement.

D. MODULAR SERVICER CONFIGURATIONS EVALUATION

The evaluation of the five servicer configurations was performed for the same three levels used in the evaluations of the ten candidate configurations presented in Section B. In Verification Level One the servicer mechanism length for each configuration was optimized using as variables: relative length of arm segments, mechanism base location, and separation distance. In Verification Level Two the five configurations were mocked up and investigated in a 3-D soft mockup. The capability of the configurations to accommodate reference module transfer trajectories was studied. In Verification Level Three the configurations were investigated for the capability to accommodate stowage of the servicer mechanism and counterbalancing. For all the configurations the servicer mechanism was hinged and thus stowed up against the face of the stowage rack. Counterbalancing approaches for the launch site and lab testing were investigated.

One of the evaluation criteria of major influence is mechanism length. The geometric relationships between spacecraft, stowage rack and module locations which affect servicer mechanism length are shown in Figure III-20. These geometric considerations were used in the level one configuration evaluation to optimize servicer mechanism length. Nine module locations (as indicated) were examined. Associated end effector locations (A-K) were used.

L is the base location of the servicer mechanism. Module location one sets the maximum length of the servicer mechanism. The straight line from A to L represents the ideal length (198 inches) that a servicer mechanism can approach. The end effector point A is 140 inches from the spacecraft centerline. The modules are 40 inch squares with ten inch end effectors. A spacecraft to stowage rack separation distance of 60 inches is shown. Module locations one through four represent radial replacement modules. Modules five through seven represent axial replacement modules, and modules eight and nine represent modules in the stowage rack.

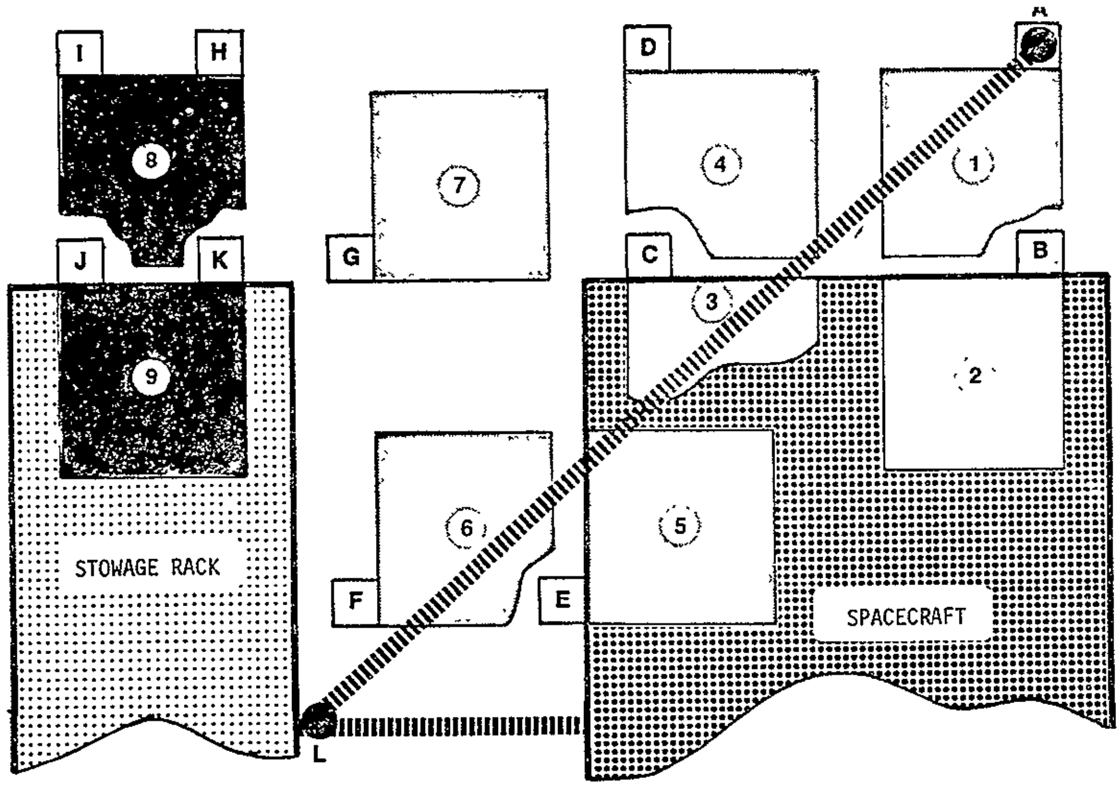

Figure III-20 Servicer Mechanism Length Considerations

The servicer mechanism length optimization started with locating the mechanism joints for each of the ten configurations so they most effectively aided avoiding the interference points at the corners of the spacecraft and module one. This initially set the segment lengths. Then the selected segment lengths were evaluated further at each of the nine end effector locations (A-K). Segment lengths and separation distance were modified when required. The configuration was then checked against module location one again. Thus, iterations through the series of module locations were performed as necessary to verify a functioning configuration.

For configurations having two clustered gimbals at the base (L) a 13 inch radial standoff was used to represent a realistic mechanization. The end effector was assumed to have a 20 inch separation between the first and second gimbals. These two factors significantly aid avoiding the interference points.

Several factors can be observed from the geometry. Configurations which use right angles between segments will not approach the ideal straight line very effectively, and thus they end up being long. Configurations which result in bringing the module up against them for axial module removal will result in greater separation distances (75 inches). The results of these geometrical relationships are summarized for the level one evaluation in Figure III-21. Each of the five servicer configurations

SCREENING CRITERIA	CONFIGURATION				
	AXIAL	AXIAL NEAR-RADIAL	NEAR-RADIAL	TWO-TIER RADIAL	AXIAL/TWO-TIER RADIAL
Module Removal Direction	Axial	Axial Radial	Radial	Radial	Axial Radial
Module Locations	C[1],E,F	C,D,E,F, G,H,K	C,D,H,K	A,B,C,D, I,J,H,K	A,C,D,E,F,G, H,I,J,K,B
Reach End Effector/Module Attach Locations	OK	OK	OK	OK	OK
Servicer Mechanism Length (Inches)	134	208	185	246	254
Ideal Servicer Mechanism Length (Inches)	101	153	140	178	216
Separation Distance (Inches)	60	60	16	24	75
Complexity					
Degrees of Freedom					
Rotational	4	5	4	6	7
Index	1	1	1	0	0
Total	5	6	5	6	7
Number of Segments	2	2	2	3	3

Figure III-21 Level One Verification Results

was investigated for the screening criteria parameters used previously. It should be noted that the objective of collecting this data on the five configurations is to verify how the servicers meet the criteria, not for comparison between servicers. Since each servicer has been designed to satisfy different spacecraft requirements, a direct comparison of servicers is not realistic.

The servicer mechanism length was optimized for each configuration considering as variables--relative length of arm segments, mechanism base location, and separation distance between the spacecraft and stowage rack. This was accomplished using the module transfer cases shown in Figure III-20. The module locations investigated for each configuration are shown on the figure. For the axial servicer the C' module location means that the C module was removed in the axial direction. The derived servicer mechanism lengths are stated. They can be compared to the "Ideal Servicer Mechanism" lengths shown. The ideal represents the minimum possible length using a straight line.

The separation distance is basically set by three factors--module length, module removal direction, and servicer arm configuration. A distance of 60 inches is required to remove and flip axially a 40-inch module with end effector attached. For radial module removal, the separation distance is set by the servicer mechanism motion envelope and stowage requirements.

The results from investigating the level two and three parameters are summarized in Figure III-22. Since each configuration is a near optimum for its group of spacecraft requirements, it would be expected that none of the configurations would have any basic problems. None of the configurations have any singular axes which would complicate the control problem.

Each has a plane-of-motion in which it can be controlled. This results in simpler control laws. The Axial-Two-Tier Radial servicer motion is planar. However, the servicer has seven degrees of freedom which implies redundancy in controlling the mechanism. After studying this condition a simple solution was determined. It was observed that the first elbow location could be held fixed (requires holding shoulder pitch constant) for each major part of a module exchange trajectory from all the various module locations. An example is the extraction of a module from the second tier of a spacecraft. With the arm up high enough to clear the spacecraft, the shoulder pitch is driven to position the first elbow at its predetermined location for this

SCREENING CRITERIA	CONFIGURATIONS: AXIAL	AXIAL/ NEAR-RADIAL	NEAR-RADIAL	TWO-TIER RADIAL	AXIAL/TWO-TIER RADIAL
Control Complexity					
Motion					
Axial	Planar	Planar	N/A	N/A	Planar*
Radial	N/A	Planar	Planar	Planar	Planar*
Singular Axes					
Axial	None	None	N/A	N/A	None
Radial	N/A	None	None	None	None
Stowage	Good	Good	Good	Good	Acceptable
Counterbalancing	Good	Good	Good	Good	Good

*Control of the seventh degree of freedom is straightforward.

Figure III-22 Levels Two and Three Verification Results

subtask. It is held fixed (constant) there. The rest of the joints are then driven to perform the module removal.

Good stowage and counterbalancing approaches were evolved on the conceptual level for each configuration. However, the axial/two-tier radial configuration has an acceptable stowage approach which is somewhat more complex. The servicer mechanism design necessitates that the first arm segment be longer than half the spacecraft diameter. This is required to reach all the module locations. This condition causes the base of the arm to require hinging off to the one side so it can be stowed.

E. CONFIGURATION SELECTION

This section summarizes the logic and considerations that resulted in the selection of a preferred configuration for more detailed design of a space version of the servicer. The selection was made by NASA and Martin Marietta concurs with the choice. The major guidelines in the selection were:

- The space design will be one of five modular forms
- The totality of the five modular forms spans the totality of servicer requirements
- It is desirable that the engineering test unit be the same configuration as the space design
- The majority of the detail analysis will be for the space design
- The other four modular forms are to be worked at the conceptual level
- The space design configuration will establish the image for on-orbit servicing--complexity, reliability, cost, capability

This section also addresses the decision to select a configuration for a pre-prototype servicer, called the engineering test unit (ETU). As noted above, this is desired to be as close as possible to a flight design but in actuality this was not really practical.

1. Space Design Configuration Selection

The initial selection process, described in Sections A and B, had been predicated on identifying a servicer mechanism configuration that would have a capability to service the majority of probable serviceable satellite configurations. As was stated earlier, the preferred configuration from this evaluation (Configuration 10) was felt to be too complex. Instead, emphasis was placed on simpler configurations for initial development. These simpler forms are the five modular forms discussed in Sections B and D. Note that the five modular forms do not compete in the usual sense, rather each satisfies a different requirement, so each of the forms represents a distinct capability associated with a complexity. So the question becomes one of picking a combination of capability and complexity (as represented by one of the five modular forms) that will convey the best image of on-orbit servicing over the next few years.

The logic in this final selection process is shown in Figure III-23.

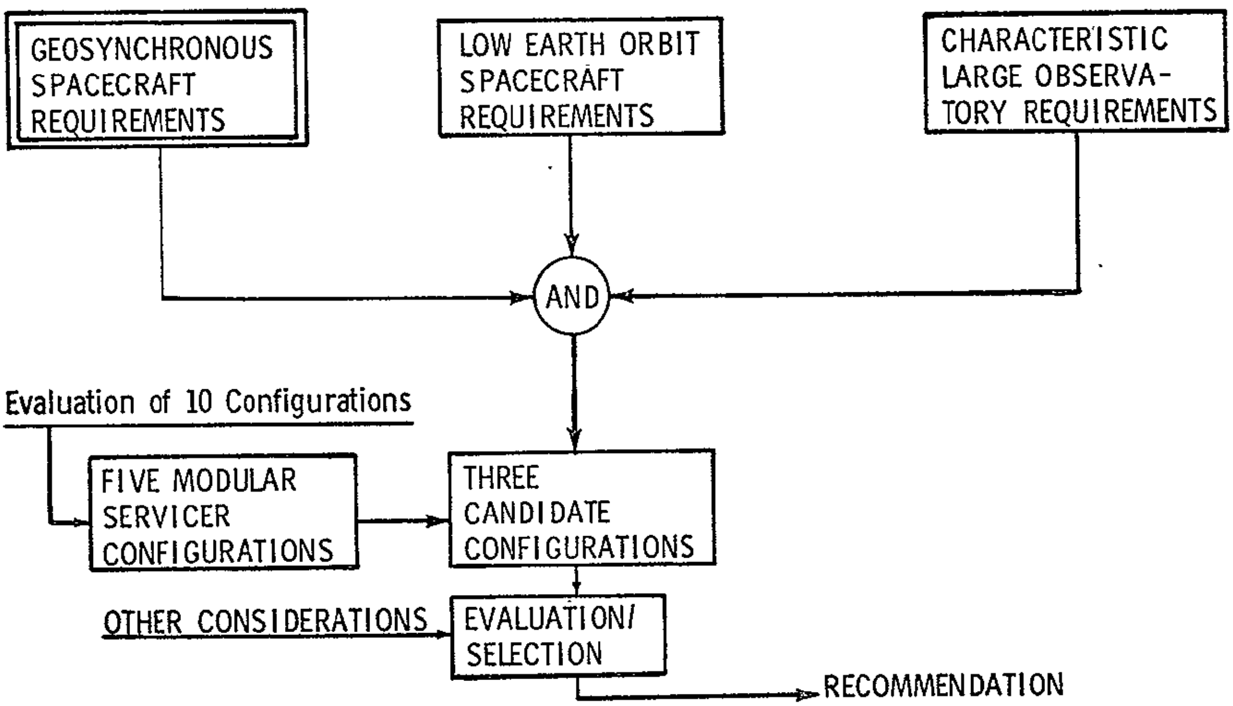

Figure III-23 Selection Logic

Three different classes of satellites were considered with the major emphasis being the geosynchronous spacecraft, DSCS II and SEOS, which were designed for servicing by TRW. The low earth orbit spacecraft considerations led to increased interest in single tier spacecraft, but no further identification of preferred module replacement directions.

The recommended configuration 10 servicer from the first analysis (Section B) was a three segment, seven degree of freedom concept. This plus our prior experience were used to generate the five modular configurations as discussed above. Of the five, three were felt to be good candidates.

These were: 1) axial; 2) axial/near-radial; and 3) axial/two-tier radial. The other two configurations--near-radial and two-tier radial--do not have an axial module removal capability. They are not compatible with the serviceable DSCS-II and SEOS satellites and are not as versatile as the other three forms. However, the near-radial configuration is the simplest of all five and shows some of the growth characteristics of the axial/near-radial configuration.

The selection between these three candidates was based on the considerations that follow. A surprisingly large number of considerations come into play in a selection process such as this. These considerations have been grouped as follows:

- Public image of servicing
- Balance of versatility vs simplicity
- Utility aspects
- Complexity aspects
- Engineering test unit aspects

A summary of the considerations in the first category is shown in Figure III-24.

- THE SELECTED CONFIGURATION BECOMES THE IMAGE FOR SERVICING SYSTEMS
  - Complexity
  - Cost
  - Capability
  - Reliability
- ALL THE GOOD ART WORK AND SYSTEM DESCRIPTIONS WILL BE FOR THE SPACE DESIGNS
- SIMPLER CONFIGURATIONS ARE EASIER TO PRESENT AND COMMUNICATION CAN BE MORE COMPLETE
  - General Introductions
  - Final Presentations
  - Movies
  - Demonstrations with Engineering Test Unit
- A CLEAR EVOLUTIONARY PATH FROM PRIOR WORK IS DESIRABLE

Figure III-24 Public Image Considerations

It should be made clear that while each of the other four modular forms was carried at the conceptual level, the prime effort was with regard to the selected configuration. This is particularly true in those presentations which were brief or where the selected configuration was used to set the stage for another discussion point. The lower the capability and simpler the configuration the easier it is to explain and the better the level of communication that results. However, too low a capability is not a good image where a spacecraft program requires and desires a greater capability.

The second set of considerations, summarized in Figure III-25, relates to the question of what is the best balance between versatility, or capability, and simplicity of the servicer image. This is felt to be a question that was best answered by NASA with their better knowledge of the agency's long-range planning and probable users of orbital servicing. The last two points on the figure are in direct opposition and illustrate the difficulty in making a selection. The criteria of "minimum constraints on spacecraft designers" was one of the major points during the first IOSS as well as in this current study. This balance between versatility and simplicity will be discussed further in Chapter V.

- SELECTION OF THE BEST BALANCE BETWEEN VERSATILITY AND SIMPLICITY FOR THE SERVICER IMAGE IS A NASA PREROGATIVE
- MINIMUM CONSTRAINTS ON SPACECRAFT DESIGN WILL LEAD TO USER ACCEPTANCE
    - Number of Tiers
    - Module Replacement Directions
- THE POTENTIAL USER SHOULD HAVE GREATER CONFIDENCE IN A SIMPLER SYSTEM AND THUS ACCEPT IT EARLIER

Figure III-25 Versatility vs Simplicity Considerations

Figure III-26 addresses the servicer utility aspects of the selection. The selection was made in the sense that the selected configuration is not the only one that will exist, rather it will represent orbital servicing for the next few years. It will be the image of orbital servicing and it will be the test case for most of the analyses and evaluations that will be conducted. As our knowledge increases over the years, it may be reasonably expected that better choices can be made.

- THE SELECTION IS NOT IN THE SENSE OF ULTIMATE UTILITY OR MATCHING TO A SPECIFIC NEED
- THE TOTALITY OF THE FIVE MODULAR FORMS SPANS THE SPECTRUM OF SERVICER REQUIREMENTS
- THE SELECTED CONFIGURATION SHOULD BE GROWABLE TO THE OTHER FOUR MODULAR FORMS
- IT SHOULD INCLUDE AN AXIAL CAPABILITY
- THE GEOSYNCHRONOUS SPACECRAFT (DSCS II AND SEOS) SERVICING REQUIREMENTS ARE SATISFIED BY A ONE-TIER/AXIAL SYSTEM
- THE MAJORITY OF POTENTIAL LEO SAVINGS CAN BE CAPTURED BY SERVICING ONE TIER
- THE CHARACTERISTIC LARGE OBSERVATORY SPACECRAFT DESIGN WAS NOT FAR ENOUGH ALONG TO INFLUENCE SERVICER CONFIGURATION SELECTION

Figure III-26 Servicer Utility Considerations

The approach was that all five modular forms would be developed, produced and available for operational use over the years. Then each spacecraft designer or program manager could select the modular form that best suits his needs. The order in which particular modular forms will be developed and enter the inventory will depend on the requirements of the first spacecraft programs to plan on using orbital servicing. However, all five forms appear to have distinct places in the eventual total spectrum of on-orbit servicing requirements.

As the five modular forms have been conceived to have a great deal of hardware commonality, any of the three candidate configurations can be grown to all five configurations. However, it is generally easier to work from the middle level of complexity to the ends rather than work from either end to the other.

Figure III-27 addresses the complexity aspects of configuration selection. If the selected configuration is simple then it is possible to perform the analysis at greater depth and in this sense learn more about the servicer configuration. This was particularly true when the limited resources of this fixed price contract were considered.

- THE SIMPLER THE CONFIGURATION, THE MORE WE CAN LEARN
- STUDY RESOURCES FOR SERVICER PRELIMINARY DESIGN ARE LIMITED. THUS, COMPLEXITY OF SERVICER CONFIGURATION INFLUENCES DEPTH OF ANALYSIS
- SELECTION OF A SIMPLE CONFIGURATION LEADS TO SIMPLICITY IN OTHER SERVICER ASPECTS
    - Primary Force Direction
    - Sequential versus Coordinated Motions
    - Mechanical Guiding versus Electrical Slaving
    - Decoupling of Control Stability from Structural Stiffness
- COMPARATIVE VOLUMETRIC CAPABILITY OF AXIAL/NEAR-RADIAL AND AXIAL
    - Same for Basic Form
    - Axial/Near-Radial is Better in Extended Forms
- ARM LENGTH OF AXIAL CONFIGURATION IS LESS THAN 55% OF OTHER CONFIGURATIONS
- WEIGHT OF AXIAL CONFIGURATION WILL BE 60 LBS LESS THAN THE OTHER CONFIGURATIONS

Figure III-27 Complexity Considerations

The module exchange task is quite simple in that there are only three kinds of trajectories involved: insert/extract, module flip, and transfer to next location. Each of these in itself is very elementary. Also all of the trajectories, end points, directions, and order of doing things is known before any mission. Thus, it is possible to use this task simplicity (as opposed to a general purpose manipulator where the tasks are not defined) to lead to a simple mechanical and control system design. Similarly the simplicity of a mechanism configuration can be carried over into many other aspects of the design and its use. This is particularly true if only one module removal direction (axial or radial) is considered.

The last three points on the figure address some benefits of the axial configuration as compared to the axial/near-radial form. However, it seems relatively easy to conceive of ways of extending the axial/near-radial configuration by simple additions. These ideas did not come as easily for the axial configuration.

Figure III-28 addresses the Engineering Test Unit (ETU) aspects of the selection process and were based on a desire to have the ETU closely represent the configuration selected for the space design. The key aspects were successful demonstrations and the limited resources available in this fixed price contract. As the ETU is to be the first step in an orbital servicing demonstration facility at MSFC, it can be expected to be expanded and updated as more is learned about the requirements of orbital servicing and the design of servicer systems.

- PREFER TO HAVE THE ENGINEERING TEST UNIT REPRESENT THE CONFIGURATION OF THE SPACE DESIGN
- THE SIMPLER THE ENGINEERING TEST UNIT; THE HIGHER THE PROBABILITY OF SUCCESSFUL DEMONSTRATIONS
    - Arm Length
    - Degrees of Freedom
    - Module Removal Directions
    - Variety of Interface Mechanisms
    - Number of Tasks to be Demonstrated

  ESPECIALLY WITH LIMITED RESOURCES
- ACCOMMODATE EXTENSION AND EXPANSION OF THE ENGINEERING TEST UNIT IN THE FUTURE AS IT IS USED IN THE DEMONSTRATION FACILITY

Figure III-28 Engineering Test Unit Considerations

Servicer mechanism arm length is a particularly important parameter in that it affects so many things which increase costs and required accuracy. These parameters include joint drive torque level, output gearing accuracy, gear housing machining accuracy, feedback sensor accuracy, servo loop gains and compensation, arm segment stiffness, electronic grounding and signal shielding, as well as system calibration.

The selected configuration should be such that it can be modified and updated in the future. It is important that the drive sizing be such that they can be used for most of the modular forms.

After carefully considering all these above considerations the axial/near radial configuration, the second modular form described in Section B, was selected. This configuration represented the best balance between capability and simplicity. It is completely comptabile with the geosynchronous spacecraft and can service most of the low earth orbit spacecraft with its more than one tier capability. The two directions of module removal provide more freedom to the spacecraft designer in terms of module location, interface mechanism applicability, and module removal direction. The natural growth options to the second tier do give a second tier capability although it is not as versatile as the axial/two-tier configuration.

The simpler axial configuration was felt to be too restrictive on the spacecraft designer even though it can service a large volume at the one-tier level. The axial/two-tier configuration is more complex and it was believed that its greater capability would not be required often enough to warrant its use as the orbital servicer image in the next few years.

2. ETU Configuration Selection

The selection of a configuration for the engineering test unit was based on the following desires:

- Must be a simple expression of a servicer
- Need to keep the demonstrations simple
  - Gives more confidence to user
  - Increases probability of good demonstrations
- Want to be able to get potential users involved
- Must keep overall size such as to fit within space available
- Prefer to have both bottom and side mounting interface mechanisms involved
- Prefer to demonstrate radial as well as axial module removal
- Resources available are marginal for axial form of servicer mechanism

The dilemma in this selection was a desire to have the ETU and the space design in the same configuration and yet stay within the contract funding limits. The probability of successful demonstrations is much greater for the shorter arm and fewer degrees of freedom of the axial configuration. Little real consideration was given to the axial/two-tier configuration because it does not look like the axial/near-radial configuration and because of its potential cost.

The resulting selection for the ETU was the axial configuration (the first of the five modular forms in Section B), however certain features were incorporated to permit realistic evaluation of near radial operations without the longer arm segments required of the axial/near radial. The shoulder and roll drives were sized larger than the axial-only servicer in order to perform radial module removal. The ability to do radial module removal was obtained by having part of the spacecraft mockup approximating a fifteen foot diameter and the other part at a diameter that will permit investigation of radial module removal for reasonably sized modules using the basic axial configuration segment lengths. The difference in direction of the end effector was accommodated by provision of a separate adapter.

An extension to this contract was proposed and approved subsequent to the decision process described above to incorporate an additional degree of freedom into the selected axial version. This will provide an axial/ near radial capability in the ETU and its controls before delivery to MSFC at the end of the follow-on contract.

IV. SERVICEABLE SPACECRAFT PRELIMINARY DESIGN

Design of an automated spacecraft that takes full advantage of the hardware and operational economies possible with orbital servicing is an important part of the overall space servicing concept. The total dollar investment in serviceable spacecraft in the Shuttle Era will be many times the investment in servicers. Therefore, spacecraft economies have a much larger potential payoff than economies in servicer design.

TRW has developed a great deal of information on serviceable spacecraft design concepts as part of a series of studies sponsored by SAMSO.[1, 2, 3] This information has been used extensively in this study.

One of the major characteristics of automated serviceable spacecraft will be the packaging of equipment in replaceable modules. A first step toward this modularization has already taken place. In recent years, spacecraft designers have developed configurations which separate spacecraft housekeeping functions from payload functions. The goal has been to minimize housekeeping hardware changes from one mission to another. The result is the "Standard Spacecraft Bus" concept. In this concept, subsystems such as power, attitude control and communications are located in a separate structure which has a simple mechanical interface to which a mission-peculiar module is attached. Additional flexibility is provided by using a central on-board computer for attitude control, command and data handling and general function management which allows large changes in function by only changing the software. Necessary hardware changes are facilitated by allowing a variety of sensors, actuators and black boxes. Further modularization is possible through the use of a data bus. This means that commands, timing and telemetry are multiplexed such that component interconnections are reduced to a few wires. A data bus facilitates accommodating a broad class

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1. "DSP Payload Study (U)," SAMSO-TR-72-266-2 (also TRW Report No. 16439-6392-RE-00), August 1972 (SECRET).

2. "In-Space Servicing of a DSP Satellite," SAMSO-TR-74-168 (also TRW Report No. 16439-6637-RU-00), March 1974. (Three Volumes - I and II SECRET, III Unclassified).

3. "Final Report, Servicing the DSCS-II with the STS," SAMSO-TR-75-135, dated September 1975 (Three Vol.), prepared by TRW Systems Group.

and number of payload elements, depending on the mission. Each element has its own data interface unit, thus greatly simplifying interconnections.

The cost effectiveness of a serviceable spacecraft is enhanced when non-repairable failures are minimized. Most of the spacecraft subsystems and mission equipment should be packaged into removable modules, called Space Replaceable Units (SRUs). It is desirable that the SRU sizes and attachments be standardized. Care must be taken in SRU sizing and configuration arrangement to ensure good thermal control. (In some instances thermal radiator area requirements dictate a minimum SRU size). Finally, the SRU attachment to the spacecraft backbone structure should be simple and provide reasonably large tolerances to misalignments.

The TRW IOSS Follow-On Study results show that it is possible to design serviceable spacecraft to perform a wide range of upcoming NASA, DoD and commercial missions. The spacecraft meet the mission performance requirements and are also designed to enhance orbital servicing. Automated payloads examined in detail are:

Geosynchronous Orbit

- Defense Satellite Communications System (DSCS-II)
- Synchronous Earth Observatory Satellite (SEOS)

Low Earth Orbit

- 1.2 meter X-Ray Telescope (NASA Payload HE-11-A)
- Large Solar Observatory (NASA Payload SO-02-A)
- Space Telescope (NASA Payload AS-01-A)

DSCS-II is representative of the many communications satellites in geosynchronous orbit. SEOS is a geosynchronous sensor mission. The three Low Earth Orbit payloads are representative of three major scientific mission areas - high energy astrophysics, solar physics and astronomy.

Separate serviceable spacecraft designs have been developed for the two geosynchronous missions, although there is a similarity in the two concepts. A serviceable Characteristic Large Observatory (CLO) has been developed to cover all three of the Low Earth Orbit missions. The housekeeping subsystem designs are sized for 1.2 meter X-Ray Telescope mission requirements, but they can be used for the Large Solar Observatory and Space Telescope missions. Only the mission equipment varies between the three missions.

In all cases, promising automated serviceable spacecraft designs have been developed that 1) perform the required missions, 2) interface with the servicer designs developed by Martin Marietta Aerospace, and 3) prove cost-effective for orbital servicing.

A. SERVICEABLE SPACECRAFT DESIGN REQUIREMENTS

There are several reasons why spacecraft will be designed for automated servicing in the Shuttle Era. Some of these are:

- Replacement of failed equipment - rather than replace an entire spacecraft due to the failure of a small number of elements, only the failed equipment is replaced, saving the cost of the other equipment.
- Update or replacement of mission equipment - rather than fly a series of 2- or 3-year missions, place the major telescope or other instrument in orbit and change the mission equipment periodically.
- Replacement of life-limited equipment - rather than expend a great deal of resources to develop extra long-life equipment, it may be more cost-effective to plan on a replacement during the satellite lifetime.
- Replacement of expendables - rather than carry a launch penalty for the large amounts of propellant required for long-life missions, such as geosynchronous satellites with North-South stationkeeping, plan for propellant replenishment after a certain number of years.
- Retrieval of recorded data - rather than transmit data by telemetry or have re-entry requirements for data packages, have the recorded data returned by a servicer vehicle.

The underlying motivation for each of these concepts is to save money over the long run. A serviceable spacecraft may be somewhat more expensive than an expendable version to perform any single short-duration mission. The cost savings is obtained by reducing the number of replacement satellites and by making more efficient use of the launch vehicle.

Satellites designed for automated orbital servicing will be quite different in outward appearance from current satellite designs (see Figure IV-1). This is because servicing adds a new set of requirements and may also change the way certain basic mission requirements are met. The internal equipment is often the same, but the packaging and general arrangement will be different. The general arrangement of an automated serviceable spacecraft must be compatible with many factors. Several of them will be discussed in the following sections.

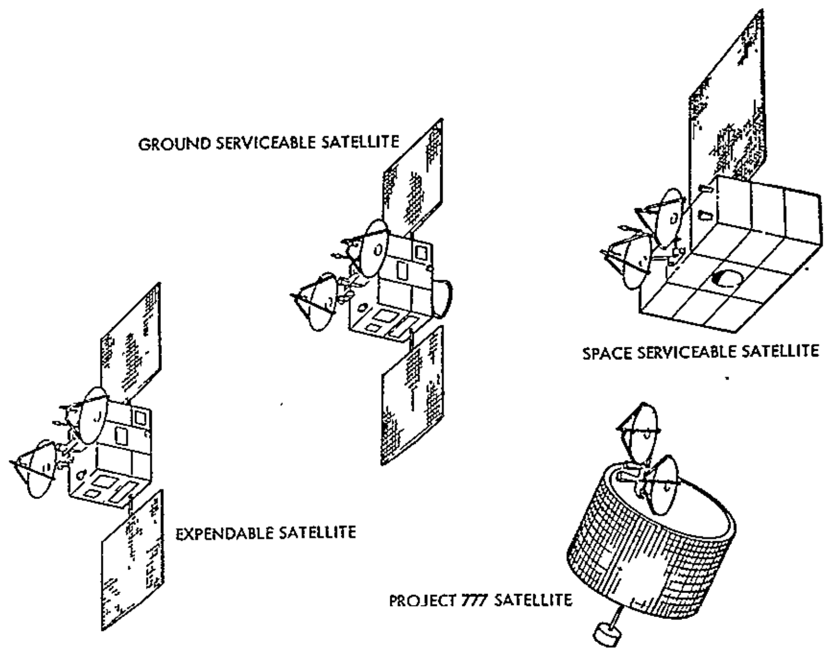

Figure IV-1 Expendable and Serviceable Satellite Configurations

1. Mission Requirements

A serviceable spacecraft must be able to perform the same basic mission as an expendable spacecraft. The major sensor or antenna may have a different design to take advantage of serviceability, but it must result in the same mission performance.

2. Housekeeping Functions

A serviceable spacecraft must provide the mission equipment with adequate attitude control, structural support, electric power, thermal control, commands, telemetry and electromagnetic compatibility. Any subsystem designs which result from serviceability requirements, such as a pigeon-hole type structure, must account for all of these considerations.

3. Space Replaceable Unit (SRU) Location and Operation

Each SRU must be located in a way that it can perform its basic function (i.e., a sensor must be able to point in its prescribed direction) but still be accessible to the servicer for possible removal and replacement. This can be a pigeon-hole concept, where the modules all face on a common plane and are removed axially, or a radial-extraction concept where modules move outward from a ring-type structure. Thermal considerations may require that a certain SRU have a face toward dark space. Also, the SRU must be designed to align the new equipment after servicing and to maintain this alignment during the remainder of the flight.

4. Servicer Interfaces

Any automated servicing will require that a servicer vehicle (Figure IV-2) rendezvous and dock with the serviceable spacecraft. The spacecraft must be capable of maintaining a passive attitude just before and

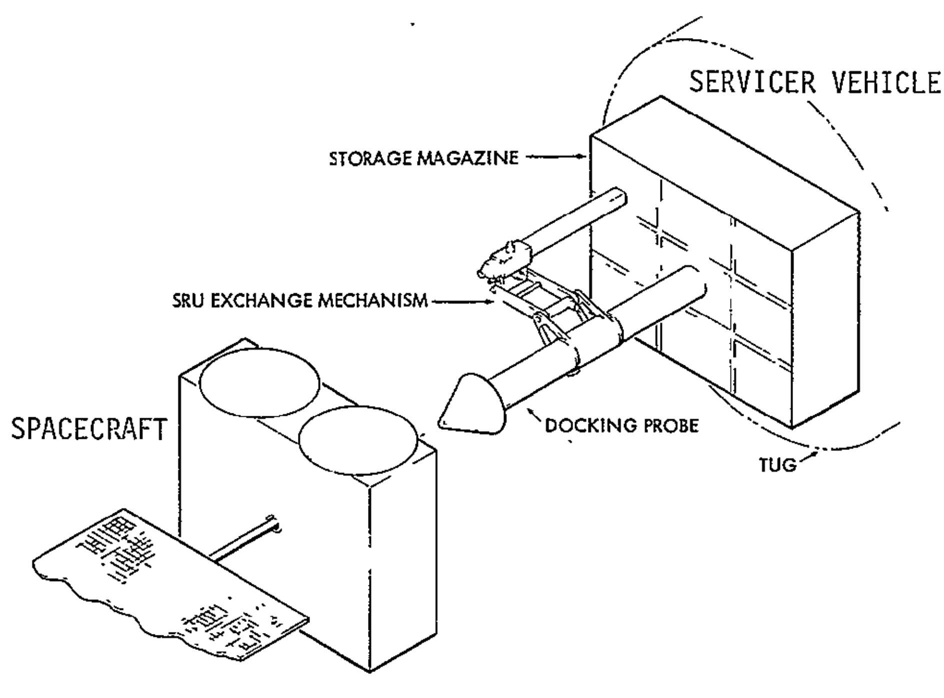

Figure IV-2 Serviceable Spacecraft and Servicer Vehicle Combination during docking, provide a docking receptacle to mate with the servicer, allow for locking to the servicer and be capable of unlocking and undocking after servicing has been completed. In addition, the SRU's must be compatible with the servicer transfer mechanism, they must not be too large or too heavy to be properly transferred and stored and they must have a common mechanical, electrical and fluid interface.

5. Space Transportation System (Shuttle) Requirements

The STS launch vehicle will be standard in the era of serviceable spacecraft and has certain special requirements that must be met. In particular, manned safety and contamination considerations may affect the serviceable spacecraft design and operational procedures. These requirements must be met by all spacecraft of the Shuttle Era, but the effect is especially true for serviceable spacecraft since they will interface with the Shuttle System several times during their lifetime.

Two of the space serviceable spacecraft from Figure IV-1 are shown mounted for launch with a Full Capability Tug from the Shuttle orbiter in Figure IV-3. The large diameter of the payload bay allows them to be mounted side-by-side.

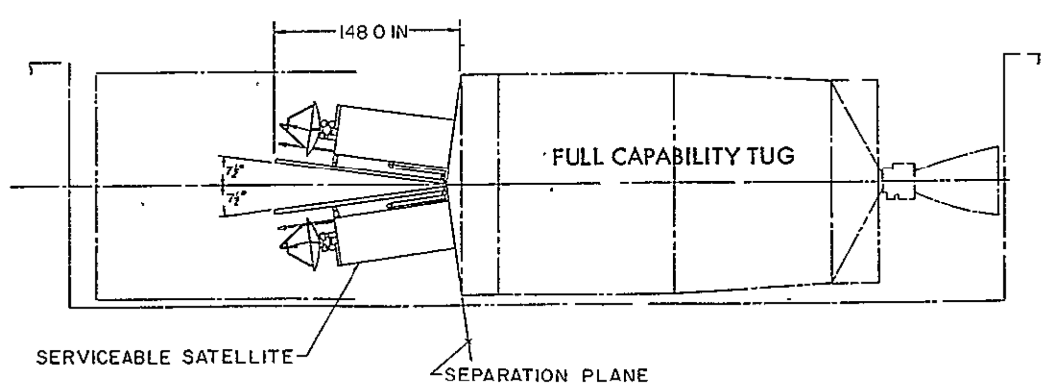

Figure IV-3 Serviceable Spacecraft Launch Configuration in Orbiter Payload Bay (Double Launch)

6. Other Considerations

Orbital servicing may impose other requirements on the spacecraft. One example is a North-South stationkeeping capability to overcome effects of nodal regression of the orbit plane. On military missions, simpler ways to overcome these effects are to reduce the pointing accuracy requirements, bias the inclination error by overcompensation at initial insertion, or compensate for the resulting pointing errors with equipment on the ground. When a fleet of satellites is to be serviced, it is very inefficient and time-consuming to have the servicer vehicle do orbit-changing maneuvers between satellites. It may be more cost-effective to pay the large propulsion weight penalty on each satellite for North-South stationkeeping. Commercial geosynchronous communication satellites will already have a North-South stationkeeping capability, so this is not a concern.

Another satellite design consideration that comes from servicing requirements is the use of a data bus. It is desirable to keep the number of wires going across the SRU interfaces as low as possible. Use of the data bus provides a substantial reduction in electrical connector pin count. It also permits an economical means of increasing the extent of fault diagnosis. Replacement of failed equipment by orbital servicing requires isolation of failures to the SRU level. The localization can probably be accomplished with no great difficulty using current techniques. However, it would be advantageous to localize failures to a much lower level so that corrective action can be taken toward product improvement.

TRW has restricted the design of serviceable spacecraft to require that they have a body-stabilized (3-axis) attitude control subsystem. It is no doubt possible to design a servicer that can capture, dock and service a spinning satellite but it is not considered to be worth the additional complexity and expense.

B. GENERAL SERVICEABILITY CONCEPTS

Detail design of serviceable spacecraft for several missions has produced several general serviceability concepts. In this section the characteristics of a Space Replaceable Unit will be discussed, as well as Non-Replaceable Units. In addition, some of the operational considerations that affect most serviceable spacecraft will be addressed.

Specific design examples will be presented in Sections C and D.

1. Space Replaceable Unit (SRU) Characteristics

The Space Replaceable Unit is the basic element of an automated serviceable spacecraft. The payoff potential of the whole orbital servicing concept can be negated if care is not taken in the design of the SRUs.

One important consideration in SRU design is to use proven equipment to minimize the design risk. A major outcome of the TRW serviceable spacecraft design studies is the determination that much current standard spacecraft equipment is usable in space serviceable designs without modification. This consideration is important in assessing serviceable subsystem feasibility, minimizing development costs and obtaining data for cost tradeoffs. Another conclusion is that high equipment reliability is still important. Although servicing can be done easily, launch costs for service flights are high and so servicing should be held to a minimum.

Modularity is a fundamental characteristic of an SRU. Related equipment is grouped in a way that failures can be detected and the whole module replaced as a unit. Equipment in a given SRU is usually from a single subsystem. This simplifies the job of assembly and integration and a single organization is responsible for the entire SRU.

In order to be able to easily service a wide variety of equipment, the SRU module should have standard dimensions and, most importantly, have a standard interface mechanism. It is often necessary to have a small number of non-standard SRU sizes on a given mission but this does not diminish the value of standard dimensions on as many SRUs as possible.

Standardization of module sizes applies to both mission equipment and housekeeping subsystems. Such standard modules could possibly be used for a number of different missions. This is one of the fundamental concepts, for example, used in the design of the Multimission Modular Spacecraft. It was not pursued in this study.

A standard SRU structural configuration, such as that shown in Figure IV-4, is based on the following criteria:

- Provide maximum functionability of SRU components during on-orbit satellite operations
- Provide a simple interface with the SRU guide and latch mechanism and the servicer arm end effector
- Be easy to slide in and out and provide adequate alignment with the spacecraft structure and other SRUs
- Be capable of axial or radial removal

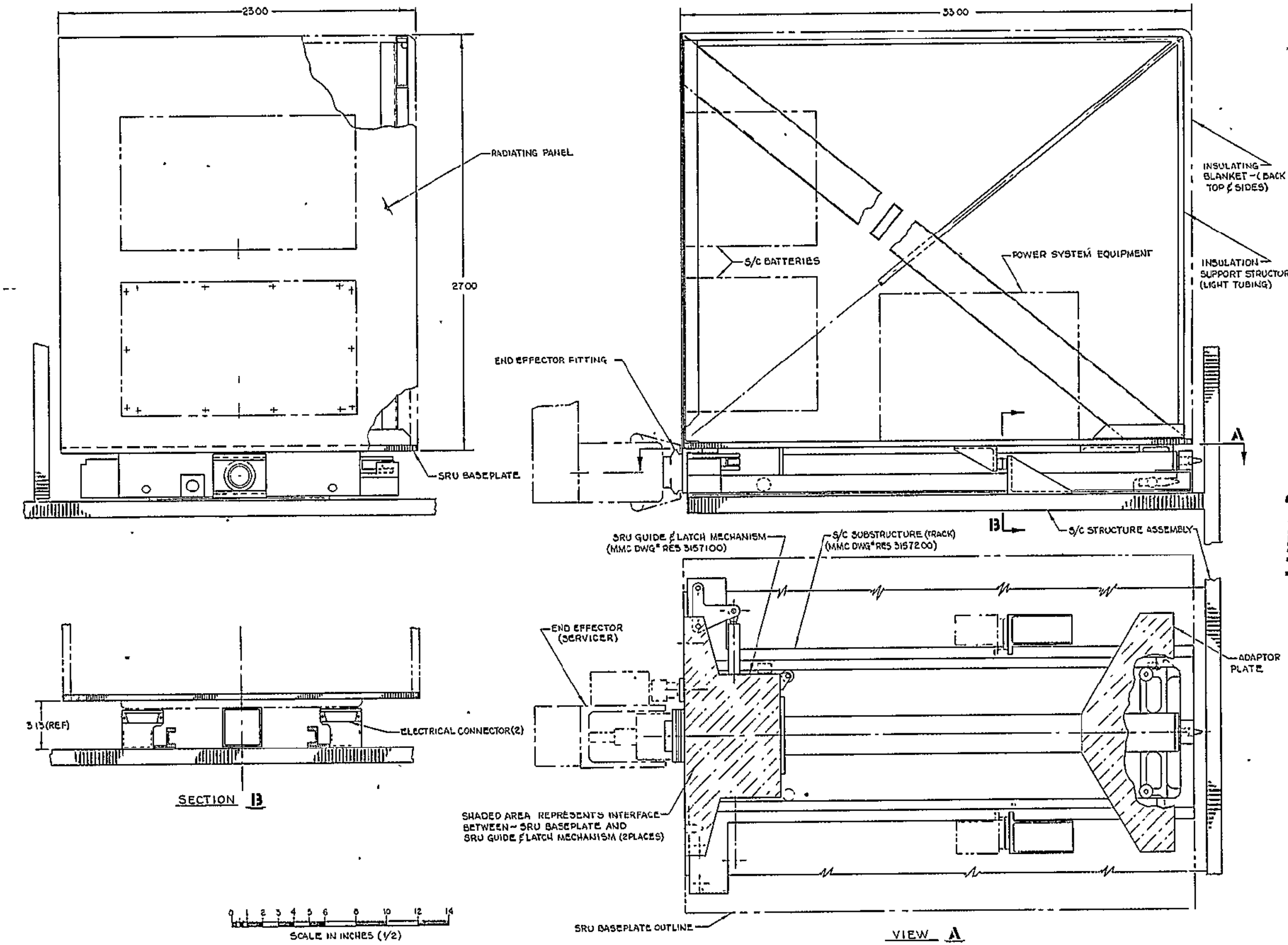

Figure IV-4 Typical Space Replaceable Unit (SRU)

- Be located in one or two tiers for easy access by the servicer arm
- Minimize the structural weight penalty

A single, minimum-access external opening in the spacecraft is used for each SRU. The SRU radiator, when required, uses this opening. Radiation panels are thermally isolated so that their radiating efficiency is preserved. This really suggests that the radiator should be structurally isolated except for its own support or any interface involvement for the particular component requiring the radiator, such as transmitter TWTs. The end effector fitting should be separate from the radiator surface and should not use up any more access opening area than necessary. A simple push-pull motion is used for SRU replacement.

The dimensions for the SRU in Figure IV-4 are 27"x35"x23". The SRU enclosure is a simple box and needs only one end and the baseplate for load-bearing structure. This particular SRU uses the side-mounting Interface Mechanism as designed by Martin Marietta. Electrical connectors and/or fluid disconnects can attach in two locations and a 50-pin plug/receptacle design is used. 50 pins should be adequate for any contingency. The usual number of pins per connector will be as follows:

3-wire data bus x 2 (redundant)	6
1-wire for power x 2 (redundant)	2
Basic for all SRUs	8 pins
Payload data or internal wiring	1 to 2
TOTAL	10 pins

The standardized interface with the end-effector of the servicer is shown in the views on the right. The SRU baseplate is constructed of a honeycomb sandwich material. It has either a conductive metal or non-conductive composite core, depending upon thermal requirements. Heat pipes can also be used to transfer heat from equipment mounted on the baseplate to a radiating wall or the front face.

Access provisions during spacecraft assembly are an important consideration in SRU design. The SRU panels have the equipment installed and checked out prior to spacecraft assembly. The principal SRU access requirement during assembly is to connect to the spacecraft wiring harness and for interconnections between the top and end panels. During integration and all-up systems testing, it is often necessary to check connections or remove malfunctioning components. It is desirable that any component on the equipment panels can be removed and replaced without the necessity of removing other components to obtain access. This is of considerable importance as unnecessary removal of equipment and connections increases the probability of incurring additional problems and historically has resulted in increased integration time and cost.

A key part in the design of an automated serviceable spacecraft is to determine the size and number of SRUs. Three variations in SRU size for axial removal only are shown in Figure IV-5 for the DSCS-II spacecraft to be discussed in detail in Section C1.

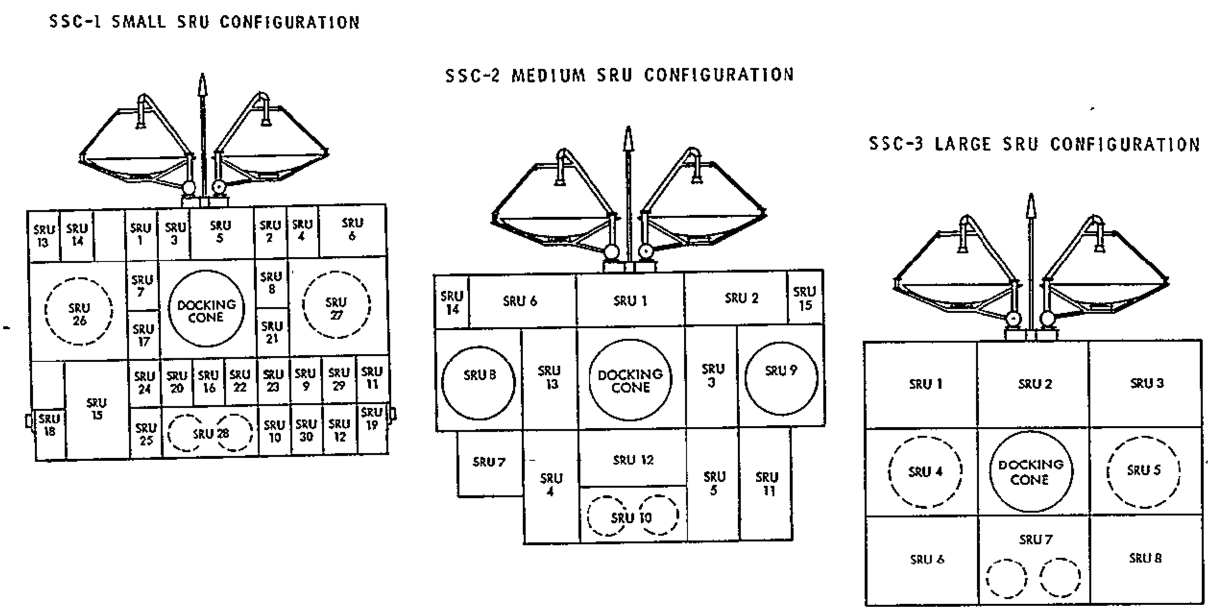

Figure IV-5 Variations in SRU Size for Serviceable DSCS-II

The SSC-1 configuration represents the smallest size module considered. The equipment is distributed so that a functional unit and its redundant member are located in an SRU wherever possible. In some instances (for example, batteries) an SRU contains only a single unit. Thermal radiation area, rather than equipment volume, sometimes dictates the minimum size of an SRU containing large heat-dissipating equipment (for example, high-level Travelling Wave Tube Assemblies).

The SSC-2 configuration has somewhat larger SRUs. Here equipment has been allocated as much as possible to minimize interfaces between functions. For example, the uplink and command chain is in one SRU, the telemetry downlink chain is in another.

The largest SRU configuration, SSC-3, has each subsystem contained in a single or, at most, two SRUs. This greatly reduces the electrical interfaces. Electric power and data signals for command and telemetry comprise the only tie between most SRUs. It should be noted that the communication antennas and their associated wave guide are packaged as SRUs in configuration SSC-3. In the other two configurations they are not replaced.

Equipment reliability is another consideration in determining what should be located within a given SRU. An attempt was made in the SAMSO studies to put the equipment with the highest failure rate for a given subsystem within a single SRU. The ability to do this depends upon the number and complexity of interconnections between SRUs, of course.

Equipment repair by replacement at levels lower than the black box (e.g., at the slice level) has not been examined in detail by TRW, but is conceptually feasible. It is not clear whether this capability would be particularly advantageous. Black boxes could be packaged so that slices could be removed and replaced. In addition, a level of fault diagnostics could be provided to pinpoint failure to a specific slice. However, it is not clear what spacecraft and servicer configurations would permit simple access to all of the hundreds of boxes.

2. Non-Replaceable Unit (NRU) Characteristics

Spacecraft equipment fails for a variety of reasons in spite of careful design reviews, the constant improvement of technology, parts screening, manufacturing process control and testing at all levels of assembly. The equipment failures occur over a period of time after the satellite starts its operation. Satellites fail or become inefficient performers as a result.

Some spacecraft failures are not repairable, either because the failed equipment is not modularized or because the failure is of such a nature that the spacecraft becomes incapable of maintaining a reasonably constant attitude for docking. These failure sources must be minimized. If many spacecraft need replacement, the whole point and cost benefit of servicing is lost. Fortunately, it has been found that only a few non-repairable failure sources must be tolerated on a serviceable spacecraft and that these failures have a minor impact on total program cost.

The Non-Replaceable Units (NRUs) on the housekeeping portion of a serviceable spacecraft can usually be limited to the primary structure, wiring harness and solar array. Each of these are high-reliability, long-life items. Depending on the mission, it may also be desirable to arrange the spacecraft configuration so that simple antennas and waveguide-coupled components are not replaced.

Some elements of the Reaction Control Subsystem and the attitude control electronics are vital to holding the spacecraft attitude angle with small rotation rates in order to dock with the servicer. These elements are replaceable but should be made highly reliable and redundant to ensure that the probability of a spacecraft being unserviceable is extremely small.

3. Operational Considerations

The design of a spacecraft for orbital servicing requires the establishment of certain operational requirements and ground rules. A good deal of effort was spent in the TRW serviceable military spacecraft studies to develop workable procedures for operation of the system. Certain overall concepts were developed and these spacecraft were designed to work within the concept.

The missions selected for study in the IOSS Follow-On Study are not drastically different from those in the previous study. Many of the operational considerations are similar. Therefore, the previous guidelines have been adopted, with only small modifications to update them.

Some critical operational considerations for a serviceable spacecraft are:

- How are failures detected and isolated?
- How is equipment shut down and turned back on before, during and after servicing?
- How are the replacement SRUs checked out?

Orbital servicing requires identification of the space replaceable unit (SRU) which has failed equipment so that the appropriate replacement SRU can be selected. Current spacecraft diagnostic techniques are probably adequate. Instrumentation is, however, generally skimpy and much time can be spent in localizing the failure. There is no doubt that the problem of localization would be greatly eased if the amount of instrumentation were increased. The data bus provides an efficient means of collecting and distributing added data.

A special investigation of failure diagnosis was conducted as part of the Serviceable DSCS-II study. The basic Project 777 satellite Digital Command and Telemetry Subsystem was used as the basis. The additional capabilities recommended to improve failure diagnosis and data recovery, plus the parts and power impacts are shown in Figure IV-6.

BOX	EXISTING DIAGNOSTICS	PROPOSED ADDITIONAL DIAGNOSTICS	PARTS INCREMENTS	ADDED POWER
EIA	3 STATUS BILEVELS (COMMAND PARITY, LENGTH, AND ADDRESS CHECKS)	4 ADDITIONAL STATUS SIGNALS (2 ANALOG, 2 BILEVEL) 1 TEST COMMAND RESULTING IN 20 BILEVEL OUTPUTS (ABOVE EQUIVALENT TO 1 RTU MULTIPLEXER)	16 IC'S 10 DISCRETES	50 MW
PCM ENCODER	1 STATUS BILEVEL (ON/OFF) 2 STATUS ANALOG	8 BILEVEL TEST INPUTS 1 ANALOG REFERENCE TEST INPUT (ABOVE EQUIVALENT TO 1/4 RTU MULTIPLEXER)	5 IC'S 8 DISCRETES	75 MW
PCM MULTIPLEXER	1 STATUS BILEVEL (ON/OFF) 2 STATUS ANALOG	8 BILEVEL TEST INPUTS 1 ANALOG REFERENCE TEST INPUT (ABOVE EQUIVALENT TO 1/4 RTU MULTIPLEXER)	5 IC'S 8 DISCRETES	75 MW

Figure IV-6 Additions to Project 777 Satellite for Increased Failure Diagnosis and Data Recovery

The serviceable spacecraft Electrical Power and Distribution Subsystem performs the Off/On function. It must permit selective shutdown of power to those SRUs which are to be replaced without shutting down the entire spacecraft and then turn on the new SRUs. It also must perform selective shutdown and turn on of those portions of itself needing replacement.

The basic design requirements for a serviceable EPDS are:

- Provide a primary power source compatible with the users' load requirements,
- Provide fault isolation for protection of the primary power bus,
- Incorporate power control switches for ease of servicing, and
- Be a simple, reliable design for minimum servicing requirements.

The first requirement is included to emphasize that the subsystem must first meet the basic requirements of any spacecraft EPDS. Fault isolation is more critical on a serviceable spacecraft, so the performance requirements for requirement 2 will likely be more severe. The third requirement will result in more switching since the criterion is used that any box must be OFF when it is removed.

The three fundamental issues in the design of a serviceable EPDS are the power bus distribution concept, the use of the disconnects and the method of power control during servicing.

The two main power bus distribution concepts which can be considered for a serviceable spacecraft are shown in Figure IV-7. The "spoke wheel" has power fed direct from the central source (Power SRU) to each SRU, and the "daisy chain" has power fed from SRU to SRU.

Figure IV-7 Serviceable Power Bus Concepts

The spoke wheel concept allows individual control of power to all SRUs. It requires more connector pins on the Power SRU and more lines in the wire harness.

A daisy chain design equalizes the number of connector pins in the Power SRU with those in the other SRUs and minimizes the total number of wires. However, the wiring required at each SRU to allow it to be removed without circuit interruption is quite complex. In addition, the power to individual SRUs cannot be controlled.

TRW has selected a spoke wheel power bus in all of its serviceable spacecraft designs. The number of pins on the Power SRU has been kept down to about ten. This allows for the use of a small, standard connector with a small connect/disconnect force.

There are three main purposes for electrical power disconnects on a serviceable satellite: 1) fault protection between the Power SRU and the using SRU, 2) removing power from the using SRU to allow it to be serviced, and 3) switching power to a standby redundant unit within a using SRU. The location of the disconnect depends on the purpose. The disconnect must be in the Power SRU for fault protection, for SRU power OFF, it could be in either the Power SRU or the using SRU, and for element switching the disconnect is usually within the using SRU. The switch to change power between redundant elements within a SRU could be located in the Power SRU, but it would require an extra set of wires.

Power during servicing can be controlled by turning everthing OFF (single bus) or by turning individual units or groups of units OFF (multiple bus). A Power Control SRU alone can be used for the simple bus concept - there is one load power disconnect switch. This concept accommodates either spoke wheel or daisy chain distribution. It is difficult to fault isolate for problems between the Power SRU and the using SRUs, however.

Adding a set of Auxiliary Power Distribution Units (APDU) allows for individual SRUs or groups of SRUs to be selectively turned OFF or ON. This multiple bus concept is more complex, requiring one power switch per using SRU or group of using SRUs. The distribution must be the spoke wheel type.

A single bus is adequate, but use of the multiple bus provides a great deal more flexibility and makes fault isolation simpler. APDU reliability is critical since it is in series with the reliability of the SRU which it serves. Its reliability must be high enough to assure that an SRU will not be lost due to an APDU switch or circuit breaker failure.

C. SERVICEABLE GEOSYNCHRONOUS SATELLITES

The spacecraft selected to demonstrate the application of automated servicing to geosynchronous satellites are:

- Defense Satellite Communications System - Phase II (DSCS-II)
- Synchronous Earth Observatory Satellite (SEOS)

These are both in the maintenance applicable set of the original study and represent the two major geosynchronous missions - communications and earth-sensing. They also have one-tier SRU configurations with axial removal.

The housekeeping subsystems for most geosynchronous spacecraft are quite similar due to the nature of the orbit. In general, body-stabilized geosynchronous spacecraft are characterized by:

- A central body containing a payload package (such as an antenna farm or sensor) that is continuously oriented toward the earth.
- A flat solar array that is rotated once a day with respect to the central body about a North-South axis to maintain a fixed sun orientation.
- A Thermal Control Subsystem that takes advantage of the north and south faces of the central body that are almost always pointing toward dark space.

These general characteristics result in fairly standard spacecraft subsystem designs. Therefore, servicing concepts developed for the two candidates will be pertinent to a wide range of other missions. For example, a preliminary check was made in the DSP In-Space Servicing Study[2] of the applicability of the "DSP concept" to two COMSATS, a NAVSAT, and a small low-altitude sensor satellite. No incompatibilities were discovered.

1. Space Serviceable DSCS-II

The Defense Satellite Communications System - Phase II (DSCS-II) mission is representative of a large class of geosynchronous communications satellites and has been selected for use in the IOSS Follow-On Study.

This section contains a summary of the serviceable DSCS-II spacecraft.[3] Each of the housekeeping subsystems will be described since they incorporate the results of all the TRW serviceable satellite studies.

DSCS-II is a military communications satellite with moderate power and pointing requirements. The current project (called Project 777) has several spacecraft in orbit. The Project 777 satellite, shown earlier in Figure IV-1, is a spinner with a despun platform which supports the 4-channel communications transponder payload and its four antennas.

Several body-stabilized configurations of the DSCS-II were developed to analyze serviceability. Body (3-axis) stabilized satellites were selected since they are much easier to service than spinners. The docking probe and transfer mechanism used in the IOSS are compatible with the space serviceable design. The body-stabilized spacecraft subsystem designs are based on FLTSATCOM, another military communications satellite now under development at TRW. The new configurations, also shown in Figure IV-1, are 1) expendable, 2) ground serviceable, and 3) space serviceable. An economic tradeoff analysis shows that the largest savings are possible with the space serviceable concept. The space serviceable DSCS-II is the only one that will be discussed here.

Three space serviceable configurations were developed, differing primarily in the size and number of replaceable modules. These differences have been discussed in Section B1. The SSC-2 configuration, with medium-size modules, has been selected for use in the IOSS Follow-On Study.

A close-up of the serviceable DSCS-II is shown in Figure IV-8. The main features of this configuration, featuring the serviceability aspects, are:

- Geosynchronous Orbit
- DSCS-II Performance
- Body-Stabilized, Single Solar Array
- North-South Stationkeeping
- Ten-Year Design Life
- 99 in. x 128 in. x 40 in.
- 2459 lb (Beginning of Life)

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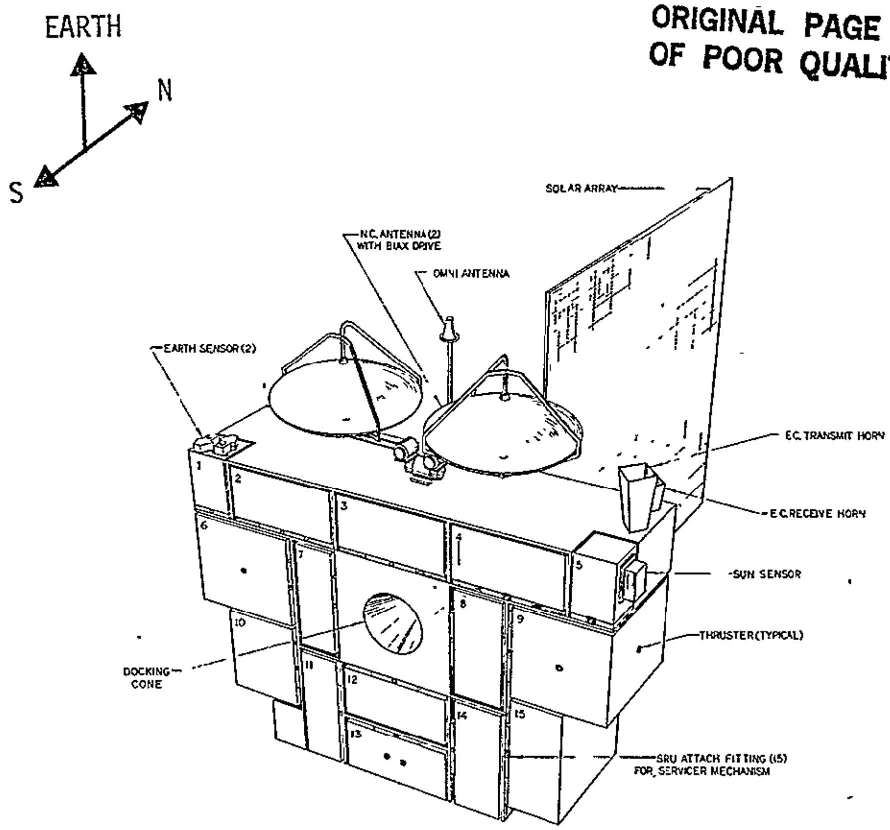

Figure IV-8 Space Serviceable DSCS-II Configuration

- 15 Space Replaceable Units (SRU), Single Tier
- Single Central Docking, Axial SRU Removal
- Largest SRU = 40 in. x 40 in. x 32 in.
- Heaviest SRU = 444 lbs
- Servicer Reach = 23 in. minimum, 72 in. maximum

This "medium size" SRU design, shown in detail in Figure IV-9, is based on co-locating equipment as possible to minimize interfaces between functions. Thus the uplink and command chain is in one SRU; the telemetry and downlink chain is in another. The two reaction wheels are in one SRU, as are the three batteries. The narrow-coverage antennas with their bi-axial drives, as well as the horn antennas, are not serviceable in this SRU configuration. If required, the antennas could be included with one or more of the Communications Subsystem SRUs.

A breakdown of the Serviceable DSCS-II mass properties, Table IV-1, shows that 1920 pounds, or 81% of the total spacecraft weight, is spacereplaceable. The major items which are not serviced are the basic structure,

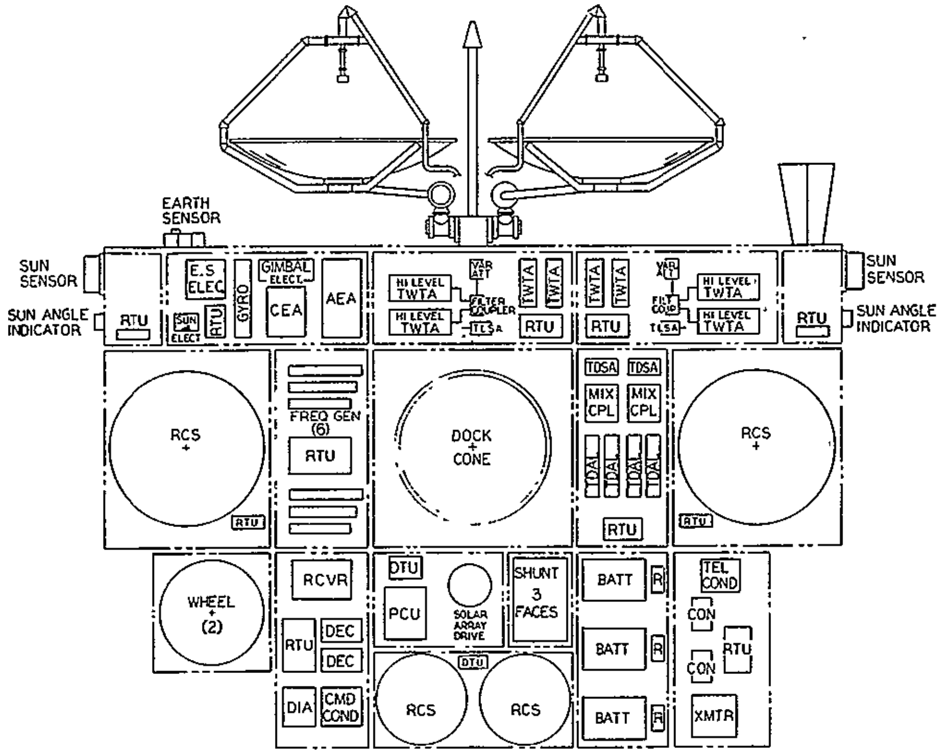

Figure IV-9 Serviceable DSCS-II Equipment Arrangement

Table IV-1 Serviceable DSCS-II Mass Properties by SRU and NRU

	Number of SRUs		Weight - Lb
SRU (Replaceable) Weight			
Communications	4		268.3
Telemetry, Tracking & Command	2		126.9
Attitude and Velocity Control	4		217.5
Electrical Power	2		206.9
Stationkeeping Reaction Control	2		899.4
Dry		(160.2)	
Propellant & Pressurant		(739.2)	
Reaction Control (other)	1		200.8
Dry		(55.4)	
Propellant & Pressurant		(145.4)	
Sub Totals	15		1919.8
NRU (Fixed) Weight			
Communications			116.9
Telemetry, Tracking & Command			8.2
Electrical Power			107.5
Structure & Thermal			212.0
Sub Total			444.6
Contingency			94.6
SATELLITE ON STATION (BEGINNING OF LIFE)			2459.0

the solar array (the solar array drive is replaceable), the narrow-coverage antennas and their biax drives, the horn antennas, the omni antenna, and the shunt element assembly.

It should be noted that the Serviceable DSCS-II is heavier than the Project 777 satellite and an expendable body-stabilized version for two major reasons. One is that North-South stationkeeping is required to keep the fleet to be serviced in a common orbit plane. To do this adds 400 lb of propellant and associated hardware for a ten-year mission.

The other major reason for a heavier spacecraft is that a single-sided solar array is used. This makes it easier to replace the critical solar array drive, but increases solar-pressure disturbance torques. The basic structure of a serviceable satellite is also less efficient than for an expendable satellite. It is impossible to assess the tare weight due to adding a servicing capability to DSCS-II because of these differences between the capabilities.

Each of the housekeeping subsystems for the Serviceable DSCS-II will be discussed separately. Maximum use is made of Project 777 and FLTSATCOM equipment. The Communications Payload and most of the Tracking, Telemetry, and Command Subsystem are directly from Project 777. The Attitude and Velocity Control and Reaction Control Subsystems are close to the FLTSATCOM designs. The major design changes for serviceability are the data bus signal distribution concept, the entire Electrical Power and Distribution Subsystem and the incorporation of North-South stationkeeping into the AVCS.

a) Structure Subsystem

The Structure Subsystem, shown in Figure IV-10, is designed to maximize the volume available for components to be carried in the SRUs, to maximize radiator area for thermal control, and to interface with the Servicer. This type of structure is less efficient than can be designed for expendable spacecraft, but the difference is not great.

The docking cone for servicing is located in the center bay of the egg-crate like structure. The walls of this bay form a fully-closed box, as do all of the internal SRU mounting structures. The walls are one-inch thick honeycomb core sandwich panels. Tubular support struts are located on each side to help support the wide upper structure. To save weight,

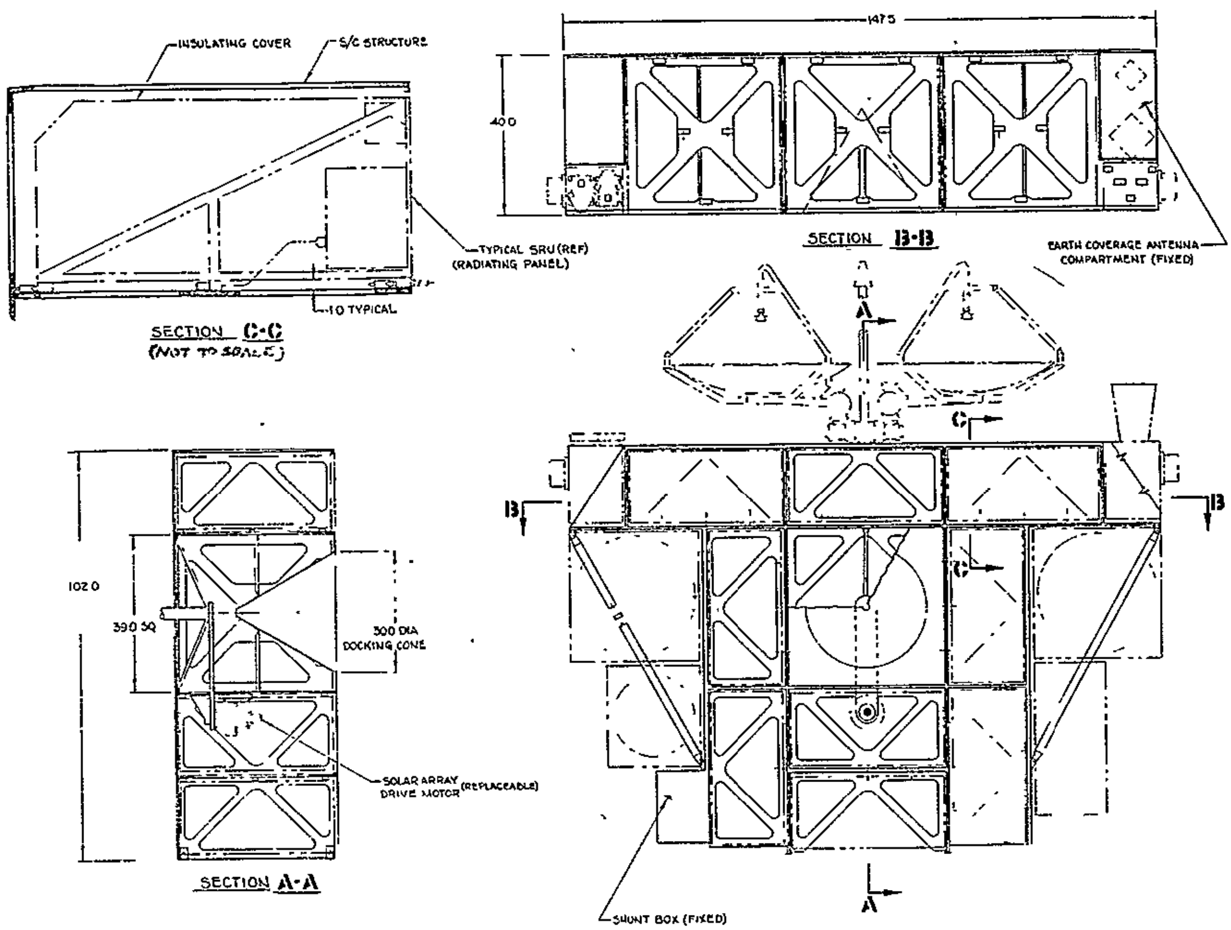

Figure IV-10 Serviceable DSCS-II Structure Subsystem

the outboard SRUs are not covered by spacecraft structure on their external faces. This is a primary consideration in the design of the thermal control for each of these SRUs. The solar array side of the structure (to the left in Section A-A) is closed by half-inch thick honeycomb panels. A lightweight tubular structure supports the folded solar array.

b) Attitude and Velocity Control Subsystem

The overall AVCS requirements for the Serviceable DSCS-II are shown in Table IV-2. The subsystem design selected, based on FLTSATCOM, is shown in the block diagram of Figure IV-11. It consists of the earth sensor, rate gyro and two sun sensors for sensing attitude, the control and auxiliary electronics for processing sensor information and generating commands, and the actuating devices, consisting of the reaction wheel and reaction control thrusters. In addition, there are drives for the solar array and two narrow coverage antennas. The secondary power converter, which is part of the EPDS, provides the required AVCS power.

Table IV-2 AVCS Requirements for Serviceable DSCS-II

MISSION

- ACS MUST OPERATE DURING ECLIPSE
- MUST OPERATE DURING SUN/MOON INTERFERENCE SEASONS
- NO GROUND INTERVENTION FOR 21 DAYS

ACS PERFORMANCE

- ACQUISITION - ACQUIRE FROM ARBITRARY ATTITUDE AND RATES OF 0.45 DEG/SEC (P, Y) AND 0.75 DEG/SEC (R)
- ON-ORBIT (NORMAL) - POINTING ACCURACY
  - ROLL - 0.15 DEG
  - PITCH - 0.15 DEG
  - YAW - 0.5 DEG
- ON-ORBIT (ΔV) - DEGRADED ACCURACY PERMITTED
- GENERAL - STRUCTURAL INTERACTION SHALL NOT DEGRADE AVCS PERFORMANCE

LONGITUDE AND LATITUDE STATIONKEEPING

- LATITUDE (NS) - ± 0.1 DEG (160 FPS/YR)
- LONGITUDE (EW) - 7 FPS/YR
- ΔV = 120 FPS FOR IN PLANE VELOCITY AND POSITION ERROR AND INITIAL DRIFT RATE CORRECTION
- ONE REPOSITIONING - ΔV = 288 FPS

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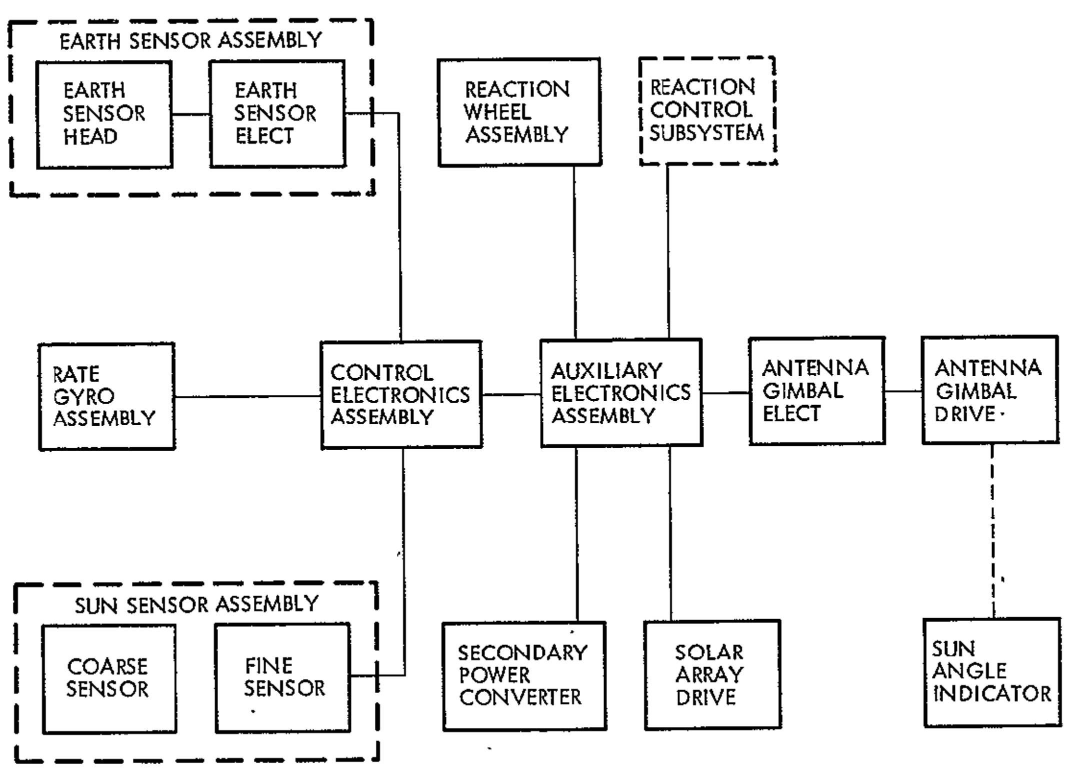

Figure IV-11 Serviceable DSCS-II Attitude and Velocity Control Subsystem Block Diagram

The body-fixed momentum wheel spins normal to the orbit plane. Its momentum provides dynamic coupling of the roll/yaw axes and precludes the need for a yaw sensor for normal on-orbit control.* The speed of the wheel is varied to provide control about the pitch axis. Periodically, the wheel speed is reduced by dumping momentum with the pitch thrusters. Roll/yaw and pitch control are in response to error signals generated by an earth sensor. Actuation for roll/yaw control is provided by offset thrusters in the Reaction Control Subsystem (RCS). Solar array positioning is accomplished by a continuous one-per-day rotation of the solar array drive.

The narrow coverage antennas are gimballed about two axes and have a nodding program built in to compensate for orbital inclination effects on pointing. The antennas, drives, electronics and sensor are the same as used on the Project 777 satellite.

The AVCS design has been analyzed to determine critical failures which could prevent holding attitude and low attitude rates and prevent docking with the Servicer. It was found that there are enough alternate sensor and control paths that the probability of this class of failure is low. Certain functions were identified as vital (e.g., yaw thrusters) and will require increased redundancy.

To allow docking to be performed during command-link failures, there is an automatic transfer to thruster control (using earth and sun sensors) if attitude errors become and remain large. It is assumed that docking will not be attempted unless the sun sensor is properly illuminated. Additional features may be required in the AVCS to reduce vulnerability to a bump during docking. Upon a completed docking, the internal thruster commands will be inhibited and the wheel speed held at its last value before docking. Wheel speed and other vital functions will be under hardline control from the Tug after docking. Control will revert to the ΔV mode immediately after separation, then to normal cruise control upon receipt of the appropriate RF command.

* In Figure IV-8 the solar array rotates about the pitch axis, the antennas point along the yaw axis and the roll axis is the orbit velocity vector.

c) Telemetry, Tracking and Command Subsystem

The up and down links used in the Project 777 satellite (shown at the left in Figure IV-12) are retained in the Serviceable DSCS-II TT&C Subsystem.

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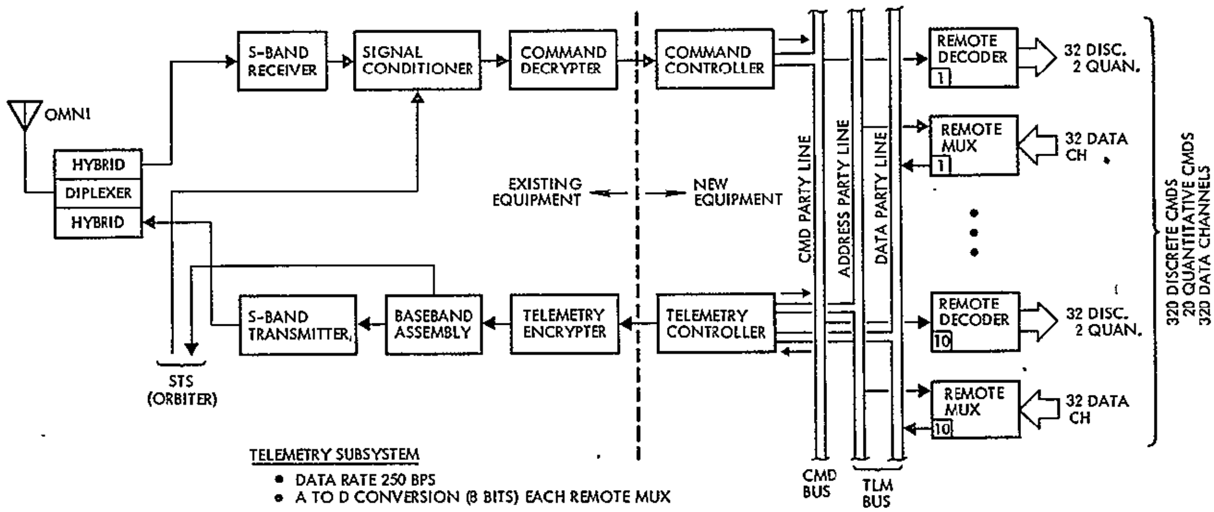

Figure IV-12 Serviceable DSCS-II Telemetry, Tracking and Command Subsystem

New equipment, shown to the right of the dotted line, replaces the Command Decoders, Electrical Integration Assembly, Switching Logic Assembly, PCM encoders and PCM multiplexers used in the Project 777 satellite. Certain switching functions packaged in the EIA and SLA are transferred to the SRUs.

This new equipment forms a modular data transfer network, or data bus. The data bus is divided into two functions - the Command Bus and the Telemetry Bus. The Command Bus consists of the command controller, the command party line and the remote decoders. Each remote decoder (RD) is capable of executing 32 discrete commands or 2 quantitative commands. This is also the basic unit module size for command expansion or contraction (e.g., add/delete 32 discrete/2 serial each time). As a command is received from the decrypter, the controller processes and formats the data for the party line transmission. Only the uniquely addressed RD will respond to the party line command transfer. Once the transfer is validated, the command is decoded and executed by the selected RD.

The Telemetry Bus contains two party lines, remote multiplexers (RM and a telemetry controller. The two party lines provide a data loop. One is the request for data or address party line and the other is the data return (data party line) or feedback to the controller. Each RM can handle 32 block selectable data channels with any mix of analog, digital and bi-level. Each RM has an integrated circuit 8 bit A/D converter for processing of analog data.

Each SRU has one or more standardized Remote Decoders and Multiplexers depending upon the number of commands and extent of telemetry data require In addition, data produced in one SRU can be transferred to another SRU by a slight modification of the system.

The telemetry controller uses a read-only memory (ROM) for storage and generation of the downlink telemetry format. The ROM provides addresses and control for selection of the individual data channels. By this means, a programmable and hence flexible format generator is obtained

The telemetry controller transfers the ROM data request or address onto the address party line. The uniquely addressed RM responds by converting (if required) and collecting the appropriate data and transferring the data to the controller via the data party line. This data in turn is down-link formatted and sent to the telemetry encrypter.

The data bus design for the TT&C Subsystem permits a large increase in telemetry data with small additional cost for Remote Terminal Units. The key features of a data bus are:

- Projected Lower Cost and Higher Reliability - The use of large scale integration (LSI) and standardized replicated functions can reduce parts count and lower both development and manufacturing costs.
- Modularity - The use of remote terminals provides flexibility to accommodate changes and advancements in requirements. This also makes a data bus amenable to totally modular spacecraft.
- Electrical Integration - The connector pin count is greatly reduced and electromagnetic interference is kept low by fewer and shorter cable runs.
- Spacecraft Integration andTest - Test, diagnostic and fault isolation data gathering is facilitated by simply connecting onto the bus with a standard remote terminal. This does require a minor controller change or to designate the RT as a controller.

- Data Management - The RT can do real time data processing on the I/O data from the subsystems and mission equipment. This can lower data rates and throughput for ground and any on-board processing.
- Redundancy Management - The data bus can conduct continuous, pre-planned diagnostics on the entire spacecraft. (It could respond to on-board failures immediately with emergency control sequences with slightly more advanced controllers and RTs.)

d) Electrical Power and Distribution Subsystem

The EPDS provides primary power to the rest of the satellite. Secondary power conversion is not centralized but is provided as needed in each SRU. This approach was adopted to minimize pin count in the SRU electrical connectors.

The general design philosophy for a serviceable EPDS has been discussed in Section B3. The design for the Serviceable DSCS-II, shown in Figure IV-13, retains the basic shunting system design used in the

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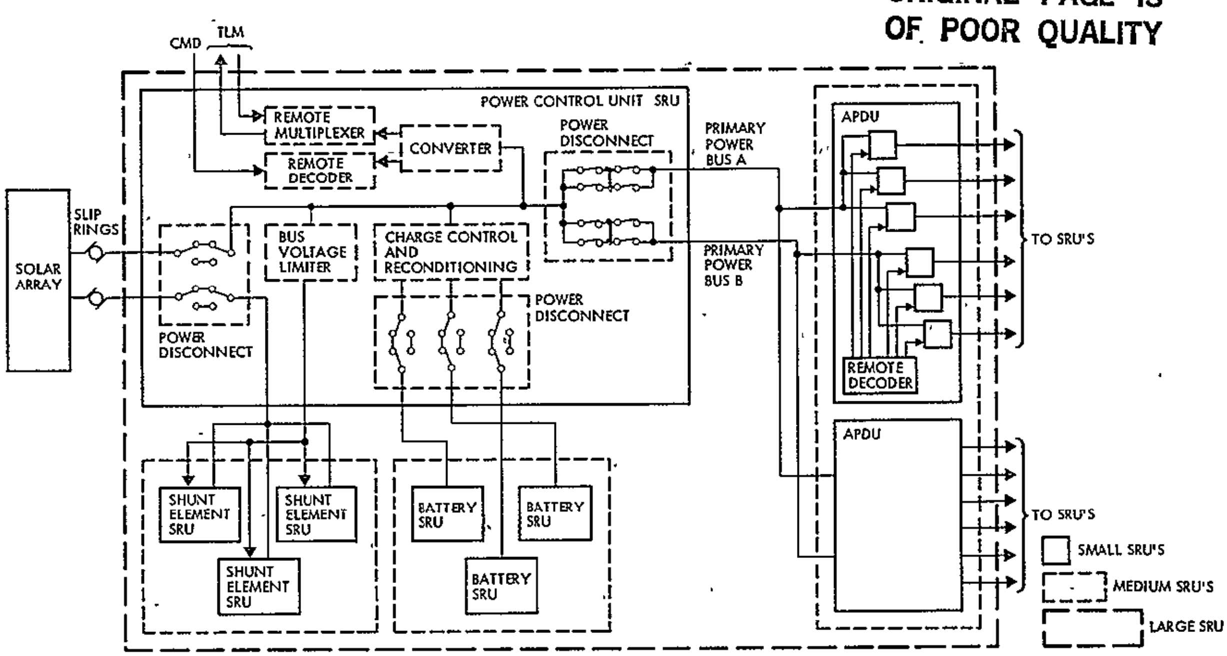

Figure IV-13 Serviceable DSCS-II Electrical Power and Distribution Subsystem

Project 777 satellite, as well as in many other satellites. The subsystem permits shutdown of power to those SRUs which are to be replaced without shutting down the entire satellite. It also permits selective shutdown of those portions of the power system if they need replacement. If the power control unit requires replacement, it can be shut down by command from the servicer. It is then reset after servicing and before the servicer disconnects from the satellite.

The entire EPDS can be packaged in one SRU, or in successively finer grained sizes. The selected configuration has the EPDS in three SRUs.

Two primary power buses are used for redundancy. The use of an Auxiliary Power Distribution Unit is recommended because of the improvement in system flexibility, but may be eliminated by a weight/cost/reliability tradeoff. Circuit breakers in the APDUs protect against a fault to ground between the APDU and user SRU. This failure mode may be precluded by careful design of the power distribution harness.

e) Reliability

The Serviceable DSCS-II is modularized to permit replacement of SRUs on orbit. The SRUs are often replaced for preventive maintenance when redundancy is lost. The reliability of each SRU is not, therefore, of fundamental importance and has not been separately computed. Total satellite reliability has, however, been computed as a matter of interest. The reliability vs. time in orbit is shown in Figure IV-14 for both

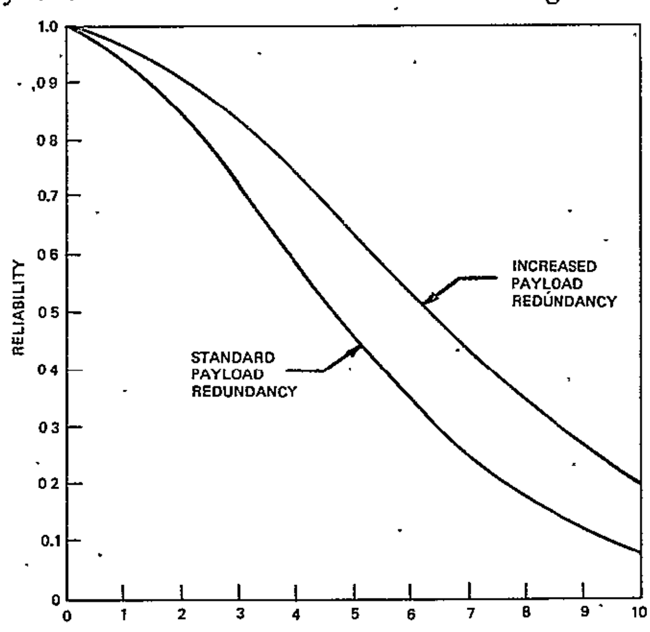

Figure IV-14 Serviceable DSCS-II Satellite Reliability versus Time On Orbit standard and increased redundancy in the communications payload. It has been found that satellite reliability is relatively insensitive to the number of SRUs, but improves measurably with increased payload redundancy.

2. Space Serviceable SEOS

The Synchronous Earth Observatory Satellite (EO-09-A) mission optically monitors environmental phenomena containing significant temporal variability. These phenomena are divided into earth resources observables and meteorological events. The earth resource observables are covered in the spectral range of 0.42 micron to 23 microns by 58 spectral bands.

Meteorological events to be monitored are severe storms, hurricane and tropical storms, flash floods, frost and freeze, clear air turbulence, fog, lake and sea breezes, air pollution, and weather modification and experiment assessment.

Perkin-Elmer conducted a Phase-A study of the SEOS mission[4] for NASA Goddard Space Flight Center, with TRW as a subcontractor for spacecraft design. The recommended Large Earth Survey Telescope (LEST), shown in Figure IV-15, uses a 1.5-meter aperture Abe Offner four-mirror anastigmat optical system with a 375 km x 750 km projected field of view operating with pushbroom arrays in the focal plane. The last mirror slides back and forth between the earth resource (ER) and meteorological (MET) sensor packages, which contain the focal plane optics and detectors.

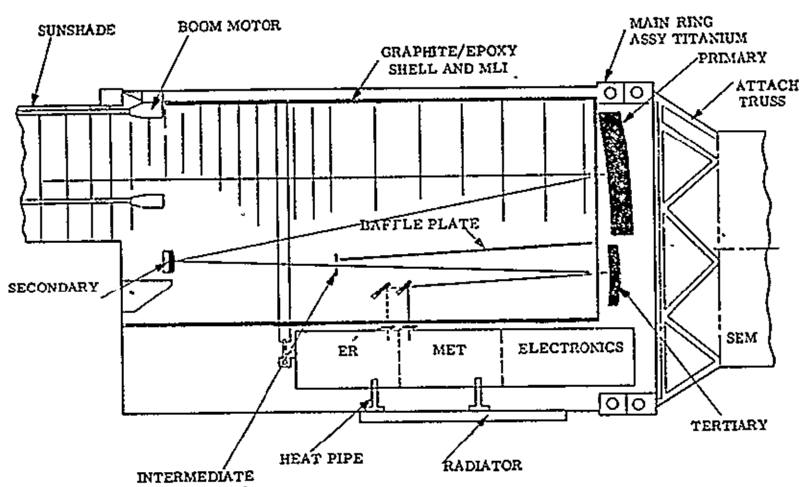

Figure IV-15 Large Earth Survey Telescope Configuration

4. "Synchronous Earth Observatory Satellite/Large Earth Survey Telescope (SEOS/LEST) Final Report," by John J. Russo, The Perkin-Elmer Corp., October 1975 (Five Volumes), NASA Contract NAS5-20075.

The LEST/sensor concept shown meets all the requirements for both types of missions. It is larger and more complex than the equipment used by General Electric in other SEOS studies.[5] However, the GE design will not meet all of the mission requirements. Technical personnel at NASA/GSFC have indicated[6] that the tighter requirements will most likely be imposed when the program is funded. Therefore, we have used the Perkin-Elmer telescope and sensor concept to develop a Serviceable SEOS.

The principal SEOS/LEST characteristics are summarized in Table IV-3 and an expendable version of the spacecraft is shown in Figure IV-16.

Table IV-3 Characteristics of Synchronous Earth Observatory Satellite with Large Earth Survey Telescope

ORBIT	GEOSYNCHRONOUS
EARTH RESOURCES & METEOROLOGICAL IMAGER	VIS TO FAR IR (0.42 MICRON TO 23 MICRONS)
3-MIRROR OFF-AXIS SYSTEM	1.5 METER EFFECTIVE APERTURE
RESOLUTION CAPABILITY	100 METERS VIS 800 METERS FAR IR
PASSIVE FOCAL PLANE COOLING	HEAT PIPE & RADIATOR
SENSOR POWER	129 WATTS AVERAGE 180 WATTS PEAK
ATTITUDE CONTROL	BODY STABILIZED

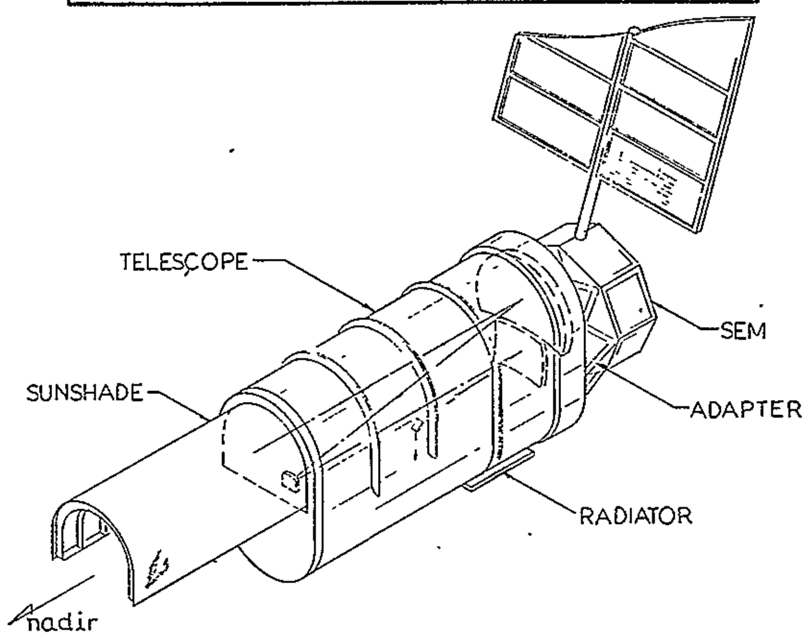

Figure IV-16 Expendable SEOS Configuration

5. "A Study of Payload Utilization of Tug," MDC G5356, prepared by McDonnell Douglas/GE/Fairchild for NASA Marshall Space Flight Center, June 1974.

6. Telephone conversations with Marvin Maxwell and Milton Ritter, NASA Goddard Space Flight Center.

This configuration uses a Service Equipment Module (SEM) to house all the housekeeping equipment. The SEM is connected to the mission equipment by a truss adapter. The sunshade is stowed during launch. A single solar array is attached to the SEM and rotates once per day. This expendable version weighs about 5000 lbs.

Design of a Serviceable SEOS is driven by the location of the earth resource and meteorological instrument packages. To eliminate the need for another mirror, ER and MET must be on the side of the telescope near the rear, as shown in Figure IV-15. It is difficult to reach these instruments from a docking port on the back of the telescope. Therefore, the docking face for servicing is located on the same side of the telescope as ER and MET. Figure IV-17 shows how the spacecraft housekeeping equipment is located in SRUs around the docking cone. The solar array is in the same location as for the expendable configuration.

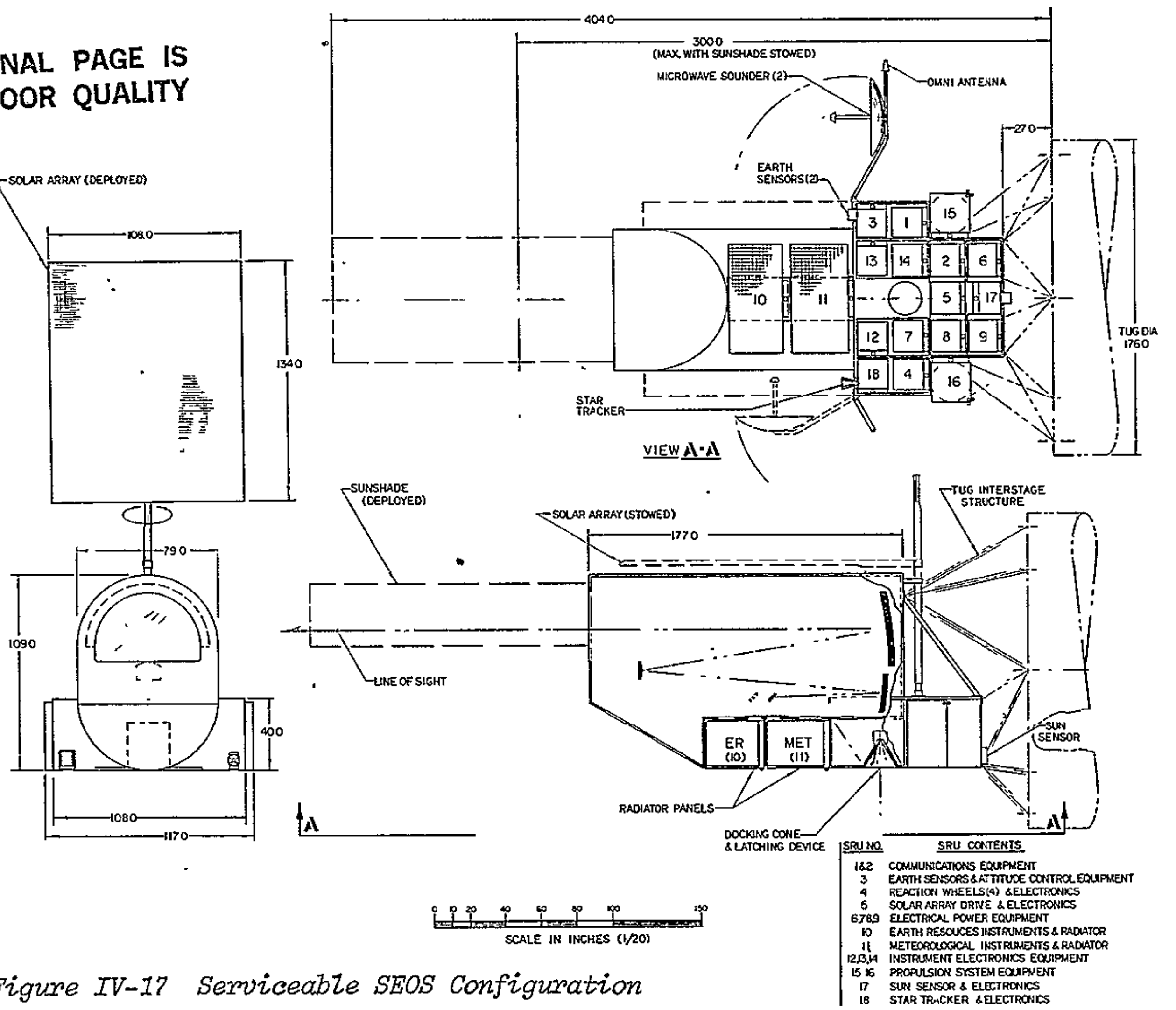

Figure IV-17 Serviceable SEOS Configuration

For launch, the Serviceable SEOS will be attached to a Space Tug by a tubular-truss interstage connecting at the rear of the telescope. The interstage, shown in Figure IV-18, will attach to the spacecraft at six points and to the Tug at eight points. The Tug will separate from the SEOS at the spacecraft-to-interstage interface.

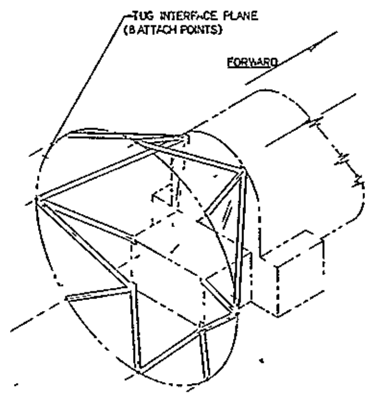

Figure IV-18 Serviceable SEOS to Tug Interstage Assembly

Spacecraft elements which fold during launch are the 1) sunshade for the telescope, 2) two microwave sounder antennas, one of which shares its mast with an omni antenna, and 3) solar array.

All of the 19 SRUs are in a single tier, are accessible with an Axial or Axial/Near Radial Servicer, and use a side-mounting interface mechanism. The largest SRU is 25 in. x 35 in. x 60 in. and the heaviest is 281 lbs. The maximum reach required for the servicer arm is 80 in. and the minimum is 30 in.

The "box and shelf" type of spacecraft structure supporting the SRUs is envisioned to be of honeycomb panel construction. Panels may be lightened in unstressed areas by use of large cutouts. The configuration has one spare SRU location and provides some growth potential. Up to four more SRUs could be added.

The solar array mast and pivot bearings are fixed to the spacecraft structure, but the drive motor and electronics are replaceable. Engagement/disengagement is provided by axial positioning of the driver/driven gear interface.

The ER and MET instruments (SRUs 10 and 11) have a severe thermal dissipation requirement. The thermal control design selected uses a multi-stage passive radiative cooler with cryogenic heat pipes to transfer heat from the instruments to the radiator panels. The selected docking face is on a North/South side of the spacecraft, so is an ideal radiative surface. Each 12 ft^2 radiator panel is an integral part of the SRU and extends out on each side of the main box. This design eliminates any breaks in the heat path link, which greatly increases the thermal control performance. Special provisions for these out-size SRUs have been provided in the servicer designs.

The GE studies of a Serviceable SEOS[5] recommend a stored cryogen method for cooling the payload. The increased-performance detectors used by P-E have a significantly greater heat load, however, and the weight penalty for a cryogenic thermal control system for them is prohibitive.

TRW's experience with passive, multi-stage radiators is not as negative as GE indicates. Direct access to the North-South face for the radiator is a critical factor in TRW's design. GE uses an East-West face for servicing and does not appear to service the radiators. The radiators will likely need replacement since contamination buildup will increase their absorbtivity.

Other thermal considerations are: (1) care must be taken to ensure that the docking maneuver does not cause excessive contamination or transient heat loads, (2) since it will take several hours for the sensors to cool down after the servicer departs, final spacecraft checkout will be delayed.

The Serviceable SEOS design shown does not include provisions for North-South stationkeeping. There has been discussion of whether such a requirement will be imposed for the SEOS mission. At this time Goddard assumes only East-West stationkeeping is required. Since there are only a few SEOS planned, perhaps only one, the need to add N-S stationkeeping for ease of multiple servicing is unlikely.

The mass properties of the Serviceable SEOS are shown in Table IV-4. The satellite weighs 5280 lbs. at the beginning of life and 37% of this is

Table IV-4 Serviceable SEOS Mass Properties by NRU and SRU

NRU (FIXED) WEIGHT				3323 LBS
PRIMARY OPTICS/MOUNT	819 LBS			
OPTICAL/NONOPTICAL STRUCTURES	1626			
SOLAR ARRAY (W/O DRIVE)	80			
STRUCTURE	566			
ANTENNAS (2 PARABOLIC, 1 OMNI)	82			
TUG INTERSTAGE	150			
SRU (REPLACEABLE) WEIGHT				1957 LBS
1. COMMUNICATIONS EQUIPMENT	163	11. METEOROLOGICAL INSTRUMENTS AND RADIATOR PANEL	189	
2. COMMUNICATIONS EQUIPMENT	140	12. SENSOR ELECTRONICS EQUIPMENT	54	
3. EARTH SENSORS AND ATTITUDE CONTROL EQUIPMENT	80	13. SENSOR ELECTRONICS EQUIPMENT	50	
4. REACTION WHEELS (4) AND ELECTRONICS	80	14. SENSOR ELECTRONICS EQUIPMENT	59	
5. SOLAR ARRAY DRIVE AND ELECTRONICS	55	15. PROPULSION SYSTEM EQUIPMENT	281	
6. ELECTRICAL POWER EQUIPMENT	56	16. PROPULSION SYSTEM EQUIPMENT	281	
7. ELECTRICAL POWER EQUIPMENT	90	17. SUN SENSOR AND ELECTRONICS	20	
8. ELECTRICAL POWER EQUIPMENT	101	18. STAR TRACKER AND ELECTRONICS	30	
9. ELECTRICAL POWER EQUIPMENT	109	19. SPARE SRU STRUCTURE	15	
10. EARTH RESOURCES INSTRUMENTS AND RADIATOR PANEL	104			
SATELLITE ON STATION (BEGINNING OF LIFE)				5280 LBS

serviceable. This percentage is lower than some serviceable spacecraft designs principally because the primary optics, its mount, and the entire optical bench structure are not to be serviced. These items are so massive (46% of the total), and have such precise alignment requirements, that servicing them is considered impractical.

The center of mass of the spacecraft is estimated to be about 1.3 inches from the docking cone. If this causes undesireable motion of the spacecraft during docking, about 50 lbs. of ballast can be added on the front of the telescope. The most massive elements are at the rear.

D. SERVICEABLE LOW EARTH ORBIT SPACECRAFT

The spacecraft selected to evaluate the servicer design for low-earth orbit is the Characteristic Large Observatory (CLO). This is a space-serviceable spacecraft that is capable of being used for several low-earth missions. The basic design analysis of the CLO is for the 1.2-meter X-Ray Telescope Payload (HE-11-A).

However, the basic concept and much of the hardware are applicable to the Space Telescope (AS-01-A) and Large Solar Observatory (SO-02-A) payloads. This payload spectrum encompasses a wide variety of low-earth missions in the post-1984 era.

The housekeeping subsystems required for the three missions are quite similar. The 1.2-meter X-Ray Telescope represents the worst-case mission in several subsystem areas.

1. Characteristic Large Observatory - 1.2-Meter X-Ray Telescope Payload (HE-11-A)

The 1.2-Meter X-Ray Telescope mission provides high resolution imaging, spectral, and polarimetric measurements of X-Ray sources. The mission equipment, Table IV-5, consists of a large telescope and eight instruments, five of which are located on a carousel that rotates them one at a time into the focal plane at the rear of the telescope. The mission equipment configuration is similar in concept to the High Energy Astronomy Observatory (HEAO)-B, but larger. NASA/MSFC is the sponsor for HEAO-B and TRW is the prime contractor.

Table IV-5 Mission Equipment for 1.2-Meter X-Ray Telescope Payload (HE-11-A)

INSTRUMENT	REQUIRED LOCATION	TOTAL WEIGHT (LB)	POWER (W)	DATA RATE (kbps)	BOX SIZES (DIMENSION IN INCHES)	NOTES
IMAGING PROPORTIONAL COUNTERS	FOCAL PLANE*	420	20	4.0	18x18x17 (2) 12x8x24 (1) 14 dia. Sphere(6)	SPHERES CONTAIN XENON (4) AND PROPANE (2)
HIGH RESOLUTION IMAGER	FOCAL PLANE*	320	100	4.0	20x12x12 (3) 20x20x16 (1)	
FOCAL PLANE CRYSTAL SPECTROMETER	FOCAL PLANE*	380	28	4.0	48x48x22 (1) 16x20x20 (1) 14 dia. Sphere(2)	SPHERES CONTAIN ARGON/CO_2
SOLID STATE SPECTROMETER	FOCAL PLANE*	400	15	4.0	21 dia. x 32 cyl. (1) 10x10x10 (1)	
POLARIMETER	FOCAL PLANE*	59	10	0.2	20x15x8 (1) 30x20x8 (1)	CONTAINS INTERNAL ROLL MECHANISM
OBJECTIVE GRATING	FRONT OF TELESCOPE, BEHIND MIRROR	140	-	-	48 dia. x 110 cyl. (1)	INCLUDES MIRROR ASSEMBLY
MONITOR PROPORTIONAL COUNTERS (2)	FRONT OF TELESCOPE, PARALLEL TO AXIS	210	22	0.9	25x18x21 (2) 14 dia Sphere(2)	ALSO REQUIRES REMOTE ELECTRONICS BOX; SPHERES CONTAIN ARGON/CO_2
ALL SKY MONITOR	EXTERNAL TO TELESCOPE	100	8	0.5	12x12 pyramid x 7.5	

* MOUNTED ON A ROTATING CAROUSEL

Data on the mission equipment and basic observatory requirements[7,8] are shown in Table IV-6, in terms of an expendable observatory. Much of the subsystem design information from the two references is pertinent to the CLO and the emphasis in this study has been on changes due to servicing. The CLO meets all of the requirements in Table IV-6 and has the characteristics listed in Table IV-7. A perspective view is shown in Figure IV-19.

Table IV-6 1.2-Meter X-Ray Telescope Payload Design Requirements

- NOMINAL 463 Km, CIRCULAR 15° INCLINATION ORBIT
- EIGHT MAJOR INSTRUMENTS:
  - CRYSTAL SPECTROMETER
  - SOLID STATE SPECTROMETER
  - IMAGING PROPORTIONAL COUNTERS
  - ALL SKY FLARE MONITOR
  - HIGH RESOLUTION IMAGERS
  - POLARIMETER
  - MONITOR PROPORTIONAL COUNTERS
  - OBJECTIVE GRATING
- 1.2-METER X-RAY TELESCOPE (NESTED ARRAY OF MIRRORS)
- SERVICING PERIOD TWO YEARS
- BODY-STABILIZED, BODY-FIXED SOLAR ARRAY
- TDRSS GIMBALLED ANTENNAS

Table IV-7 Characteristics of Characteristic Large Observatory with 1.2-Meter X-Ray Telescope Payload

- MEETS ALL 1.2 METER X-RAY TELESCOPE DESIGN REQUIREMENTS
- ESTIMATED 22,000 LB (BEGINNING OF LIFE)
- 150 IN. x 150 IN. x 516 IN.
- 25 SPACE REPLACEABLE UNITS (SRU)
- TWO SINGLE-TIER DOCKING PORTS
- AXIAL REMOVAL OF 13 HOUSEKEEPING SRU'S AT AFT DOCKING PORT A
- RADIAL REMOVAL OF 5 CAROUSEL-MOUNTED MISSION EQUIPMENT SRU'S AT AFT DOCKING PORT A
- AXIAL REMOVAL OF 6 SRU'S AT SIDE DOCKING PORT B
- ALL SRU'S ON SPACECRAFT USE SIDE-MOUNTING INTERFACE MECHANISM

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7. "High Energy Astronomy Observatory (HEAO) Block II Study," December 1975, prepared by Program Development, NASA Marshall Space Flight Center

8. "HEAO Spacecraft Modifications Definition Study for an Advanced Missions Program," Report No. 29882-6001-RU-00, TRW Defense and Space Systems Group, 15 October 1976.

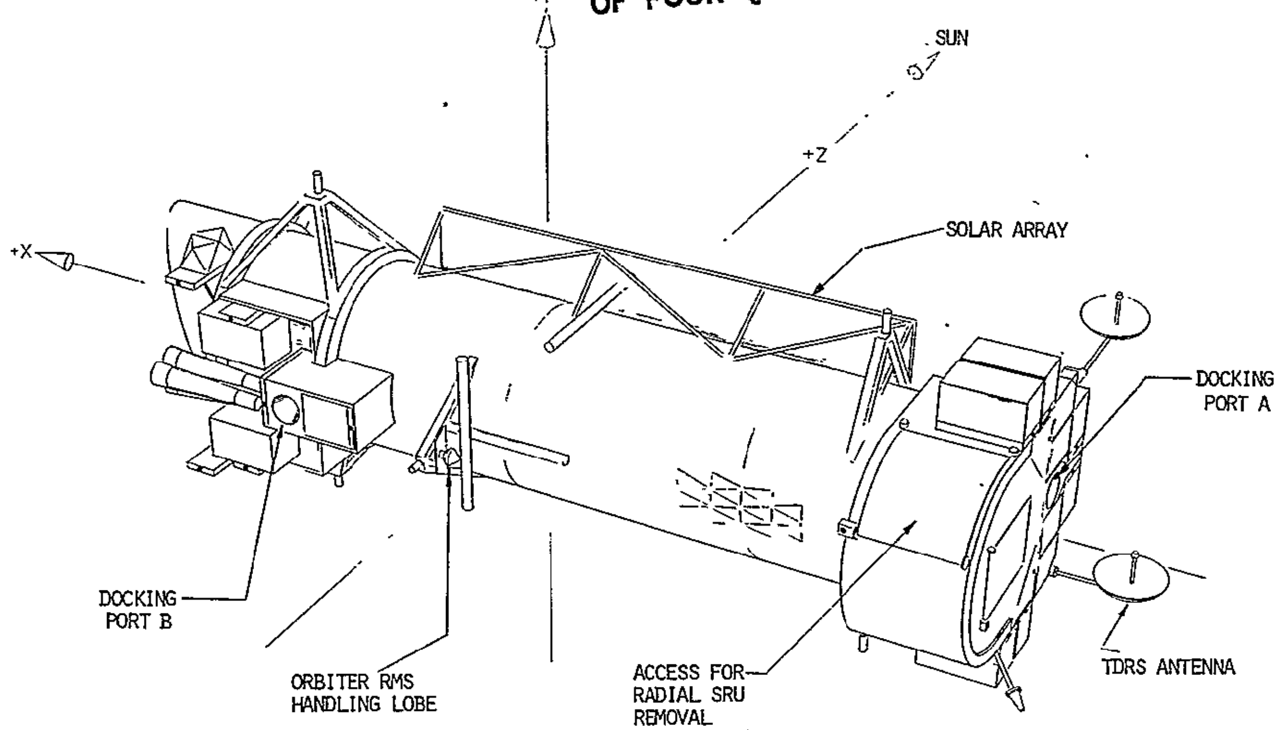


Figure IV-19 Characteristic Large Observatory Configuration

The telescope housing and internal optical bench provide the basic structure of the observatory. The High Resolution Mirror Assembly is located at the front end of the telescope and a carousel carrying five experiments is at its base. The tubular shell which houses the telescope is fitted with trunnion structures which pick up five attach fittings on the orbiter for launch. A flat solar array is attached to the telescope housing permanently, as are the sun sensor instruments and the Shunt Radiator Assembly, shown as a series of triangles.

Two docking ports are required for the CLO since mission equipment is required at both the back end of the telescope (Docking Port A) and near the High Resolution Mirror Assembly (Docking Port B). These two locations are too far apart to reach during a single docking. It is planned that the Shuttle Remote Manipulator System (RMS) arm will be used to capture and position the spacecraft during docking and redocking operations with the orbiter. A handling lobe is provided on the spacecraft to furnish a suitable grasping point for the end effector mechanism of the RMS.

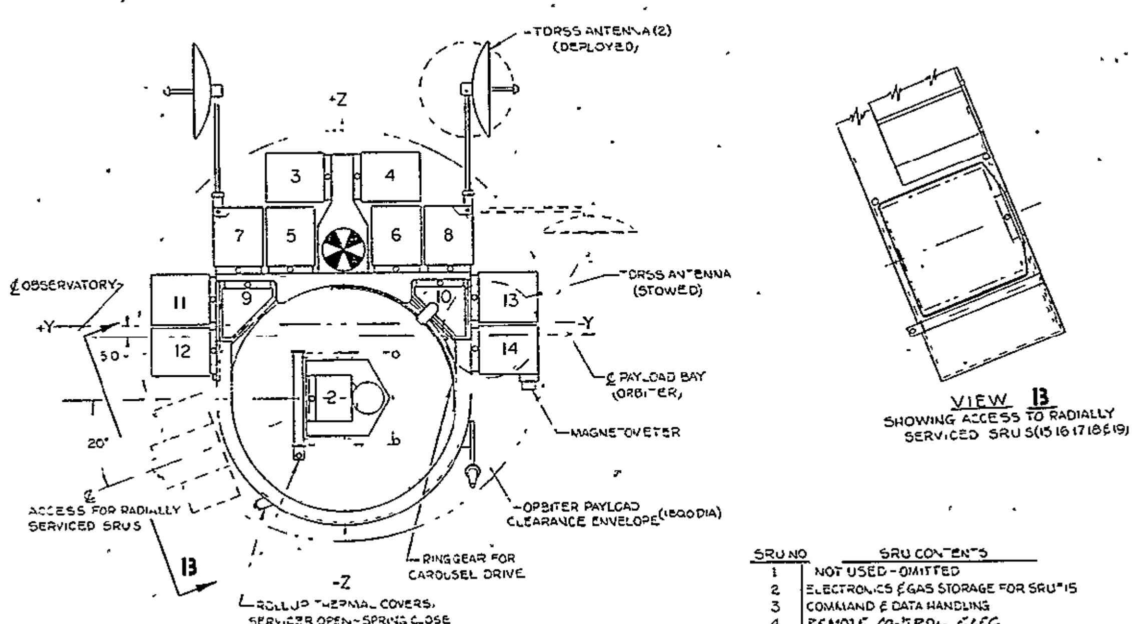
The Attitude Control and Determination Subsystem of the CLO uses reaction wheels for stabilization, star sensors, and magnetic torquer coils for momentum dumping. No reaction control gas is included.

All of the mission equipment but the Objective Grating will be replaceable. The Objective Grating is highly reliable and is tied directly to the High Resolution Mirror Assembly. It would be difficult to sever, retie and realign this attachment.

Other Non-Replaceable Units are:

- High Resolution Mirror Assembly
- Optical Bench Structure
- Omni Antennas (2)
- Aspect Sensor Upper Sunshades (3)
- Magnetic Torquer Coils (3)
- Shunt Radiator Assembly
- Carousel Bearings and Supports (2)
- Solar Array
- Sun Sensors (3)

Details of Docking Port A are shown in Figures IV-20 and 21. The 13 external SRUs have a single servicing plane for axial removal. Most of the housekeeping subsystem SRUs, such as command and data handling, attitude control and electrical power, are located here, as are the gimballed



SRU NO	SRU CONTENTS
1	NOT USED - OMITTED
2	ELECTRONICS & GAS STORAGE FOR SRU #15
3	COMMAND & DATA HANDLING
4	REMOTE CONTROL ELEC
5 6	ELECTRICAL POWER EQUIPMENT
7 8	TDRSS GIMBALED ANTENNA & ELECTRONICS
9	DIGITAL PROCESSOR FOR ACDS
10	CAROUSEL DRIVE
11 12 13 14	REACTION WHEEL (4) & ELECTRONICS
[illegible]	MAGNETOMETER

Figure IV-20 Characteristic Large Observatory SRUs at Aft Docking Port

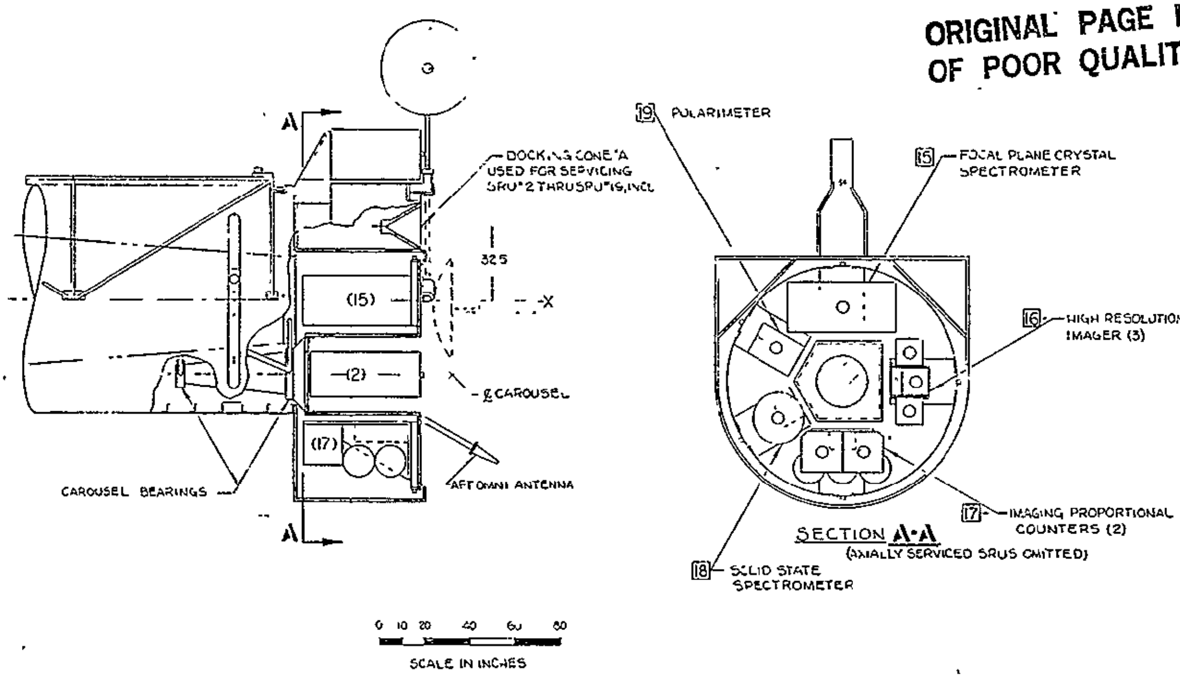


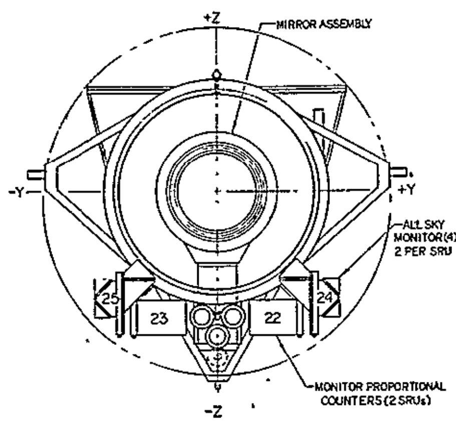
Figure IV-21 Characteristic Large Observatory SRUs Mounted on Carousel

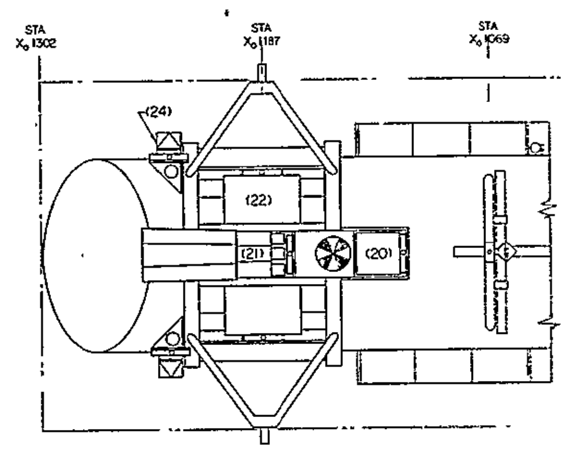
dish antennas for communication with the Tracking and Data Relay Satellite System (TDRSS). The mission equipment carousel has a serviceable rim drive mechanism, located in SRU 10, but the carousel bearings and supports are non-replaceable and must be redundant and highly reliable.

The electronics and gas storage for the carousel-mounted Focal Plane Crystal Spectrometer are removed axially from the center of the carousel (SRU 2). Fluid disconnects are required between SRU 2 and SRU 15, but there is no flexible gas line, as would be required if SRU 2 was located on the fixed spacecraft structure. The opening for SRU 2 requires a thermal closure during normal spacecraft operations. To meet this requirement and still be open for servicing, a "window shade" thermal blanket is operated by a worm gear system activated by the servicer end effector. After servicing, backdriving the worm gears permits spring cartridges to return the shade to its closed position.

Five of the mission equipment instruments are mounted on a carousel which serially positions each instrument detector in the telescope focal plane. All five of these instruments are arranged in compact groups permitting each to be an SRU. The mission equipment SRUs are removed radially through an opening in the carousel shell in the +Y, -Z quadrant. A window shade device with thermal blanketing is also used to close this opening for normal operation. The servicer end effector is used to open and close the shade.

The remainder of the SRUs are located at Docking Port B, located near the front of the telescope on the side opposite to the solar array, Figure IV-22. A second docking port is required because the sun aspect sensors for attitude determination must be rigidly mounted to the mirror assembly. Other equipment is placed here to relieve crowding at Docking Port A.





SRU NO.	CONTENTS
20	REFERENCE GYRO ASSEMBLY
21	ASPECT SENSORS (3)
22, 23	MONITOR PROPORTIONAL COUNTERS (1 EACH)
24, 25	ALL SKY MONITORS (2 EACH)

Figure IV-22 Docking Port B and Front of CLO Telescope

The use of radial-extracted SRUs was examined for the front end of the telescope. This concept, shown in Figure IV-23, has the advantage of docking at the end, which appears more convenient for orbiter-based servicing. The concept was not pursued, however, since a fixed telescope sunshade would interfere with docking and an extendable shade adds complexity to the spacecraft. In addition, cutouts are required in telescope sunshade to accommodate the All Sky Monitor SRUs.

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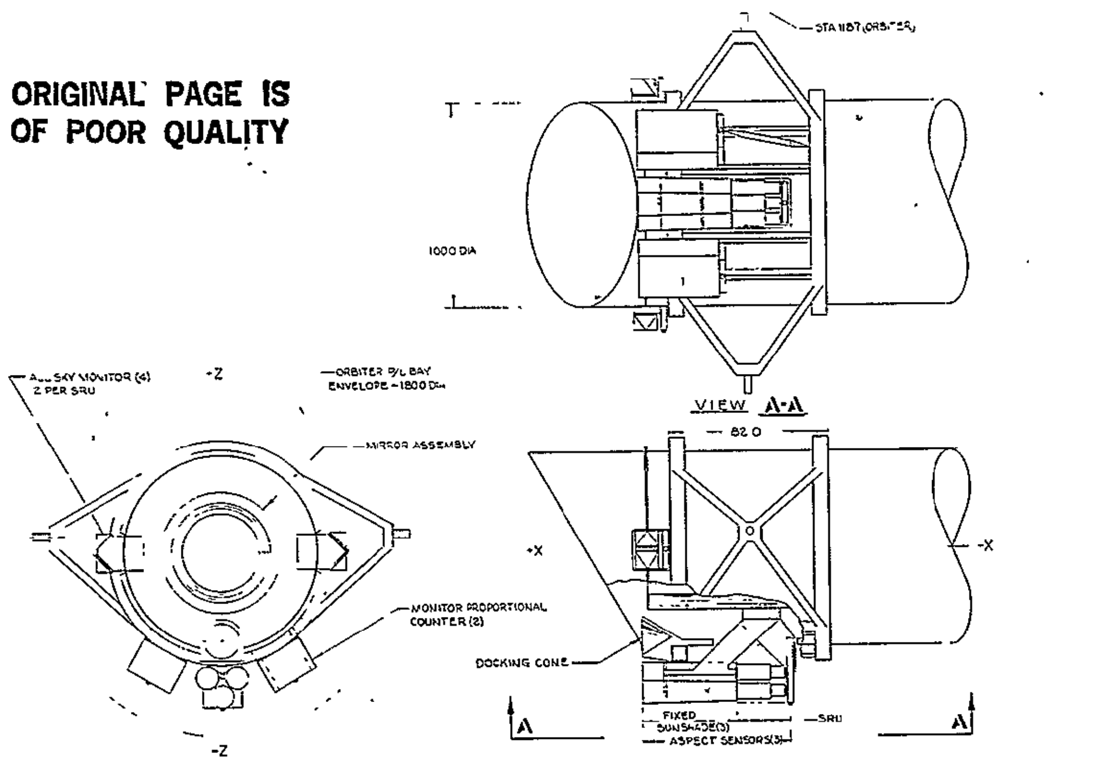

Figure IV-23 Radial SRU Configuration at Front of CLO Telescope

The coordinate axes for the CLO are defined in Figure IV-24. The observatory center of mass is 277.5 inches from the servicing face for Docking Port A and 3 inches in the -Z direction from the centerline of

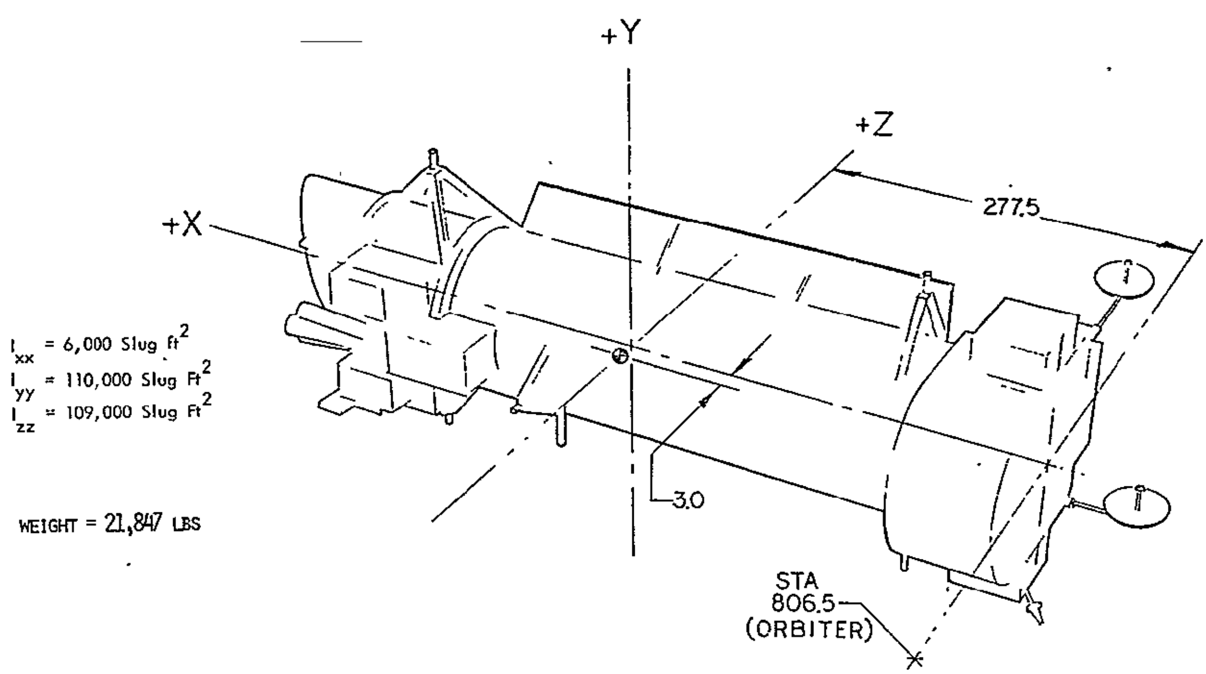

Figure IV-24 CLO Mass Characteristics

the main telescope. The total weight and moments of inertia about the center of mass are as indicated. The CLO is only slightly more massive than the non-serviceable configuration,[8] due largely to the fact that 80% of the CLO weight is non-replaceable.

The detailed CLO mass properties have been separated into SRUs and NRUs in Table IV-8. The individual SRUs range in weight from 61 lb for SRU 3 (command and data handling) to 444 lb for SRU 17 (imaging proportional counters). The NRUs far outweigh the SRUs due to the extremely heavy optical equipment that cannot be realigned in orbit. The mirror assembly alone is 46% of the total observatory weight.

Table IV-8 CLO Mass Properties by SRU and NRU

SPACE REPLACEABLE UNITS

ITEM	PAYLOAD WEIGHT	STRUCTURE & LATCH WEIGHT	NET WEIGHT
CAROUSEL-MOUNTED SRUS (RADIAL ACCESS, DOCKING CONE A			
#15 FOCAL PLANE CRYSTAL SPECTROMETER	250 LB	24 LB	274 LB
#16 HIGH RESOLUTION IMAGER (3)	320	24	344
#17 IMAGING PROPORTIONAL COUNTERS (2)	420	24	444
#18 SOLID STATE SPECTROMETER	400	24	424
#19 POLARIMETER	60	24	84
			1570 LB
-X END SRUS (AXIAL ACCESS-DOCKING CONE A)			
#2 FPCS ELECTRONICS AND GAS	130	20	150
#3 COMMAND & DATA HANDLING	32	29	61
#4 REMOTE CONTROL ELECTRONICS	330	29	359
#5 ELECTRICAL POWER EQUIPMENT & BATTERIES (2)	172	29	201
#6 ELECTRICAL POWER EQUIPMENT & BATTERIES (2)	172	29	201
#7 TDRSS ANTENNA	50	29	79
#8 AND ELECTRONICS	50	29	79
#9 DIGITAL PROCESSOR FOR ACDS	24	22	46
#10 CAROUSEL DRIVE	40	22	62
#11 REACTION WHEEL	100	29	129
#12 AND ELECTRONICS	82	29	111
#13	82	29	111
#14 REACTION WHEEL & MAGNETOMETER	86	29	115
			1704 LB
+X END SRUS (AXIAL ACCESS - DOCKING CONE B)			
#24 ALL SKY MONITORS (4)	50	20	70
#25	50	20	70
#22 MONITOR PROPORTIONAL COUNTERS (2)	105	20	125
#23	105	20	125
#21 ASPECT SENSORS (3)	65	22	87
#20 REFERENCE GYROS (3)	78	20	98
			575 LB
TOTAL OF ALL (24) SRUS			3849 LB

NON-REPLACEABLE UNITS

ITEM	NET WEIGHT
TELESCOPE ASSEMBLY	
OPTICAL BENCH ASSEMBLY	953
ASPECT SENSOR SUNSHADE ASSEMBLY	80
OBJECTIVE GRATING SPECTROMETER	140
BROADBAND FILTER SPECTROMETER	140
FORWARD THERMAL PERCOLLIMATOR	25
MIRROR ASSEMBLY	8,738
FIDUCIAL LIGHT SYSTEM	37
ISOLATION MOUNTS	60
	10,173 LB
SPACECRAFT ASSEMBLY	
OMNI ANTENNAS (3)	2
SUNSHADE ASSEMBLY (SCOPE)	25
SOLAR ARRAY (18 PANELS)	240
SHUNT RADIATOR (16 PI SEGMENTS)	44
MAGNETIC TORQUER (3)	390
HARNESS INSTALLATION	250
SUN SENSOR	3
STRUCTURE AND THERMAL	3,591
	4,545 LB
CAROUSEL ASSEMBLY	
BEARINGS AND SUPPORT	184
WHEEL ASSEMBLY	166
CABLING	50
MODULE EXCHANGER TRACKS (6 SETS)	35
FLEXIBLE CLOSURE AND DRIVE	7
	442 LB
TOTAL OF NRUS	15,160 LB

TOTAL OF NRUS AND SRUS =	18,997 LB
CONTINGENCY (15%)	2,850
TOTAL OBSERVATORY	21,847 LB (9910 KG)

a) Attitude Control and Determination Subsystem

The ACDS performance requirements for the CLO are the same as for the HEAO Block II Study,[7] summarized in Table IV-9. In addition, serviceability requirements must be met.

Table IV-9 1.2-Meter X-Ray Telescope Payload Attitude and Velocity Control Requirements

ORBIT ALTITUDE	250 N.M.
ORBIT INCLINATION	0° - 28.5°
VIEWING MODE	THREE AXIS POINTING
SOLAR AVOIDANCE CONE	$\pm 45^{\circ}$
LUNAR AVOIDANCE CONE	N/A
EARTH AVOIDANCE CONE	1° DARK, 10° LIGHT
POINTING ACCURACY (LONG TERM)	$\pm$ 30 ARC SEC/HR
POINTING JITTER (STAB. RATE)	$\pm$ 0.5 ARC SEC/SEC
POINTING ASPECT (NON-REAL-TIME)	0.5 ARC SEC
REACQUISITION ACCURACY	0.5 TO 1 ARC MIN (2σ)
ROLL MANEUVERS FOR SCIENCE	NONE
SLEW RATE	10°/MIN (DESIRED)
MAX. POINTING TIME PER TARGET	24 HOURS
MAX. FLARE REACTION TIME	1 DAY
RETRIEVAL/REVISIT	2 YEARS
POINTING ACCURACY	$\pm$ 30 ARC SEC

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The ACDS configuration developed in the HEAO Block II Study is amenable to servicing, so has been selected with the only change being the omission of the Reaction Control Subsystem. TRW recommends[8] that momentum dumping be accomplished by the magnetic torquer coils alone.

The moments of inertia of the CLO are about the same as for the HEAO Block II configurations. The additional weight for serviceability is distributed more toward the center than the other approach of attaching the complete HEAO Service Equipment Module (SEM) in one spot. Therefore, the same Sperry Model 400 Reaction Wheel Assembly, which has 400 ft-lb-sec momentum at 4000 RPM, is used.

The nonserviceable ACDS configuration[8] is shown in Figure IV-25. A functional definition of the nonserviceable ACDS assemblies is provided in Table IV-10.

Aspect sensors for attitude determination must be located near the telescope's primary mirror for precise alignment. The Invar structure which supports the primary mirror also warps the Earth's magnetic field. The three-axis magnetometer is therefore located near the support module (SM). Three-axis stabilization of the telescope and maneuvering is accomplished through momentum exchange with a cluster of four reaction wheels located in the SM. Momentum unloading is provided by the interaction of the Earth's magnetic field with three orthogonal magnetic torquer coils which are located midway between the aspect sensors and the magnetometer.

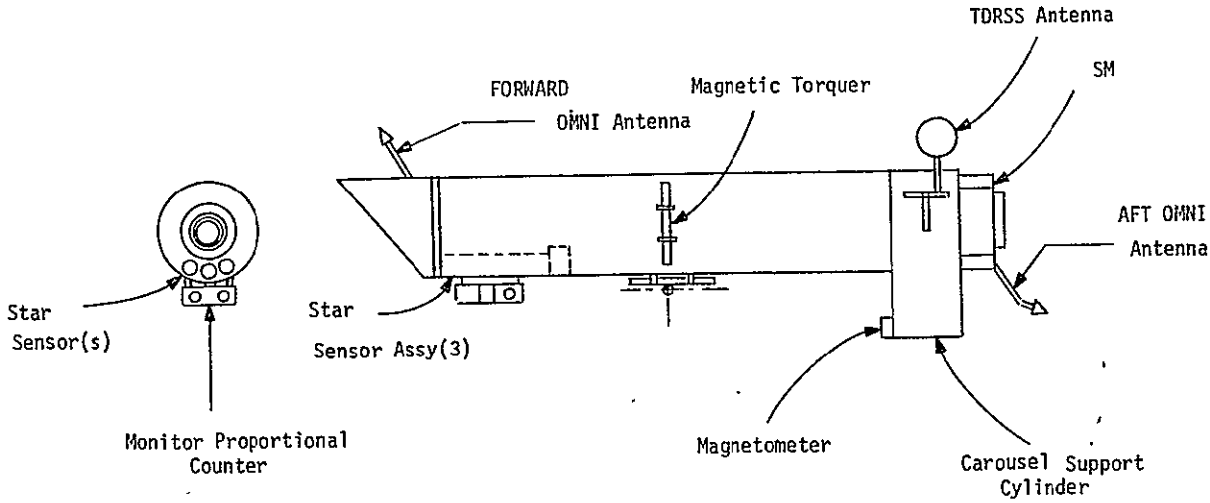

Figure IV-25 Expendable HEAO Block II X-Ray Telescope Configuration

Table IV-10 CLO Attitude and Velocity Control Subsystem Function Definition

ASSEMBLY/FUNCTION	FLIGHT QUANTITY	NUMBER OPERATIONAL
DIGITAL PROCESSOR (DPA)	2	1
SENSOR PROCESSING		
RW CLUSTER CONTROL LOGIC		
MAGNETIC DESATURATION LOGIC		
ANTENNA POINTING LOGIC		
ATTITUDE DETERMINATION SOFTWARE		
TRANSFER ASSEMBLY (TA)	2	1
DPA A/D, D/A CONVERSION AND MEMORY LOAD		
TELEMETRY MULTIPLEXING		
SENSOR MULTIPLEXING		
RW MOTOR AND MTS COIL DRIVE DIGITAL INTERFACING		
COMMAND DECODING		
POWER CONVERSION		
Y SUN SENSOR ASSEMBLY (YSSA)	2	2
Z SUN SENSOR ASSEMBLY (ZSSA)	1	1
REFERENCE GYRO ASSEMBLY (RGA)*	3	3
MAGNETOMETER ELECTRONICS ASSEMBLY (MEA)	2	1
PROCESSES ANALOG SIGNALS FROM REMOTE		
3-AXIS MAGNETOMETER		
REACTION WHEEL ELECTRONICS (RWEA)	4	4
POWER DRIVE AND WHEEL SPEED PROCESSING		
POWER CONVERSION (AC/DC)		
DIGITAL DATA I/O		
REACTION WHEEL ASSEMBLY (RWA)	4	4
COIL DRIVE ELECTRONICS (CDE)	3	3
PROVIDES GRADUATED ELECTRONIC DRIVE TO TORQUER COILS		
POWER CONVERSION		
MAGNETIC TORQUER COIL (MTC)	3	3
PROVIDES TORQUE FOR CONTROL AND WHEEL UNLOADING		
GIMBAL DRIVE ELECTRONICS (GDE)	2	2
STEP MOTOR DRIVE		
PULSE METERING AND COUNTING		
DIGITAL I/O		
POWER CONVERSION		
THREE AXIS MAGNETOMETER	2	1
ANTENNA DRIVE ASSEMBLY (ADA)	2	2
CONTAINS MECHANICAL GIMBALS, MOTORS, ANGLE PICKOFF, GEARS, AND CABLE WRAP FOR HIGH GAIN ANTENNA ARTICULATION.		
FIXED HEAD STAR TRACKER	3	2 of 3
CONTAINS SHADE, OPTICS, AND ELECTRONICS FOR ATTITUDE DETERMINATION SENSING		

* TWO (2) GYROS AND ELECTRONICS PER ASSEMBLY

For the serviceable CLO, the Transfer Assembly is replaced by a Data Bus located in the Command and Data Handling Subsystem. Reconfiguration of the ACDS equipment for servicing is influenced by:

- ACDS assembly/component groupings
- ACDS functions,
- Servicer characteristics, and
- Serviceable spacecraft design guidelines.

The allocation of ACDS equipment to SRUs is shown in Table IV-11. The three star aspect sensors are combined in an SRU with their lower shades,

Table IV-11 CLO Attitude and Velocity Control SRU Assignments

ACDS ASSEMBLY	SRU NO.	DOCKING PORT
• STAR ASPECT SENSORS (THREE IN ONE SRU)	21	B
• DIGITAL PROCESSOR (REDUNDANT)	9	A
• TDRSS ANTENNA DRIVE AND GIMBAL ELEC.	7, 8	A
• RW PLUS RWEA	11, 12, 13, 14	A
• MAGNETOMETER PLUS MEA	14	A
• MAGNETIC COIL DRIVE ELECTRONICS	11	A
• RGA (THREE IN ONE SRU)	20	B

·EACH UNIT IS MADE SELF-REDUNDANT

DOCKING PORT A: AFT

DOCKING PORT B: FORWARD

optics, and electronics in order to preserve their relative alignment integrity. The redundant digital processors are together in a separate SRU. Each of the two high-gain TDRSS antennas is packaged separately with its gimbal drive electronics to form two separate SRUs. Each reaction wheel is combined with its electronics assembly to form four (4) separate SRUs. Each RWEA contains a power converter. The MEA, which processes the analog signals from the magnetometer, is combined with the magnetometer and located in one of the reaction wheel SRUs. Similarly, redundant magnetic coil drive electronics is located in another reaction wheel SRU. The three reference gyro assemblies, each containing its own power converter, are mounted in one SRU at the front end to save space at Docking Port A. The omni antennas are considered highly reliable and therefore are Non-replaceable Units. Redundant sun sensors are also non-replaceable.

The ACDS is sized on a per axis basis. The minimum RW sizing criteria is 1.5 times the peak cyclic gravity momentum. The maximum RW sizing criteria is 1.25 times the maximum accumulated secular momentum per orbit.

The minimum is 300 ft-lb-sec per axis and the maximum is 800 ft-lb-sec per axis. Using a skewed mounting arrangement, four 400 ft-lb-sec wheels are recommended. The Sperry Model 400 RW is a derivative of a model used on the DSP satellite that uses AC motors. The AC motor and electronic driver is replaced with a brushless DC motor and its electronic driver.

The NASA-recommended magnetic torquer dipole size is 6000 A-m^2/axis and is three times the size required for the Space Telescope. By making the magnetic coil and drive electronics self-redundant, no singlepoint failure will jeopardize the mission. The core need not be redundant, only the coil and drive electronics. The coils must be positioned such that their residual field at the aspect sensors and the ACDS electronics assemblies is minimized. This means that the MTCs cannot be located near either the fore or aft docking port. The redundant coils can be attached in a way that they can be replaced by EVA, but it is unlikely this will be necessary. Redundant coil drive electronics is located in SRU 11.

The CLO must be passive for final rendezvous and docking with the orbiter. Therefore, the ACDS will be disabled about an hour before final rendezvous to allow the reaction wheels to slow down and stop. The spacecraft transient response to ACDS disable has not been examined, but no problems are anticipated.

b) Command and Data Handling Subsystem

The CDHS for the CLO uses a centralized computer. This computer may be reprogrammed, via the uplink, to operate with any equipment suit. A multiple Data Bus is also included in the system architecture for flexibility and to minimize connector pin requirements in each SRU. The selected computer is the Advanced On-Board Processor (AOP) design being developed by NASA/GSFC. A modified version of this has been selected as the NASA Standard Spacecraft Computer (NSSC). Dual redundancy in all units and the generalized capabilities in the computer allow for high reliability of the CDHS. There are several work-around provisions possible.

Throughout its service lifetime, the spacecraft may be provided with several different equipment suits. The CDHS must be capable of functioning with any configuration of equipment suits, as new equipment may have considerably different command and telemetry requirements.

Several alternatives are available to achieve the required flexibility. The spacecraft equipment may be constrained to a specific command and telemetry format, or the CDHS may be modified every time the spacecraft equipment suit is changed. Clearly, neither of these alternatives is desirable.

A flexible and adaptable CDHS has been created by the inclusion of a centralized computer. This computer may be reprogrammed, via the uplink, to operate with any equipment suit. The programming will not have to be changed when a faulty SRU is replaced with an identical functioning SRU. The inclusion of a central computer does not preclude processing equipment within the various spacecraft subsystem equipment. Existing equipment containing computers may be used and central computer will act merely as a command executive. However, new or existing equipment designed to be operated in spacecraft system configurations containing centralized processing can make use of central computers processing capabilities.

The CDHS block diagram in Figure IV-26 shows the three data busses; the Command Bus, the Telemetry Bus, and the Priority Interrupt Bus. The CDHS is made up of the command decoder, central computer, and central telemetry unit. Remote Telemetry Units (multiplexers) are incorporated within the other SRUs, such as ACDS, Power and Experiments.

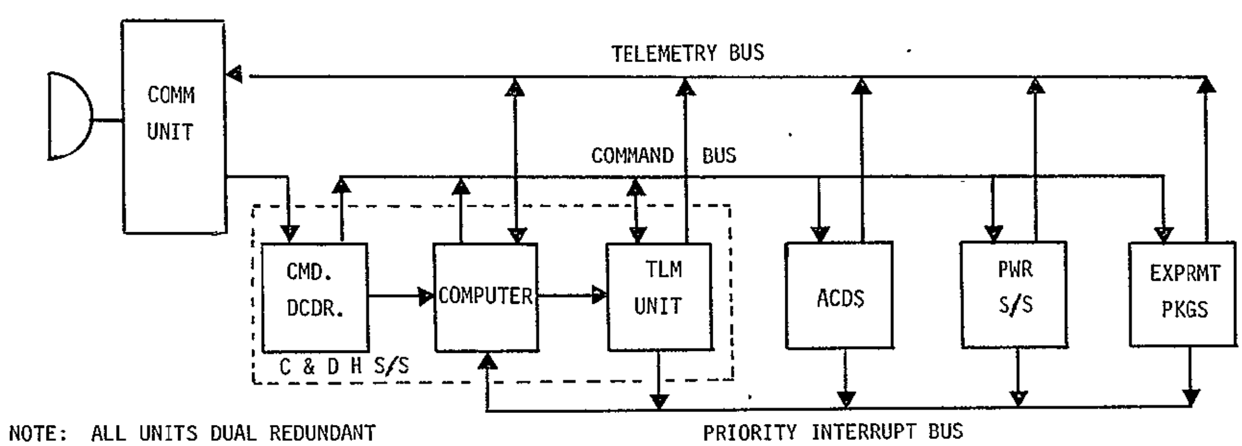

Figure IV-26 CLO Command and Data Handling Subsystem Block Diagram

In normal operation, uplinked commands are received from the ground or the TDRSS multiple-access antenna by the communications unit (Comm Unit). It is then decoded and polynomial checked by the command decoder, and further processed by the computer. The computer then transmits a properly formatted command, via the Command Bus, to the appropriate spacecraft element. These commands are interleaved on the Command Bus with the normal telemetry requests from the telemetry unit. Telemetry responses from the various spacecraft elements, are transmitted to the Comm Unit for downlink transmission via the prime Telemetry Bus.

In the absence of the uplinked commands, the computer outputs commands to the various elements, from its stored command programs, as required to maintain proper operation of the observatory. Should the computer require data from a spacecraft element, the data response is transmitted via the redundant Telemetry Bus, using the redundant RTU. Thus the normal downlink telemetry format is not disturbed.

If a spacecraft element requires that a special function be performed by the computer (because of some extraordinary circumstance, etc.), an interrupt signal is sent from the element to the computer via the Interrupt Bus. An interrupt signal causes the computer to stop what it is doing and dedicate itself to the interrupt requester. In the event of several simultaneous interrupts, the computer is equipped with priorization logic within its stored programming. It will service the highest priority interrupt first, proceed to the next highest priority interrupt, and so forth until all interrupts have been serviced. Then the computer returns to where it was prior to the receipt of the interrupt.

As an example, consider the normal changeover from one experiment to another. From its previously stored program, the computer issues commands via the Command Bus to shut down the present experiment, rotate the carousel, and turn on the new experiment. Once these have been accomplished, the experiment might require certain commands for proper configuration. It then sends an interrupt to the computer. Upon receipt of the interrupt, the computer branches to a service routine for that equipment and issues data request commands via the Command Bus. In response to these commands, the affected redundant RTU turns on, gathers the requested data, and transmits the data to the computer via the redundant Telemetry Bus. The computer acts upon this data and issues commands to the experiment via the Command Bus. Once the interrupt servicing has been completed the computer returns to where it left off in its normal routine.

The spacecraft is launched with an initial nominal set of station-keeping and equipment service routines stored within the computer. In addition, the initial nominal telemetry format is stored within the telemetry unit. If one or more of the routines must be changed, the computer is reprogrammed via the uplink. The computer software is modular, and hence only the affected modules need be altered. Changes in the telemetry format are stored within the computer and then transmitted to the telemetry unit for internal storage.

Changes of the spacecraft system configuration during servicing require reprogramming of the computer and/or the telemetry unit. This is accomplished in the manner described, via the uplink. Virtually the entire suit of spacecraft equipment can be changed to different equipment, and the computer reprogrammed from the ground to operate these equipment.

The system operation described above considers no equipment failures. However, this configuration is remarkably resistent to the effects of failures within the CDHS equipment. Loss of a command decoder and/or a computer and/or a telemetry unit results in switching to the affected redundant unit(s). Since the required power and signal switching are successfully provided, these failures are transparent to the ground. Redundant unit take-over can be accomplished either automatically aboard the spacecraft, or via uplink command from the ground. For automatic switchovers, equipment status telemetry is sent, informing the ground of the switchover.

Loss of an RTU results in a change in system performance characteristics. Computer data request replies are then handled by the affected redundant RTU along with normal telemetry data. Thus the computer data request replies would be interleaved with normal telemetry data on the same Telemetry Bus. The exact impact of a failure such as this on the overall system response is difficult to assess as it depends upon several presently undefined factors, such as the telemetry rate, the telemetry format, and specifically which RTU has failed. However, the overall impact can be small. The computer would have to wait one telemetry data word time before receiving its requested data.

Loss of both computers would severely affect system performance but commands can still be uplinked to the spacecraft via the command decoder. All commands, including station-keeping, would have to be transmitted by this means until servicing could be accomplished.

The computers considered for the CDHS are shown in Table IV-12. Some of the criteria used for selection are:

- Sufficient operating speed and capacity to execute all necessary algorithms.
- Availability in a space-qualified model.
- Possibility of achieving status as a standard spacecraft computer

Table IV-12 CLO Command and Data Handling Subsystem Computer Candidates

DF224	Autonetics
ML-1	IBM
HTC	IBM
*CDC 469	Control Data Corporation
SUMC	RCA
HDC701P	Honeywell
D216	Autonetics
CP-16	General Electric
CP-24	General Electric
CP-32	General Electric
NNF (Also called APL)	RCA
*AOP	Developed by GSFC
MSC	Raytheon
* Best Candidates	

Although many of the listed computers could be used for this application, the best choices are the CDC 469 (used on HEAO), and the AOP (NSSC). Either computer will work. The CDC 469 is slightly larger and consumes slightly more power than the AOP. NSSC is listed in the NASA Standard Equipment Catalog (No. 4001), so is given preference.

All of the CDHS modules will fit within a single SRU, including redundant elements and weigh a total of 30 lbs. About 40 watts power is required with the prime system on and the redundant system in standby.

c) Electrical Power and Distribution Subsystem

The EPDS configuration for the CLO is essentially that shown for the Serviceable DSCS-II in Figure IV-13, with a fourth battery and fourth shunt element. The number of APDUs is determined by the exact redundancy configuration of the total spacecraft and the number of SRUs requiring primary power. The Spacecraft Integration Assembly and the Experiment Accommodation functions of the HEAO EPDS go into the individual SRUs.

The 1.2-meter X-Ray Telescope mission power budget[8] is shown in Table IV-13. The budget is based on an average experiment bus load of 272 watts

Table IV-13 Power Budget for CLO with 1.2-Meter X-Ray Telescope Payload

	WATTS
EXPERIMENT AVERAGE	272
MISSION EQUIPMENT THERMAL CONTROL (DISABLED DURING ECLIPSE)	240
SPACECRAFT LESS ATTITUDE CONTROL AUTHORITY	215
MAGNETIC TORQUING	30
REACTION WHEEL DRIVE:	
DISTURBANCES	101
FIVE 90-DEG MANEUVERS PER DAY	55
CONTINGENCY AT 5 PERCENT OF ALL EXCEPT SPACECRAFT	35
TOTAL LOAD IN SUNLIGHT, P_S	948
REQUIRED SOLAR ARRAY POWER, P_A	1619

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including individual instrument heaters as necessary, and upon maintaining the observatory Z-axis oriented toward the sun. An additional 240 watts is allocated for general thermal control of the mission equipment. Battery life is extended by disabling this thermal control power during eclipse. The 215 watts carried for the spacecraft includes TDRSS communication antenna steering functions. The magnetic torquing allocation includes magnetometer load and coil drive. The total reaction wheel drive load, averaged over the worst four orbits of the mission, is only 101 watts due to gravity, aerodynamic and residual magnetic disturbances. Maneuver power is allocated at 55 watts, based on five 90-degree excursions per day with 100 ft-lb-sec momentum residuals and 950 kilojoules per maneuver. The 5 percent contingency allocation is arbitrary, and in practice can be augmented by the experiment module heater power when necessary.

Four 20-Ah batteries are assumed, with a 19.5% maximum depth of discharge (DOD) if the mission equipment thermal control is disabled during eclipse. If the spacecraft is required to operate with only 2 of 4 batteries, the DOD could get up to 39%, which is very high for low earth orbits. A third operating battery reduces the maximum DOD to an acceptable 26%. A fifth battery may need to be added to increase the likelihood that three will remain operating.

The solar array conservatively requires 18 standard large HEAO modules to provide 1619 watts after two years in orbit. These are shown in the configuration drawings and mass properties statement. Since the array is assumed to be non-replaceable, additional array modules will likely be required. Depending on the specific array technology used, it is estimated that up to 22 array modules are required for a 7-year lifetime. Each module weighs 15 lbs. Another alternative is to make the array serviceable. It does not lend itself to the standard SRU configuration, but the array modules could possibly be replaced by EVA at the orbiter.

With the mission equipment thermal control off, there will be a 673 watt load shortly after eclipse. With 22 array modules, this would require a fifth shunt module. It may be possible to operate with four shunt modules if the mission equipment thermal control is turned back on near the end of eclipse.

For servicing, the batteries are contained in two SRUs of two each. The rest of the EPDS equipment is in two other SRUs.

d) Thermal Control

The most critical CLO thermal control problem is the dissipation of heat from the carousel-mounted mission equipment, as illustrated in Figure IV-27. Heat pipes will be required within the individual SRUs to transfer heat from the equipment to the back (right hand) face of the SRU and then to the carousel structure.

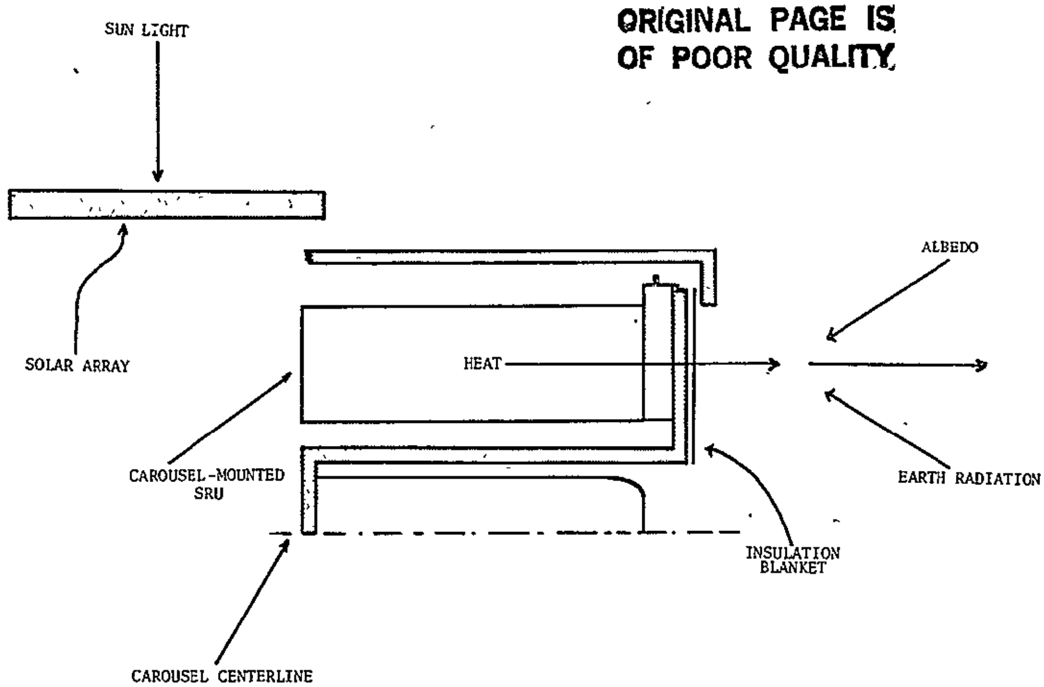

Figure IV-27 Thermal Control of CLO Carousel-Mounted Mission Equipment

Since the back end of the observatory is nominally at right angles to the sun, a simple insulation blanket, such as aluminized Kapton, is sufficient for passive control with albedo and earth radiation effects. This configuration should keep the equipment temperatures to less than 90°F and the mounting surface below 80°F.

The assumptions for this estimate are:

- Radiator (carousel surface) area = 63 ft^2 (9 ft diameter)
- Surface degraded to $\alpha = 0.4$, $\varepsilon = 0.86$
- Heat Load (200 n mi orbit)
  - Electronics = 170 W (worst case)
  - Albedo = (0.3) (442) (0.9) αA
  - Earth radiation = (79) (0.9) εA

Other areas of the observatory use standard passive thermal control techniques.

e) Reliability

The CLO configuration provides a highly reliable approach for a periodically-maintained system. Block diagrams for the three critical subsystems (Command and Data Handling, Electrical Power and Distribution, and Attitude

Control and Determination) are shown in Figures IV-28, -29 and -30 respectively. Numerical predictions are shown for two years, the assumed time between servicing visits. The unit failure rates (λ), in units of failures

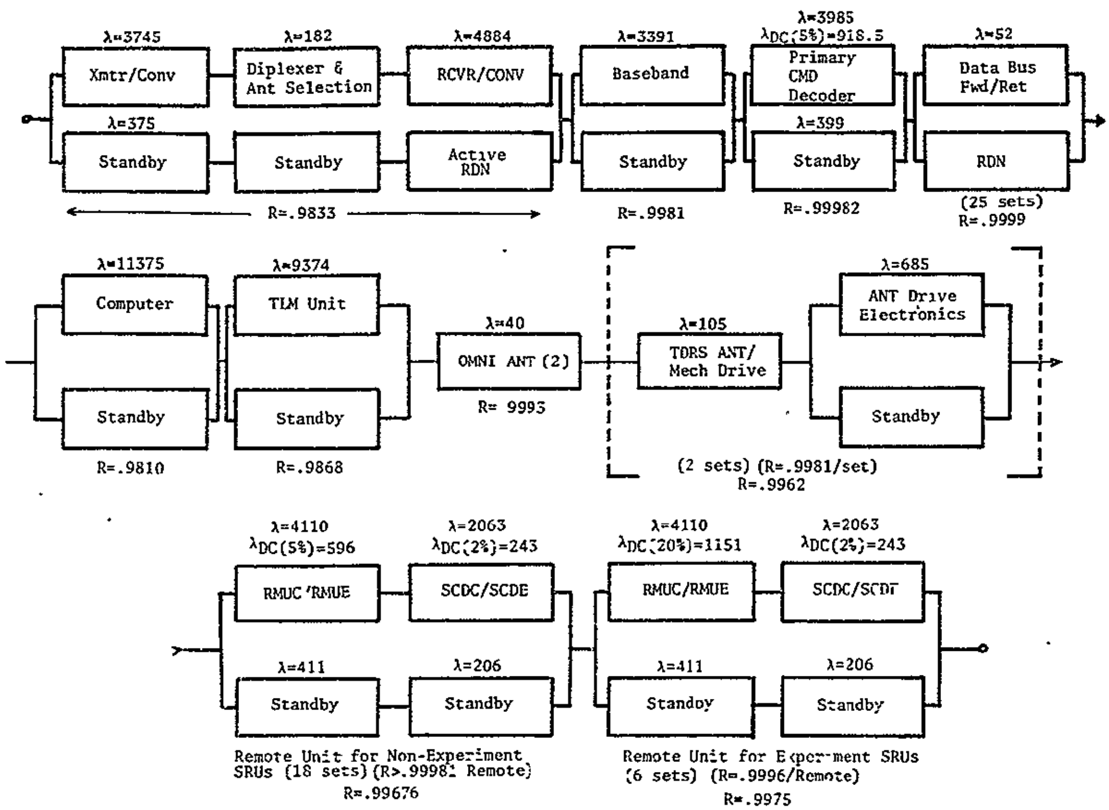

Figure IV-28 CLO Command and Data Handling Subsystem Reliability

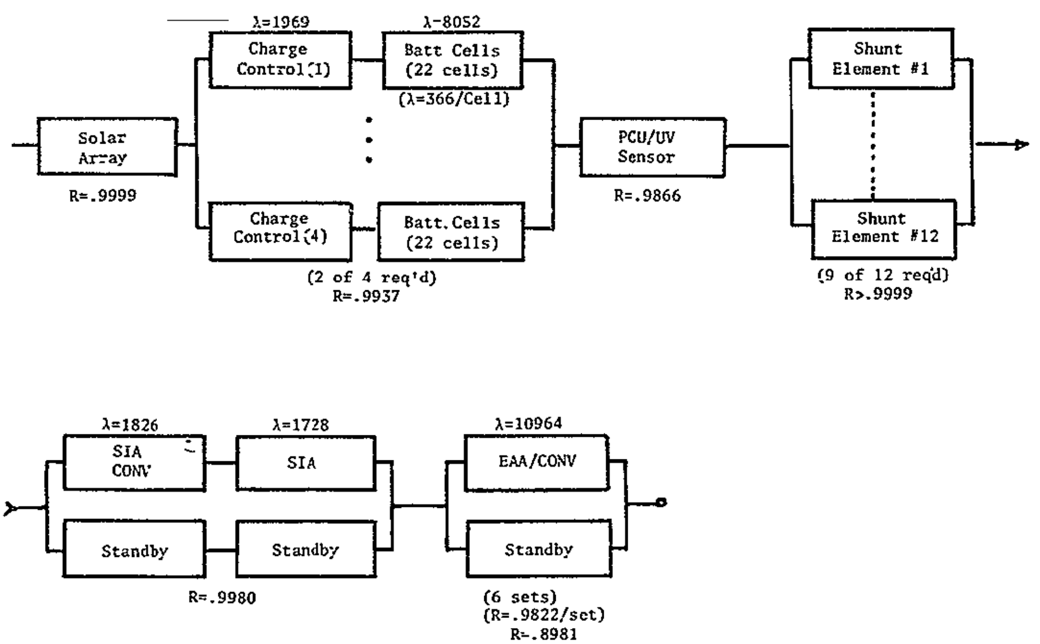

Figure IV-29 CLO Electrical Power and Distribution Subsystem Reliability

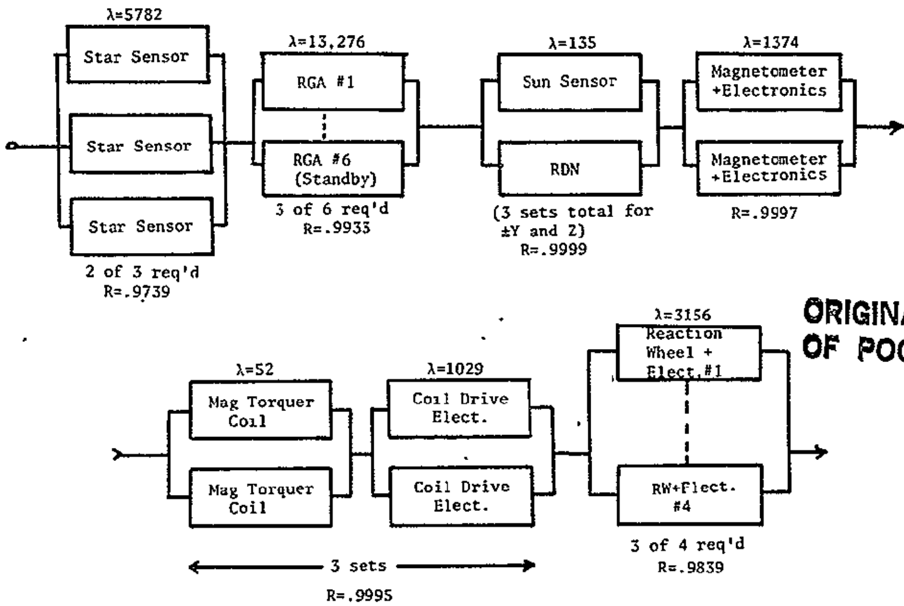

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Figure IV-30 CLO Attitude Control and Determination Subsystem Reliability

per 10^9 hours, are based on TRW hardware experience, with duty cycles (λ_{DC}) to account for the particulars of the serviceable mission.

The CDHS, Figure IV-28, uses a centralized computer and a multiple data bus. Each of the twenty-four SRUs has its own redundant Remote Telemetry Unit. The antenna to communicate with the TDRSS is included in the CDHS reliability diagram.

The critical CDHS elements are the computer and telemetry units. Each has a redundant unit. A conservative estimate was used in the calculations. There are work-around modes that could be used for continued operation with degraded performance, if necessary.

In the EPDS, Figure IV-29, the 18-panel solar array is a Non-Replaceable Unit, as are the shunt elements. Both have high reliability. The most critical elements, the batteries, are shown for 2 of 4 operating being satisfactory. As pointed out in EPDS discussion, it is desirable to have at least 3 operating. The combined reliability of the charge control and batteries for two years is .8989 for 3 of 4 and .9677 for 3 of 5.

Figure IV-30 shows the interaction of the major ACDS equipment. The basic concept is active reaction wheel control with magnetic unloading. The star sensors are located at the front of the telescope, as are the reference gyro assemblies. The sun sensors, which are extremely reliable, are mounted as redundant Non-Replaceable Units on the observatory outer structure. The critical ACDS elements are the star sensors, reference gyros and reaction wheels. Reliability is made sufficient by using a configuration that allows 2 of 3 star sensors, 3 of 6 reference gyros and 3 of 4 reaction wheels and their electronics.

An overall housekeeping subsystem reliability of 0.7846 after two years is indicated as follows for the worst battery requirement (3 of 4):

CLO Subsystem	R (Two Years)
Command and Data Handling	.9401
Attitude Control and Determination	.9510
Electric Power and Distribution	.7947
Thermal and Structures	.9990 (est.)
Net	.7097

There is no reliability requirement on the CLO, but 0.7097 compares with the HEAO-B reliability requirement of 0.78 after one year. If the 3 of 5 battery option is selected, the net reliability becomes 0.7641.

No good mission equipment reliability data has been obtained. TRW is not responsible for this data on HEAO, so does not have direct access. It appears that this is not a design requirement, so is not analyzed in any detail.

f) SRU Design

A fluid disconnect is required between SRUs 2 and 15. Information on this subject was obtained from Fairchild Stratos Division, which has a contract with NASA MSFC to design and develop standard payload fluid disconnects for Space Shuttle. The disconnects are being designed to connect and disconnect various lines between the orbiter and its payloads while in orbital flight, including in-orbit remote servicing of payloads. The disconnects must handle toxic or cryogenic gases and fluids, have low flow resistance, extremely low leakage when disconnected and must survive the space environment.

Under the NASA contract, Stratos will establish functional and operational requirements and design, fabricate and test a prototype fluid disconnect. Discussions with their project manager and other engineering personnel indicate that the requirements on the CLO fluid disconnect are within the capabilities of their prototype.

The inertial mass properties of the largest SRU on the CLO, SRU 17, have been calculated to verify the performance of the servicer arm. This is the imaging proportional counter, located on the carousel and is removed radially from Docking Port A. The moments of inertia are shown in Figure IV-31 about the end effector fitting, where the SRU is grabbed by the servicer.

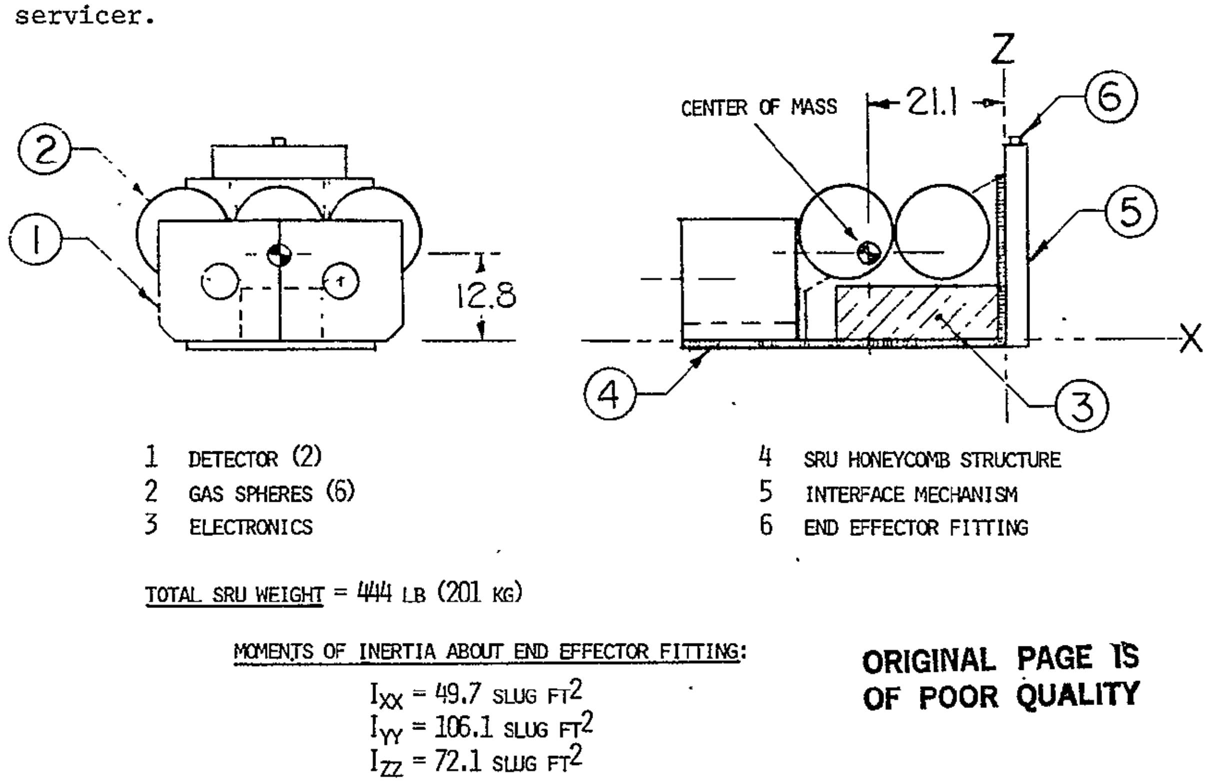

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Figure IV-31 Mass-Properties of Largest CLO Space Relaceable Unit (SRU 17)

The sun aspect sensors require a special SRU design. The design proposed is a new application of concepts from other technologies. The aspect sensors are the same as used on HEAO-B. Each of the three sensors has a very long sunshade. The 75-inch total length of the equipment greatly exceeds the maximum SRU dimension desired by the Servicer.

The "upper" sunshade assembly is constructed of composites and contains knife-edge surfaces. This surface does not see any significant loading conditions, has no moving parts and is highly reliable. Since likelihood of it failing is very small, it will not be serviced. The three "lower" sunshades, made of aluminum, and the three tracker assemblies, will be serviceable in a single SRU. The special feature of this SRU is the tight requirement on lateral alignment as well as in-and-out position.

Precise alignment of the aspect sensors with the telescope mirror assembly is critical; therefore, every means is employed to achieve and maintain precision of alignment. The mirror and optical bench assemblies are rigid entities but are attached to the telescope shell through flexures which provide for thermal excursion.

Although other Docking Port B SRUs are structurally mounted to the shell structure, the aspect sensor equipment is not. Both the SRU portion of the aspect sensors and the fixed sunshade assembly are mounted directly to the mirror assembly. A clearance cutout in the telescope shell structure permits the aspect sensors to bypass the flexure arrangement and behave as a rigid appendage to the mirror assembly.

The aspect sensor SRU design, shown in Figure IV-32, features a sliding guide plate and spring-loaded sleeve. The SRU moves radially (in terms of the telescope) to make or break the optical alignment while the fixed sunshade sleeve travels axially to eliminate any gap between the fixed and moveable components.

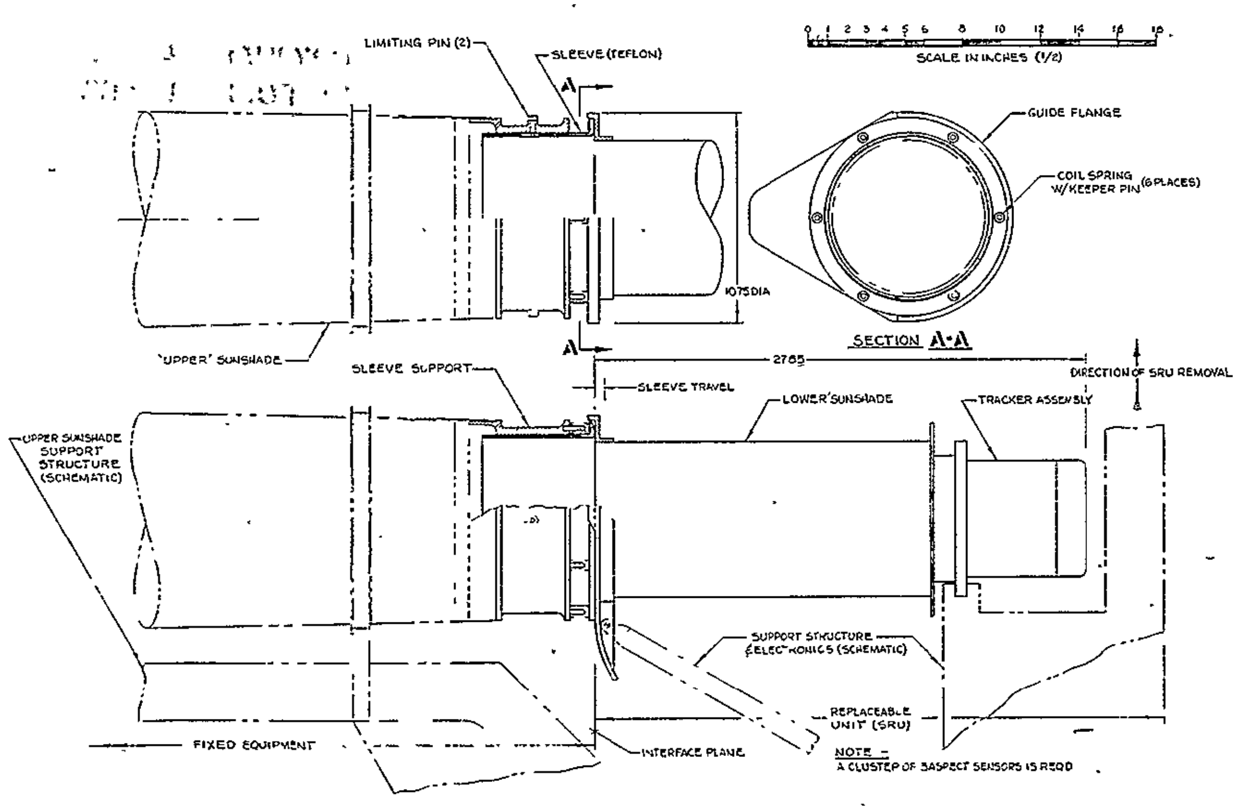

Figure IV-32 Details of SRU 21 (Sun Aspect Sensors)

2. Characteristic Large Observatory - Large Solar Observatory Payload (SO-02-A)

The Large Solar Observatory mission and representative spacecraft[9] characteristics are summarized in Table IV-14. NASA/GSFC is the development agency. Personnel there have supplied or verified most of the data used in this part of the study.

Most of the LSO mission equipment will be flown earlier on Spacelab as sortie payloads,[10] especially the Dedicated Solar Sortie Mission (DSSM). The mission equipment that has been proposed to date for SO-02-A is shown in Table IV-15.

Table IV-14 Characteristics of Large Solar Observatory Mission (SO-02-A)

LARGE SOLAR OBSERVATORY (NASA PAYLOAD NO. SO-02-A)	
PURPOSE	MULTISPECTRAL OBSERVATIONS OF QUIET SUN AND SOLAR ACTIVITY
ORBIT ALTITUDE	350 KM (189 N MI)
ORBIT INCLINATION	30°
DESIGN LIFETIME	24 MONTHS
SPACECRAFT LAUNCH WEIGHT*	9825 KG (21,660 LB)
MAXIMUM DIAMETER*	4.57 M (15 FT)
MAXIMUM LENGTH*	16.2 M (53 FT)
CORE INSTRUMENT	PHOTO HELIOGRAPH (150 CM)

*NON-SERVICEABLE VERSION

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Some of the proposed Large Solar Observatory mission equipment will fit easily into a single standard SRU as developed for the CLO. About a dozen

9. "Summarized NASA Payload Descriptions, Automated Payloads - Level A Data," prepared by Program Development, NASA Marshall Space Flight Center, July 1975.

10. "Payload Description - Volume II, Book 1 - Sortie Payloads, Level B Data," prepared by Program Development, NASA Marshall Space Flight Center, July 1975.

Table IV-15 Proposed Mission Equipment for Large Solar Observatory

NO.*	NAME	DESCRIPTION	MEASUREMENT OBJECTIVE/FUNCTION	DIMENSIONS (M)	MASS (KG)	PEAK POWER (W)	STANDBY POWER (W)	DATA RATE (BPS)
1	CORONAGRAPH, EXTERNALLY OCCULTED	IMAGES CORONA BY OCCULTING SOLAR DISK	OBSERVE CORONAL BRIGHTNESS 2000-7000 Å, 1.5 TO 6 SOLAR RADII	0.60 X 0 60 X 4.60	204	100	16	1.0 E + 06
2	PHOTO-HELIOGRAPH,** 150 CM	TELESCOPE WITH SPECTROGRAPHS	HIGH SPATIAL AND SPECTRAL RESOLUTION IMAGES OF SUN	1 47 X 1.47 X 7 11	1256	800	300	1.2 E + 07
3	SPECTROGRAPH, ULTRAVIOLET	SLIT-GRATING SPECTROGRAPH, TV AND FILM	SOLAR SPECTRUM, 1000-2200 Å	0.50 X 0 50 X 4.00	250	100	7	2 5 E + 03
4	SPECTROHELIOMETER, EXTREME UV	TELESCOPE, SPECTROMETER, POINTING	MAP SUN EMISSION, 280-1700 Å TO 0.1 Å AND 1 ARC-SEC	0 66 X 0 61 X 3.70	270	120	19	1 1 E + 05
5	SPECTROMETER/SPECTROHELIOGRAPH	WOLTER TELESCOPE, GRAZING INCIDENCE	MAPS AND SPECTRAL LINE SHAPES, 100-600 Å	0 20 X 0 40 X 2.00	150	20	2	1.0 E + 03
7	SPECTROMETER/SPECTROHELIOGRAPH, SOFT X-RAY	ODA COLLIMATORS, GRAZING TELESCOPE, FLAT CRYSTAL	CORONAL MAPS AND LINE PROFILES, 0.5-25 Å	0 60 X 1.00 x 4.00	270	100	37	2.0 E + 4
8	PHOTOMETER, GRID-COLLIMATOR ACQUISITION	TWO ALIGNED GRIDS, PHOTON COUNTERS	LOCATE FLARE TO 2 ARC-SEC IN 10-30 SEC	0 25 X 0.25 X 2.00	30	15	0	2.0 E + 2
9	COLLIMATOR, MODULATION	MULTIGRID COLLIMATOR, PHOTON COUNTER	MAP OF HARD X-RAY EMISSION VERSUS ENERGY	0 20 X 0.20 X 3.10	50	15	0	2.5 E + 3
11	FLARE DETECTOR, SOLID STATE	ARRAY OF COOLED Si DETECTORS	FLARE X-RAY SPECTRUM HIGH TIME RESOLUTION	0.50 X 0 50 X 0.50	90	20	0	3.0 E + 03
12	BURST DETECTOR, X-RAY	ARRAY OF SCINTILLATION SPECTROMETERS	DYNAMIC RANGE OF FLARE X-RAY FLUX VERSUS ENERGY, TIME	1 00 X 1 00 X 1.00	300	20	0	3.0 E + 03
13	SPECTROMETER, X-RAY/GAMMA RAY	SCINTILLATION SPECTROMETERS	CONTINUUM FLARE BREMSSTRAHLUNG AND GAMMA RAYS	1.00 X 1.00 X 1.00	900	20	0	1 0 E + 03
14	SPECTROMETER, GAMMA RAY	SCINTILLATOR AND SOLID-STATE SPECTROMETER	NUCLEAR GAMMA-RAY LINES FROM FLARES	1 00 X 1.00 X 1 00	700	20	0	4 0 E + 03
15	POLARIMETER, SOLAR X-RAY	CRYSTAL Li AND Be SCATTERERS, PROPORTIONAL COUNTERS	POLARIZATION OF FLARE X-RAYS, 1-200 keV	1 00 X 0.50 X 1.00	100	15	0	3 0 E + 03
16	POLARIMETER, BRAGG REFLECTION CRYSTAL	CRYSTAL REFLECTOR, PHOTON COUNTER	POLARIZATION OF SOFT X-RAYS FROM FLARES, 1-10 keV	1.00 X 0.50 X 1 00	40	35	0	1 0 E + 03
17	SOLAR NEUTRON EXPERIMENT	TWO SOLID SCINTILLATORS, 32 PHOTOMULTIPLIERS	SOLAR FLARE NEUTRONS, 2-100 MeV	1.00 X 1.00 X 1 30	230	22	0	1.0 E + 02
18	DETECTOR, HIGH ENERGY GAMMA RAY AND NEUTRON	MULTIPLE SCINTILLATOR DETECTOR	SOLAR NEUTRONS AND GAMMA RAYS ABOVE 10 MeV	0.40 X 0.40 X 0 40	100	5	0	5.0 E + 02
-	SPECTROHELIOGRAPH, HIGH SPECTRAL PURITY	PLANE GRATING, ZONE	MAPS IN SELECTED LINES, 20-100 Å	← N/A →				
-	TELESCOPE, X-RAY	GRAZING INCIDENCE FOCUSING TELESCOPE	HIGH SPATIAL RESOLUTION MAPS IN COARSE X-RAY REGIONS	← N/A →				
-	DETECTOR, SOLAR GAMMA RAY	SPARK CHAMBER	DECAY GAMMA RAYS FROM FLARE PIONS	← N/A →				

*1 = SO 001, 2 = SO 002, ETC.

**CORE INSTRUMENT

experiments, however, exceed the baseline servicer capabilities as a single SRU and may need to be repackaged.

The major items which cannot be serviced easily are the large telescopes, optical benches, precisely aligned optical surfaces and long collimators. These are the same types of items that are not serviced for the 1.2 meter X-Ray Telescope mission.

A TRW solar physics specialist and spacecraft mechanical designer have developed repackaged serviceable configurations for each equipment that exceeds the nominal SRU capabilities. Examples of repackaged mission equipment are shown in Figures IV-33, -34, and-35.

The Externally Occulted Coronagraph, Experiment No. 1 in the mission equipment list, is a long optical cylinder with recording and support equipment located in boxes at the rear of the Coronagraph and along the side (Figure IV-33). The length of the cylinder (4.6 m) makes it difficult to service the entire experiment as a unit. TRW recommends that the

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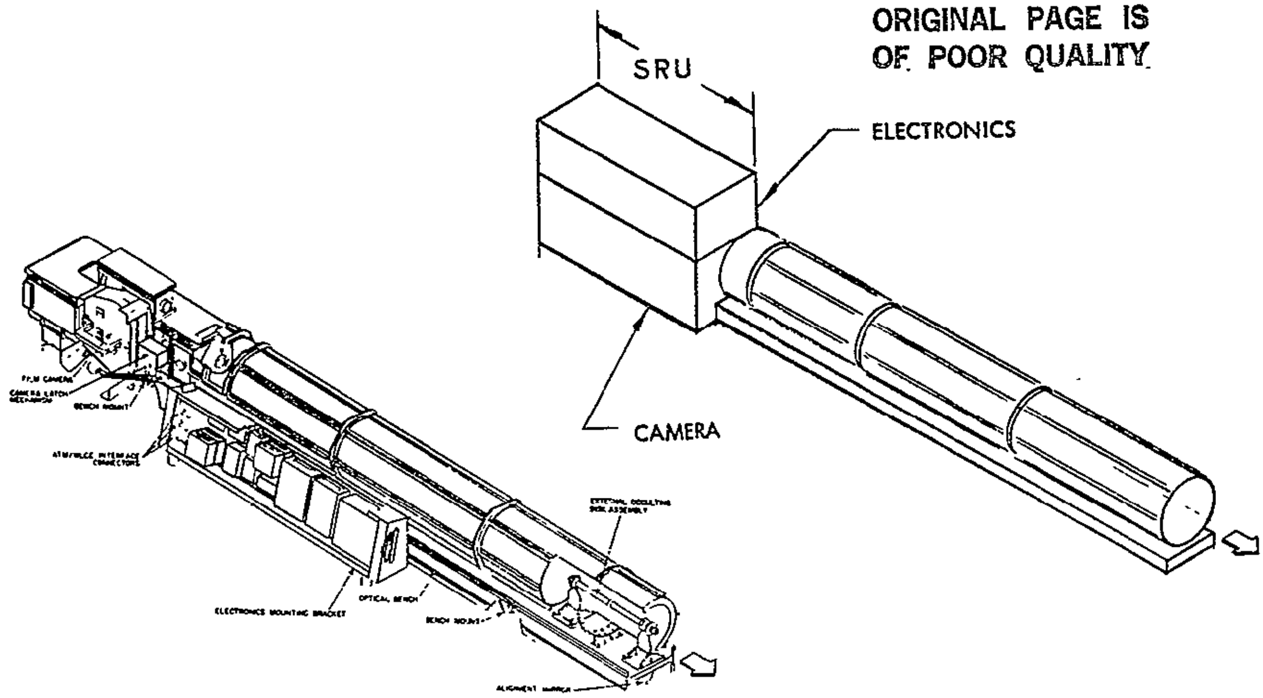
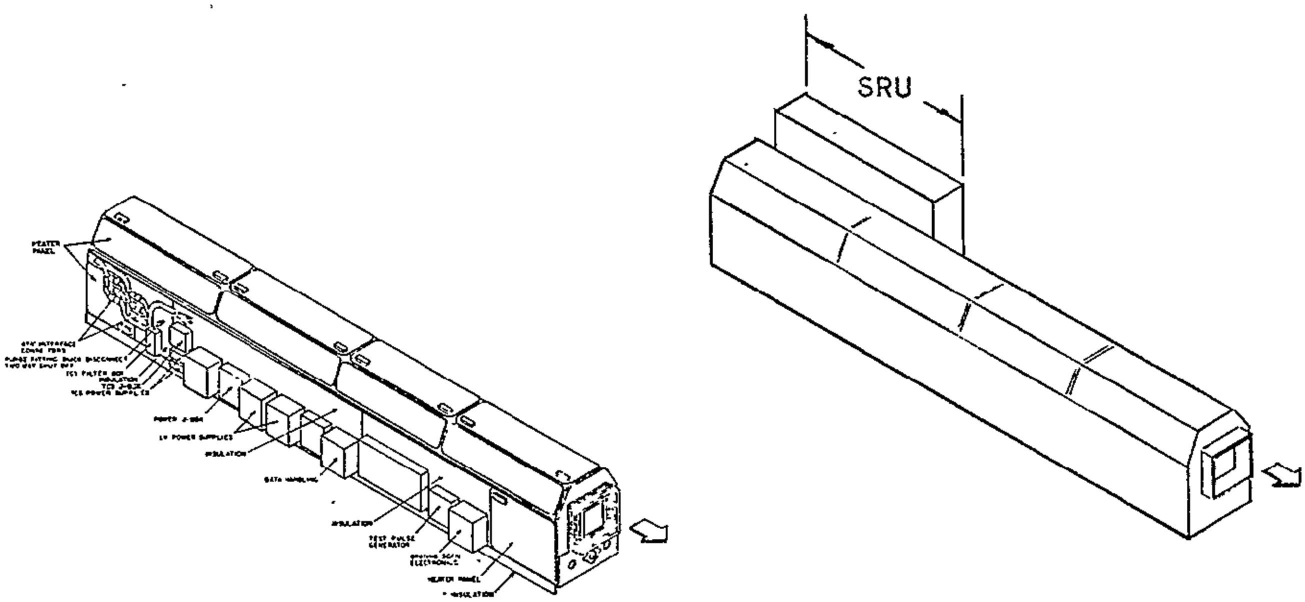


0.6M x 0.6M x 4.6M
204 KG

Figure IV-33 Repackaging of Externally Occulted Coronagraph for Servicing



0.5M x 0.5M x 4.0M
250 KG

Figure IV-34 Repackaging of Ultraviolet Spectrograph for Servicing

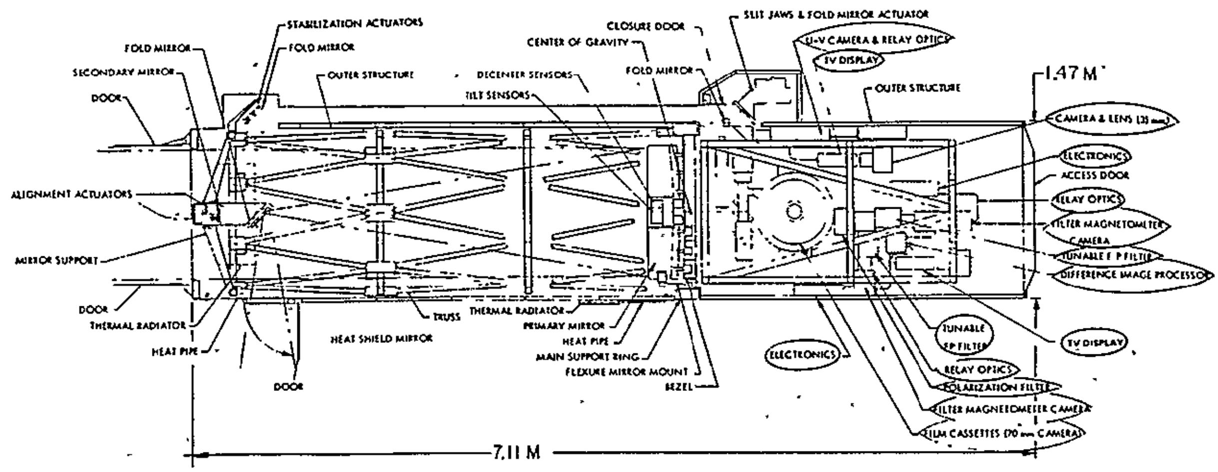


Figure IV-35 Repackaging of Photoheliograph for Servicing

camera and electronics be re-packaged as one or more SRUs and mounted at the rear of the optics. The optical cylinder would be a Non-Replaceable Unit.

Experiment No. 3, the Ultraviolet Spectrograph, is currently packaged with the support equipment contained in many small boxes along the length of the Spectrograph (Figure IV-34). In our opinion most or all of this equipment can be re-packaged for serviceability at one location, perhaps at the rear as shown. An important design problem will be the vacuum seals.

The Extreme UV Spectroheliometer (No. 4) and High Spectral Purity Spectroheliograph are similar in design to the Ultraviolet Spectrograph. They can be re-packaged for serviceability in much the same manner.

The Photoheliograph, Figure IV-35, exceeds the nominal SRU maximum in both mass and dimension. It is unlikely that the optical structure on the left part of the sketch can be removed and realigned in orbit. The recording equipment on the right can probably be serviced, however. In particular, all the equipment whose titles are circled.

It will probably be necessary to re-package the Photoheliograph equipment into several SRUs. The center of gravity location indicates that at least half the 1256-kg instrument mass is at the right end. Depending on the interactions between units, this experiment may require two or three separate SRUs.

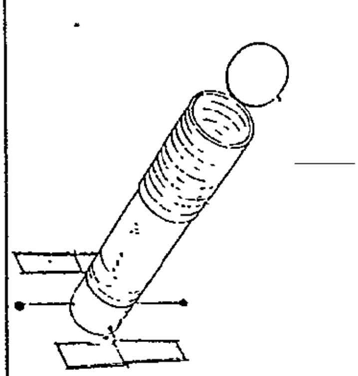
Since SO-02-A subsystem requirements are lower than for HE-11-A, the same housekeeping SRUs can be used. Therefore the CLO concept can be used for the Large Solar Observatory mission.

3. Characteristic Large Observatory - Space Telescope Payload (AS-01-A)

The 2.4 meter Space Telescope mission,[9,11,12,13] has recently been designated as a major NASA new start. This mission will represent a quantum jump in the ability to carry out astronomical studies in a favorable space environment. The data used in this section do not reflect details developed as part of the current hardware procurement process, but are representative of this class of missions.

The Space Telescope mission is summarized in Table IV-16 and the seven (7) candidate Scientific Instruments (SI) for the mission are described in Table IV-17. Four SI will be selected for each flight.

Table IV-16 Characteristics of Space Telescope Mission (AS-01-A)



PURPOSE	PRECISION OBSERVATIONS OF ALL TYPES OF PLANETARY, GALACTIC AND EXTRA-GALACTIC OBJECTS.
ORBIT ALTITUDE	500 KM (270 N MI)
ORBIT INCLINATION	28.2°
DESIGN LIFETIME	36 MONTHS
SPACECRAFT LAUNCH WEIGHT	9500 KG (20,900 LB)
MAXIMUM DIAMETER	4.3 M (14.1 FT)
MAXIMUM LENGTH	13 M (42.7 FT)
TELESCOPE	RITCHEY-CHRETIEN, 2.4 METER APERTURE
CORE INSTRUMENT	FIELD CAMERA -F/24

11. "Payload Descriptions, Volume I - Automated Payloads, Level B Data (Preliminary)," Marshall Space Flight Center, July 1975.

12. "Large Space Telescope Science Instrument Final Review - Goddard Space Flight Center," prepared by Perkin-Elmer, July 1975.

13. "The Space Telescope", NASA SP-392, National Aeronautics and Space Administration, 1976.

Table IV-17 Proposed Mission Equipment for Space Telescope

NAME	DESCRIPTION	MEASUREMENT OBJECTIVE/FUNCTION	DIMENSIONS (M)	MASS (KG)	PEAK POWER (W)	STANDBY POWER (W)	DATA RATE (BPS)
FIELD CAMERA, F/24*	70-MM SECONDARY ELECTRON CONDUCTION ORTHICON (SECO) CAMERA	DISCOVER NEW TARGETS	ALL SPACE TELESCOPE SCIENTIFIC INSTRUMENTS ARE DESIGNED TO FIT IN A STANDARD MODULE ENVELOPE. THE ENVELOPE APPROXIMATES A 90-DEGREE CYLINDRICAL QUADRANT AND CAN FIT WITHIN A RECTANGULAR SOLID OF 0 8 X 0 8 X 2 1 METERS.	210	130	70	N/A
FAINT OBJECT SPECTROGRAPH (FOS)	SLIT-GRATING SPECTROGRAPH, SLIT JAW CAMERA	DISTRIBUTION OF ENERGY FROM FAINT CELESTIAL OBJECT VERSUS WAVELENGTH, 90-800 NANOMETERS		208	132	56	1.0 E + 06
HIGH SPEED POINT/AREA PHOTOMETER (HSAP)	ULTRAVIOLET-VISIBLE PHOTOMETER OPERATING IN A PHOTON-COUNTING MODE WITH OPTIONAL ANALOG MODE	PRECISE MEASURE OF CONSTANT OR TIME VAPIABLE INTENSITIES OVER A WIDE DYNAMIC RANGE OF ASTRONOMICAL SIGNAL STRENGTHS		145	91	26	1.0 E + 06
INFRARED PHOTOMETER	DEWAR, SUPER-COOLED INTERNAL DETECTORS, EXTERNAL OPTICS	PHOTOMETRY OF POINT OR EXTENDED SOURCES IN MID- AND FAR-INFRARED WAVELENGTHS		309	31	23	N/A
HIGH RESOLUTION SPECTROGRAPH (HRS)	INTERCHANGEABLE ECHELLE AND FIRST-ORDER GRATINGS, TWO SECO CAMERAS	IMAGING SPECTROSCOPY OF POINT OR EXTENDED SOURCES		270	132	22	1.0 E + 06
ASTROMETER	RECORDS POSITIONS USING THE ASTROMETRIC MULTIPLEXING AREA SCANNER (AMAS) TECHNIQUE	MEASURE PARALLAXES, PROPER MOTIONS, ANGULAR DIAMETERS AND INDIVIDUAL MASSES		136	60	35	N/A
PLANETARY CAMERA, F/48/96	TWO SELECTABLE FOCAL LENGTHS IMAGING ON A COOLED CHARGE-COUPLED DEVICE (CCD) DETECTOR	IMAGING PHOTOMETRY AT HIGH ANGULAR RESOLUTION FOR HIGH SURFACE BRIGHTNESS OR MULTIPLE BRIGHT SOURCES		126	90	50	N/A

*CORE INSTRUMENT

As noted, all of the equipment is designed for manned on-board serviceability. The standard module is designed to fit as a quarter of the cylindrical section for experiments along the telescope axis. The equipment masses vary from 126 to 304 kg, well within the standard SRU limit. The only parameter that exceeds the baseline servicer capabilities is the 2.1 meter length. If this cannot be accommodated, it is possible to repackage the electronics on several of the SIs so that they can be serviced by the baseline servicer. Examination of the instruments has shown that the same considerations for re-packaging that were developed for the Large Solar Observatory are true for Space Telescope.

In summary, it appears the CLO serviceability concepts can be applied to Space Telescope missions.

V. ON-ORBIT SERVICER SYSTEM

This chapter will discuss the overall servicer system development approach and more specifically the approach followed during this contract in carrying that development process through a preliminary hardware design phase. While the somewhat philosophical nature of this section would seem to be more appropriate in the introduction of the report, it was considered to be pertinent here for the following reason. A better understanding of the approaches and design philosophy in this chapter is possible with the perspective provided by the servicer requirements in Chapter II, the selection of a basic servicer configuration in Chapter III and the design of maintainable spacecraft in Chapter IV.

The overall servicer system development plan will be presented in Section A to better understand the pivotal nature of the servicer work currently being done. The servicer design approach, discussed in Section B, will show how the multitude of information that was collected and derived in Chapters II, III and IV was utilized to make the tradeoffs and develop the concepts necessary to arrive at an optimized design that met the desires of both user and supplier of the servicing system. Some of the thought processes that went on and the driving requirements and dominant influences that eventually evolved will be presented. In essence, Section B is a reexamination of a number of basic guidelines that were established at the outset of the contract and an expansion of them into a form necessary to proceed to the more detailed level of servicer design in Chapters VI (mechanical) and VII (controls).

This chapter will conclude in Section C with a description of specific requirements and/or characteristics of those elements of the Space Transportation System (STS) that are necessary to support on-orbit servicing. While not related directly to the discussions in A and B, the STS is essential in a servicing mission and must be well understood before hardware design is initiated. The data in this section goes beyond the STS requirements in Chapter II, providing modes of operating, available capabilities and constraints on the STS as well.

A. SERVICER SYSTEM DEVELOPMENT APPROACH

The overall servicer system development is pictorially represented in Figure V-1. Many studies for many agencies have been conducted by many companies prior to MMC's first IOSS. Most of them were directed at a specific spacecraft design. The objective of the first IOSS included placing these many studies on a common basis. The primary tasks of that study are shown in the first block. The conclusion was that orbital maintenance is indeed cost effective. Interface mechanisms were designed and fabricated and a preliminary servicer design accomplished.

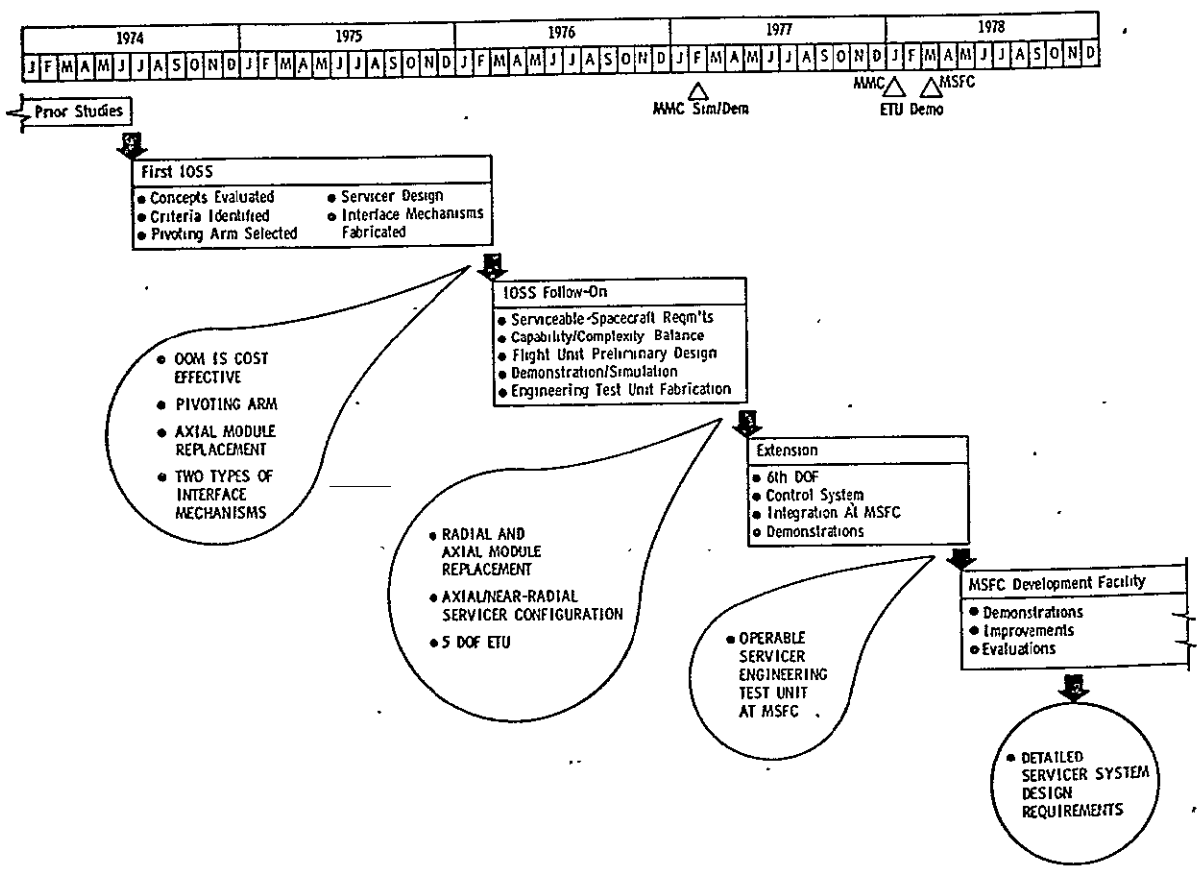


Figure V-1 Servicer System Development Approach

The study being reported on here is a follow-on to that first IOSS. Its objective was to carry the work of the first study to a more detailed level of design.

The systems design approach which guided most of the effort of this follow-on started with a set of requirements, expanded the concepts of the first IOSS, then allocated functions so that a total working system was formed. The functions were allocated to the set of elements shown in Figure V-2. Requirements were prepared for each element. Designs were prepared for each of the major elements at various levels of detail.

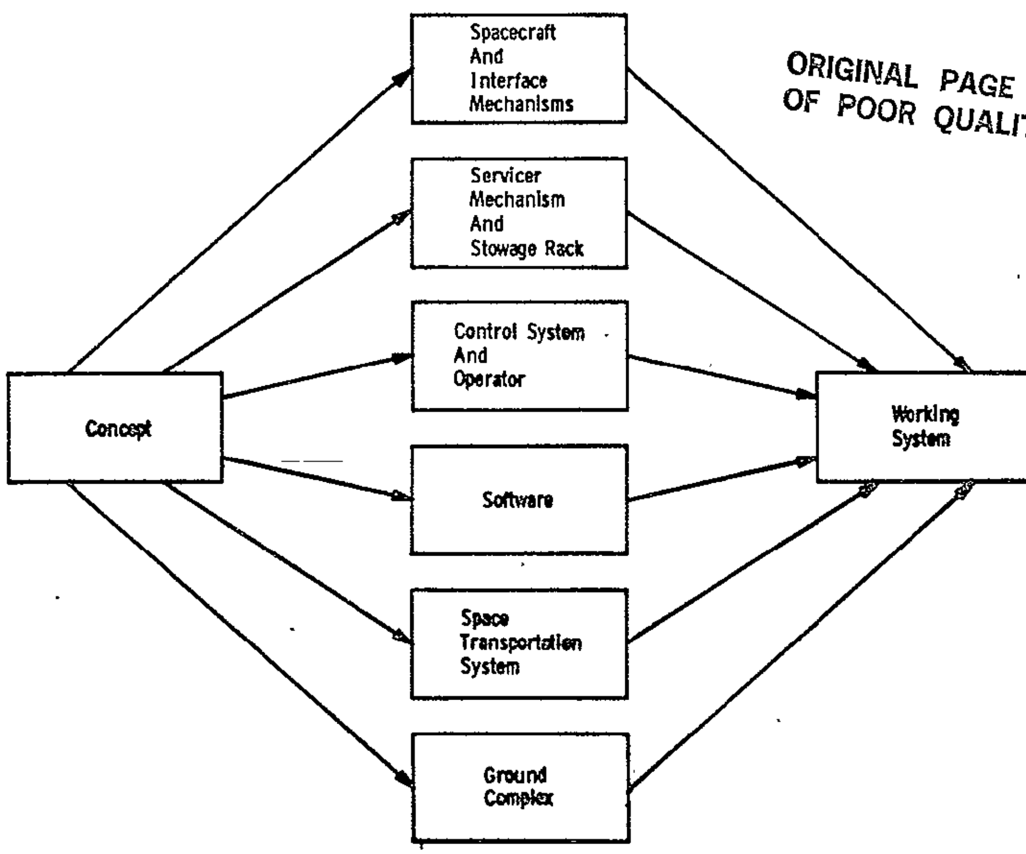


Figure V-2 On-Orbit Servicing Elements

The follow-on contract included a simulation/demonstration that went a long way towards verifying the concept and design feasibility.

The challenge in subsequent phases of development is to build representatives of the various blocks in Figure V-2 and to evaluate interactions between the blocks. The results of these evaluations will then be used to define the next steps in the on-going development plan. While the study resources have been effectively used to define the various blocks and the preliminary analyses indicate the interactions are understood, it is well within the range of possibility that unexpected events can occur or that new avenues for system improvement may open up.

The first of these subsequent phases is shown on Figure V-1 as an extension of the current IOSS contract. It is planned to continue the hardware development. Control electronics will be fabricated instead of simulated, and a full six degrees of freedom will be incorporated in a preprototype servicer mechanism called the engineering test unit (ETU). A demonstration again will be conducted with the more flight-representative equipment. The ETU and control electronics will be delivered to MSFC for use as a demonstration and evaluation tool. The remaining activities planned in this NASA servicing system development are centered on use of the tool at MSFC to continue evaluations of the design approach and refinement of characteristics and requirements. The downstream objective is arriving at enough detail in design approach to prepare a well justified servicing system design specification.

A separately documented Implementation Plan has been prepared as MCR-76-258 by Martin Marietta, April 1978, which relates and expands upon the servicer system development activities shown in Figure V-1. It discusses in particular, those activities that should be carried out beyond the scope of those shown on the figure. It emphasizes the need for parallel development of the servicer system and a serviceable spacecraft, the need for an on-orbit demonstration as part of the Orbiter Flight Test program, and the need for a continuing program which will help to develop user confidence in the viability of the concepts.

When the servicer system reaches an operational stage, then the manager of each new spacecraft program will need to assess whether his spacecraft should be made serviceable. If he decides to incorporate servicing, then he and his spacecraft designer must determine which of the five modular forms best suits their needs and will minimize their total life cycle costs. Figure V-3 summarizes, at a top level, the process and considerations in such a selection.

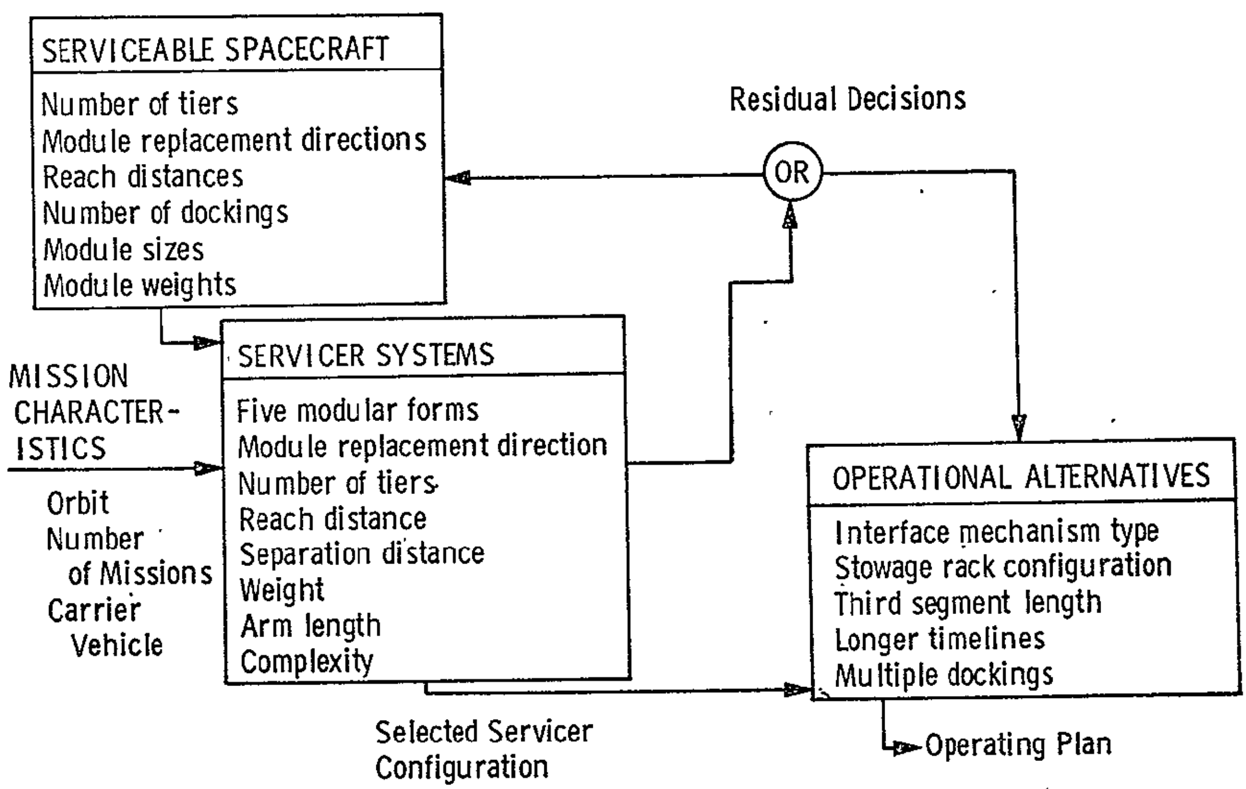


Figure V-3 Selection of Servicer Modular Form to Match Satellite Requirements

A preliminary definition of the spacecraft is first made. This is then compared with the servicer capability available while keeping in mind the mission characteristics. A first selection of a servicer configuration is made and any residual decisions are noted. A first estimate of life cycle costs can then be made. The process is iterated until satisfactory life cycle costs are obtained.

Note that the residual decisions can be addressed either through spacecraft redesign or the operational alternatives listed. When all residual decisions have been resolved, an operating plan can be prepared and the resulting life cycle costs determined. This discussion has been included to illustrate how the proposed servicer system modular forms introduced in Chapter III can be used by the spacecraft program manager to develop program options to save money and by the spacecraft designer to prepare spacecraft configuration alternatives that will evolve to a best satisfaction of mission requirements.

B. SERVICER SYSTEM DESIGN APPROACH

The primary role of the servicer is to provide a service to the spacecraft community to increase the lifetime of their spacecraft through on-orbit replacement of failed components, whether they are subsystem or mission equipment hardware. To be cost effective, the servicer system must be applicable to a wide range of spacecraft programs. The work of Chapter III was directed towards identifying a single servicer system that could satisfy the needs of the vast majority of spacecraft programs while not being unduly influenced by a few unique special cases. The servicer then must accommodate potential users to the fullest extent. In fact, without a very high degree of accommodation of the spacecraft community's needs the servicer concept will probably never fly. So these desires must be adhered to. At the same time the servicer designer and developer has some desires that are in conflict with those of the user. Further compounding the situation is the continued uncertainty regarding the specifics of the user's desires and requirements. Fortunately, a probable upper bound on serviceable spacecraft requirements was evolved in Chapter II.

It is resolution of these conflicts, while minimizing total servicer life cycle costs, that is of primary interest in this chapter. Some of these have already been resolved in the configuration selection of Chapter III. The more obvious requirements of the user, as well as supporting systems such as STS and ground operations, are:

- High capability and versatility;
- Minimum impact on spacecraft design;
- Minimum impact on supporting systems;
- High reliability;
- Life cycle cost savings;
- Simple operations;
- Continuity of development and operations.

On the other hand the servicer design would like:

- Simple configuration;
- Simple design;

- Low development risk;
- Minimal, simple interfaces.

The real conflict reduces to simplicity on the part of the servicer designer vs the desire for a highly capable and versatile servicer design with minimum spacecraft impacts on the part of the user.

The first step in resolving this conflict was to review the servicer system requirements of the first IOSS and updating them using the results from Chapters II, III, and IV. The resulting set of requirements was separated into four sets. The system level requirements are shown here as Table V-1. The servicer mechanism, interface mechanism, and stowage rack requirements were expressed as design characteristics and are given in Chapter VI.

Table V-1 Servicer System Level Requirements

- Impose minimum restrictions on the spacecraft and module designers by allowing flexible and efficient packaging of modules on spacecraft.
- Be compatible with most automated serviceable spacecraft.
- Compensate for tolerances/misalignments in 6DOF.
- Avoid initial module to opening close-fit requirements.
- Operate interface mechanism latches.
- Interface mechanism to provide connector make/break forces.
- Interface mechanism components shall be mechanically passive.
- Be compatible with EVA.
- Generate operational status signals.
- Minimize sliding friction areas.

A major step towards resolution of the conflict of simplicity vs versatility was selection, in Chapter III, of a simple, single tier, servicing arm, constrained to motion along cylindrical coordinates. High capability and versatility were provided through modularity or simple add-on segments that provided a potential for more tiers or larger diameters. The objective was to carry that same theme forward through the remaining design phase.

The geometry of the servicing task was a major influence in the configuration selection and continued to guide the remaining design as well. This geometry is illustrated in Figure V-4.

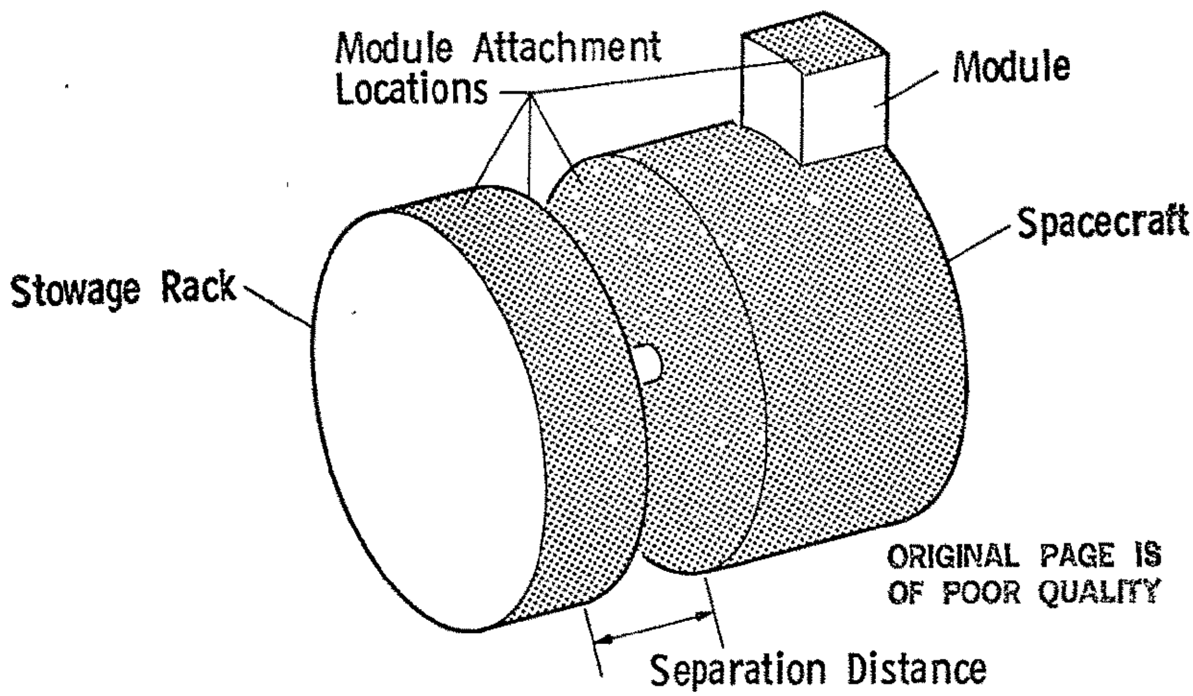


Figure V-4 Servicing Task Geometry

Note that the module attachment locations form a surface of revolution about the spacecraft centerline. The base of the servicer arm is conveniently mounted on the axis of the cylinder which also becomes the docking axis. The radially mounted modules extract ideally along the radius direction of the cylinder while the axial mounted modules extract along the other coordinate -- the axial direction -- of the cylindrical elements.

It was found that this is a useful characteristic, not only in evolving the selected configuration into mechanical hardware design, but also in the control system design, permitting the trajectory to be broken into simple sequences that minimize the need to drive more than one joint at a time.

As compared to the capabilities of a general purpose manipulator, such as configuration 10 in Chapter III, the task to be performed by a servicer system can be arranged to be relatively simple. All modules effectively look the same to the servicer and they only need to be removed, flipped, relocated, inserted, attached, and latched. There are no other actions and each module sees the same actions. This simplicity of task was recognized and used over and over again in the preliminary design process.

Another key feature influencing trade studies and designs is also apparent from that same geometry. All the elements of the servicing task are completely and accurately defined prior to the flight. The stowage rack physical dimensions and envelope are very accurately known; the spacecraft has, of course, been dimensioned in detail prior to flight. The vehicles are aligned within 0.1° (1σ) in all axes by the docking system on orbit. Consequently, all the module locations and hazards are known ahead of time. This permits complete trajectory and sequence definition prior to flight. This is an obvious advantage to the control system which must implement the necessary steps to complete the exchange. Simple, accurate, automated sequences are possible (remove, flip, relocate and insert). Note also that there is no great time criticality so emphasis can be placed on slow, deliberate and smooth motions.

Many of the design trades revolved around the interactions illustrated in Figure V-5. As noted in Chapter II the serviceable spacecraft configurations and their requirements are the primary drivers on servicer mechanism configuration selection and requirements. These are expressed in terms of module sizes, number of tiers, module replacement direction, module location, module weight and number of dockings. The stowage rack configuration and requirements are primarily driven by the servicer mechanism and only slightly by the spacecraft configuration.

C-3

The point is that the stowage rack configuration need not be the same as the spacecraft configuration, rather it should be designed to suit the servicer configuration. This is reflected in Chapter VI where the servicer mechanism can handle radial module removal, but the stowage rack is primarily an axial module removal configuration. The stowage rack configuration is affected directly by certain spacecraft module data, namely number, size, and weight.

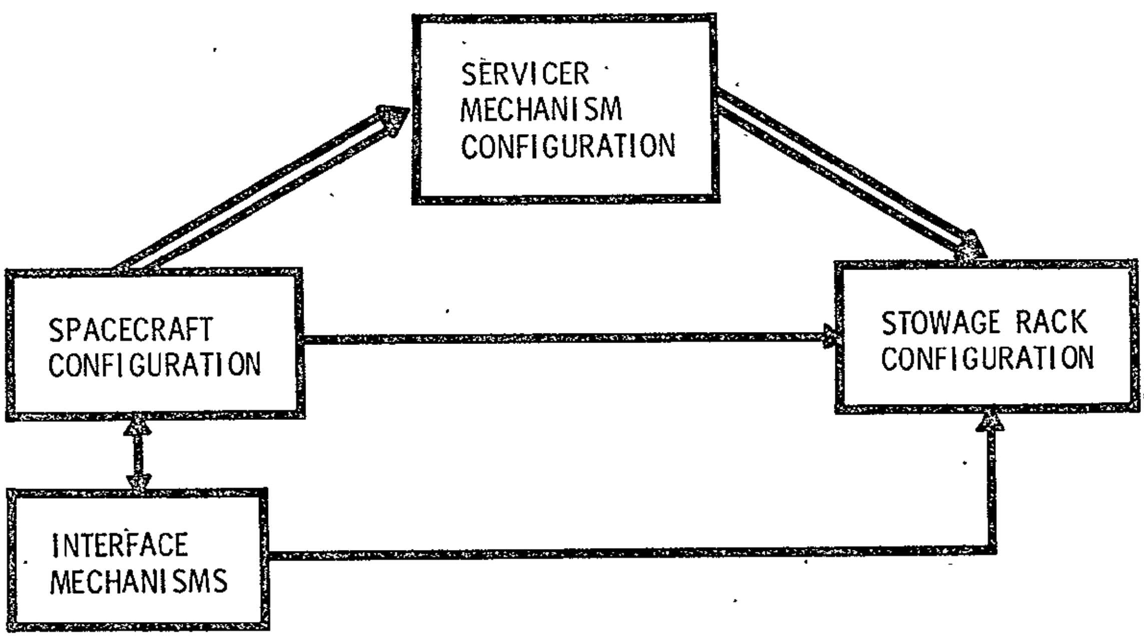


Figure V-5 Element Interactions

The module interface mechanism configurations are more closely related to spacecraft design. They need only have consistent interfaces with the servicer mechanism and with the stowage rack. A variety of types and sizes of module interface mechanisms are possible and perhaps the spacecraft designer should be given the option of using standard interface mechanisms or developing his own design. This is the recommended approach.

How these features and the selected configuration, to be further detailed in Chapters VI and VII, come together into a proposed design is shown in Figure V-6.

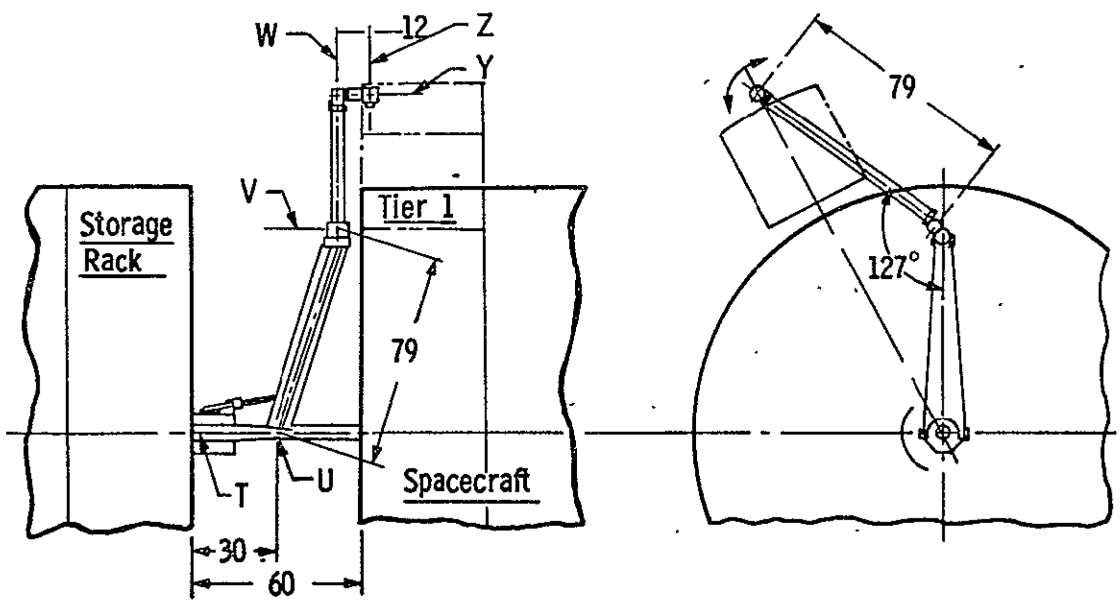


Figure V-6 Selected Servicer Configuration

From the figure the potential for growth through modularization is apparent. Arm segments can be replaced with longer or shorter length segments. The number of degrees of freedom and arm segments can be reduced or increased.

The spacecraft servicing requirements groups, shown in Table V-2, can all be accommodated by modular changes to the design. Refer to Chapter III, Section C for more details on how this is accomplished.

Table V-2 Servicing Requirements Groupings

Removal Direction	Number of Tiers	Designation
Axial	1	Axial
Axial and Radial	1	Axial/Near-Radial
Radial	1	Near-Radial
Radial	2	Two-Tier Radial
Axial and Radial	2	Axial/Two-Tier Radial

The modularized design could cost somewhat more to develop but peak funding can be spread over more years while an initial, simple version is evolved into broader capabilities. Ultimately, spacecraft tailoring of servicer capabilities to each spacecraft should decrease the per mission servicing costs and keep weight at a minimum as well. Note that the axial/near-radial configuration, which was selected for near term development, lies in the middle of the range of complexity and capability. It can be readily incremented up and down to the other four modular forms.

Another area where good design practice can lead to a simple, yet versatile servicer design is in the proper allocation of the conduct of the servicing functions between the man, the control system and the mechanisms. The mechanisms can carry some of the load of the servicing operations by providing backdriving joints that can automatically correct for strain relief and misalignments. Proper selection of arm lengths and joint orientations can simplify the control system design and ease the operator's training. The control system bears much of the responsibility of accomplishing the servicing activities. Its implementation can do much to simplify the operator's tasks. Sequences can be stored and automated to whatever degree is desired. Procedural displays can be incorporated via the use of software. Failure detection and hazard avoidance features can be included. Others are discussed in Chapter VII. The value of man in the loop to add flexibility and versatility yet maintain simplicity will become more obvious in Chapter VII. Man is invaluable in placing value judgments on successful completion of the tasks in a module exchange operation. With a properly configured control system he can also provide effective yet highly capable backup control of the servicer.

In conclusion, the major conflict of simplicity vs high capability and versatility has been very effectively resolved by taking full advantage of the unique geometry of the servicing tasks and by optimizing the role of the three major elements in servicing -- control system, man and mechanism. A number of examples of how this theme was carried out will be seen as the detailed design is discussed in the subsequent chapters.

C. SUPPORTING SYSTEMS DESIGN

This section will describe the characteristics and/or requirements of the major systems supporting the servicer system during a typical on-orbit servicing mission. This data, while necessary in the design process of the servicer, also helps to understand some of the capabilities as well as constraints that aid or restrict the servicing function. The discussion is divided into support equipment provided by: (1) the carrier vehicle, in general; (2) by the Shuttle Orbiter uniquely; and (3) by the ground and flight operations elements. Characteristics of the docking system are discussed in Section (4).

1. Support Equipment Provided by the Carrier Vehicle

The support equipment required by the on-orbit servicer and provided by either the Orbiter or Upper Stage (Tug) is addressed first. The on-orbit servicer can readily be designed to be compatible with both carrier vehicles with only minor changes. The sources of data for this evaluation are the *Space Shuttle System Payload Accommodations*, JSC-07700, Volume XIV, and the Final Report of the *Space Tug Avionics Definition Study*, Report No. CASD-NAS75-012, Contract NAS8-31010, by General Dynamics.

a) Electrical Power - The servicer shall be designed to operate from standard spacecraft power provided by the carrier vehicle. 28 VDC power will be supplied to the servicer through a two-wire system providing a power return. Power ground will not be thorough the spacecraft skin. A separate backup power interface will be provided. Each power source will be capable of providing the total power requirements of the servicer. The operational power requirement of 290 watts for the servicer is based on three drives operating simultaneously -- drive heaters, video system, and electronics. The standby power of 50 watts is based on heater and partial electronics power only. Both power requirements include a 20% growth margin. These power requirements can be met by any of the five orbiter payload bay power interfaces and the upper stage payload power allocation. Ripple will not exceed 0.9 volts peak-to-peak at any single frequency and 1.6 volts over the broad band. Any requirement for higher quality DC or AC power shall be satisfied by the servicer.

b) <u>Telemetry</u> - Both the Tug and Orbiter provide the downlinking of data required by the servicer. The characteristics are as shown below in Figure V-7. The Orbiter interface is simplified due to the extensive payload services provided by the Orbiter. Formatting for downlink is provided as well as a real-time video channel. The Tug provides submultiplexing of housekeeping data with Tug data for transmission. The video data, however, is transmitted on a lower rate (256 Kbps), hard-wired channel. This requires the servicer to format and provide the necessary header data compatible with TDRS, STDN or Orbiter for the video data.

- The carrier vehicle shall provide --
  - downlinking of servicer data in real time
  - clock
- Space Tug
  - 10 Kbps housekeeping data submultiplexed with Tug data
  - 256 Kbps hardwired telemetry link for video data. The servicer shall format the video data for transmission.
- Orbiter
  - 64 Kbps channel for housekeeping
  - composite video channel
  - the Orbiter shall provide multiplexing, formatting, block interleaving of data as required for display, record, and/or transmission to the ground

Figure V-7 Telemetry

c) <u>Command</u> - The carrier vehicle shall provide:

- Command reception, demodulation, error detection, correction, and command distribution;
- Commands shall be issued in real time or from memory;
- Command table programming and variable update;
- Sequence initiate commands;
- Real time discrete commands for remotely manned control.

It is required that the command system for the servicer be commanded under several modes of operation. Commands may be issued from a preprogrammed sequence within the servicer initiated by a command from the ground or the Orbiter Payload Specialist Station. An example of a preprogrammed sequence is the servicer checkout sequence. Command capability to alter a sequence such as this or to update information in memory is available if it is required. Operation of the servicer in a remotely manned control mode requires a capability of the carrier vehicle to process real time discrete commands. The carrier vehicle will provide the capability to receive uplink commands and distribute them according to address.

d) Attitude Stabilization - The servicer will not have any attitude control capability. All requirements will be met by the carrier vehicle. The error analysis performed during this IOSS follow-on study assumed that the maximum error in docking accuracy was 0.1 deg in all axes. This accuracy may be met through final mechanical latching techniques, however, the carrier vehicle must provide docking capability to permit a final alignment to within these tolerances. Attitude during servicing operations shall be maintained with less than a 1500 ft-lb moment.

The carrier vehicle shall provide thermal maneuvering such as a "rotisserie" mode to minmize thermal gradients during orbit transfer.

2. Support Equipment Provided by the Orbiter

The particular support equipment provided by the Orbiter are as follows:

a) Caution and Warning - Caution and warning (C&W) capability is provided by the Orbiter for payloads to alert the crew to anomalies which require flight crew attention. With respect to servicing, C&W parameters may originate in the servicer, the serviceable spacecraft, or the replacement modules. As a goal, servicer hazards shall be constrained such that actual or impending failures will not propagate into the Orbiter subsystems or cause potential injury or harm to the crew or vehicle. If required for the servicer, serviceable satellite, or replacement modules, C&W parameters will be distributed to the Orbiter through the Servicer/Orbiter C&W interface channel.

b) Video System - The Orbiter shall provide: 1) servicer video display and record; 2) real time video transmission to the ground; 3) additional payload bay cameras may be used as required. The Orbiter closed circuit television (CCTV) system provides for monitoring of payload bay and cabin area activities and accepts up to three simultaneous standard input signals from an attached payload. The payload TV must utilize horizontal and vertical synchronous signals provided by the Orbiter CCTV for synchronization. The composite video signal is transmitted through the video switching network to the Orbiter monitors, the FM signal processors for S-band FM transmission, and to the Ku band signal processor for transmission via the Ku-band communications link.

c) Control Station Location - The Shuttle Aft Flight Deck contains the Payload Specialist Station which is varied from flight to flight to accommodate the particular needs of the payload. These panels are the logical location for the C&D functions of servicing. The capabilities required for servicing are: 1) software programmable for specific flight display requirements; 2) TV monitors; 3) three degree of freedom hand controllers; 4) tailored panel for lights, meters and controls for monitoring performance and backup control. Greater visibility on the payload specialist station and its function and role are provided in Chapters VII and VIII.

d) Shuttle Remote Manipulator System - The SRMS can be used to aid and augment on-orbit servicing from the Shuttle Orbiter. Modules too large for the servicer can be exchanged using the SRMS. The primary use of the SRMS is the capture, docking to the servicer, and deployment of the serviceable satellites. It will also be used to relocate and reorient the servicer system and adapter within the Orbiter cargo bay.

3. Ground and Flight Operations

Some areas of interest regarding this category are provided below. More detail in several of the following operations areas is supplied in Chapter IX.

a) Rendezvous - The servicer is essentially a passive payload in all operations except the actual servicing of spacecraft. As such, the capability to rendezvous and dock with serviceable satellites is provided by the carrier vehicle. For low earth orbit spacecraft, the Shuttle is utilized for rendezvous. In high earth orbit an upper stage will provide the rendezvous capability.

b) Operational Timelines - The total time to change out a module in the supervisory mode has been estimated to be 9.2 minutes. This was determined based on the selected joint maximum rates and accelerations during a sequence of activities for module exchange using an axial servicer for axial module removal. It is estimated that six module exchanges will be performed on the average per servicing. By allowing 30 minutes for servicer unstow and checkout, and 15 minutes stow, the average servicing will require approximately two hours.

Timelines for maintenance mission types were presented in the Final Report of the first IOSS study. These timelines estimated that from 30 to 100 hours were available for servicing. With two hours required for servicing, a well planned servicing mission would easily have adequate time for additional rendezvous and multiple servicings.

c) Communication With Serviceable Satellite - During servicing it is necessary that a command and data link be set up between the serviceable spacecraft and the servicer. The command link will allow the servicer to place the spacecraft in a quiescent mode for servicing and power the vehicle to operational configuration after servicing. Nominally, a powered-down configuration would be the status during servicing. A data channel and partial power may be required under the circumstances of servicing a spacecraft with hazardous components necessitating the monitoring of C&W parameters.

d) Contamination - The Orbiter payload bay will be at a Class 100,000 cleanliness level. This will be adequate for the servicer. The exchange modules -- particularly in the case of optical instruments -- will probably require Class 10,000 cleanliness levels. Several methods exist for providing the cleaner levels — overpressurizing the modules, a continuous gas purge, and bagging are a few. Additional services required for the cleaner levels will be payload chargeable and not the responsibility of the carrier vehicle.

e) <u>Checkout of the Servicer</u> - Servicer checkout on-orbit prior to servicing will be performed from the Payload Specialist Station for Orbiter operations and from the ground for upper stage operations. A preprogrammed command sequence and data limit checks will be performed with man in a supervisory position.

4. <u>Docking System</u>

During the first IOSS the servicer mechanism was designed for axial module exchange only. Thus, it was compatible with either central or peripheral type docking systems. However, the servicer system requirements analysis of Chapter II established the need for both radial and axial module removal on the same mission. No simple way was found to handle the radial and axial requirement when a peripheral docking system is involved. The peripheral system acts as a fence dividing the axial region from the radial region. The use of a generalized peripheral docking system would force the development of two servicer mechanisms and stowage racks. Their operational utility would also be diminished. On the other hand, if a central docking system is used, then all of the advantages of the modularized servicer system immediately accrue. In effect a lighter, more versatile servicer system with a lower life cycle cost can be developed.

The first IOSS economic analysis showed that ground return of spacecraft was not economical. If this is accepted, then spacecraft retrieval is no longer a significant mission. Rather, the significant use of a docking system becomes the servicing mission. As the servicing mission is better satisfied by a central docking system, development by the NASA of a docking system should emphasize the central approach

VI. SERVICER MECHANISM PRELIMINARY DESIGN

Two servicer mechanism designs were prepared during the current contract. These were a preliminary design for a space version of the on-orbit servicer mechanism and a hardware design for the engineering test unit of the servicer mechanism structure. It is the preliminary design of the space version that is discussed in this chapter.

The design which is presented here has been evolved through a series of iterations during which a very wide range of alternatives were considered. The result is believed to be sound, it meets all of the requirements, and it can be carried to a flight design. The first IOSS screened and evaluated 15 different on-orbit servicer concepts from the literature. No significantly different concepts have been identified from the literature since then. Those concepts which have appeared recently were directly relateable to the categories of the first IOSS. That study selected a pivoting arm system which was optimized for axial module exchange and emphasized simplicity and light weight.

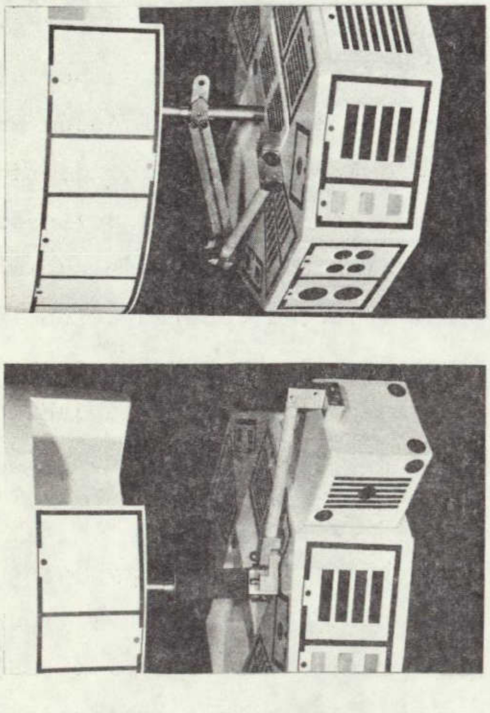
The current activity statement of work reopened the question of the need for radial module removal. The results of the reevaluation in terms of servicer requirements are given in Chapter II. In Chapter III a very large number of candidate mechanism configurations (number and type of degrees of freedom and number of arm segments) were identified, categorized and evaluated. Additionally, a set of criteria for selecting a configuration was identified and defined. This led to selection of the axial/near-radial configuration by NASA, a selection in which we concur. As discussed in Chapter III, the selected configuration is one of a family of five modular forms. The approach taken in the first design stages was to ensure that the selected design was compatible and extendable to all five modular forms. The intention is to keep open the option of developing all five modular forms at minimum cost.

The details of the mechanism design, and particularly the rotary joint designs, has evolved in a manner similar to the mechanism configurations. Our first attempts at the design of the Shuttle Remote Manipulator System (SRMS) joints considered a wide range of alternatives. This led to our avoidance of telescoping or sliding arm segments with their low rigidity and high weight.

The dual path drive evolved from detail consideration of a range of rotary drive designs including external spur gears and harmonic drives. The dual path rotary drive configuration was used in several applications, the most recent of which was the Proto-Flight Manipulator Arm (P-FMA) delivered to MSFC in March of 1977. The drives are light in weight, stiff, have high torque, minimum backlash, incorporate all the necessary feedback and safety elements and are backdriveable. The P-FMA design approach was used in three of the drives for the space version. However, the learning and experience was carried over into the designs for all of the drives.

The simplicity and function allocation approaches discussed in Chapter V were mostly applicable to configuration selection and overall system design. However, these concepts were also carried into the mechanism design details whenever applicable.

The major characteristics of the selected axial/near-radial servicer configuration are shown in Figure VI-1. Serviceable spacecraft designers have been provided with a great deal of freedom. There are very few restrictions on module replacement direction, spacecraft diameter, interface mechanism type, or module



- Axial Module Replacement
- Radial Module Replacement
  - Attach locations in a common plane
- Tip Force - 20 pounds
- Maximum Operating Radius - 7.5 ft
- Module Mass - 10 to 700 lbs
- Module Size - 17 in. cube to 40 in. cube
- Time to Replace One Module - 10 min.
- Stowed Length - 71 in.
- Mechanism Weight - 140 lbs
- Stowage Rack Weight - 309 lbs
- Degrees of Freedom - 6
- Operable in one-g with bolt-on counterbalance

Figure VI-1 On-Orbit Servicer Major Characteristics size, weight, or shape. The capacities selected were based on an evaluation of 28 serviceable spacecraft designs from the literature.

The one-tier capability is very compatible with all geosynchronous spacecraft requirements such as the DSCS II communications spacecraft and the SEOS earth observation satellite. The servicer requirements analysis of Chapter II showed that the majority of servicing benefits could be obtained for multiple tier spacecraft if the least reliable components were located in the one tier accessible to the servicer. Alternatively, multiple one-tier dockings can be used to service all replaceable modules as has been done with the Characteristic Large Observatory.

Characteristics of the first IOSS pivoting arm have been retained in terms of tip force, module replacement time and a low stowed length including the module stowage rack. A significant reduction in stowage rack weight has been achieved by going to an open truss type structure. Ground turn-around checkout has been facilitated by provision of a bolt-on counterbalance system that permits operation of the servicer in one-g without modules, but with lightweight interface mechanisms.

The axial/near-radial configuration for space design is shown on Figure VI-2. A major point in the selection was that the eventual form of servicer that will become accepted and used cannot be identified now, so the selection had to be made on the basis of a best estimate of the probable situation. It was decided to go with a relatively simple configuration that has natural and easy growth options.

This design has only two major components: (1) a servicer mechanism, and (2) a stowage rack for module transport. A docking mechanism is also shown for reference and so that the mechanical interface aspects can be more easily visualized. The servicer mechanism and the stowage rack were designed separately with interfaces for individual removal and replacement. This allows for simple removal for maintenance and also for quick ground reconfiguration. Stowage racks can be configured and loaded for particular flights prior to attachment to the carrier vehicle. It may be desirable to have available several stowage racks for this purpose. The stowage rack shown mounts directly to an upper stage. A flight support structure has been designed to adapt the stowage rack shown to

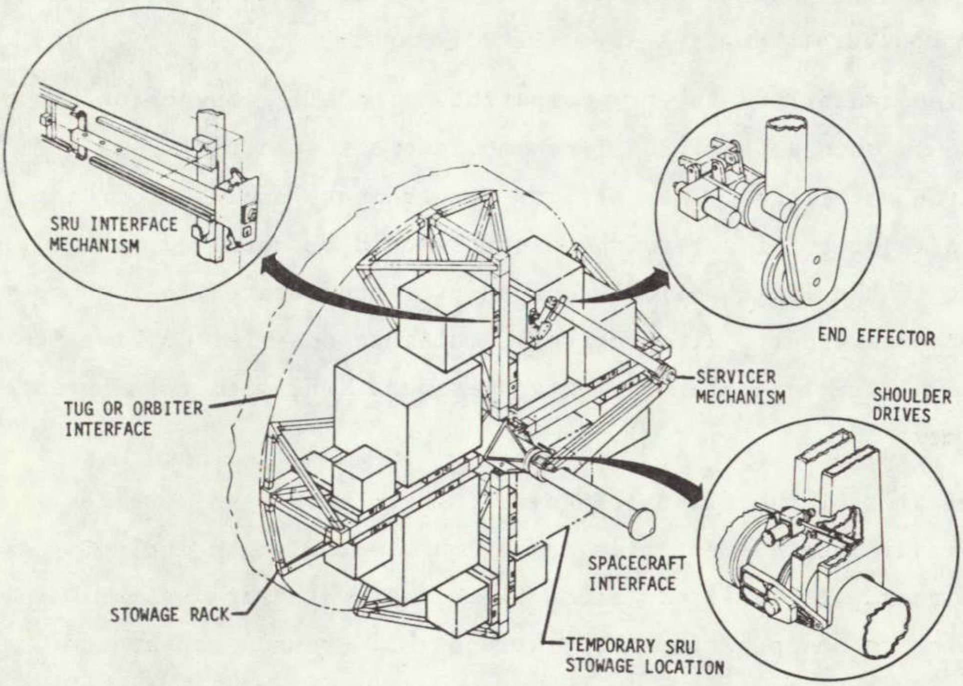


Figure VI-2 On-Orbit Servicer - Space Design

the Orbiter.

A major difference between this design and the first IOSS is the use of a parallelogram linkage form of upper arm. This approach keeps the lower or fore-arm parallel to the stowage rack and spacecraft faces as did the translational drive of the first IOSS. However, it is a stiffer and lighter weight mechanism.

The on-orbit servicer has been designed to satisfy the requirements established in the first IOSS contract and updated by our more recent work. Those requirements have been divided into three groups: servicer mechanism, stowage rack, and interface mechanism. The three groups are compatible and are reported on separately in this material in terms of system characteristics. The primary servicer mechanism characteristic as noted in Table VI-1 is the ability to replace modules in both axial and radial directions as well as off-axis directions and combinations of directions. The restriction on radial module replacement is that all of the interface attach points must be contained in a common plane whose normal is parallel to the docking axis. For the configuration designed, this plane can be no farther than 63 inches from the front face of the stowage rack. This dimension can be increased by changing the length of the third arm segment.

Table VI-1 On-Orbit Servicer Mechanism Characteristics

- Axial Module Replacement
- Radial Module Replacement -- attach locations in a common plane
- Maximum Operating Radius - 7.5 feet
- Module Mass - 10 to 700 lbs
- Module Size - 17 in. cube to 40 in. cube
- Degrees of Freedom - 6
- Stowed Length - 27 in.
- Tip Force > 20 lbs in worst configuration
- Attach/Latch Actuator Located in End Effector
- Time to Replace One Module - 10 minutes
- Be Compatible with Supervisory and Remotely Manned Control
- Probability of Mission Success = 0.98
- Reusable for 100 Missions
- Lifetime of Five Years
- No Ability to Exchange Modules in One-g
- Operable in One-g with Counterbalance and No Modules
- Lightweight

The module size limits shown are representative for cubes. Smaller sizes can be handled, but they are not efficient. Some larger sizes can be handled at lower rates and with appropriate stowage rack configurations. The 20 lb tip force can be obtained in either the radial or the axial direction. For some mechanism configurations, the resulting forces will be larger due to shorter radii or the toggle effect. The 10 minute module exchange time corresponds to the nominal joint rates and a significant time allowance between motion segments. The actual time used will depend on control system and training specified.

It was decided not to include an ability to exchange modules in one-g as this would have resulted in very significant weight penalties. The mechanism is designed only to move modules in zero-g and thus cannot even support itself except over a limited range of joint angles. However, a bolt-on counterbalance system has been designed so that the mechanism may be exercised over its full range of operation between flights.

An analysis of the applicability of the selected servicer configuration to the three serviceable spacecraft designs of Chapter III was made. TRW was kept aware of the mechanism capabilities as they evolved and TRW included these aspects in their spacecraft configuration considerations. As the TRW designs became available checks were made on module sizes and interface mechanism attach point locations. Methods of handling and stowing the outsize modules of the SEOS and the CLO were identified and validated using two dimensional models. An early version of the CLO had modules located outside the reach envelope of the axial/near-radial configuration. It was relatively easy for the servicer and the spacecraft designers to work together and to evolve a CLO configuration which could be readily serviced. It should be noted that the CLO probably presents the most complex and sophisticated serviceable spacecraft design challenge that has been identified. Yet its configuration evolved in a rapid and straightforward manner. Thus the capability and utility of the axial/near-radial configuration was demonstrated.

The design approach of Section A builds on the system approach of Chapter V and addresses the effects of the five modular configurations on the servicer mechanism design. It also discusses the approaches used in electro-mechanical component selection. Section B introduces details of the servicer mechanism design at the overall and specific joint levels. Mechanism stowage and test counterbalancing are also covered. The recommended interface mechanism approach and rationale are discussed in Section C and three standard sizes are identified. Details of the new truss type stowage rack are covered in Section D. The compatibility of the selected stowage rack design with the three serviceable spacecraft of Chapter IV are also addressed. The final section, E, addresses the implications of the docking system design on the servicer mechanism.

A. DESIGN APPROACH

The servicer mechanism design approach builds on the simplicity and function allocation approaches of Chapter V. The selected servicer mechanism configuration involves six electromechanical drives or joints. These are shown in Figure VI-3 as

T - Shoulder Roll	W - Wrist Yaw
U - Shoulder Pitch	Y - Wrist Pitch
V - Elbow Roll	Z - Wrist Roll

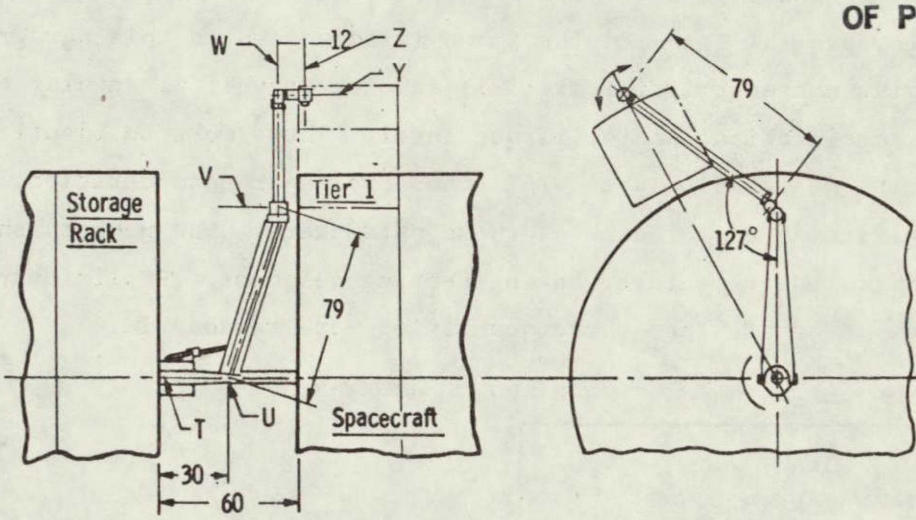


Figure VI-3 Axial/Near-Radial Servicer Configuration Layout

The lengths shown on the figure are in inches. The symbol X is used for a second elbow joint which is used in some configurations. These symbols are used extensively in the rest of this chapter.

Chapter III identified the value of modularized forms of servicer mechanisms. These modular forms are servicer configurations that span the range of capability (module removal direction and number of tiers) with five levels of mechanism complexity. The Chapter III configuration selection became a question of selecting the most appropriate combination of complexity and capability. Each of the five modular forms had been optimized as the most effective configuration to satisfy the capability requirement. Also all five used similar design approaches and shared common hardware elements so that life cycle costs of all five would be minimized. This section discusses the identification of the common hardware elements in paragraph 1. The second paragraph discusses the approach to selection of components for the electromechanical drives and tabulates the selected components.

1. Joint Drive Characteristics

Identification of the type and characteristics of the fewest number of joint drives necessary to span the five modular forms is addressed in this paragraph. The basic joint drive characteristics are: 1) angular travel; 2) angular slew rates; 3) angular acceleration; and 4) torque level. These must be identified before a joint can be designed. Table VI-2 shows the major considerations in selecting the specific values for each of these parameters. The numbers shown are based on specific requirements or on engineering judgment. Preliminary calculations were made to assure that these quantities were reasonable.

Table VI-2 Joint Drive Requirements Identification

<u>ANGULAR TRAVEL</u>

- Axial and radial module replacement
- Full range of module locations
- Module flip

<u>ANGULAR RATES</u>

- Module exchange time < 10 minutes
- Relative rates based on experience

<u>ANGULAR ACCELERATIONS</u>

- Occur in small part of travel time = 1 second
- Acceptable stopping distance < 4 inches
- Acceptable torque levels ≈ 700 lb module

<u>TORQUES</u>

- Tip force = 20 lbs
- Module and mechanism acceleration
- Worst module locations
- Misalignment forces
- Reasonable combinations

The angular travel data was calculated from the configuration layouts and verified using the servicer and spacecraft model. The angular rate discussion is presented in Chapter VII. The acceleration/deceleration stopping time and distance are reasonable for remotely manned control as well as for supervisory control. The torque calculations required care because of the variety of conditions, module motion directions, and module locations that had to be considered. Estimates of mechanism inertia were made to ensure that valid acceleration torques were calculated. These four parameters were determined for each joint of each of the five modular configurations.

The joint drive requirements data for the axial/near-radial configuration are shown in Table VI-3. Similar data was prepared for the other four modular configurations. The moment of inertia data includes the effect of the mechanism itself and have been calculated for a worst case. However, the module was not extended out past the wrist unless it was necessary to accomplish a specific trajectory. Two inertial configurations were used for elbow roll. The larger number corresponds to the module extended out past the wrist and the second corresponds to the module tucked in near the forearm.

Table VI-3 Servicer Mechanism Joint Data - Axial/Near-Radial

	Moment of Inertia (slug-ft^2)	Angular Rate		Acceleration (rad/sec^2)	Torque from Acceleration (ft-lbs)	Torque from Tip Force (ft-lbs)	Joint Design Torque (ft-lbs)
		deg/sec	rad/sec				
Wrist Roll (Z)	100.49	12	0.21	0.21	21.1	--	21.1
Wrist Pitch (Y)	117.32	12	0.21	0.21	23.4	--	23.4
Wrist Yaw (W)	188.13	6	0.10	0.10	18.8	--	18.8
Elbow Roll (V)	1,297.98 540.27	8 8	0.14 0.14	0.14 0.14	181.7 75.6	162.0 162.0	181.7 162.0
Shoulder Translation (U)	1,841.5	4	0.07	0.07	129.0	172.6	172.6
Shoulder Roll (T)	1,909.5	6	0.10	0.10	191.0	191.0	191.0

While only acceleration and tip force torques are tabulated, several other combinations involving misalignments were checked to see that they did not cause a higher requirement. The joint design torque shown is the larger of the acceleration and tip force torques as these two should never be required at the same time. The tip force considerations involved both axial and radial module removal considerations using worst case module locations with the tip force direction being parallel to the module removal direction. This resulted in the elbow torques being comparable to the shoulder torques. The wrist torques are much lower. Similar tables were made for the other four mechanism modular configurations.

The joint drive requirements for all 29 of the drives used on the five modular forms are shown in Table VI-4. Generally the larger inertia related torques were used. In some cases, two different angular rates are required for the different configurations. The joint travel data is coded, in parentheses, with the number of the applicable configuration as noted above the configuration name.

Table VI-4 Servicer Mechanism Joint Data - Composite

Joint Axis	TORQUE (foot-pounds) (1) Axial	(2) Axial/Near Radial	(3) Near Radial	(4) Two-Tier Radial	(5) Axial/Two-Tier Radial	Angular Rate (rad/sec)	Acceleration (rad/sec^2)	Joint Travel (degrees)
T	65.2	191	191	225	221	0.10	0.10	380 (1,2,3,4,5)
U	173	173	--	--	308	0.07	0.07	60 (1,2,3,4) 135 (5)
V	75.8	182	179	205	164	0.07* 0.14	0.07* 0.14	175 (1,2,3) 180 (4,5)
W	14.1	18.8	17.8	162	162	0.10 0.14*	0.10 0.14*	200 (1,2,3) 250 (4) 290 (5)
X	--	--	--	26.7	26.7	0.10	0.14	180 (4) 240 (5)
Y	--	23.4	23.4	23.4	23.4	0.21	0.21	380 (1,2,3,4,5)
Z	21.1	21.1	21.1	21.1	21.1	0.21	0.21	380 (1,2,3,4,5)

*Two-Tier Radial Configurations

The purpose of preparing the composite table was to identify the smallest number of joint types that could reasonably be used to satisfy all 29 joint requirements. The rules used were:

1) Common joint type - rotary or ball screw

2) Similar joint travel

3) Similar joint rate

4) Similar torque level

Where a higher rate was assigned to a specific joint, the equivalent acceleration torque was calculated and checked against the torque level assigned. It is anticipated that adapters may be required in some instances to be able to use some joint types in some applications.

The result of the joint drive commonality analysis is shown in Table VI-5. The upper part of the table shows the axis assignment by type. Subscripts show specific drive type, e.g., T_1, T_2 and a - means that a drive is not required. For the Z and Y axes, the requirements for each are common across all using configurations because the application is common. The X drive is only required for two configurations and has a requirement that was close enough for the Y_1 drive to be used. The W axis is an indexing drive for the W_1 assignment, but is an elbow drive where V_2 was assigned.

Table VI-5 Configuration/Joint Assignments

CONFIGURATION / JOINT AXIS	AXIAL	AXIAL/ NEAR-RADIAL	NEAR-RADIAL	TWO-TIER RADIAL	AXIAL/ TWO-TIER RADIAL
T	T_1	T_2	T_2	T_2	T_2
U	U_1	U_1	-	-	U_2
V	V_1	V_2	V_2	V_2	V_2
W	W_1	W_1	W_1	V_2	V_2
X	-	-	-	Y_1	Y_1
Y	-	Y_1	Y_1	Y_1	Y_1
Z	Z_1	Z_1	Z_1	Z_1	Z_1

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CONFIGURATION / JOINT TYPE	AXIAL	AXIAL/ NEAR-RADIAL	NEAR-RADIAL	TWO-TIER RADIAL	AXIAL/ TWO-TIER RADIAL
T_1	X				
T_2		X	X	X	X
U_1	X	X			
U_2					X
V_1	X				
V_2		X	X	X (2)	X (2)
W_1	X	X	X		
Y_1		X	X	X (2)	X (2)
Z_1	X	X	X	X	X

For the V axis, the elbow does not have to generate a tip force for the axial configuration, but it must for each of the others. The U_1 drives are to be screw jack type drives, but the U drive for the axial/two-tier radial configuration is a regular rotary drive. The situation for the T axis is similar to that for the V axis where the T_1 drive does not have to generate a tip force, but the T_2 drives do.

From the cross plot data of the lower part of the table, it can be seen that the axial configuration uses two drives (T_1 and V_1) that are not used in any other configuration. Each of the axial/near-radial drives are used in at least one other configuration. This point was included in the configuration selection logic. The symbol, (2), in the last two columns means that two of these drives are required for the specific configuration. The conclusion is that nine joint drives can be adapted to cover all 29 drive requirements across the five modular forms.

The specific characteristics selected for each of the nine joint drive types are shown in Table VI-6. Generally, the largest of the composite requirements was used and then rounded upwards slightly. The power data was computed on the basis of rate and the torque occurring at the same time. This generally will not occur, but is a useful approximation at this stage of the design. As can be seen, the power levels are relatively low and are readily obtainable from

Table VI-6 Drive Characteristics

DRIVE	RATE (rad/sec)	TORQUE (ft-lbs)	TRAVEL (deg)	NO. OF JOINT USAGES	HORSEPOWER	WATTS
T_1	0.10	65	380	1	0.0118	8.8
T_2	0.10	225	380	4	0.0409	30.5
U_1	0.07	175	60	2	0.0223	16.6
U_2	0.07	300	135	1	0.0382	28.5
V_1	0.14	80	180	1	0.0204	15.2
V_2	0.14	200	300	6	0.0509	38.0
W_1	0.10	20	200	3	0.0036	2.7
Y_1	0.21	30	380	6	0.0115	8.6
Z_1	0.21	25	380	5	0.0095	7.1

permanent magnet DC torque motors. The smallest requirement, W_1, is an indexing drive and is more in the category of a permanent magnet DC gear motor. The low power levels result from the low joint rates selected. For those joints where the torque level is based on module acceleration, a doubling of a joint rate results in a fourfold increase in motor power.

The requirements listed in Table VI-6 are those used in the detail drive designs. The slightly higher torque capabilities can be used to provide higher rates and tip forces than were used in analyzing the different modular configurations. The use of nine drives instead of twenty-nine should make for lower life-cycle costs in terms of development, production, spares and servicer system maintenance. Note that any configuration could be developed first, with the others following naturally.

2. Electromechanical Component Selection

The electromechanical joints of a servicer mechanism are similar to those of a general purpose manipulator in that each tends to be a special requirement and that they are not available as catalog items. While each is special, our experience has led to a definite approach to designing these mechanical joints. The major factors considered are listed in Table VI-7. Each was considered in detail for each joint of the axial/near-radial configuration which again becomes the discussion reference.

Several other joint types, such as the external spur gear, planetary, eccentric, and harmonic drive, were considered earlier in our learning process. However, each is generally inferior to the internal gear, dual path, drive arrangement that is used for half of the joints. This approach is compact, lightweight, low backlash, backdriveable, smooth, and very stiff. The worm gear approach is used where backdriveability is not required, and the linear actuator will be used for the shoulder pitch drives where the angular travel is less than 60 degrees. It was desired to use the dual path approach for the wrist roll drive. However, the result would have been a poor package. Instead a conventional external spur gear drive was selected to minimize the wrist dimension parallel to the wrist roll axis. Increases in this dimension force increases in the spacecraft to stowage rack separation distance or decreases in module

Table VI-7 Joint Design Factors

TYPE SELECTION	CONSIDERATIONS
• INTERNAL GEAR - DUAL PATH • WORM GEAR • LINEAR ACTUATOR	• SIMPLICITY • STIFFNESS • WEIGHT • COST • RELIABILITY • DESIGN MARGINS • MOTOR DERATING
ELEMENTS INVOLVED	
• GEAR RATIO • MOTOR TYPE • POSITION SENSOR • RATE SENSOR • BRAKE • WIRING HARNESS	

maximum size for axial module replacement.

Softness of a drive can make a significant contribution to loss of servicer mechanism stiffness. It is most important that the shoulder and elbow drives be stiff. This is accomplished by keeping effective shaft diameters to nearly the full housing diameter, double ending the connection if possible, derating bearings, and designing gears and housings for stiffness rather than wear or breaking strength. These approaches also lead to more reliable drives because of the greater stress margins and effective derating.

Gear ratio selection is influenced by maximum joint output rate, motors available, and joint size and weight requirements. For instrument servo systems where the load inertia is relatively constant, the concept of motor inertia equal to reflected load inertia is often used to select a gear ratio. This concept results in very large gear ratios and thus expensive joints when applied to manipulators. The question of proper gear ratio was addressed some time ago in terms of a large motor and low gear ratio versus a small motor and high gear ratio. A large part of the drive weight is associated with the output gear mesh, housing, output bearings, and the need to keep stiffness high. Thus, gear ratio selection has little effect on drive weight until ratios go below 50 to 1. The consideration then becomes the number of gear passes and the effect on reliability and cost. It takes three stages of gearing to get the desired minimum ratio, which in turn leads to the range of ratios normally used (≈100 to 1) when DC permanent magnet torque motors are used. Where high speed DC permanent magnet motors are used, as for the end effector jaw drive, then higher ratios are more appropriate. These ratios are selected to match motor speed to load speed.

The permanent magnet DC torquer type of motor has been selected for the majority of applications. It has been used extensively in space and is very reliable when two sets of brushes are used. It is also lightweight and can be readily packed into the drives along with the tachometer generator and brake which all operate at the same speed. Figure VI-4 shows the shoulder roll drive motor requirements (dashed lines) and the candidate motors available from the catalogs. The numbers in parenthesis are the motor weights in lbs. The selected motor is the Magnetic Technology 5125C-135 which has its maximum power point close to the drive requirements point. Derating is effected by realizing that the maximum torque and speed requirements do not occur at the same time. Thus, motor current can be limited to prevent overheating. The ratio of stall torque to friction torque is greater than the acceptable 40 to 1 ratio. Similar analyses were conducted for five of the six joint drives.

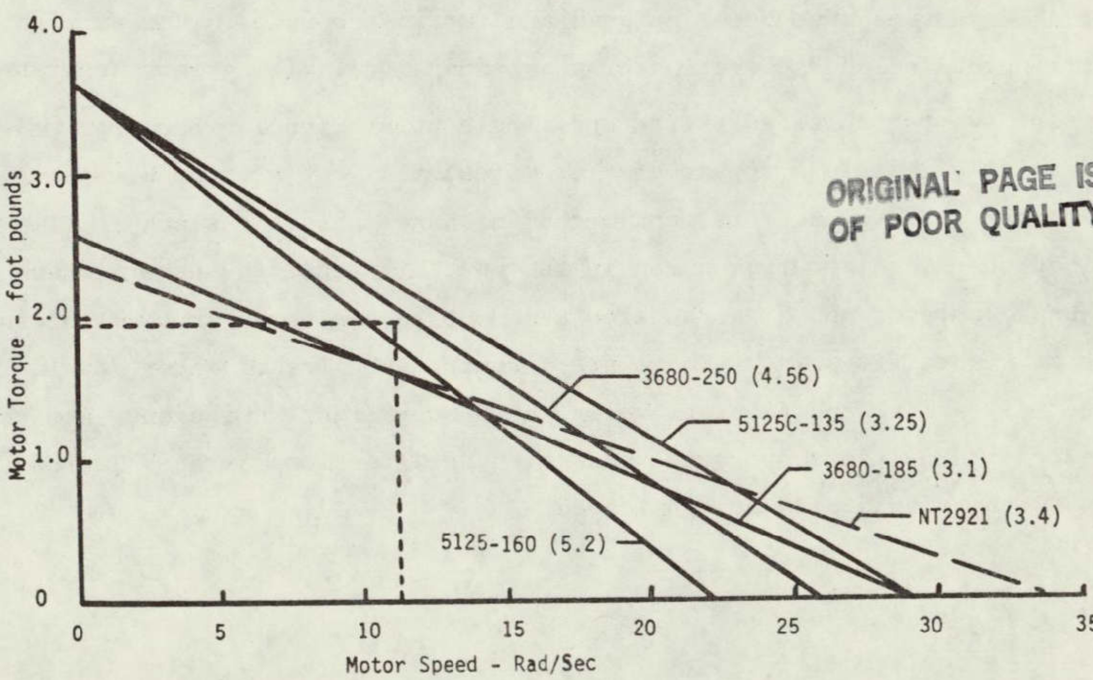

Figure VI-4 Shoulder Roll Drive Motor Candidates

The sixth (W) drive uses a worm drive and a high speed DC permanent magnet motor with gearhead which was selected on the basis of providing the required torque and speed. The motors for the end effector jaws and the interface mechanism drive were shown to be high speed DC permanent magnet motors with gearheads. This type of drive has adequate power, packages well, is space qualified, and suitable gear ratios are available.

Selection of the position and rate sensors used for each drive is discussed as part of the control system analysis of Chapter VII. Brakes are used on each backdriveable joint (T, V, Y, and Z) to keep the joint from moving inadvertently. The brakes are normally engaged and require current to release so that the joint may move. The logic to provide this current is part of the control system design. Each brake is mounted on the high speed motor shaft to minimize brake size which was selected so that the brake can resist the motor stall torque.

The electromechanical components used in the various drives along with some of their important characteristics are listed in Table VI-8.

The wiring harness has not been designed at this time. Several manipulator system wiring harnesses have been designed and fabricated by us. The straightforward attributes include separation of power and signal wiring into separate cables, use of twisted shielded pairs, and single point signal grounding. Alternatives exist with regard to use of flat wires, flat cables, round cables, number and location of connectors, and method of handling cable slack to allow for joint motion. The latter question is the most difficult to answer. Each approach has advantages and disadvantages and must be evaluated in terms of the parameters at a specific joint. These are: number and size of wires, cable type, number of cables, and joint travel. The Proto-Flight Manipulator Arm used flat cable constrained in sheet metal cans with unidirectional wrap. The reverse wrap system should also be considered.

Table VI-8 Electromechanical Component Summary

			MOTOR				POSITION SENSOR			RATE SENSOR			BRAKE	
Joint Designator	Drive Type	Gear Ratio	Type	Vendor Part Number	Stall Torque in. oz	No Load Speed rad/sec	Type	Vendor Part Number	Accuracy arc-min	Type	Vendor Part Number	Maximum Speed rad/sec	Vendor Part Number	Torque in. oz
Shoulder Roll (T)	Dual Path Internal Gear	113	DC Torquer	Magnetic Technology 5125C-135	700	29	Resolver	Singer Kearfott CZ41093001-0	3	DC	Magnetic Technology 2813C-088	60	Electroid FSB-35	560
Shoulder Pitch (U)	Ball Screw	2262	DC Torquer	Magnetic Technology 1500C-250	100	415	Potentiometer	Computer Instruments Corp. Model 205	2	DC	Magnetic Technology 1500E-050	800	Delevan BF-10-24-4	80
Elbow Roll (V)	Dual Path Internal Gear	127	DC Torquer	Magnetic Technology 5125C-135	700	29	Resolver	Singer Kearfott CZ41093001-0	3	DC	Magnetic Technology 2813C-088	60	Delevan BF-20-24-6	240
Wrist Yaw (W)	Worm Gear and Planetary	8271	DC High Speed	Globe Industries 5A545-22	3.6 (51% eff.)	1100	Potentiometer	Computer Instruments Corp. Model 100	10	DC	Globe Industries N/A	N/A	N/A	N/A
Wrist Pitch (Y)	Dual Path Internal Gear	110	DC Torquer	Magnetic Technology 3000C-065	85	65	Potentiometer	Computer Instruments Corp. Model 205	6	DC	Magnetic Technology 1937E-040	520	Delevan BF-10-24-4	80
Wrist Roll (Z)	External Spur Gear	50	DC Torquer	Magnetic Technology 4590B-063	170	39	Potentiometer	Computer Instruments Corp. Model 7810	60	DC	Inland Motors TG-2139	128	Electroid FSB-7	112
End Effector Jaws	Ball Screw and Planetary	81	DC High Speed	Globe Industries 5A505-7	7.0 (51% eff.)	1456	Microswitches	N/A	N/A	N/A	N/A	N/A	N/A	N/A
Interface Mechanism	Planetary	148	DC High Speed	Globe Industries 5A509-7	7.0 (51% eff.)	1456	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A

B. MECHANICAL DESIGN DETAILS

The purpose of this section is to present the details of the servicer mechanism drives, how the servicer mechanism and docking probe can be configured for stowage during the launch and reentry phases, a method of counterbalancing the mechanism for ground checkout, and a weight statement. The interface mechanism and stowage rack are discussed in subsequent sections.

The NASA selected axial/near-radial on-orbit servicer configuration is shown in Figure VI-5. This design has only two major components: 1) a servicer mechanism, and 2) a stowage rack for module transport. A docking mechanism is also shown for reference and so that the mechanical interface aspects can be more easily visualized. The servicer mechanism and the stowage rack were designed separately with interfaces for individual removal and replacement. This allows for simple removal for maintenance and also for quick ground reconfiguration.

As the specific joint designs used in this configuration are part of the common set described in Section A for the five modular servicer configurations, it will be possible to build from this configuration to the other four configurations.

Characteristics of the first IOSS pivoting arm have been retained in terms of tip force, module replacement time and a low stowed length including the module stowage rack. However, the translational drive of the first IOSS has been replaced by a parallelogram linkage and a ball screw drive to provide greater rigidity at lower weight and a concept which is applicable to a wider range of configurations. Also the end effector roll drive has been redesigned to obtain a better balance between gearing weight and motor weight and a wrist pitch drive has been added to bring the total number of degrees of freedom to six. A three-view installation layout of the servicer mechanism is shown in Figure VI-6. The major dimensions in inches are:

Separation distance between stowage rack and spacecraft - 60

Separation distance between stowage rack and U joint effective pivot axis - 30

Upper arm (Four bar linkage) - 79

Lower Arm - 79

Lower arm centerline to end effector centerline - 12

Wrist pitch drive centerline to end effector attach interface - 8

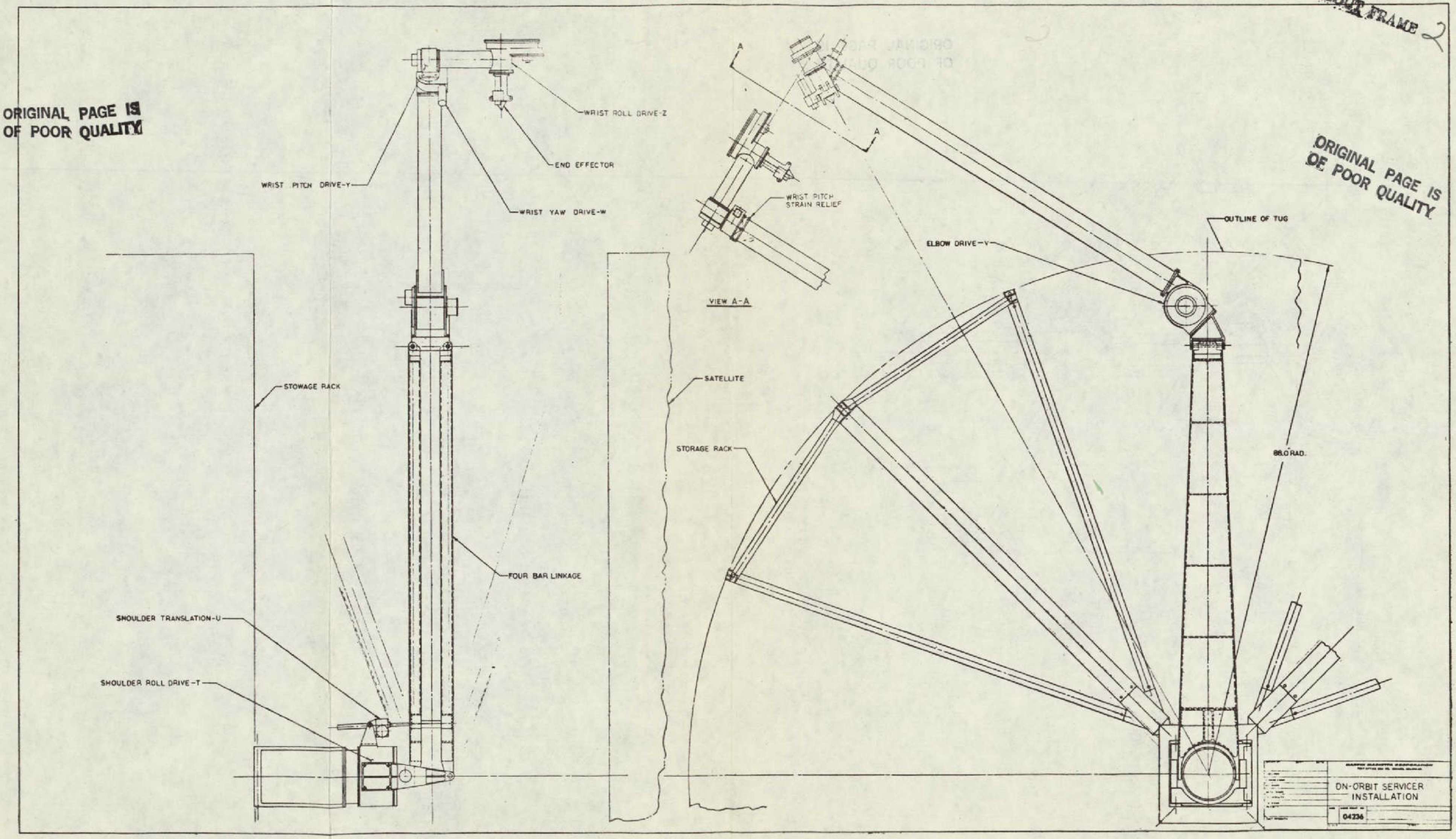

Figure VI-6 On-Orbit Servicer Installation

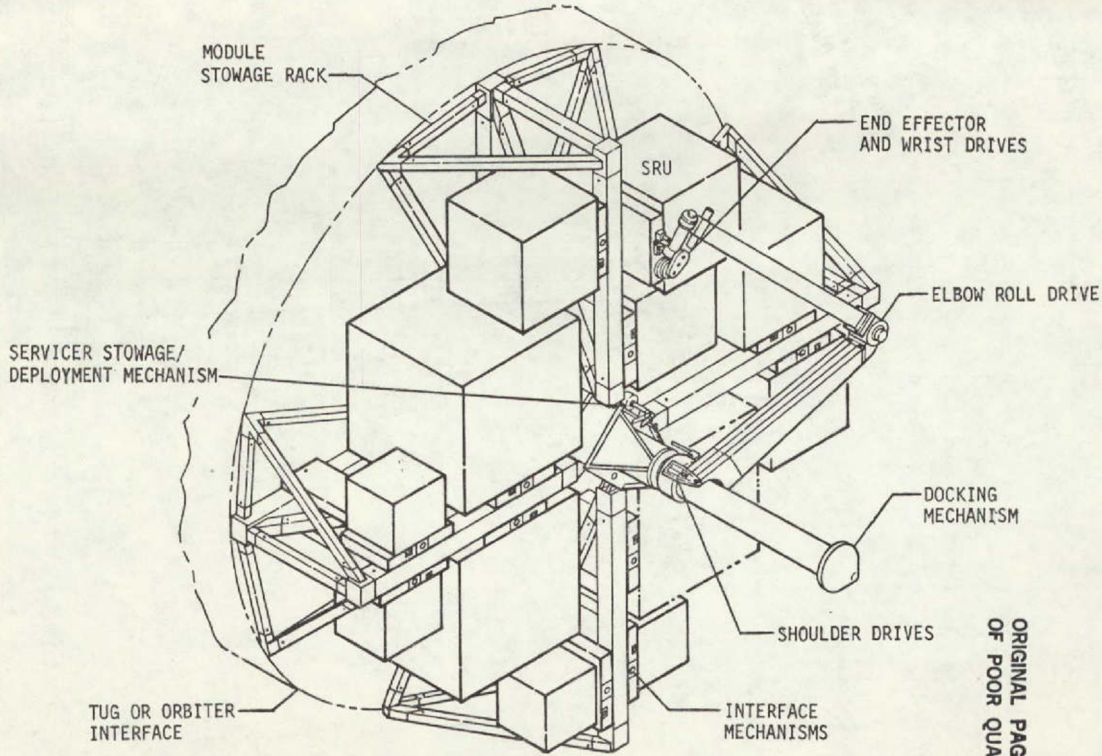

Figure VI-5 On-Orbit Servicer Mechanism

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A wrist pitch strain relief mechanism is provided at the base of the wrist yaw drive (W). This strain relief is to provide a controlled compliance to compensate for misalignment tolerances when the interface mechanism is in the guides. A similar mechanism is included as part of the wrist yaw drive. These two strain reliefs are at 90 degrees to each other and thus provide relief in two of the three degrees of freedom. The third axis is the elbow roll drive axis which is backdriveable. So the two strain relief mechanisms and the backdriveable elbow roll drive provide full three axis compatibility for small (≈ 2 degree) misalignments. Each of the two strain reliefs are designed to prevent any motion until the 20 lb tip force is exceeded. For forces greater than 20 lbs (when the spring preload is exceeded) a low spring rate comes into play at essentially constant force to a firm stop after two degrees of relative motion. The wrist pitch strain relief mechanical design consists of a pair of shoulder screw pivots and a set of eight compression springs along with the necessary mechanical stops. A Bellville spring approach is used in the wrist yaw strain relief and it is discussed below.

The four bar linkage, or translation members, are built up from machined end fittings, side channels, and facing plates. Each member is thus a riveted assembly with appropriate stiffeners. The original concept used open construction to simplify the assembly operations. However, a strain analysis showed that the members could be stiffened significantly if the boxes were closed, so this was done. The critical stiffness configuration is with the lower arm at 90 degrees to the translation members so that the translation members are being twisted about their longitudinal axes. The closed boxes and the member spacing selected provided the necessary stiffness.

The shoulder roll drive shown in Figure VI-7 has a large diameter because of its interface with the docking mechanism and a decision to use the accurate internal-gear dual-path construction. The docking probe tube is bolted to the gear plates which in turn will be connected to the stowage rack through the folding mechanism. The two outer bars are the supports for the upper arm parallelogram. Large diameter, small cross section output bearings are used to provide a high level of structural stiffness.

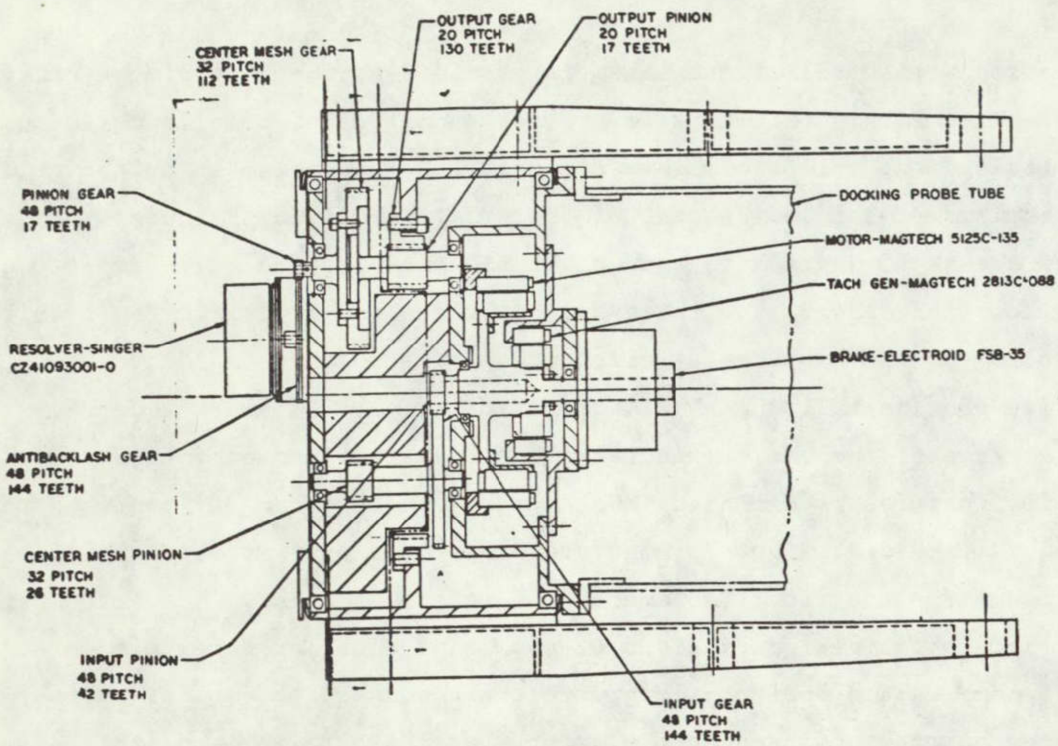

Figure VI-7 Shoulder Roll Drive (T)

The permanent magnet DC torque motor is shown on the center shaft along with a tachometer generator and a fail-safe brake. The overall drive ratio of 113 to one is obtained in three steps in a dual path approach. The center mesh gears are made adjustable on their shafts and are used to remove much of the drive backlash. After adjustment, the center gears are pinned to their shafts. This straight spur gear construction is backdriveable at a low torque level. Seals or covers are provided for all bearings.

The position indicator selected is a Singer resolver identical to those used for the proto-flight manipulator arm and is driven through an anti-backlash gear. The high accuracy of a resolver is required at the shoulder drives because of the long arm length. The resolver drive gearing provides over a full revolution of travel of the joint.

The linear translational drive of the first IOSS was replaced by the shoulder pitch drive shown in Figure VI-8. Use of the two translation members in a parallelogram configuration results in the lower arm of the servicer being kept parallel to the front of the stowage rack. This simplifies the hazard avoidance problem. However, as the shoulder pitch drive alone is operated, the end effector moves along an arc instead of along a straight line. This is not too important when the end effector alone is being moved, but it is important when a module is being removed from the interface mechanism baseplate receptacle (guides). For the module to move in a straight line parallel to the docking axis, the shoulder pitch, shoulder roll, elbow roll, and wrist roll joints must move in synchronism. When the module is in the guides and the shoulder pitch joint is providing the basic motion, then the other three joints will backdrive and should automatically move in the proper synchronism.

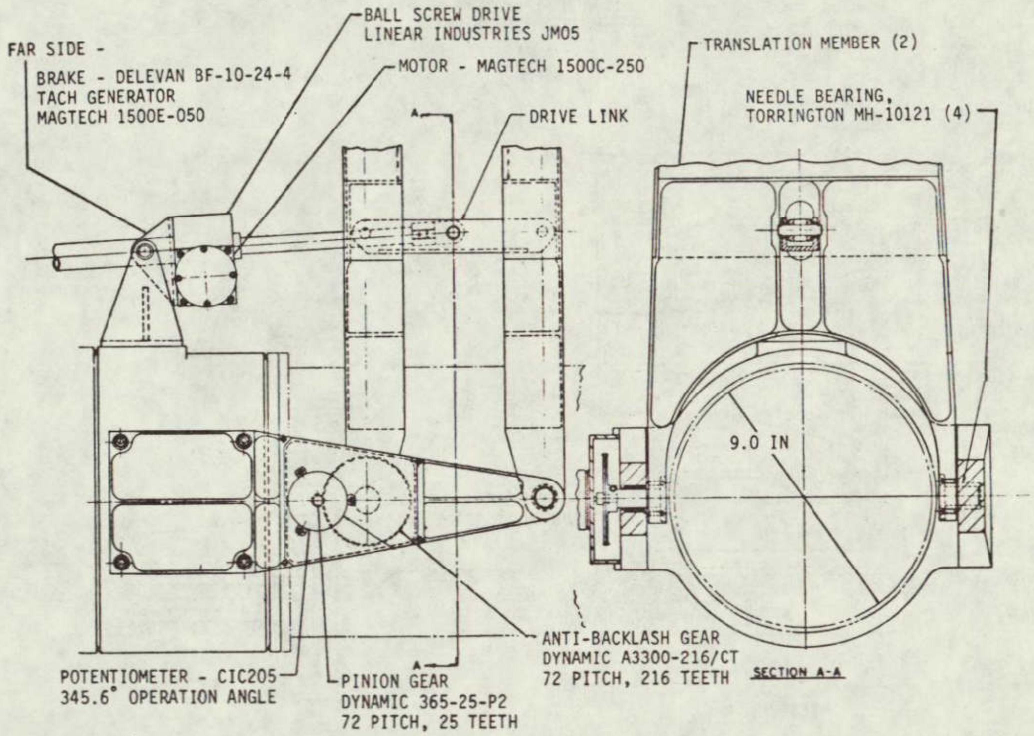

Figure VI-8 Shoulder Pitch Drive (U)

The shoulder end of the translation members are C shaped to provide clearance for the docking probe. The gearing is effected through a Linear Industries ball screw drive with a worm reduction. This joint is not backdriveable and it need not be. The permanent magnet DC torque motor is mounted on one side of the worm drive gearbox with the tachometer generator and the fail-safe brake on the far side. Seals and covers are provided for all bearings and for the rearward extension of the ball screw.

The position indicator selected is a single turn potentiometer that has been geared up to provide almost a full turn of travel. The small angular motion of the shoulder pitch drive resulted in a lower ratio of full travel to allowable angular error and thus permitted use of the potentiometer. Antibacklash gearing to the potentiometer is provided.

The elbow roll drive shown in Figure VI-9 has been patterned after the internal-gear dual-path drives of the proto-flight manipulator arm. The drive is mated to the parallelogram translation linkage through an angle fitting. The joint centerline is mounted offset from the centerlines of the translation members and the outer arm segment so that these arms may be folded back towards each other with a minimum of interference.

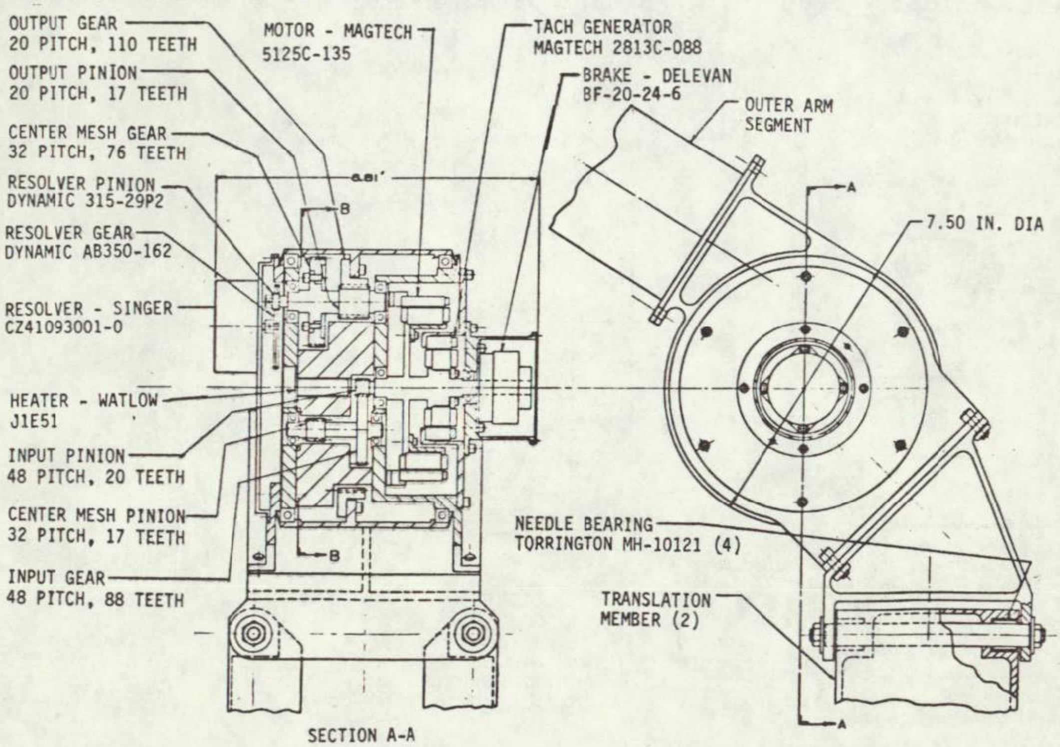

Figure VI-9 Elbow Roll Drive (V)

The drive is powered by a permanent magnet DC torque motor mounted on the input shaft. The input shaft pinion drives a dual mesh, three-stage gear reduction. This gear train can be traced by following the torque transmission through two shafts. The final gear stage terminates with the internal gear which is fixed to and drives the outer housing. The tachometer (rate sensor) is mounted to the

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input shaft, giving the maximum voltage level for rotational speed. The fail-safe brake is also mounted on the input shaft which requires the minimum torque, and therefore power to restrain the drive if motor power were interrupted. The overall drive ratio of 127 to one is obtained in three steps. The center mesh gears are made adjustable as for the shoulder roll drive. This straight spur gear construction is backdriveable at a low torque level. Seals or covers are provided for all bearings.

The Singer resolver (position sensor) is driven through an antibacklash gear from the third shaft. This reduces the position error due to gear backlash in the output gear stage. The high accuracy of a resolver is required at the elbow drive because of the long arm length. The resolver drive gearing allows full joint motion. The limit switch is provided as an indicator that the drive has reached its maximum travel. The heater is required in the cold thermal case to prevent the drives from going below -100°F.

The wrist yaw drive, located at the wrist end of the outer arm, is shown in Figure VI-10. Its purpose is to turn the modules end for end so they may be

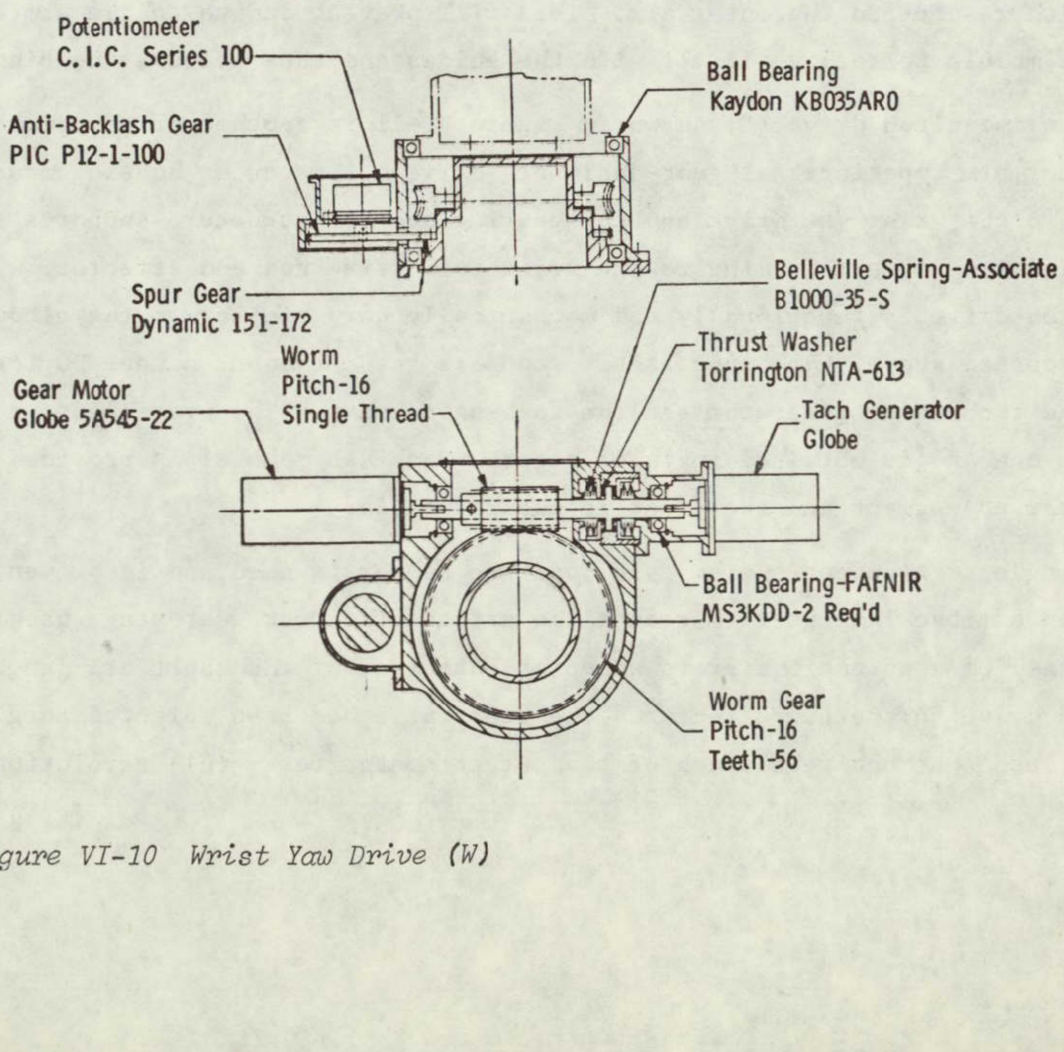

Figure VI-10 Wrist Yaw Drive (W)

placed in the spacecraft or stowage rack. This capability is required for both axial to radial and radial to radial module exchange. The drive is thus basically an indexing and not a servo drive. It is required to drive at a constant rate during the module flip and thus a tachometer generator has been provided. A 56:1 worm gear ratio is used along with a gearhead on the DC permanent magnet motor to provide the necessary slow speed from a small high-speed motor. A brake is not necessary because of the inherent non-backdriveability of the worm drive. A potentiometer is used to provide an indication of which index point the joint is at as well as a basis for generating the other drive signals as the module is flipped.

To allow for misalignment of the module as it is entering the guides on the spacecraft or stowage rack, a Belleville spring preload assembly has been provided for strain relief. If the guide misalignment force is less than 20 lbs, then the springs hold the worm firmly in place and the end-effector does not move with respect to the arm. If forces greater than 20 lbs are experienced, say due to binding in the guide, then the worm will move and thus let the end effector rotate with respect to the outer arm. This will prevent damage to the arm and allow the module to realign itself with the guides and thus relieve the binding.

The wrist pitch drive (Y) shown in Figure VI-11 is another of the proto-flight manipulator arm type internal gear-dual path drives. Its outer housing mounts directly to the wrist yaw drive and the gearing support structure supports the short third arm segment leading to the wrist roll drive and end effector. The wrist pitch drive is functionally and mechanically very similar to the elbow roll drive discussed above. The input shaft supports the permanent magnet DC torque motor, the tachometer generator and the fail-safe brake. The overall drive ratio is 110 to one and is obtained in three steps. Backlash removal is provided by center gear adjustment and the joint is backdriveable.

A single turn potentiometer type position sensor is used and is driven through an antibacklash gear to reduce the effect of output gear stage backlash. The accuracy of a potentiometer is acceptable because of the short arm length from this drive to the end effector. The gear ratio has been selected to give slightly less than one revolution of the potentiometer for a full revolution of the drive.

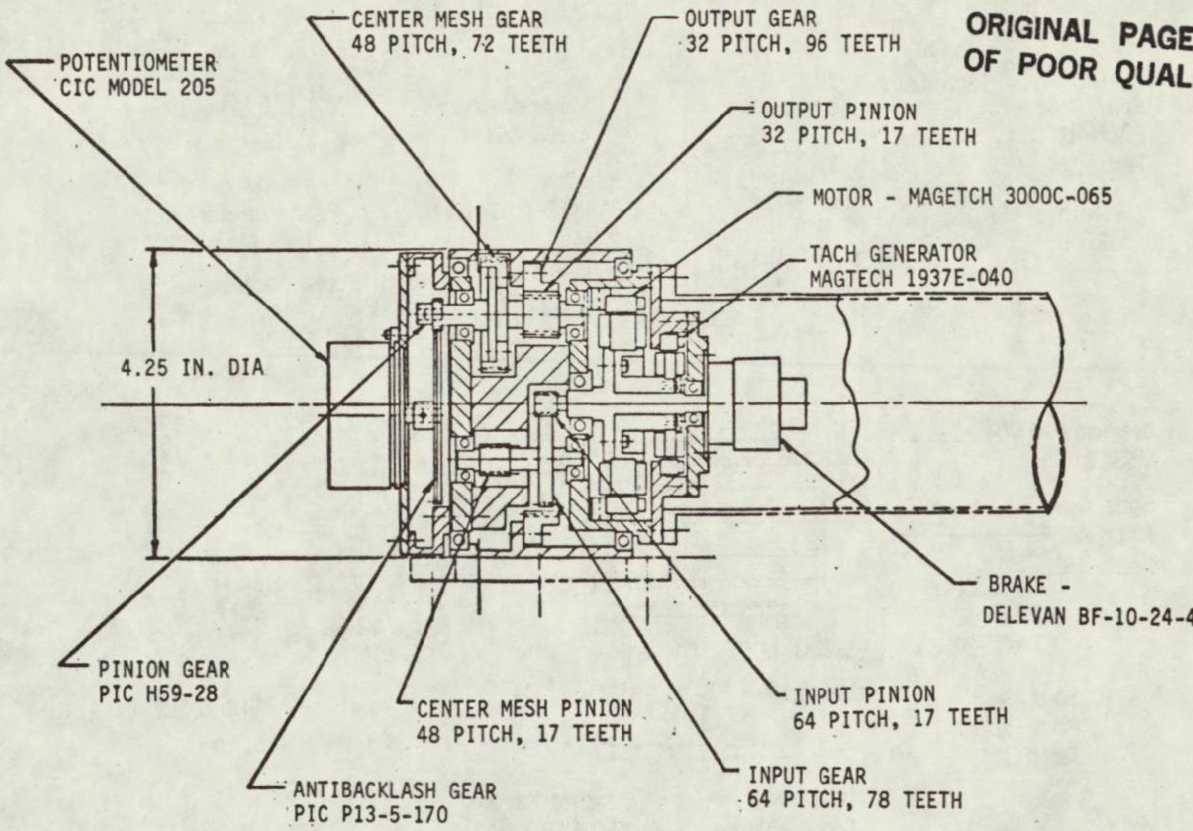

Figure VI-11 Wrist Pitch Drive (Y)

The wrist roll drive design shown in Figure VI-12 was driven by the desire to minimize length so that the operating length for axial module exchange would be minimized. The outer form of the end effector is cylindrical and thus could be readily mounted in large diameter, small cross-section bearings as is desired. The drive then took the form of a large gear mounted on the end effector and driven through a further gear reduction from the motor shaft. Other drive elements--brake and tachometer generator--are on the motor shaft, while the potentiometer (not shown) is geared directly to the large gear. Note that a full 360 degrees of travel are provided in the wrist yaw drive so that any module can be positioned in any orientation on the spacecraft.

The end effector concept is an extension of our prior work on general purpose manipulators and is designed to mate with either of the two interface mechanisms. It accomplishes two things: 1) it attaches the servicer mechanism to the module; and 2) it operates the latching mechanism. End effector attachment

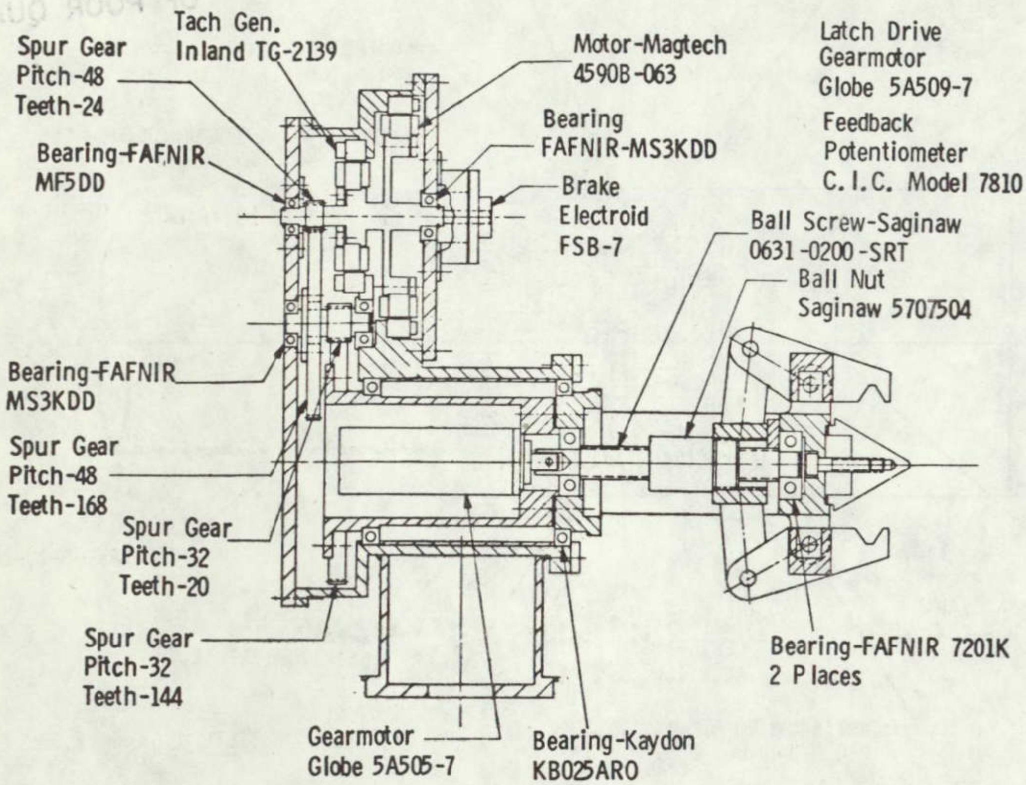

Figure VI-12 Wrist Roll Drive (Z) and End Effector

is accomplished by two closing jaws grasping a rectangular shaped baseplate grip. 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 latch drive mechanism (not shown) is an integral part of the end effector attach drive. It is operated by an electric motor through a gear head. The motor and gear train are designed to produce an operating torque of 28 in.-lbs with a stall torque of 50 in.-lbs. Both the end effector and interface mechanism latch drive designs are similar to those of the first IOSS and to the hardware delivered on the first IOSS.

Stowage of the servicer mechanism for major engine firing of the carrier vehicle involves two systems. One to deploy/stow the center post with docking mechanism and the second to latch the arm in place near the front of the stowage rack. This should be done with a minimum of additional actuators and latches, should maintain the servicer's basic structural stiffness in the deployed position, and should minimize stowed length so that Orbiter cargo bay space is not wasted. The selected system is shown in Figure IV-13.

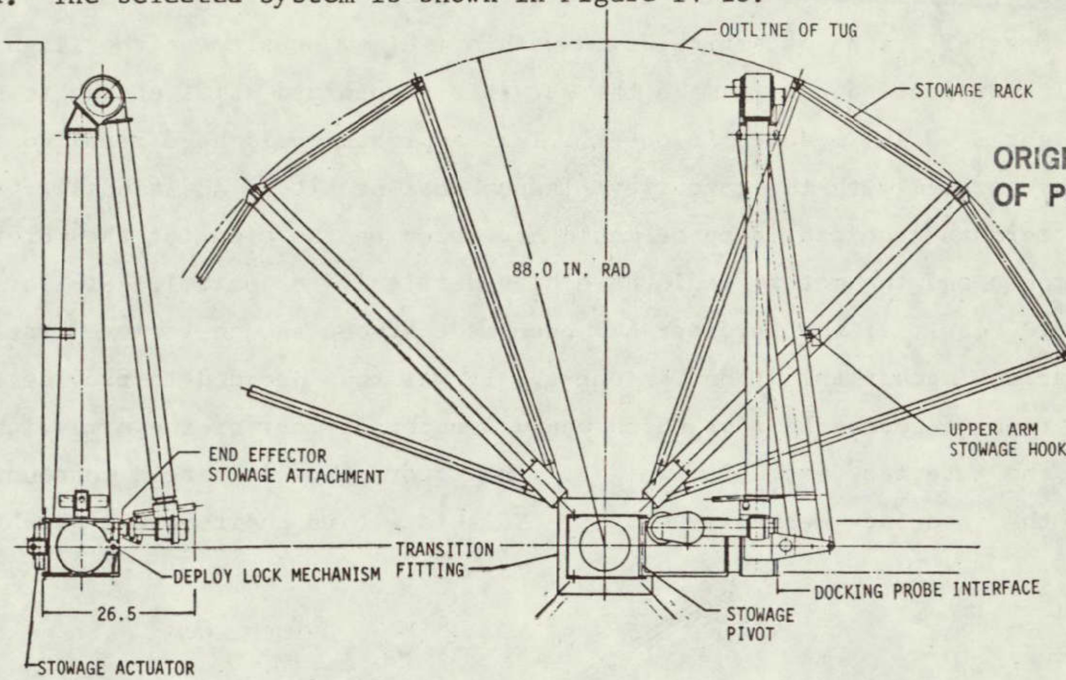

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Figure VI-13 Servicer Mechanism Stowage

The deploy/stowage operation involves a stowage pivot, a lock mechanism, and an actuator. The pivot is in the transition fitting at the center front of the stowage rack. An overcenter double pin deploy lock mechanism with a separate actuator is used. The stowage actuator that actually folds the mechanism is a ball-screw drive.

The method of latching the servicer mechanism in place uses the servicer drives themselves. The shoulder roll drive rotates the translation members to where they are close to the front of the stowage rack. The shoulder pitch drive then forces the upper arm stowage hook under a latch on one of the stowage rack upper beams. The elbow and wrist drives are then used to line the end effector up with the end effector stowage attachment. The end effector motor is then used to tighten the jaws on the end effector stowage attachment. These operations will hold the servicer mechanism tightly in place with no need for additional actuators. The axial distance required for this method of stowage is 26.5 inches. It will be necessary to provide a method of latching the docking probe in place when its design details become known.

A servicer mechanism counterbalance system for use between flights is shown in Figure VI-14. It is intended to provide a method of moving the mechanism through its full range of articulation with a minimum penalty on the flight version. It was decided not to make the servicer strong and stiff enough to handle full weight (700 lb) modules in one-g. That approach would have resulted in a very heavy system with the concurrent launch cost penalty. Additionally, the control torques required in space would have been on the order of the motor threshold torques and the motors would have been difficult to control. Similar problems would result if the servicer had been made strong enough to move itself and the interface mechanisms around in one-g. It was thus decided to provide a bolt-on type of counterbalance which would counteract most of the arm weight and part of the interface mechanism weight. The motors thus only need to counteract half of the interface mechanism weight. This is within their normal capability.

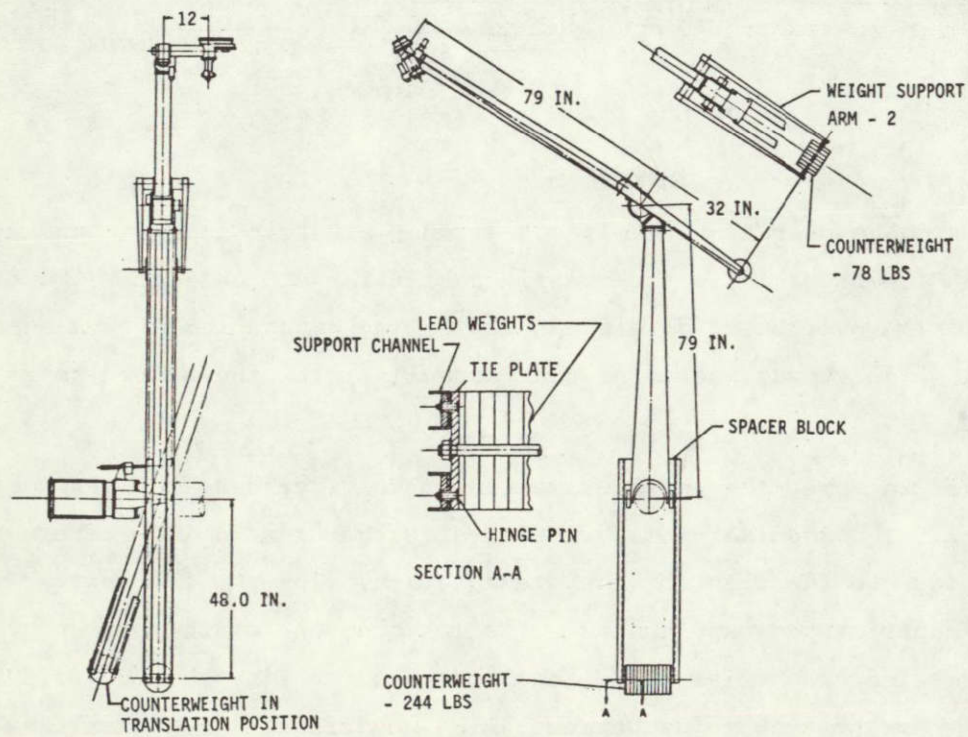

Figure VI-14 Servicer Mechanism Checkout Counterbalance System

The counterbalance weights are segmented and can be added or removed in increments. Use of this property permits counterbalancing the mechanism, with or without the interface mechanism attached, more exactly in selected configurations for specific tests. The entire counterbalance system is removed for flight and thus only a minimum weight penalty results.

A weight statement for the servicer mechanism alone is shown in Table VI-9. The total weight is within 3% of that for the pivoting arm of the first IOSS. This axial/near-radial configuration is longer, has more degrees of freedom, and a greater module removal capability than the pivoting arm. The weight of the mechanism deployment/stowage hardware has been included while the weight of the docking probe, its retraction mechanism, and its stowage latches have not been included. The joint weights appear larger than usual due to the inclusion of the joint-to-arm attachment fitting weight into the joint weight.

Table VI-9 Servicer Mechanism Weight Statement

END EFFECTOR/WRIST ROLL (Z)	12.0
WRIST YAW JOINT (Y)	9.0
WRIST PITCH JOINT (W)	7.5
LOWER ARM TUBE	3.7
ELBOW JOINT (V)	27.0
FOUR BAR LINKAGE	16.0
SHOULDER TRANSLATION (U)	4.0
SHOULDER ROLL (T)	32.0
ELECTRICAL WIRING	15.0
INTERCONNECTING TUBE FITTING	6.0
STOWAGE DRIVE	4.0
DEPLOY LATCH MECHANISM	3.0
	139.2 pounds

To obtain a total flight weight it is necessary to add the weight of several items to the weight of the servicer mechanism. These items include: the control electronics at 30 lbs, the stowage rack (Section C), the stowage rack Orbiter flight support system (Section D), if appropriate, and the weights of the modules to be flown.

C. INTERFACE MECHANISMS

The module--or space replaceable unit--interface mechanism provides 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 which is fastened to the module and a baseplate receptacle which is fastened to the spacecraft or to the stowage rack. The baseplate receptacle is passive. The baseplate has the linkages, cams, and rollers which latch the baseplate into the receptacle. The baseplate mechanism is mechanically driven from the servicer end effector. The interfaces of the interface mechanism are thus with the modules, the servicer end effector, the spacecraft, and the stowage rack as discussed in Chapter V.

The first IOSS interface mechanism analyses and the serviceable spacecraft configurations of Chapter II were reviewed to identify a logic for selecting a single interface mechanism as a standard. The data did not lead to such a logic, rather it indicated that a variety of interface mechanisms are possible and could be useful. The disadvantage of multiple interface mechanism alternatives is probable higher cost. This analysis is summarized in Table VI-10.

Table VI-10 Interface Mechanism Analysis Summary

SERVICEABLE SPACECRAFT CONFIGURATION AND MODULE DATA WERE REVIEWED
FIRST IOSS INTERFACE MECHANISMS WERE REVIEWED TWELVE DESIGNS FROM LITERATURE TWO NEW DESIGNS
MANY USEFUL INTERFACE MECHANISM ALTERNATIVES CAN BE GENERATED
GUIDELINES POSTULATED IN THE FIRST IOSS STILL ARE VALID
THE INTERFACE BETWEEN THE SERVICER END EFFECTOR AND THE INTERFACE MECHANISM SHOULD BE STANDARDIZED
INTERFACE MECHANISM RECEPTACLES MUST BE COMPATIBLE WITH THE STOWAGE RACK STRUCTURE
WITHIN THESE GENERAL LIMITS, INTERFACE MECHANISM DESIGN ALTERNATIVE SHOULD BE PERMITTED

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.

A set of module interface mechanism design criteria were prepared in the first IOSS. These included a requirement for nonredundant module support so that spacecraft loads would not couple into the module structure and so that spacecraft or module structural deflections due to loads, thermal effects, or one-g load relief would not couple into each other. This approach simplifies the structural analysis. The arrow symbols of Figure VI-15 represent the forces that can be reacted by, or transmitted through various types of fasteners. The left hand side symbol represents two plates butting together where only a compressive force can be transmitted. The right-most symbol represents a bolt in a tight hole where all three force components in both directions can be reacted. There are a number of useful configurations that will provide a nonredundant fastening.

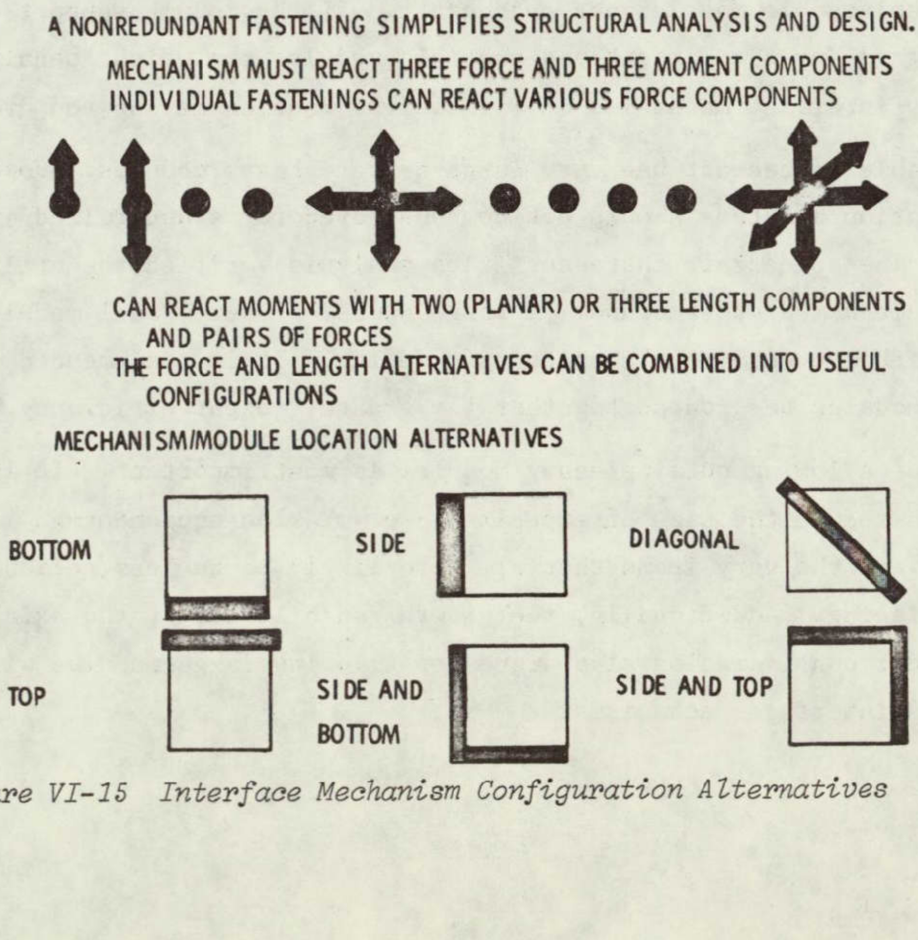

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Figure VI-15 Interface Mechanism Configuration Alternatives

The first IOSS resulted in the design and fabrication of two interface mechanisms--one for bottom mounting, and one for side mounting. The sketches near the bottom of Figure VI-15 are intended to show additional interface mechanism location possibilities. Various nonredundant attachment configurations can be used with each of the location alternatives. This wide variety of potentially useful interface mechanism alternatives with no apparent approach to selecting a "best" led to the recommendation for leaving it up to the spacecraft designer to make his own selection.

It is also felt appropriate to establish "standard" interface mechanisms that a spacecraft designer can buy off the shelf if he desired. To that end an analysis of module size and weight distributions was made to identify useful standard sizes of interface mechanisms.

The first IOSS suggested the development of an interface mechanism as a two-part kit in perhaps three sizes. These standard interface mechanisms could be made available to spacecraft designers. Each designer could then make his choice within his own set of design and economic constraints. The following discussion provides a basis for selecting the three sizes.

Interface mechanism weight is an important consideration because of the large number of replaceable modules per spacecraft. If the module support mechanism is designed for the largest and heaviest module, the weight penalty in using the heavy interface mechanism on lightweight modules may be too great.

Some serviceable spacecraft use very large or very heavy modules. However, they are the exception and thus should not be considered for standardized interface mechanisms. The spacecraft characteristics analysis verified the utility of the 40-in. module as an upper bound. A large number of very small modules were identified with one serviceable spacecraft design. It is recommended that the smaller modules be grouped together for greater weight efficiency.

The concept of allowing outsize/heavy modules is most important. It is often difficult to reduce the size of experiment--or mission equipment--modules and these are the very items that are more likely to be less reliable and thus need replacing. Additionally, the growth capabilities of the axial/near-radial servicer configuration also argue for allowing large modules with specially-designed interface mechanisms.

The majority of the available information on module sizes and weights from serviceable satellites was referenced in *The Effect of Serviceable Spacecraft Design on Servicer System Requirements*, Martin Marietta Corporation, May 1976. That information plus the module data for the TRW version of SEOS resulted in data on 683 modules from 30 different serviceable spacecraft. Eleven size categories were identified as: <15 in., 40 to 100 in., >100 in., and eight others in a geometric progression from 15 to 40 inches. The modules were then grouped into the categories based on the largest module dimension. The number of modules per category varied from four to 180. The largest number in specific categories were: 81 from Space Telescope were <15 in., 165 from Geosynchronous Platform Study by RI were 22-25 in., and 60 from Payload Utilization of Tug by MDAC were 28-31 in. There were no modules less than 2.5 in. Only four modules were in the >100 in. category.

The resulting data is plotted in Figure VI-16 in cumulative form where the abscissa corresponds to the largest dimension of the category; e.g., 40 in. for the 35-40 in. category. The effect of the large quantities of modules in specific categories can be seen. The 40-in. module size captures 92 percent of the total and still seems to be a valid upper boundary. The 26-in. size nicely includes the large number of 25-inch modules and leaves about one-third of the modules for the 40-in. size to capture. The third size was selected at 17 in., which puts the three sizes in a geometric progression and captures slightly less than one-third of the modules. It was felt desirable to have a size that would not unduly burden the very small modules and thus the 17-in. size is slightly smaller than that for the 30% figure.

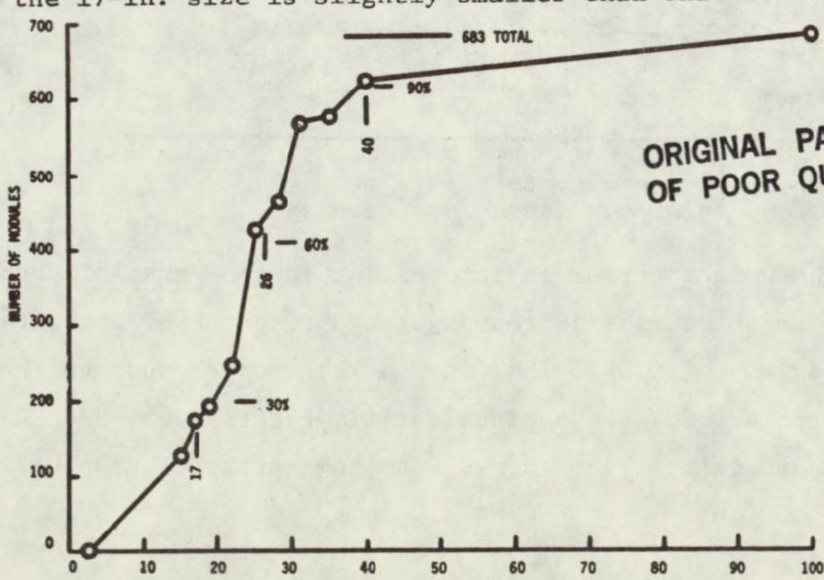

Figure VI-16 Module Size Distribution

The recommended interface mechanism standard sizes thus become--17 in., 26 in., and 40 in. These correspond to modules no larger than a cube of the indicated dimension. Modules of smaller dimensions can of course be accommodated.

The module weight data is plotted in Figure VI-17 for most of the 683 modules. Data in three categories were deleted as the number of modules in the sample were too small and the data was not believed to be representative. Average weight for all the modules is plotted as is a weight representing an upper bound for 90 percent of the modules in that category. Again, the emphasis is on being able to satisfy the major part of the requirement, yet not be driven by extremes. Design module weight values have been selected to be slightly over the 90% curve. These module weights are significantly less than previously and are:

40 in. interface mechanism - up to 400 lb modules;
26 in. interface mechanism - up to 200 lb modules;
17 in. interface mechanism - up to 75 lb modules.

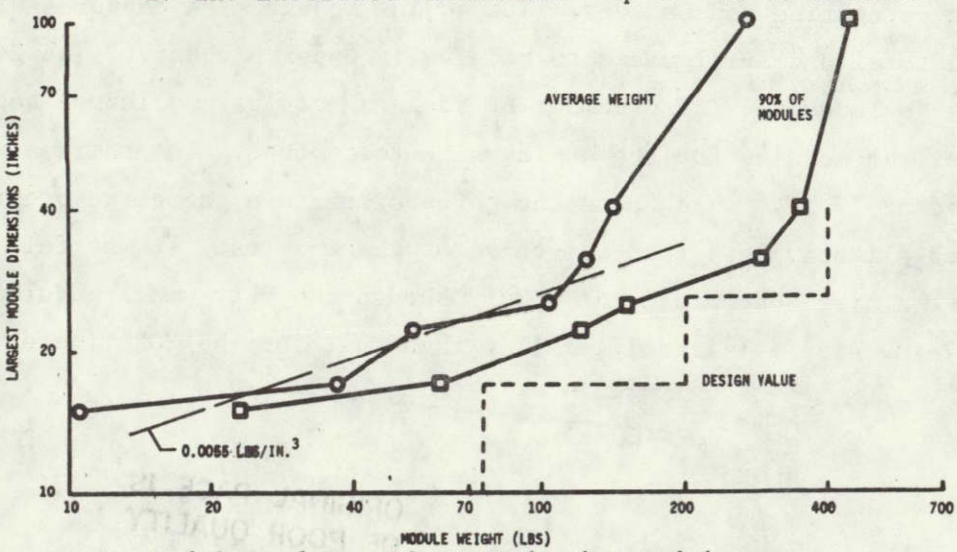

Figure VI-17 Module and Interface Mechanism Weights

The shape of the weight curves is interesting in that the left hand part of the average weight curves is reasonably represented by a constant density (cubical modules) of 0.0055 lbs/ in.^{3}. This corresponds to the average density of spacecraft such as Intelsat II, Fleetsatcom, and RCA Satcom. The right hand part of the curves seem to represent an upper limit on module weight of 500 lbs.

Preliminary estimates of flight unit weight for the three sizes of interface mechanisms in both the side- and bottom-mounting configurations were made. The reference information was the weight of the interface mechanism engineering test units delivered to NASA as part of the first IOSS. These weights were adjusted for design improvements and flight qualification. The weight for the three sizes was then estimated considering loads, fixed end effector attach geometry, fixed connector travel, and fixed end effector torque limits.

Interface Mechanism Weights (lbs)

	Bottom-Mounting		Side-Mounting	
Size	Receptacle	Baseplate	Receptacle	Baseplate
17 in.	2.6	12.8	3.4	9.0
26 in.	3.5	17.0	4.5	12.0
40 in.	5.3	25.5	6.8	18.0

For completeness, the interface mechanism characteristics are shown in Table VI-11. The interface mechanisms--both side and bottom mounting--were designed to satisfy the requirements established in the first IOSS. Our recent work has not indicated any need to update those requirements. The totality of on-orbit servicer characteristics is addressed in this material in three separate groups: servicer mechanism, stowage rack, and interface mechanism.

Table VI-11 Interface Mechanism Characteristics

- HAVE STANDARD INTERFACE WITH SERVICER END EFFECTOR
- BE COMPATIBLE WITH STOWAGE RACK STRUCTURE
- ACCOMMODATE A WIDE RANGE OF MODULE SIZES AND MASSES
- BASEPLATE TRANSMITS ALL FORCES AND MOMENTS
- ACCOMMODATE A RANGE OF CONNECTOR TYPES AND FORCES
- ACCOMMODATE MISALIGNMENT IN SIX DEGREES OF FREEDOM
- SMALL AND LIGHTWEIGHT
- BE COMPATIBLE WITH OPERATION BY ASTRONAUT
- PROVIDE NONREDUNDANT MODULE SUPPORT
- ALLOW FOR THERMAL AND STRUCTURAL DEFLECTIONS
- PROVIDE A TWO-STAGE ENGAGEMENT--CAPTURE AND LOCKUP
- PROVIDE SEPARATION FORCES
- PROVIDE POSITIVE LOCKUP DEVICE

Our spacecraft and servicer requirements work indicated that a variety of interface mechanism designs should be permitted providing they each satisfy the appropriate characteristics of Table VI-11. Three sizes for each of the side and bottom mounting interface mechanisms were derived above. It is recommended that these sizes be made available for those spacecraft designers who elect to use them.

The characteristics shown provide for separate structural analyses of module, spacecraft and stowage rack and avoid stresses due to thermal deflections in the mounting structures. The concept of having a mechanical drive interface in the end effector which avoids electrical interconnections and the concept of all latching forces being contained in the interface mechanism are felt to be significant in terms of evolving a workable system.

A bottom-mounted SRU interface mechanism based on an MSFC concept and a side-mounted interface mechanism were designed as part of the first IOSS and engineering test units of each were built and delivered. They are shown in Figures VI-18 and 19.

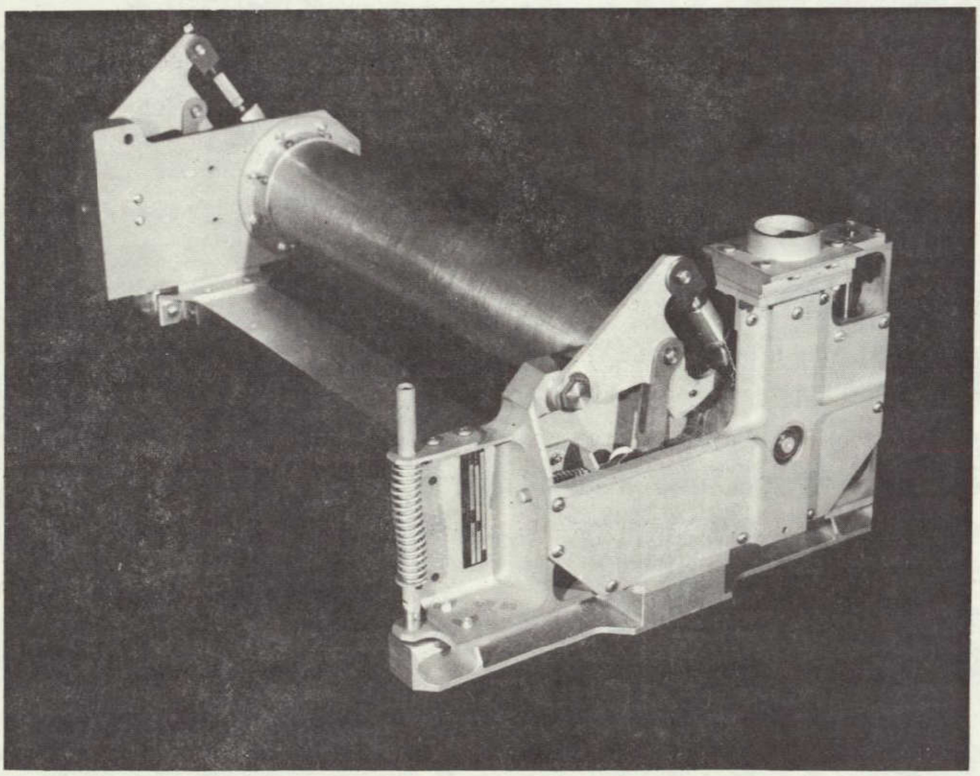

Figure VI-18 Bottom-mounted Interface Mechanism Engineering Test Unit

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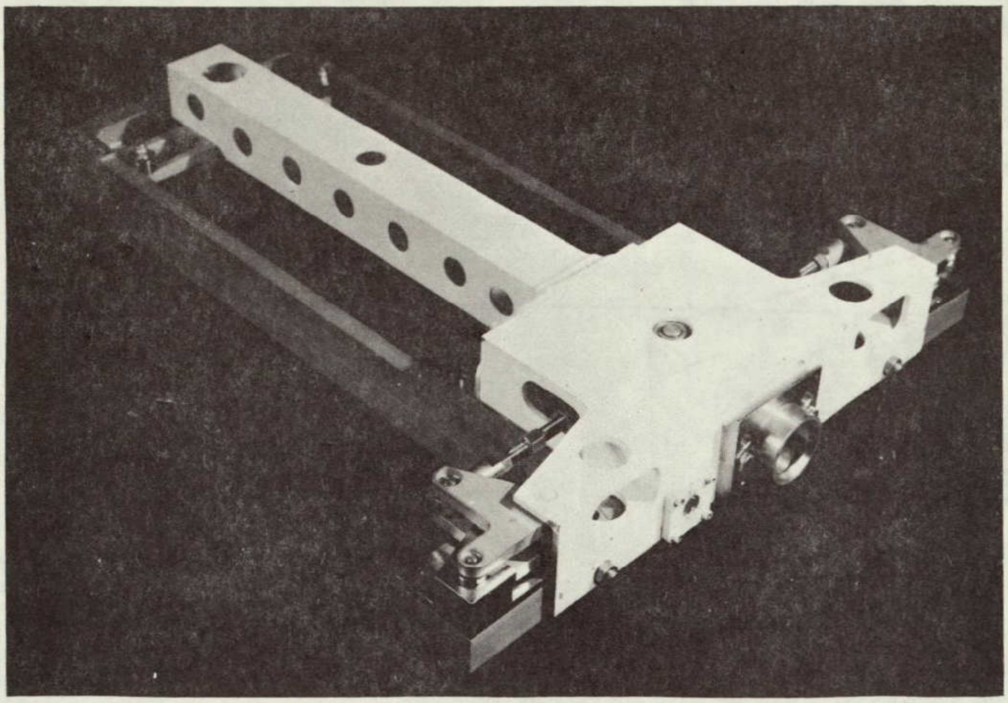
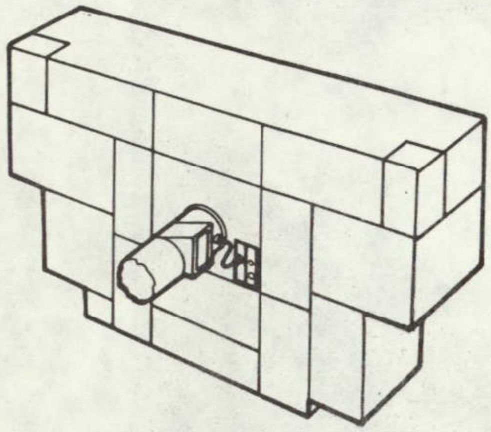


Figure VI-19 Side-mounted Interface Mechanism Engineering Test Unit

The space designs from which these two engineering test units were adapted satisfied all of the interface mechanism characteristics of Table VI-11.

A hardwire connection between the servicer and the satellite is required to perform several functions: 1) control spacecraft power; 2) verify module connector engagement; 3) verify interface mechanism engagement; 4) verify power - ON in modules before undock; and 5) control spacecraft attitude control system. It is necessary to be able to turn spacecraft power off so that arcing does not occur at the connector when modules are being replaced. It is also advantageous to be able to verify that connector continuity exists, the interface mechanisms are fully engaged and that the modules are receiving power before the servicer is undocked. Satellite checkout is recommended to occur after undocking and will use the normal ground control techniques used for initial satellite deployment.

Rather than have the hardwire connection made as part of the docking operation, it was decided to mount the connector, with a cable to the docking probe, on a shortened interface mechanism, as shown in Figure VI-20. The servicer can handle this interface mechanism just like any other interface mechanism while allowing for the length of the interconnecting cable. The connection is made after docking is complete, but before module exchange starts and the connector is stowed on the docking probe after all modules have been exchanged and before undocking.



- Connector mounted on short interface mechanism
- Servicer handles connector
- Connector stows on side of docking probe
- Short cable to docking probe

Figure VI-20 Servicer to Spacecraft Hardwire Connection

To keep the number of pin connections down, it is suggested that the multiplex, or data bus, approach be used. This system would be basically separate from the satellite data bus system so that it can be operated with satellite power off. The multiplexer power would come from the servicer. The multiplexer signals would go to each spacecraft module connector, to the spacecraft power terminals within each module, and to a microswitch at each interface mechanism. Multiplex connection to control (on/off/standby) the spacecraft attitude control system can also be made.

Should more extensive satellite checkout be desired before undocking, then additional separate connectors and cable systems, mounted in the same interface mechanism, can be provided. These additional connections would be routed through the servicer to the carrier vehicle (Orbiter or Tug).

D. STOWAGE RACK

The purpose of the replacement module stowage rack is to provide structural support for the replacement modules during launch and reentry as well as for the servicer mechanism and docking probe as shown in Figure VI-2. Its design is complicated by the uncertainty in the number and weight of modules that will be carried to orbit on specific missions. The serviceable spacecraft analysis of Chapter II did provide certain guidelines which are given in Table VI-12. A review of average parts factors from the first IOSS and average densities leads to a high design margin in terms of volume available for a single spacecraft servicing. This calculation used a stowage rack diameter of 14.7 feet and a length suitable for one tier of 40 in. modules. Statistical variations in parts factor, density variations, spacecraft weight, and the desire for multiple spacecraft servicing will all reduce the margin for specific missions.

Table VI-12 Stowage Rack Design Guidelines

DESIGN TO BE COMPATIBLE WITH SERVICER MECHANISM - Module replacement direction - Alternative forms of modularized servicer mechanism - Provide for servicer mechanism stowage - Provide for docking probe stowage
PROVIDE 40-in. AXIAL LENGTH TO ACCOMMODATE MODULES
USE MAXIMUM ALLOWABLE DIAMETER = 14.7 ft
PROVIDE ONE TIER OF MODULE STOWAGE - Average volume margin = 6.5
SELECT STRUCTURAL TYPE FOR MINIMUM WEIGHT
DESIGN TO BE COMPATIBLE WITH VARIETY OF INTERFACE MECHANISM CONCEPTS
IT MAY BE DESIRABLE TO HAVE MORE THAN ONE STOWAGE RACK DESIGN TO MINIMIZE TUG MISSION WEIGHT

Note that the stowage rack configuration was defined after the servicer configuration was selected as noted in Chapter V. This approach is more logical and has fewer pitfalls than forcing the servicer mechanism configuration to accommodate to arbitrarily selected stowage rack requirements. The selected approach results in interesting results such as radial module removal from the spacecraft and axial module removal from the stowage rack.

The major contributor to servicer system weight in the first IOSS was the stowage rack. Thus its design was examined more carefully in this study.

While our initial considerations left open the possibility of heavy, low-cost stowage racks for Orbiter missions and lightweight, higher-cost stowage racks for Tug missions, it was found that a single lightweight design could be developed for both applications.

The on-orbit servicer has been designed to satisfy the requirements established in the first IOSS contract and as updated by the Chapter II analysis. As noted in Chapter V, those requirements have been divided into four groups: system level, servicer mechanism, stowage rack, and interface mechanism. The groups are compatible and are reported on separately in this material in terms of system characteristics. Those for the stowage rack are listed in Table VI-13. The stowage rack configuration has been selected for a direct interface with the full capability tug (FCT) at its forward spacecraft attach ring. Compatibility with the Orbiter is obtained by addition of a flight support system which is described below. Compatibility with the Earth Orbital Teleoperator System (EOTS), the Interim Upper Stage (IUS), the Solar Electric Propulsion System (SEPS), or other applicable upper stages will be accomplished by adapters. The basic premise of a 176 in. diameter cylinder should make the stowage rack readily compatible with most of these carrier vehicles. In the case of the smaller EOTS a special stowage rack may be more appropriate.

Table VI-13 Module Stowage Rack Characteristics

- COMPATIBLE WITH OPERATIONS AT ORBITER, TUG (IUS, FCT), EOTS
- MULTIPLE SPACECRAFT CAPABILITY PER MISSION
- PROVIDE FAILED MODULE TEMPORARY STOWAGE
- PROVIDE MODULE ENVIRONMENTAL CONTROL
    - Thermal, Radiation, Contamination
- WITHSTAND ORBITER CRASH LOADS
- BE COMPATIBLE WITH ORBITER/TUG/EOTS ELECTRICAL POWER
- STOW MODULES OF THE FOLLOWING SIZE CHARACTERISTICS

    Large -- 40 x 40 x 40 inches - 400 lbs
    Medium -- 26 x 26 x 26 inches - 200 lbs
    Small -- 17 x 17 x 17 inches - 75 lbs
- REPRESENTATIVE MODULE COMPLEMENT

    40 in. size - 3*
    26 in. size - 5
    17 in. size - 4
    *One of these spaces is left open for temporary module stowage
- LIGHTWEIGHT

The module sizes and complement shown are intended to be representative of a typical upper bound mission. Many spacecraft will require fewer modules for a single repair mission. There are no constraints on which modules may be used first. For each specific mission, the module complement must be identified and the interface mechanism receptacles appropriately located.

The Orbiter crash loads have again been found to be the critical strength requirement. The desire for a lighter weight system has resulted in a quite different approach to the stowage rack structure design. It must be remembered that outsize modules, different styles of interface mechanisms, or other special requirements may make it a necessity to design other stowage rack configurations. However, the concepts and approaches used in the following should simplify the design process.

The representative set of 11 modules and one temporary stowage location identified in Table VI-13 are shown in Figure VI-21. The basic stowage rack configuration is a truss work consisting of four frames that connect the central transition fitting to the circumferential tug structure and which supports the modules through the interface mechanisms. The servicer mechanism attaches to the transition fitting. The outer ends of the four trusses are stabilized to the tug structure through sets of braces.

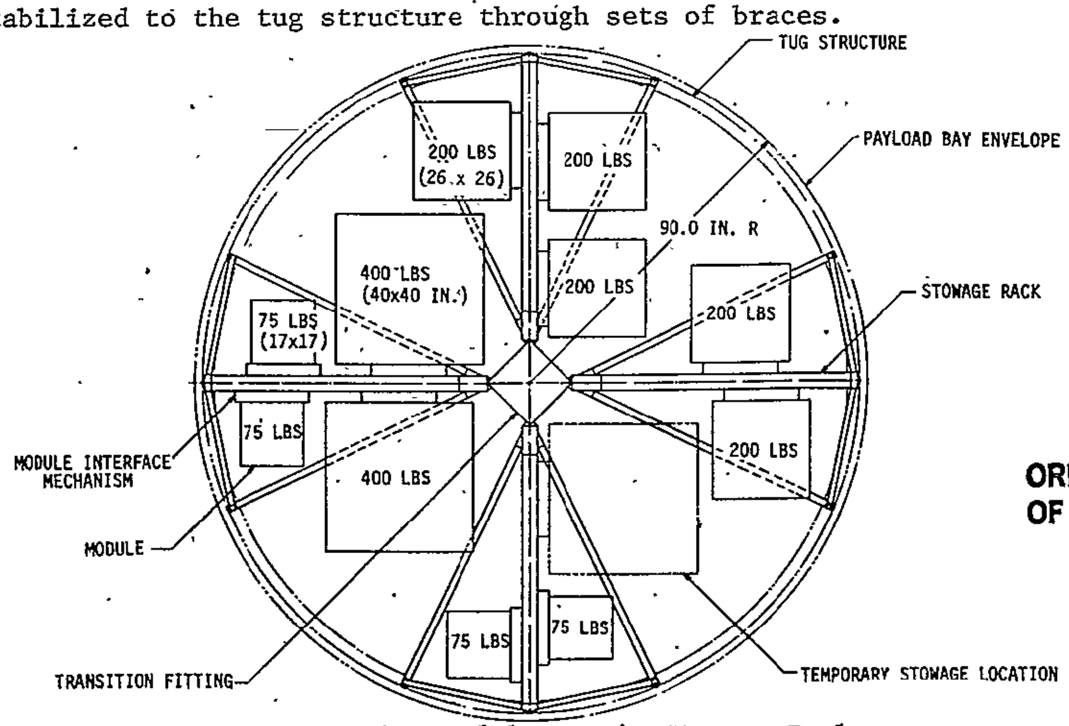


Figure VI-21 Representative Module Set in Stowage Rack

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The weights for the various modules shown in the figure are those used in the stress analysis and are adequate for 90 percent of the modules as was identified above. The module plan view sizes are shown. The depth of the modules can be anything up to 40 inches. As the interface mechanism receptacles, or guides, mount to the upper and lower beams, they must be long enough (or have support extensions) to span the 40 inch distance between these beams. The module temporary stowage location was made as large as the largest module carried. Each failed module is first stored in the temporary location and then moved to the good module's previous location. This opens up the temporary location again. It, of course, can be used for temporary stowage of smaller modules as the guide spacing is compatible for all side mounting interface mechanisms.

The truss structure and module configuration shown are appropriate for axial module replacement (which does not imply only axial module replacement in the spacecraft) and the side mounting interface mechanism. Radial module stowage and/or use of bottom mounting interface mechanisms would require adapters or a different truss configuration.

The stowage rack truss design is shown in Figure VI-22. It is made up of four identical trusses and a central box. The figure shows a side view of one truss, an end view of a second truss, and a partial side view of a third truss, as well as the central box structure. All four trusses are identical. The servicer mechanism mounts to the transition fitting at the top of the central box. The main truss elements are the upper and lower box beams which also provide the attachments for the interface mechanisms. The upper and lower beams are cross-braced and supported laterally at the outer ends. The outer ends of the lower beams mate to the full capability tug (FCT) spacecraft attach points as do the lateral braces. The lateral braces are tied to the outer end of the lower beam with column members. Additional column members tie the inner ends of the lower beams to the outer ends of the lateral braces.

Attachment of the stowage rack trusses to the FCT is via 12 of the 16 spacecraft deployment fittings, one of which is shown in Figure VI-22. These are hydraulically powered. This feature may be disabled for any tug flight where it is intended to return the servicer, or they may be retained as an emergency backup system should the servicer fail to stow itself at

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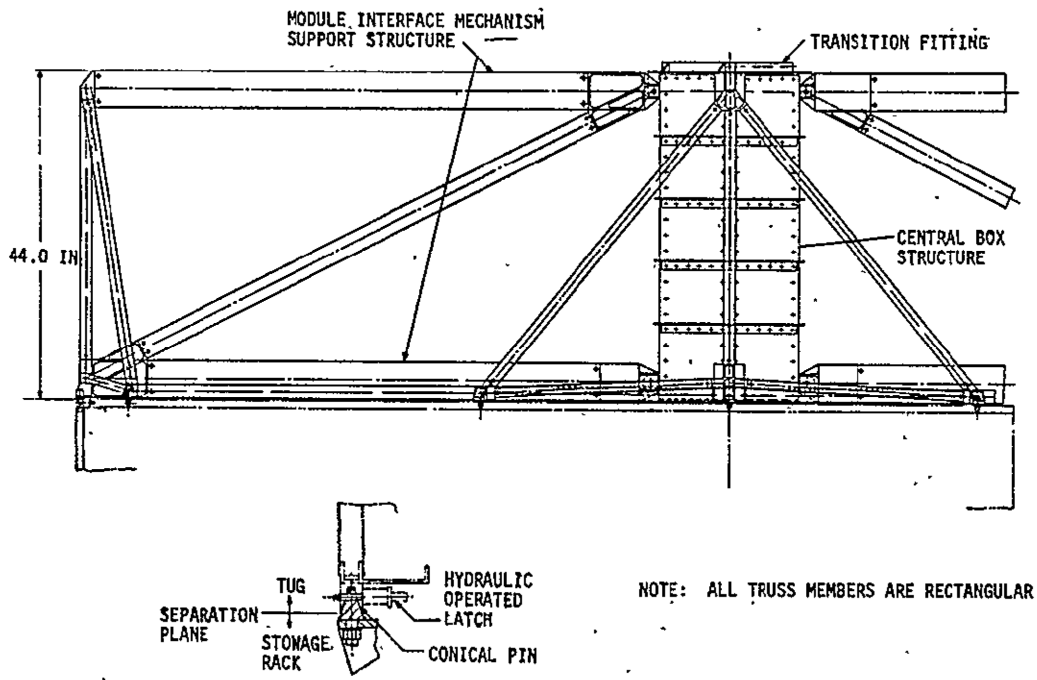

Figure VI-22 Stowage Rack Design

the completion of the servicing activity. Note that the cost analyses of Chapter IX indicate that it is cheaper to leave the servicers and stowage racks in geosynchronous orbit.

A stress analysis was used to aid the stowage rack design process and to determine member sizes and thus weight. This analysis is capsulized in Table VI-14. A review of the first IOSS stowage rack design requirements led to the decision to delete the requirement for Orbiter crash landings with a spacecraft mounted to the front of the stowage rack. This is a very unlikely event and can be accommodated with special structure in the Orbiter if the special case should arise. It was also decided to avoid the need for a skin around the outside of the stowage rack if possible as that approach resulted in a relatively heavy stowage rack in the first IOSS study.

The general criteria for the stowage rack and a representative module complement are defined above. A single 400 lb module mounted at the middle on one side of a truss was also considered as a design load condition. As with most lightweight truss type structures, beam column and local skin buckling effects were found to be significant. The Orbiter crash loads sized most members.

Table VI-14 Stowage Rack Stress Analysis

- MODULE COMPLEMENT
  - Three 400-lb modules
  - Five 200-lb modules
  - Four 75-lb modules
- CONDITIONS
  - Full capability tug attach points
  - Tension, compression, beam column, and local buckling effects
  - Worst combinations of module locations
- LOADS
  - LXRT docking interface moment
  - Orbiter launch and reentry loads
  - Orbiter crash loads
- RESULTS
  - Critical conditions were column and local buckling
  - Margins of safety
    - 9 to 30 percent for crash loads
    - 40 percent for operating loads

Use of standard sizes of tubing resulted in the margins of safety, based on material ultimate strength, shown for crash loads. The conventional 40% margin of safety was used where the operating loads were critical.

The stowage rack was designed to mount directly to the full capability tug as that application is significantly more weight critical than is the Orbiter application. A configuration for an adapter between the stowage rack and the Orbiter is shown in Figure VI-23 as it would be in the launch and return parts of a low earth orbit servicing mission. The module stowage rack mounts to a large ring using the same attach locations as for the tug. A simple bolt-on attachment rather than the conical pin system of the tug is planned. The large ring is connected to the truss structure through a pair of pivots (one near each Orbiter cargo bay sill). A launch/return position latch is used to hold the ring in its proper position during these mission phases. The truss uses four cargo bay sill attachments (two on each side) and a keel fitting to transfer the servicer loads to the Orbiter structure.

The sill attachment spacing is such that the remotely operated attachment fittings can be used. These fittings are provided as part of the Orbiter system. Their use will permit release of the entire servicer system and jettison should the servicer system ever fail in a mode which prevents closing

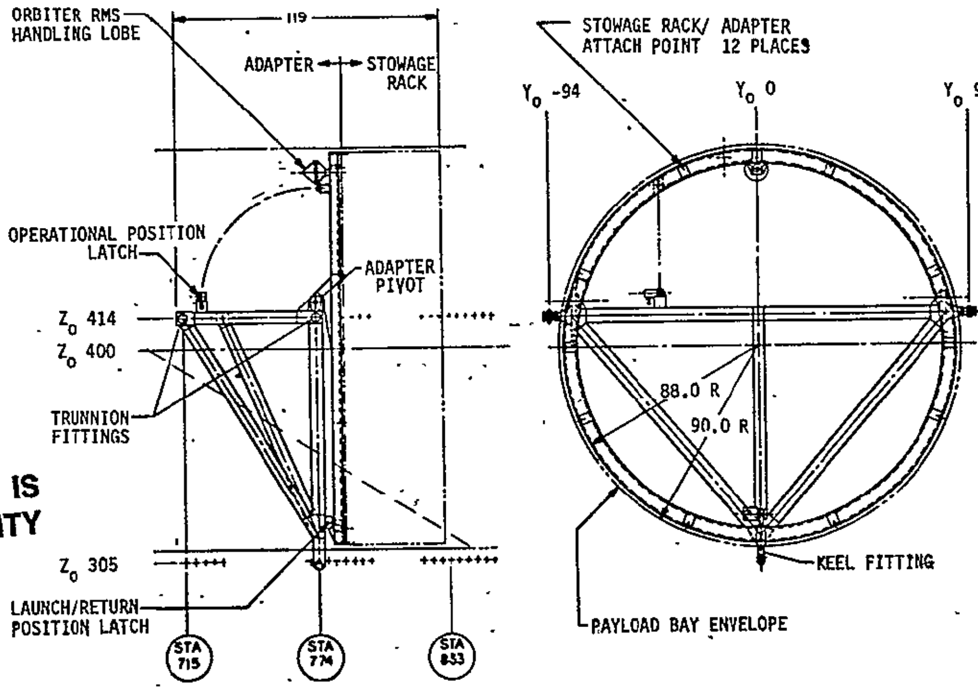

Figure VI-23 Stowage Rack to Orbiter Adapter - Launch/Return Configuration

the cargo bay doors. Use of the remotely operated fittings also permits launch and return of the servicer in one cargo bay location and use of it in another cargo bay location. It can be moved from one cargo bay location to another by the shuttle remote manipulator system (SRMS). The SRMS handling lobe is shown. Should the servicer location be changed in this way, then it will be necessary to remotely break and make the necessary electrical connections. The latches shown are to be electrically operated from the Payload Specialist Station on the Orbiter aft flight deck.

The stowage rack is rotated from the launch/return configuration to the operational configuration, as shown in Figure VI-24, by the Shuttle remote manipulator system. Use of the SRMS avoids the need for a separate actuator. An operation position latch is used to hold the large ring and stowage rack with respect to the adapter truss during module exchange. The operational loads are small (mainly inertia reactions to firing of the Orbiter attitude control system engines) and thus a single latch is felt to be adequate. However, a second latch can be provided if it is necessary.

A preliminary assessment of the SRMS reach and viewing capability was made and it appears that the servicer can be located as far forward as shown. The forward location is desirable so that the module exchange operation can

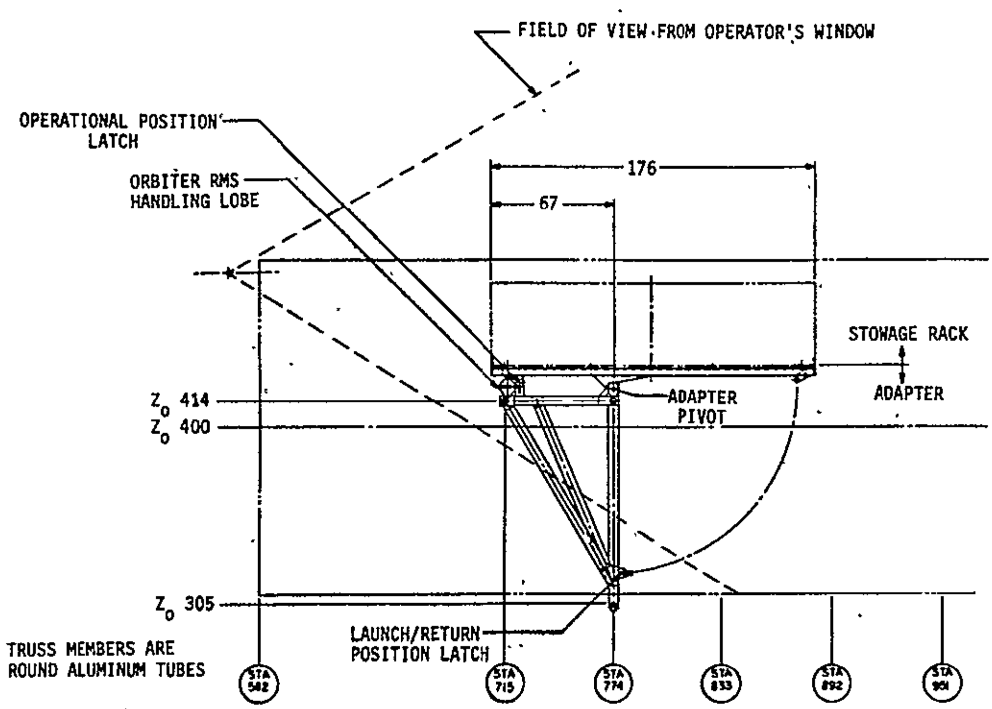

Figure VI-24 Stowage Rack to Orbiter Adapter - Operational Configuration

be viewed directly and closely from the cargo bay windows of the aft flight deck. This direct view cannot be used for control of the module exchange process because the servicer control panel is at some distance from and to one side of these windows. However, they can be used by a second astronaut for monitoring or problem assessment.

The forward operating location is also desirable so that the SRMS operator can view the servicer docking probe when he is docking a failed spacecraft to the servicer as well as during deployment of a repaired spacecraft.

A weight statement for the stowage rack with a complement of 12 interface mechanism guides, is shown in Table VI-15. The total weight is 175 lbs or 36 percent less than the weight of the first IOSS stowage rack. As can be seen the major weight contributors are the main beams and the interface mechanisms. Should fewer than 12 interface mechanisms be required on a mission, the extras can be removed and the weight appropriately reduced.

It is important to minimize the weight of the basic stowage rack as it goes to and returns from geosynchronous orbit where the per pound launch costs are high. It is not so important that the adapter to the Orbiter (or

Table VI-15 Stowage Rack Weight Statement

STOWAGE RACK		ADAPTER TO ORBITER	
Center Support	12.5	Ring Frame	112
Upper Main Beams	57.7	Basic Triangle	153
Lower Main Beams	68.8	Horizontal Beams	70
Outer Tubes	22.8	Braces	66
Lateral Tubes	48.6	Fittings and Mechanisms	81
Fittings and Fasteners	42.6		482 pounds
Interface Mechanism Guides (12)	56.5		
	309.0 pounds		

flight support system) weight be minimized as the cost to low earth orbit and return is not so important. For these reasons, the design of the adapter to the Orbiter has not been weight optimized. It is recommended that a brute-force approach be used to save design and test costs while ensuring adequate safety margins.

The servicer mechanism and control electronics weights are shown in Table VI-9.

A representative set of modules was selected for each of the three spacecraft described in Chapter IV. These were then fitted into the stowage rack, one set at a time, to determine if there were any problems. It was found that two sets of modules for either the DSCS-II or the SEOS could be accommodated in the stowage rack. There were some difficulties with the CLO, but one set of CLO-replacement modules could be accommodated in the stowage rack.

A representative set of modules was selected for a Defense Satellite Communications System - Phase II (DSCS II) maintenance mission. The selection was based on including the most difficult module (SRU) to store, one of each type of module, and a parts factor which is representative for this class of spacecraft. The space replaceable units (SRU) or modules selected are:

SRU	MODULE TYPE	WEIGHT LBS.
1	Communications	65.4
5	Telemetry, Tracking and Command	59.1
6	Attitude Control	66.7
8	Stationkeeping/RCS-Loaded	449.7
10	Reaction Control - Loaded	200.8
12	Electrical Power	66.0
	TOTAL	907.7

The largest and heaviest SRU on the DSCS II is the loaded stationkeeping/ RCS module. The resulting parts factor is 0.37, which is a little high for communications satellites. The large parts factor is due to including SRU 8. This SRU would not normally be replaced as it has been sized to have enough propellant for the full 10-year mission duration.

As can be seen from Figure VI-25, the modules are easily accommodated in the stowage rack. There is sufficient room for a second set of DSCS II modules so that two spacecraft could be serviced on one mission. The modules shown have been located only from geometric and stowage rack strength considerations. However, there is sufficient room to allow for additional constraints such as center of gravity control on the tug part of the mission. The SRUs may be located back-to-back or slightly offset from each other as may be desired. In general the communications spacecraft missions are not very demanding on stowage rack volume.

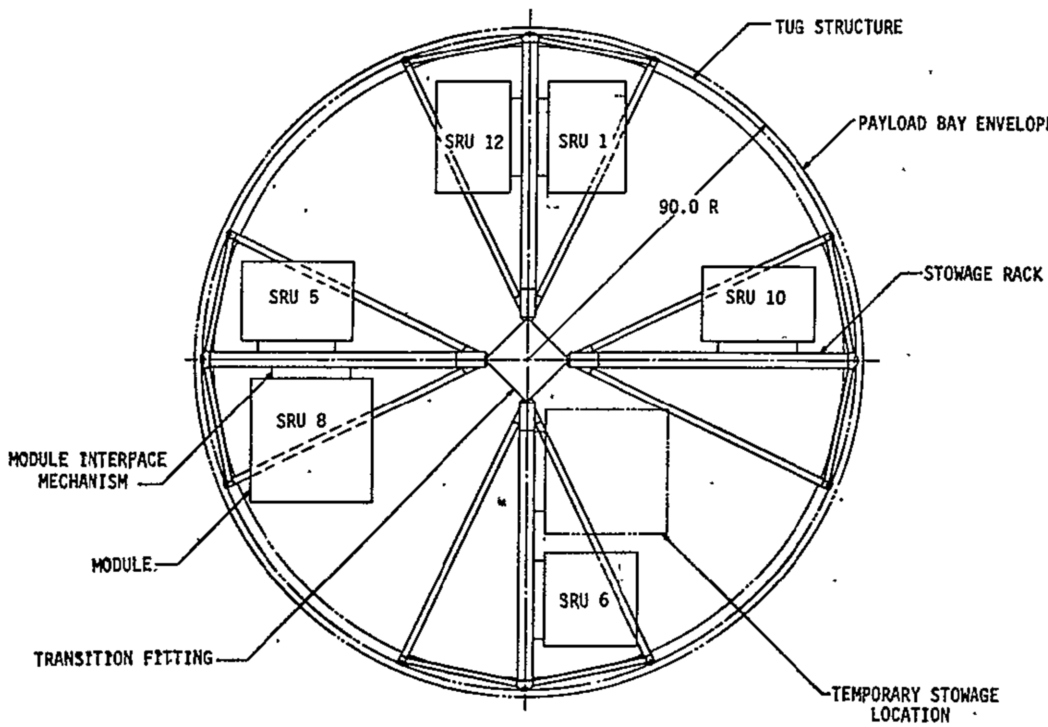

Figure VI-25 Stowage Rack - DSCS II Maintenance Mission

A representative set of modules was selected for a Synchronous Earth Observatory Satellite (SEOS) maintenance mission using the same criteria as for the DSCS II mission. The selected space replaceable units (SRU) or modules are:

SRU	MODULE TYPE	WEIGHT LBS.
1	Communications Equipment	163
4	Reaction Wheels and Electronics	80
8	Electrical Power Equipment	101
11	Meteorological Instruments and Radiator Panel	189
13	Sensor Electronics Equipment	50
15	Propulsion	281
	TOTAL	864

The SEOS modules are generally smaller and slightly heavier than the DSCS II modules. The two SEOS instruments pose a challenge in that their radiators exceed the 40-in. dimension that is used as a normal upper limit for module size. However, because of the orientation of the radiators with respect to the interface mechanism, it is possible to handle and stow these outsize instruments. The parts factor for the above complement of modules is 0.16. This compares favorably with the 0.15 parts factor used in the first IOSS for this same satellite.

As can be seen from Figure VI-26, the selected complement of modules is easily accommodated in the stowage rack. Note the large frontal area that must be allocated for temporary stowage of the meteorological instruments module with its large radiator panel. As with the DSCS-II, there appears to be adequate room for a second set of modules so that two SEOS spacecraft could be serviced on one mission. The stowage rack volume is also adequate for servicing one DSCS-II and one SEOS on the same mission. The module locations shown on the figure can easily be adjusted to provide center-of-gravity control for the tug portion of the mission. From the communications satellite (DSCS II) and the geosynchronous earth observatory (SEOS) evaluations, it appears that the selected stowage rack configuration is suitable for most geosynchronous maintenance missions.

A representative set of modules was selected for a Characteristic Large Observatory (CLO) maintenance mission using the same criteria as for the DSCS II and SEOS missions. The selected space replaceable units (SRU) or modules are:

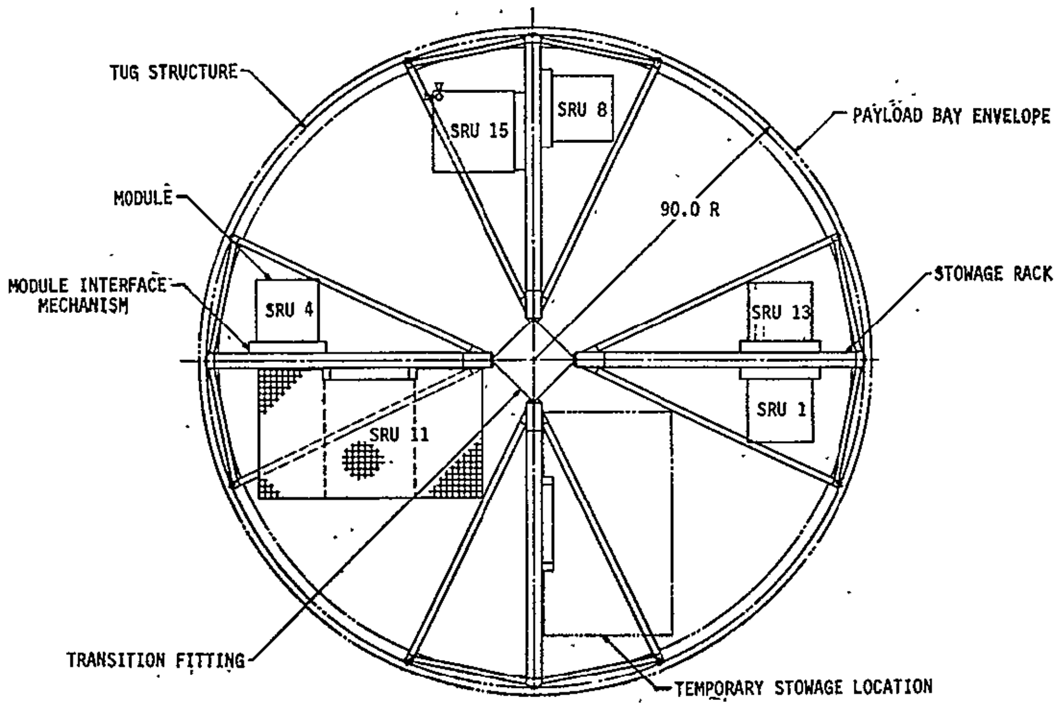

Figure VI-26 Stowage Rack - SEOS Maintenance Mission

SRU	MODULE TYPE	WEIGHT LBS.
2	Electronics and Gas Storage	150
5	Electrical Power Equipment	201
7	TDRSS Gimbaled Antenna and Electronics	79
11	Reaction Wheel and Electronics	111
15	Focal Plane Crystal Spectrometer	274
17	Imaging Proportional Counters	444
21	Aspect Sensors	87
	TOTAL	1346

The CLO modules selected are generally larger and heavier than either the DSCS II or SEOS modules and one extra module has been selected. The parts factor is 0.062 which is less than the 0.09 value used for HE-11-A in the first IOSS but quite comparable with the 0.06 used for the similar HE-01-A. As shown in Figure VI-27 four modules exceed the 40-in. dimension limit. SRUs 15 and 17 can be fitted in because the large dimensions do not occur in the interface mechanism direction. SRU 2 is slightly long in the direction of the interface mechanism, but it could be mounted so that it projects through the truss work into the open space behind the stowage rack. The interface mechanism for SRU 2 probably should be limtied to the normal 40-in.

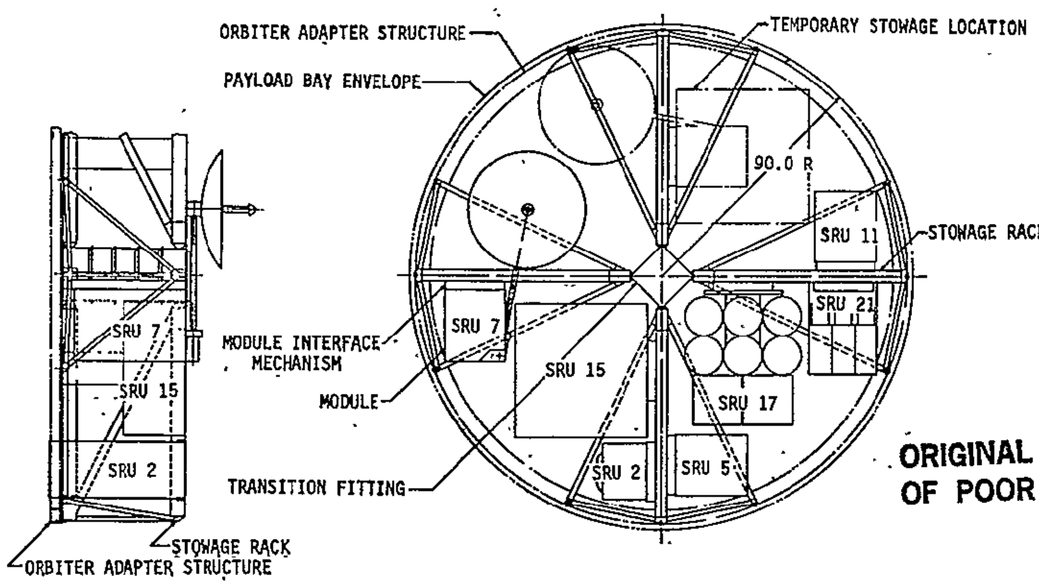

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Figure VI-27 Stowage Rack - CLO Maintenance Mission

The TDRSS antenna and its module, SRU 7, is 65-in. long and thus will not pass through the 60-in. spacecraft to stowage rack separation distance. The approach is to recognize that SRU 7 (and SRU 8) is mounted on the outside of the CLO with no exterior structure. The concept is to use a shorter interface mechanism length, say 26 in. Then the module can be moved axially until the interface mechanism is disengaged. There is adequate separation distance for this. Then the SRU would be moved out radially to clear the spacecraft where it can be flipped end for end. It could then be maneuvered radially within the separation distance and around the stowage rack truss work to the temporary stowage location. These kinds of approaches make it possible to handle outsize modules.

Stowage of replacement modules for the CLO is a much more demanding task than were the DSCS II or SEOS modules. However, it is possible to handle any of the CLO modules and to stow a reasonable complement of them--30 percent or the 24 SRUs or 35 percent of the total SRU weight.

E. DOCKING SYSTEM CONSIDERATIONS

The effect of the docking system on servicer system design was reviewed and the six aspects of Table VI-16 were identified as possible impacts. If on-orbit servicing becomes a significant mission for the Space Tug, or its equivalent, then the needs of servicing should be strongly considered. Each of the five modular forms works best with a central docking system and some cannot work with a peripheral system. However, it is possible to use outboard stiffening struts (3 or 4) and then for the servicer to reach between them.

Table VI-16 Docking System Effect on Servicer System Design

- SYSTEM TYPE
    - Central, small systems are preferred
    - Outboard stiffening struts can be accommodated
- RELATIVE ENERGY ABSORPTION
    - Space is available for shock absorbers in three configurations
    - Adaptation required for the two radial-only configurations
- MULTIPLE OPERATION MISSIONS AT HIGH EARTH ORBIT
    - Deploy before service is compatible
    - Multiple servicer is compatible
    - Retrieval not compatible with servicer return
- ACCURACY
    - Can be made acceptable
    - Errors can be measured
- PROBE STIFFNESS
    - Adequate for servicing small spacecraft
    - May require additional support for large spacecraft
- SENSORS
    - Few, small sensors required
    - Docking TV camera can aid servicing

A docking system mechanism will be provided to absorb the relative energy between the servicer and satellite during contact. Three of the servicer configurations provide at least 24 inches of space for this function. However, the radial-only configurations (2) have reduced the separation distance to a minimum. For these two configurations, either the separation distance can be increased or the docking cone can be recessed into the spacecraft.

While multiple operation missions (deploy, retrieve, and service) can be readily accomplished at the Orbiter, some combinations do not appear possible for high earth orbit missions.

Preliminary calculations indicate that a nine inch diameter docking probe will provide adequate stiffness for large spacecraft docked to the Orbiter. Should additional analysis later prove otherwise, the large spacecraft can be stabilized by using several outboard stiffening struts from the stowage rack to the spacecraft. Consideration can also be given to use of the Shuttle remote manipulator system to support the spacecraft laterally. The two-manipulator Orbiter configurations may be more appropriate for this application.

The serviceable spacecraft designs nominally avoided interference between appendages and docking by locating the solar arrays and antenna booms perpendicular to the docking direction. An alternate TRW design of the DSP satellite used a single solar array located on the opposite side of the vehicle from the docking cone. This method has the disadvantage of asymmetric solar wind loading forces on the spacecraft. Where arrays are located normal to the docking direction, the use of longer booms or designing the array with cutouts may be used to avoid violation of the docking excursion envelope.

Slightly more than half of the spacecraft designs of Chapter II did not provide for retracting appendages. The proportion of nonretracting designs was higher for the low earth orbit spacecraft. This is the opposite of what might be expected. Extended appendages could be a hazard to the Orbiter. The possible interferences are less for Tug based servicing. However, the main rationale for an appendage retraction capability is to be able to return the spacecraft to earth. This capability can also be obtained by incorporating a method of severing the appendages in an emergency.

The Space Telescope design includes a manual override for retraction of appendages by a suited crewman under contingency conditions. This capability is readily adaptable to operation using the servicer end effector in a manner similar to that discussed for the CLO thermal blankets in Chapter IV.

VII. SERVICER CONTROL SYSTEM

This chapter will present the important considerations that were addressed in evolving the servicer control system design and will describe the proposed design.

The control system is a vital ingredient in satellite servicing. Its design is influenced directly by the servicer arm configuration and with good design it can be made to exploit and enhance the mechanical design features in order to achieve simplicity, yet maintain all desired capabilities. The control system is also important in that it provides the interface between the servicer and the operator. Consequently, human factors aspects under operational conditions must be considered as well.

The servicer system configuration for which the controls in this chapter are designed has been selected previously in Chapters III and V. Detail mechanical design of the arm is depicted in Chapter VI. The culmination of all the design activities is the simulation demonstration described in the next chapter, VIII.

In the discussion that follows, the general approach for the design process will be presented first. The key requirements impacting controls design are discussed next. The conceptual design derived in earlier studies will be introduced and expanded upon. Two major servicing characteristics that influence controls design directly--the trajectory sequence and the reference coordinate systems--are defined in detail. Finally, the controls related hardware selections that evolved from the design process are described. The chapter is concluded with a description of how each of the three proposed control modes would be utilized in an operational scenario.

Before proceeding into any details, it is important to emphasize some unique features of the servicing task that have influenced all aspects of the servicer design but have particularly enhanced the control system. The first such feature relates to the fact that all the elements of the servicing task are completely and accurately defined prior to the flight. The stowage rack physical dimensions and envelope are very accurately known; the spacecraft has, of course, been dimensioned in detail prior to flight. The vehicles are aligned within $0.1°$ (1σ) in all axes by the docking system on orbit. Consequently, all the module locations and hazards are known ahead of time. This permits complete trajectory and sequence definition prior to flight. This is an obvious advantage to the control system which must implement the necessary steps to complete the exchange. Simple, accurate, automated sequences are possible (remove, flip, relocate and insert). Also, a wide range of ground-computer aided visual simulations can be devised when explicit detail of the physical elements is known beforehand.

Another unique feature is the distinct cylindrical geometry of the stowage rack as well as many of the spacecraft. The center of the servicer arm is conveniently mounted on the axis of the cylinder. The radially mounted modules extract ideally along the radius direction of the cylinder while the axial mounted modules extract along the other coordinate--the axial direction--of the cylindrical elements. The control system has been configured to exploit this geometry, permitting the trajectory to be broken into simple sequences that minimize the need to drive more than one joint at a time.

A. APPROACH

The approach that was followed in this study's servicer controls design may be better understood if the overall control system development plan is examined first. In Chapter V, the overall servicer development plan was introduced and discussed. From that plan, the control system development plan has been extrapolated. Figure VII-1 shows the important activities and milestones that are envisioned to ultimately bring the servicer control system to a state of readiness for Phases C and D. The study that is documented in this report shows up on the figure as the IOSS Follow-on. The time span is from February 1976 to March 1977. This period represents an important part of the effort leading toward control system development. Note that at the beginning of this phase a controls trade study had already been completed and a control system concept proposed. This shows up in the "output teardrop" summarizing the controls related conclusions from the first IOSS.

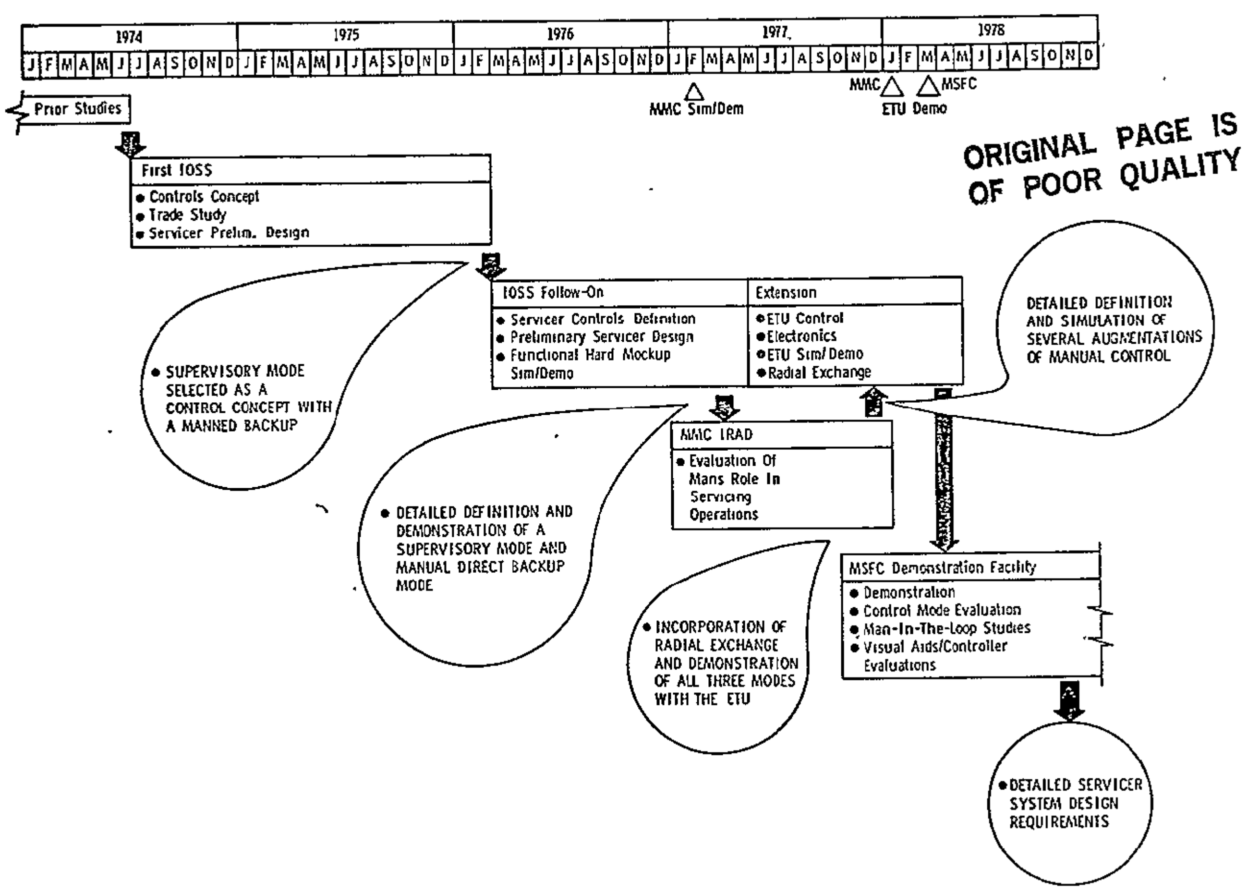

Figure VII-1 Control System Development Plan

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The servicer control system design had a first evaluation and verification in the functional hard mockup simulation demonstration of the IOSS follow-on. This is described in the next chapter, VIII. A still further step in development will be taken during the contract extension shown on the plan. Breadboard electronics will be designed and fabricated for controlling the servicer engineering test unit (ETU). The output of that phase will be a demonstration of two modes of control of the 6 DOF servicer arm and electronics. A third mode of control, manual augmented, will be evaluated during this same extension period as part of an IRAD that is investigating man's role and capabilities in remote control of on-orbit maintenance activities. The result, then, at the end of the contract extension is delivery to MSFC of the ETU and a fully-checked out control system in all three proposed modes of control.

The next phase of development, shown for 1978 and beyond, utilizes this tool in the MSFC demonstration facility for a number of development test activities, many of them controls related as seen by the list in that block. The eventual outcome of this phase will undoubtedly be a number of changes and improvements together with an increase in confidence in the capability of the evolved design to successfully control an on-orbit servicing system.

The remaining discussion will be concentrated on how the design evolved during the IOSS follow-on, what that design is and how it should be used. The specific approach that was followed to arrive at this is illustrated in Figure VII-2. This figure depicts the various decision points in the control system design task and their interrelationships. The order of their completion is generally from the top down on the figure. Essential prerequisites are: (1) a compilation of all requirements from Chapter II that affect the controls, and (2) selection of a candidate arm configuration; see Chapter III. With these as a base, each of the areas on the chart will be discussed in more detail in this chapter. Two relatively independent and parallel sequences of tasks should be noted on the figure. One involves establishing optimum trajectory sequences. This, in turn, permits development of timelines and the accompanying acceleration and velocity of the joints. These will be discussed below under D. The other path depicts definition of the selected control modes--manual vs automated, or some combination in between--followed by selection of the coordinate reference frame(s) in which the arm and the module will be driven.

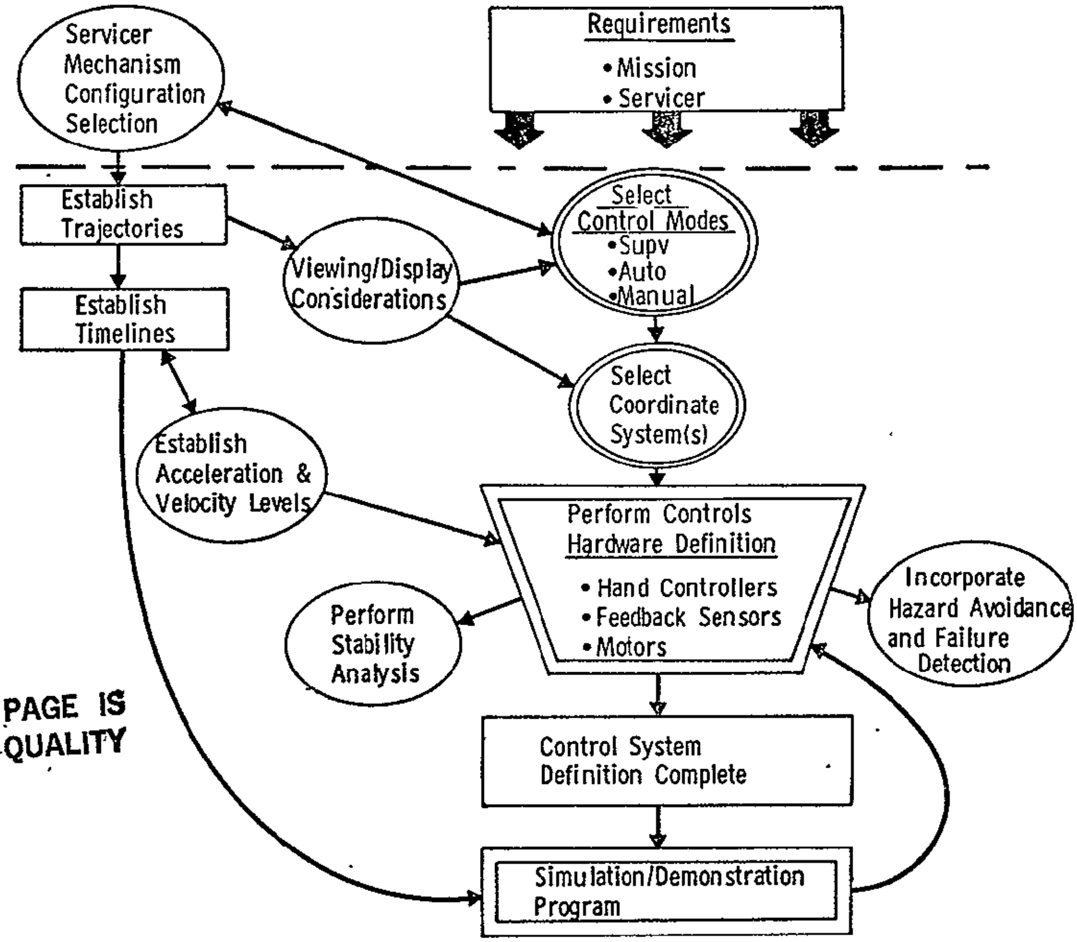

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Figure VII-2 Control System Definition Approach

These are discussed in sections C and E, respectively. The coordinate reference frame is strongly influenced by visual requirements.

The focal point for all these activities is the hardware definition activity. The motors, feedback sensors, controllers, displays, and electronic design take shape at this point and become mated in an integral system configuration. This will be discussed in section F.

With the hardware defined, the next level of activity includes a stability analysis, definition of hazard avoidance techniques and of failure detection/correction methods, and redundancy requirements.

The final step in the approach is the simulation and demonstration activities. These are essential in verifying the selected designs and determining where design changes or improvements are required. They will be discussed in Chapter VIII.

B. CONTROL SUBSYSTEM REQUIREMENTS

The overall servicing system requirements were introduced in Chapter II. Not all requirements are pertinent to the controls subsystem. Those that are, plus any others not previously defined, are documented here.

1. Mission Derived Requirements

The key mission related requirements are shown. These dictate the replaceable module sizes, weights, and locations and define the potential carrier vehicles for the servicer--Orbiter, geosynchronous carrier vehicle, etc. Automated servicing must be a consideration for potential military missions. It is possibly the only effective means of control at high orbits where data and communications are limited.

- One module handled at a time
- Module size - 40-in. cube max
- Module weight - 700 lbs max
- Axial and radially mounted modules
- Up to 25 modules/flight
- Varying sizes and weights on a flight
- More than one spacecraft/flight
- Maximum orbit - geosynchronous
- Compatible with Shuttle bay mounting
- Provide for manned monitoring of operations

2. Servicer Related Requirements

This next level of requirements becomes defined as the servicer system design begins to evolve. Those relevant to controls are:

- Stow and unstow servicer arm
- Provide hazard avoidance
- Provide failure detection consistent with safety/ redundancy
- Remote control compatible with TV transmission rate as slow as three frames/minute
- Arm accuracy - ±1.3-in. radial, ±0.75-in. axial

Servicer accuracy must be established to design the module cavity envelopes and attachment mechanisms. The control system plays a key role in achieving safe operation by providing hazard avoidance and failure detection functions. Servicer interface definition finds constraints arising such as data transmission rate limits. These can drive the form of the control system and its capabilities. For example, the Tug carrier vehicle at geosynchronous orbit can transmit no more than 50 kbps of data. This limits the refresh rate of typical TV pictures being transmitted on that line to three per minute. This is a dominant factor in effective manual control of the servicer from the ground. The controls must also provide for arm stowage. Certain spacecraft modules may require unique and accurate alignments with respect to the spacecraft. The control system will have to provide the bulk of this requirement.

3. Controls Related Requirements

In this category of requirements are those that, though they may be derived from higher level requirements, are in a form specifically directed at the control system design characteristics. There are also requirements in this category that are independent of the mission or servicer system but are just good engineering design goals for any control system. In this category are the stability and phase margins, those shown being typical for most control systems throughout the industry. The coordinate transformations are an integral element of the control system and easily implemented within the control system hardware.

- Provide manual backup control modes
- Provide all coordinate transformations
- Provide compatibility of controls and visual displays
- Control loop gain margin, >6 dB
- Control loop phase margin, >30°

4. Effect of Servicer Mechanical Design on Controls Requirements

The preferred servicer arm mechanical configuration presented in Chapter VI has some unique characteristics that have a definite effect on the controls design. The key mechanical characteristics are:

- Six independent controllable joints
  - four backdriveable joints
  - two spring preloaded joints

- Four bar parallelogram linkage for longitudinal drive
- Module locations and trajectories correlate to cylindrical coordinates
- Both radial and axial removals with common configuration

To elaborate on some of the above; backdriveable joints permit arm motion to be commanded from a single, or at most, two joints, while other axes "float free" during a module retraction and insertion. This presents a good potential for simplifying the controls.

The four bar linkage, while simpler than a translational drive for + and -X motion, does require some coordinated joint rotation to accomplish a pure translational motion at a constant radius. This can be done in the controls mechanization. During module insertions and retraction, this can be maintained through backdriving of those joints as well.

The backdriving feature in four of the degrees of freedom and the spring preloaded strain relief in the remaining two permit the arm and its segments to be relatively stiff, which simplifies stability and allows good response and accuracy of controls, yet does not impose any severe strains on any elements of the arm.

The mechanical configuration of the arm can be the same for either the radial or axial exchange tasks; however, the relationship between hand controller axes and any visual system requires a change in the electronic mixing of the command signals from one configuration to another.

C. CONTROL MODE SELECTION/DEFINITION

As pointed out earlier, the previous servicer study phase had done a trade study and selected an overall concept for control. That selection should be reviewed before discussing specific modes. Considered in that selection were such factors as autonomy, the form of man's participation where required, and the carrier vehicle with its representative mission profiles. The concept resulting from that trade featured a supervisory type of control for the servicing task, particularly during the early development years. This supervisory concept encompasses both automated control and manual control, thereby carrying along the development of all the ingredients of any number of other concepts should they be found to be more desirable at a later date.

Supervisory control is characterized as a semi-automatic control mode that accomplishes each of the segments of a module exchange trajectory automatically while man monitors that automatic activity for success and provides a command to proceed to the next step only after assured of success in the last. In the event of a failure or undesirable outcome, the man selects the remotely manned mode and, via TV, accomplishes the remaining operations manually. It can be seen the only operations lacking to provide a completely autonomous system are some automatic means of monitoring the sequences for safety and success and a subsequent automatic "go" when a sequence is successfully completed. This is the most difficult operation to mechanize and yet one of the simplest for man to implement. In light of a strong desire to include man for observation and possible backup control, anyway, the supervisory mode was a highly desirable selection. It maximizes automation technology yet minimizes new state-of-the-art or risky developments. It also includes development of a completely manual control capability for backup operation. Some additional rationale for the selection of this configuration are:

- Well-defined and accurate trajectories are possible with the normally preprogrammed automatic module manipulation;
- The undesirably low TV refresh rates (as few as three/minute) impact only the backup mode, selected after a failure when much slower motion and sequencing can be tolerated;

- Reliable status monitoring, hazard monitoring, and failure detection, which is very difficult to implement for completely autonomous operation, can be performed simply and effectively by man in a supervisory role;
- The backup mode has a high degree of functional redundancy; i.e., utilizes as little common hardware as possible with the supervisory mode.

The selected concept for the servicer embodies many of the features of both an autonomous system and a manually controlled system. From the standpoint of flexibility, both in configuration modifications during development and during mission operations, this is highly desirable. It does, however, require a considerable number of capabilities of the control system. Some of these are listed below. How each of these is provided for and to what extent will be presented in more detail in the subsequent pages.

- A method of automating the trajectory sequence is required;
- On-orbit TV is required;
- Both rate (manual modes) and position (automatic) control of joints are required;
- A ground visual display that is compatible with hand controllers is required;
- Comprehensive status data must be transmitted to the ground;
- Augmentation of hand controller commands for coordination with visual scene (manual backup) is required;
- Computer aided visual scene generation is desirable (manual backup).

To implement the proposed concept, three specific control modes quickly became obvious.

- Supervisory
- Manual Direct
- Manual Augmented

A summary of the implementation of these and accompanying rationale is provided in Table VII-1.

Table VII-1 Control Modes Summary

MODE	IMPLEMENTATION	RATIONALE
Supervisory (Prime)	• Automated segments • TV Monitor • Manual "Go's" - step by step • One joint at a time driven	• Man performs evaluation and status monitoring • Well defined, safe trajectory • Module location required pre-flight
Manual Direct (Backup)	• Joint driven directly from panel	• Minimum hardware/software required • Provided for failure case
Manual Augmented (Alternate)	• Manual control from hand controllers • Arm motion coordinated with visual displays	• Capable of acquiring any target of opportunity within reach • Representative of "conventional" teleoperator control approaches

The supervisory mode of control is the normal mode of operation. All servicer arm motions and trajectories are determined before flight and stored on board. The computer implementing this mode will sequence from one segment of the trajectory to the next, but only when the man has evaluated the state and provided a "go".

The manual direct mode is provided as a totally unsophisticated means of backup control. It sends rate commands from panel switches directly to the joints themselves. Commands are one joint at a time. Motion is with respect to each joint's mounting base rather than with respect to the TV display coordinate system, making the control task somewhat awkward for some configurations. The primary feedback displays are error meters which are one to one with the control inputs. Manual direct mode uses are: 1) as a possible normal control mode for certain simple arm configurations that lend themselves to direct joint control; 2) as a backup in the event of a failure in the ground computations or downlink used in the augmented manual mode; or 3) in the event a joint failure has occurred that can be worked around but the normal coordinate transformations either onboard or on the ground are not valid.

The manually augmented mode has man doing most of the arm control as in the direct mode above only using hand controllers instead of panel switches. Also the computer is still in the loop to facilitate the direction of motion of the arm and provide optimization of its motion with respect to the displays provided. Its most useful role is to perform unscheduled motions to previously unidentified targets of opportunity. In most of the discussion that follows regarding the enhancement techniques of manual control, it is this mode that is being referred to, not the manual direct.

It should be stressed that the selection of these three modes is not final. They have been deliberately chosen to span the spectrum of sophistication of the types of control envisioned for the servicer arm. From the evaluations conducted with these, during the remaining development phase, quantitative data will be available to make a sound selection of the ultimate flight system control modes.

More detail on how each of these three modes is accomplished from the hardware and software standpoint will be provided in sections H, I and J of this chapter along with the operational aspects of using each of these three modes.

A more detailed yet significant concern regarding control mode selection is the type of control command to be used to drive the servicer joints; i.e., should it be rate, acceleration or position servo-driven? Table VII-2 shows some of the reasonable alternatives for the servicer system and the significant characteristics of each.

Table VII-2 Types of Control

TYPE	BASIC CHARACTERISTICS	REMARKS
Acceleration	• Motor current (torque) proportional to command volts • Motor stops when command is removed	• Harder to handle than rate • Rate feedback not required • Like rate control for "fine tuning" (minimum impulses)
Acceleration plus rate	• Like acceleration command, but motor rate maintained until negative acceleration commanded	• Generally "pulse" type of commands • Slower response • Rate feedback not essential • Rate "fine tuning" is possible • More difficult to stop arm
Rate	• Arm rate proportional to commanded volts • Arm position held when command is removed	• Safe control - no command, no motion • Good manual response possible
Position	• Arm position follows commanded position (volts) • Rate is constant or a function of position error magnitude	• Position feedback required • Rate feedback desirable • No arm drift

The easiest of the above to implement is acceleration control. In this case the arm is driven directly with no real form of feedback or control. It is a feasible approach but from a manual operator's viewpoint there are other methods that provide much better controllability. The acceleration-plus-rate and the rate mode shown are two typical types. There are certainly other combinations that may be feasible as well. The first three types on the chart are usually considered in reference to manual control. The requirements of a man in the loop generally drive the control type and can often be very sensitive, hence, the variety of types of control that have been conceived for such an application.

The last type on the chart--position control--is thought of more with respect to an automated application. In this case the response and sensitivity are all of little consequence. The arm is to reach a given position and how it gets there is of no great concern. A position feedback sensor is, of course, required. Rate feedback is also desirable to insure some controlled rates during traverse. In any event, the options for position control are not that many and generally concern themselves with the secondary issues of just how position control is mechanized--what kind of position comparator, type of feedback device, relative positioning vs absolute, etc.

The conclusion is that position control is required for the semi-automated supervisory operations with joint rate controlled as well until the desired position is achieved. Control in either of the manual modes is recommended to be done using a rate type of control.

D. TRAJECTORIES AND SEQUENCE OF EVENTS

In developing a preferred module exchange trajectory the following criteria were used as a guide:

- Take maximum advantage of natural joint motion;
- Minimize need for driving several joints concurrently;
- Compatible with automated and manual operations;
- Maximize hazard clearances;
- Simplify hazard avoidance;
- Common trajectory regardless of module size;
- Capable of exchange within Orbiter bay confines.

The general theme of these criteria is simplicity in implementation and use as well as maximized margins for safe operation. Since the automated sequences to accomplish the necessary trajectories are implemented in onboard software for the supervisory mode, the simpler these sequences are and the more common each can be with other configuration's sequences the easier the software development and verification will be.

Straightforward coordinate transformations and driving a single joint at a time wherever possible all tend to simplify the implementation of the trajectory.

Maximizing the distance of the module from any hazard eases the accuracy in defining a given trajectory and also relaxes the TV monitoring response requirements. Servicer accuracy during the trajectory can impact hardware design requirements beyond that needed for meeting an accuracy at an end point only; e.g., sample rates, rate feedback, etc.

The one criteria that is counter to simplifying the controls task is the last--exchanges in the Orbiter bay confines. It has a negative impact on achieving simplicity in design yet it is a desirable feature and the capability will be provided. The discussion to follow will treat it as a separate problem, offering some specialized techniques in accomplishing the desired trajectory safely.

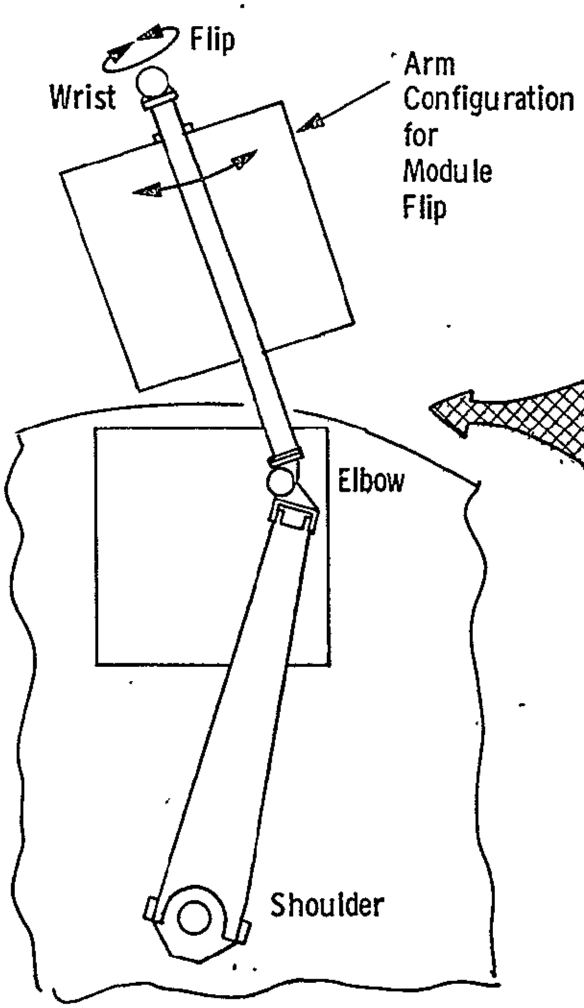
The following pages will first discuss the radially mounted module exchanges. That will be followed by a discussion of the trajectory for axial type of module exchanges. The one special case--axial exchange within the cargo bay confines--will follow. Other trajectories are certainly possible for a given exchange but are too numerous to discuss in detail. They are generally just slightly different combinations of the steps in the three basic trajectories.

The TV viewing considerations for these basic radial and axial trajectories in both the supervisory and manual control modes is also presented in this section.

The section closes with a typical sequence of events that is applicable to either an axial or radial exchange.

1. Module Trajectory Sequence - Radial

The steps to accomplish the initial attachment, withdrawl, flip and reinsertion of a module that is mounted on the circular periphery of the spacecraft and stowage rack are listed in a brief form in Figure VII-3.



1. Position Wrist to Module Attach Point
2. Retract in Radial Direction to Engage Module
3. Secure End Effector to Module and Disengage Module From Rack
4. Extend Arm in Radial Direction Until Fully Extended
5. Translate Longitudinally to Center Module Between Spacecraft & Stowage Rack
6. Rotate Module to Align with Upper Arm
7. Rotate Module 180^{0} about Upper Arm
8. Rotate Arm About Shoulder to Predefined Angle
9. Rotate Module at Wrist to Align with r Vector
10. Translate in Longitudinal Direction to Align Module with Guides
11. Retract Radially to Insert Module in Cavity
12. Secure Module To Rack And Disengage End Effector

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Figure VII-3 Module Trajectory Sequence - Radial

The position of the arm during the flip (steps 6 and 7) is depicted pictorially on the figure. This phase of the sequence warrants emphasis at this time. Note that with the arm nearly fully extended, and the module above and clear of the cylindrical envelope of the spacecraft and stowage rack, the proximity of hazards are minimized and the accomplishing of the flip is a simple, single-joint drive about the upper arm axes. Even for the largest module considered, the clearance between module and outer spacecraft perimeter is over four inches. Exact location of the arm in the longitudinal direction is not critical though if it is near center the longest possible extension, and thus greater clearance, is provided.

Note that for radial exchanges the retraction and insertion actions must be along the radius vector. The module axes can be maintained along this r vector during retraction by backdriving the elbow and wrist joints as the shoulder roll joint forces the module out along the guides. For insertion, however, it is desirable that all these joints be driven in a coordinated manner so the module will move along the radius vector regardless of whether it is in the guides or not. Any visual scenes will certainly be enhanced. Implementation of this coordinated motion is not difficult in the cylindrical geometry of the servicing task. It is assumed this capability is provided for primary operation and will be discussed in more detail in section E.

2. Module Trajectory Sequence - Axial

Figure VII-4 depicts the exchange of a module mounted on the face of the spacecraft and inserted into a cavity on the face of the stowage rack. This requires retraction and insertion parallel to the cylindrical axes of the two elements, therefore the term "axial exchange".

The figure again depicts the position of module and arm for the flip sequence (steps 6 and 7). The point to be made here is that the sequence is the same as for the radial exchange, thereby capitalizing on the advantages of hazard avoidance and single joint drive that were discussed on the previous figure.

Motion along the r vector is not necessarily required for axial exchanges; consequently, the coordinated shoulder and elbow joint drive can be eliminated. On the other hand, it may be desirable to add some coordination of joint drives

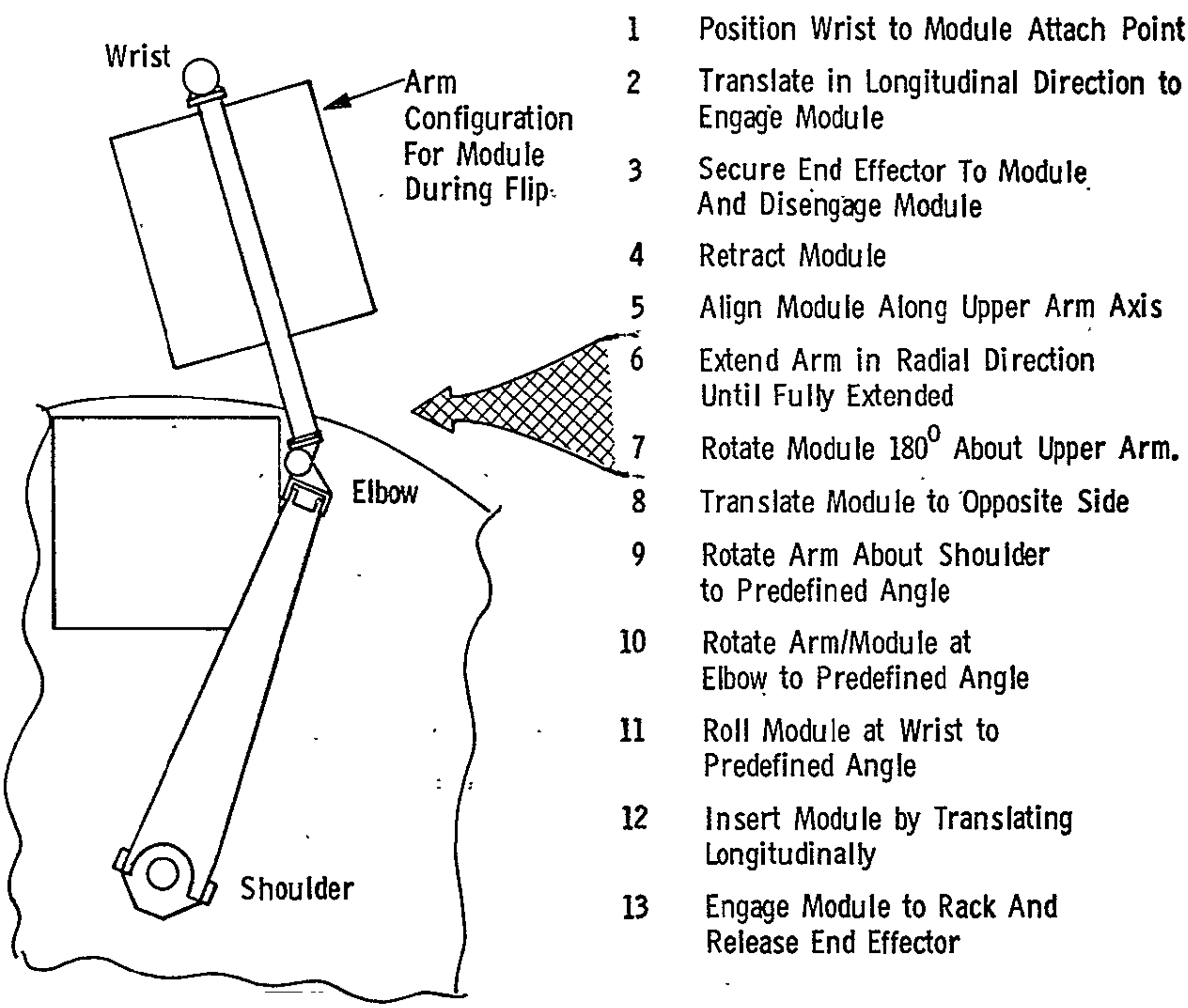


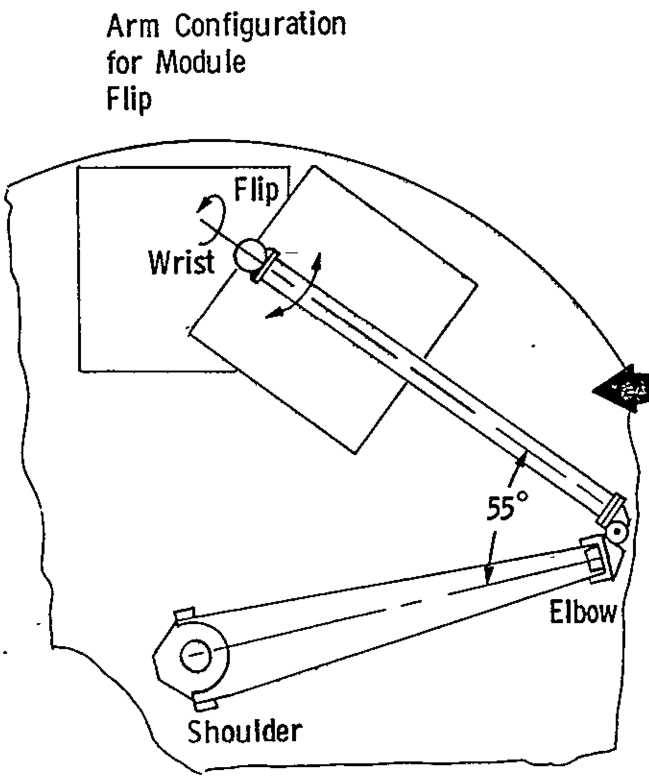
Figure VII-4 Module Trajectory Sequence - Axial

to account for the effect of the four bar linkage on the arm radius as translation motion is conducted. As before, this can be done through backdriving but coordinated joint drive is more optimum and recommended for primary control.

The sequence on this figure assumes that there are no restrictions on projecting the arm and module out beyond the spacecraft periphery. This is assumed to be typical and the normal operational scenario; however, some future missions may require deviation from this. One case is use of the servicer in the Shuttle Orbiter Bay with all operations restricted to within the periphery of the spacecraft/stowage rack cylindrical envelope. That unique application and proposed solution is discussed next. The solution is typical of any application restricting exchanges within the spacecraft/stowage rack periphery.

3. Module Trajectory Sequence - Cargo Bay Constrained (Axial)

For satellite servicing within the spacecraft/stowage rack it must be assumed that modules will be located on the faces of the spacecraft and stowage rack and exchanges will be conducted in the axial direction. Extension of the arm or a module beyond the spacecraft/stowage rack periphery will not be permitted. All normal axial sequences can be completed without violating this constraint with one exception--the module flip. The sequence of operations is shown in Figure VII-5. They are essentially those of Figure VII-4, except for steps 5, 6 and 7. The module flip in the Orbiter Bay is proposed with the module positioned as depicted in Figure VII-5. Joint angles must be more accurately defined to avoid module protrusion out of the envelope, or interference with the lower arm during the flip. For commonality it is recommended the module axis always be aligned with the upper arm axis. The V joint is then positioned to ≈55° for a 40" square module. This angle may vary from one module geometry to another but can be easily calculated before flight.



1 Position Wrist To Module Attach Point
2 Translate +x To Engage Module
3 Secure End Effector And Disengage Module From Rack
4 Retract Module
5 Drive Elbow To ≈55°
6 Drive Wrist Roll To Align Module With Upper Arm
7 Flip Module About Upper Arm While Translating Wrist In x Direction Per Predefined Profile
8 Rotate Arm About Shoulder To Predefined Angle
9 Roll Module At Wrist To Desired Angle
10 Drive Elbow To Desired Angle
11 Insert Module By Translating Longitudinally
12 Secure Module To Structure And Release End Effector

Figure VII-5 Module Trajectory Sequence - Cargo Bay Constrained (Axial)

There may be some tailoring of the sequences shown, depending on where the module has been extracted from. If the V joint angle is larger than 55° (edge mounted attach point) the V joint will have to be driven to ≈ 55° before the module is aligned with the upper arm in order to avoid penetrating the peripheral constraint. Basically this is reversing steps 5 and 6 or 9 and 10. Again, this is easily determined and accounted for in preflight programming.

The flip sequence is a far more delicate operation than for the radial configuration, particularly when the module size approaches the maximum of 40 inches. A multiple joint rotation profile is required. This sequence is illustrated in more detail in Figure VII-6.

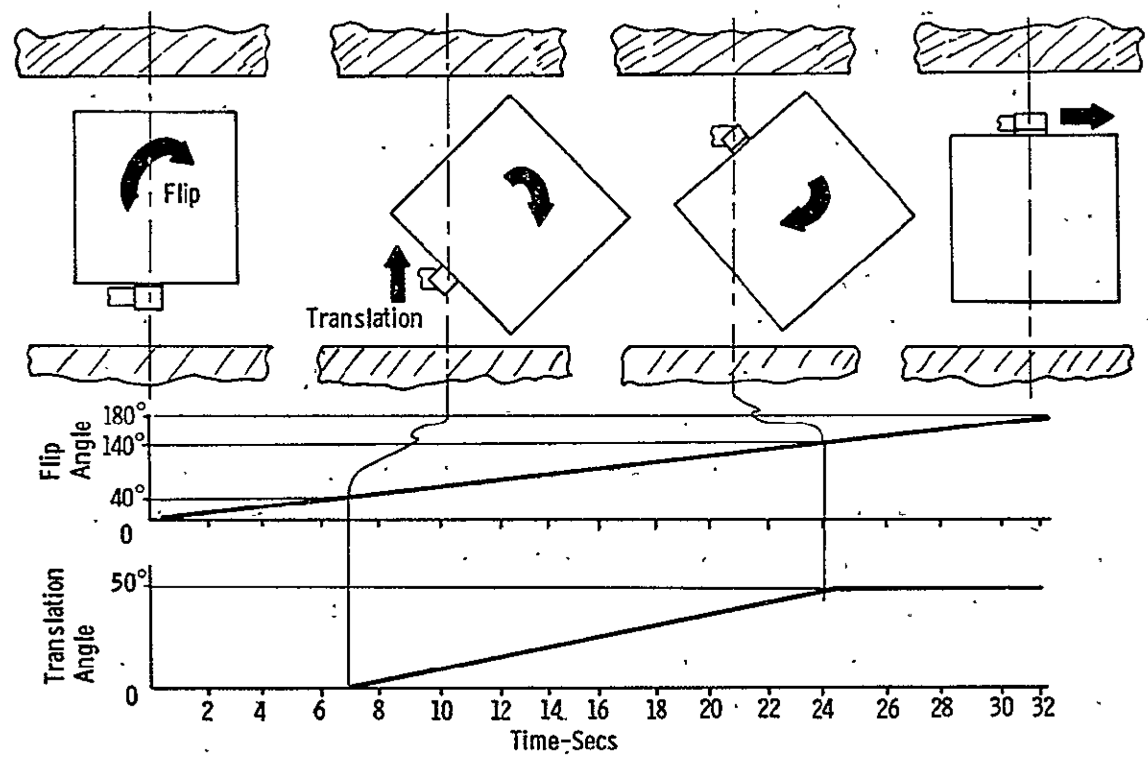


Figure VII-6 Module Flip Trajectory - Cargo Bay Constrained (Axial)

The figure shows a 40 inch square module (maximum) during the 180° flip sequence that takes place between the spacecraft and stowage rack in the Orbiter Cargo Bay. The module axis has been aligned with the upper arm axes. The view is looking down at the end of the upper arm as shown in Figure VII-5. The remaining arm segments are not shown for simplification. The module positions are shown at the beginning or end of key phases within the flip segment shown. These phases are derived from the time profile of the flip translation joint drive motors shown at the bottom of the figure.

The relationship shown between the two joints is necessary to accomplish rotation of the module about the upper arm without impacting any surfaces. The complicating factor is that the axis of flip rotation must be translated from the one surface to the other at the same time; however, it must be delayed at the beginning while the rotation is started and finished before the rotation is completed to avoid impact.

Note that when the preferred near-radial arm configuration is used for axial operations with full size modules within the cargo bay constraints, the short 12 inch extension needed for reaching the radial attach point must be removed to accomplish the flip within the required envelope. This is a simple task in the proposed modular design of the preferred servicer arm.

4. Viewing Considerations

It is apparent from the previous discussion on trajectories that the location of a TV camera(s) for effectively monitoring the module during the variety of movements required is a complex decision. Whether the motion is automated or manually controlled, some idea of successful hazard avoidance and safe operation should be provided to the ground, preferably by TV. At a minimum the TV should permit sighting visual cues that verify successful completion of each segment of the exchange trajectory. A second major requirement on TV evolves from the manual augmented control mode. This mode is designed around the presence of a TV image for control purposes. The selection and implementation of this manual control mode and its corresponding coordinate system--discussed later in this chapter--is influenced by the location of the camera and the visual scenes it is expected to transmit.

A somewhat generalized summary of the type of views desired during various phases of an exchange for both the supervisory mode and manual control modes are shown in Table VII-3. The conclusions to be drawn are that the supervisory mode viewing requirements are not excessive. If a TV picture at trajectory segment completion to verify success can be accepted as adequate, a single TV camera with state-of-the-art field-of-view (FOV) characteristics is satisfactory. End effector mounting appears feasible. Viewing for manual augmented control represents the driving requirements. The axial exchange is the worst case, but only if a broad search of an entire spacecraft face must be conducted to locate the proper target. Two separate cameras are one solution--one on the end effector and one with wide FOV coverage from somewhere on the stowage rack. In the radial case the search routine can be restricted to a narrow strip around the periphery of the spacecraft possibly alleviating the need for the second camera. A similar single-camera solution may be feasible for the axial case if it is assumed the camera, with its widest FOV, is driven in circles of decreasing radii about the shoulder.

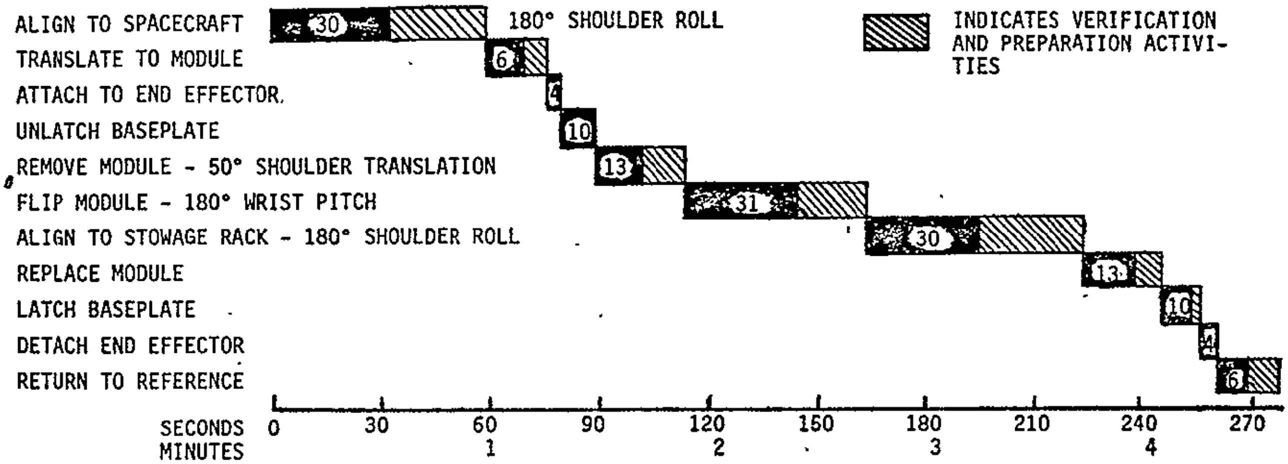
Table VII-3 Viewing Considerations

Configuration	Travel to and Align with Module	Engage Module and Retract	Orient for and Perform Flip	Orient to Insert Module	Insert Module
Supervisory Mode (Radial or Axial)	Viewing at sequence completion for verification	Narrow FOV; end effector mounted camera is adequate	Simple radial extension and unobstructed rotation for flip. No stringent viewing requirements	Viewing at sequence completion for verification. Wide FOV monitoring desired	Simple +X translation or radial retraction. Narrow FOV adequate for monitor
Manual Augmented (Axial)	Wide FOV Desirable for target acquisition; multiple circular scans is feasible alternate	DRIVING REQUIREMENTS		Wide FOV desirable for target acquisition and hazard avoidance; multiple circular scanning is feasible alternate	
Manual Augmented (Radial)	Radial scan with narrow FOV feasible for target acquisition				

Viewing requirements for the Orbiter constrained axial exchange are not considered here. Requirements will be greater, but Orbiter TV cameras will undoubtedly aid the problem.

5. Module Exchange Sequence of Events

Figure III-7 shows the timeline for a typical module exchange. The maximum arm motions are shaded with the worst case time required, in seconds, in the block. The remainder of the bar for each segment is the allocation for verification of proper completion plus any preparation for the next segment. Note that the segments are all reasonably uniform in length and a total exchange can be accomplished in less than 10 minutes.



MAXIMUM JOINT RATES AND ACCELERATIONS

	Shoulder Roll	Shoulder Translation	Elbow Roll	Wrist Pitch	Wrist Yaw	Wrist Roll
Rate, deg/s	6	4	8	6	12	12
Accel.* deg/s²	6	4	8	6	12	12

*Based on 1 second to achieve maximum rate

Figure VII-7 Module Exchange Sequence of Events

The joint's angular rates and accelerations upon which the timeline is based are given at the bottom of the chart.

Some of the criteria, and/or requirements, that influenced selection of the maximum rates and accelerations are:

- Replace failed module with a good one in < 10 minutes;
- Activity durations should be somewhat evenly distributed;
- Decelerate in < 10% of typical sequence duration;
- Distance to accelerate should be less than four inches
- Avoid excessive drive sizes/weights.

Some of these criteria are derived from projected mission applications and estimates of time allocations for the maximum on-orbit module exchanges. Others are in the category of engineering goals. They may hopefully permit cost effectiveness through commonality, e.g., drive sizes, and provide ease and safety in manual control operations.

A module exchange in 10 minutes permits six exchanges per hour or probably less than two hours at an orbital station for a single spacecraft servicing completion. This appears well within typical mission profiles of the Tug or the Orbiter. More than one spacecraft may be serviced on a single mission.

Note that none of the joints require excessively high rates. There is also good potential for commonality. The accelerations shown are not really critical nor is meeting them exactly of any great concern. The general objective was to not spend a large percentage of the travel time accelerating. The arbitrary one second of acceleration time is only 3% of most of the 180° rotations in the timeline which is considered more than adequate. Even for a short travel of 20° the acceleration time is less than 30% of the travel time. It should be pointed out that the above estimates assume a maximum rate is commanded. For automated operations, the rates commanded will be reduced proportionately as the remaining angular travel becomes smaller. This is for reasons of safety, accuracy for visual scene monitoring and to avoid overshoot at the target. The time to accelerate to a lower commanded rate will obviously be less than for the high rates.

Simulation/demonstration showed some of the above time spans are quite optimistic, particularly for development phase activities. For operational conditions, however, the times are still considered suitable.

E. COORDINATE SYSTEMS

This section is devoted to the selection of the coordinate systems for the servicer. Selection and definition of the coordinate system(s) is important to the controls because it will be the controls system task to implement motion of the servicer arm in the directions of whatever coordinates are selected. This can require complex coordinate transformation equations impacting electronics design and computer software. The coordinate system also guides the trajectory sequence and consequently is a factor in achieving simple, straightforward exchanges that also appear as such to the man controlling the servicing. Because of these ramifications the following discussion will cover considerably more than coordinate systems--touching on TV viewing, displays and hand control coordination and other man-in-the-loop considerations. How these semirelated topics all tie together and how they will be presented is shown below in Figure VII-8. The order in which the steps in the selection process were conducted and will be presented is reflected in the numbers on the blocks.

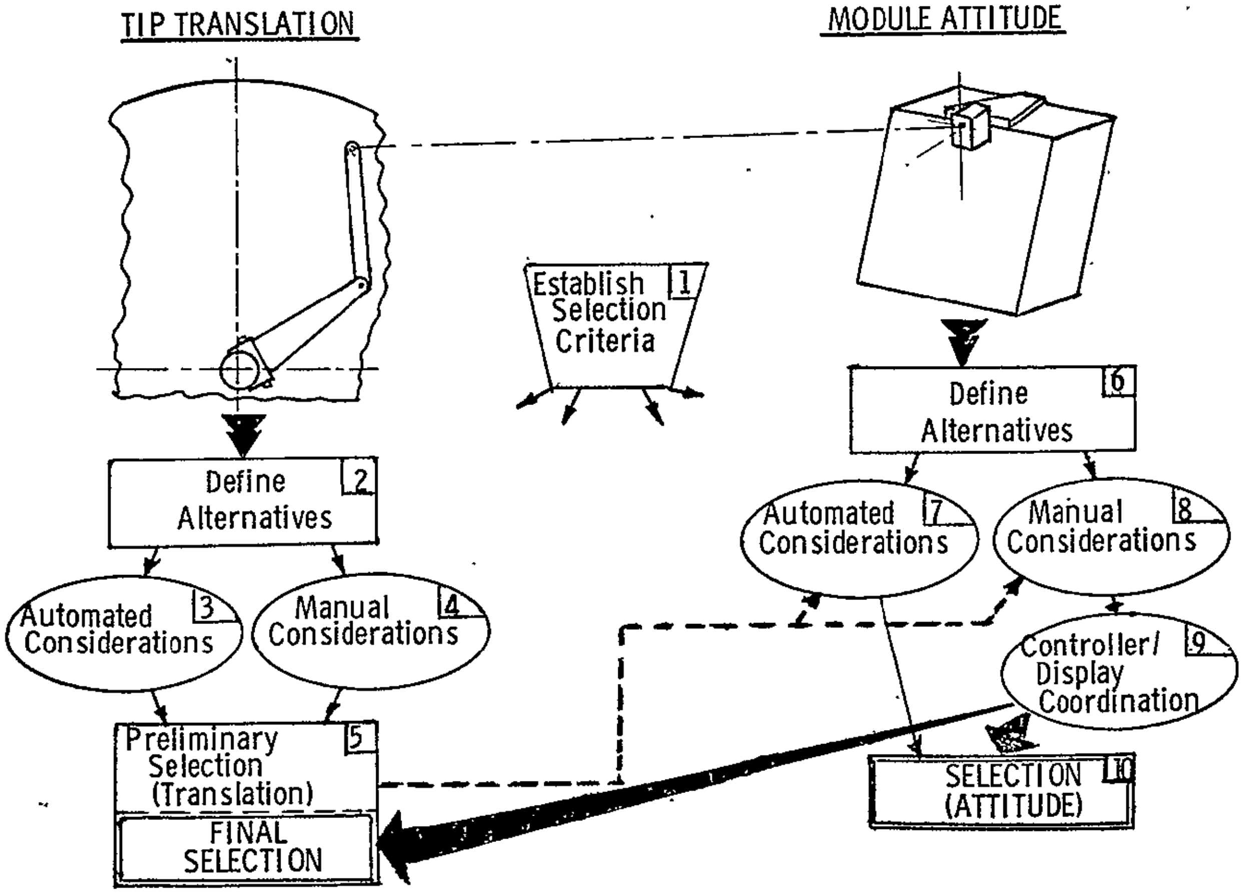


Figure VII-8 Coordinate System Selection Approach

Note that initial considerations of the coordinate systems were expedited by separating the tip translation function from the module attitude. The proposed typical trajectories will, in general, utilize these two functions independently. Obviously, there is an interrelationship between the two, but primarily in the mechanization of the equations of motion. Selection of a system for tip translation simplifies the module attitude evaluations. That information feedback is shown dotted between blocks 5 and 8 on the flow chart.

It will be shown that in order to accomplish the desired coordination between hand control commands and the visual displays, the tip translation coordinate system must be driven as a function of module attitude. The impact of that mechanization on the tip translation motion is shown as the final feedback path (heavy arrow) at the bottom of the chart, and is a necessary step before final selection of the tip translation coordinate system.

The criteria (block No. 1 on Figure VII-8) used to guide the necessary selections are:

- Compatible with automated operations;
- Compatible with direction of module motions;
- Compatible with module location geometry;
- Should provide off-axis remove/replace capability;
- Compatibility with manual operations;
- Compatible with hand controller/visual system coordinates;
- Ease of implementation;
- Ease of operator training.

In attempting to simplify the subsequent discussion, several points can be made here. First, the selection of a given coordinate system explicitly defines the direction in which the end or action point of the arm will move. The motion of the end effector "attitude" can be independent of the arm's end point motion or it can be slaved directly to it.

Most of the more desirable motions of a manipulating arm require coordinate systems that are not related directly to the individual joints that accomplish that motion. Instead, they require that multiple, coordinated joint motion be provided. The relationship of the joints being driven to the desired motion at the end of the arm is accomplished through a coordinate transformation which is merely a series of trigonometric relationships in equation form that are mechanized in electronics or computer software. The simpler the coordinated transformation, the simpler its mechanization. The servicer task is simplified by considerable use of simple, one-at-a-time, joint drives.

One of the more important factors that impact the type of coordinate system is the geometry of the physical elements. Another is how involved man is in performing the manipulation and, in particular, what his visual cues are while accomplishing the task. The latter is of particular significance since the more directly relatable the motion of the hand control is to the apparent motions of the arm as seen from the visual cue the more quickly can the operator be trained and the more effective he will be. Again, it should be pointed out that the servicer system cylindrical geometry permits a corresponding simpler control mechanization than most typical manipulator arm applications.

1. Tip Translational Motion

In examining first the tip translational motion (left-hand path on Figure VII-8), a number of alternative directions of motion are quickly apparent. These are illustrated in Figure VII-9. Other motions are undoubtedly feasible, such as spherical, but they do not seen even remotely applicable nor are they easily incorporated.

The simplest form of control is to drive each joint directly; i.e., independent of other joints or of any base system coordinates. For the servicer task, direct control is not entirely undesirable. The T-joint on the axis does rotate the arm around the axis at a constant radius. The tip motion caused by the V-joint is in a direction that is purely a function of the V-joint angle at the time. It is not very easily resolvable as a visual cue. Note that for an application where the tip direction of motion is of little concern and only the ultimate destination is, such as in an autonomous mode where all angles are precalculated, the direct system is a feasible and desirable alternative. Only one feature of the servicer arm tends to complicate the direct approach. That is the four-bar linkage for module insertion/retraction motion. It results in Y or Z motion, as well as $\pm X$, thus making either one of the following alternatives more attractive.

The cylindrical coordinate system (right side of Figure VII-9) requires a mathematical transformation to relate hand control commands in the servicer arm radial and angular direction to the T, U and V joint commands. The transformation is relatively

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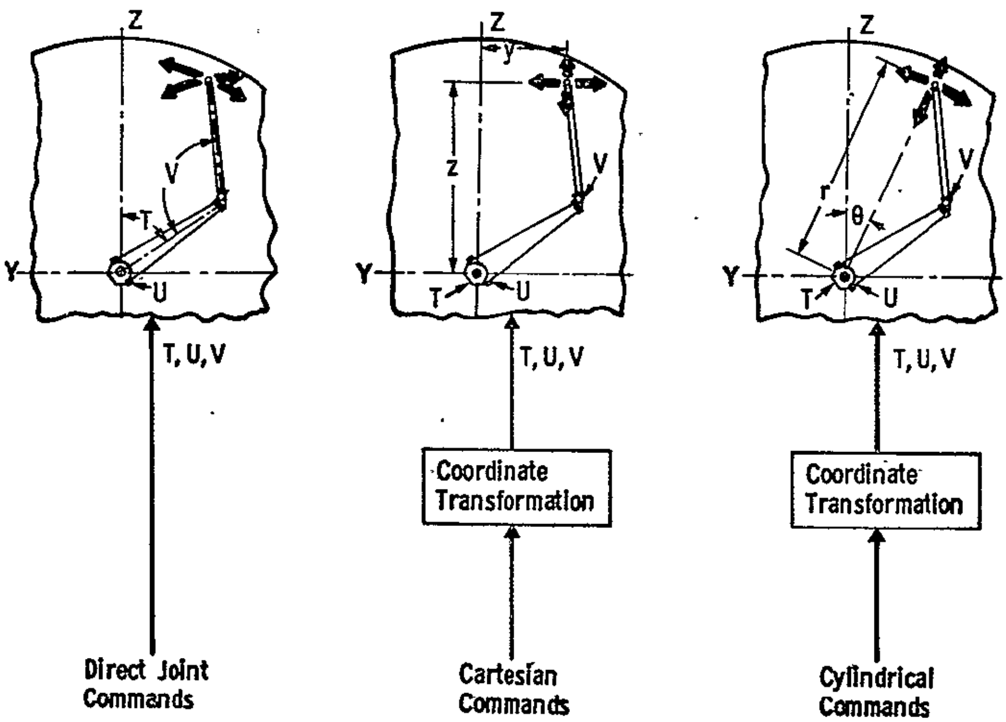

Figure VII-9 Coordinate System Alternatives for Tip Translation

straightforward because of the cylindrical geometry of the servicing systems physical elements. The arm tip motion is consistently as shown regardless of the location of the tip. The cylindrical system is quite satisfactory for axial operations and ideal for modules mounted radially on the circular periphery.

The cartesian coordinates (center of the figure) utilize a transformation that always moves the arm tip in orthogonal directions with respect to a specified orthogonal reference. From a visual display standpoint this is desirable, particularly for modules mounted orthogonally on the face of the spacecraft or stowage rack. It is also valuable for removal of orthogonally oriented modules on the periphery but not on the spacecraft axes. The coordinate transformation is consistently more complex. The system is not particularly suited to radial removal of modules mounted around a circular periphery, unless centered on an orthogonal axis.

It could be concluded at this point that cylindrical coordinates are the more desirable. There is, however, another aspect that should be considered before making any final conclusion. That relates to how the arm motion will actually be accomplished in the semiautomated supervisory mode and in the manual modes. There are unique characteristics in each mode that tend to influence the coordinate system selection.

a) Tip Motion Considerations - Automated Trajectory - Figure VII-10 below summarizes the key functions that must be accomplished when the arm or arm-plus-module is in a totally automated sequence. It represents the simplest scenario. All angles, rates and the time sequence of operations are calculated on the ground prior to flight. In the strictest sense, no man or visual aspects are necessary. The impact on the presence of these will be discussed later.

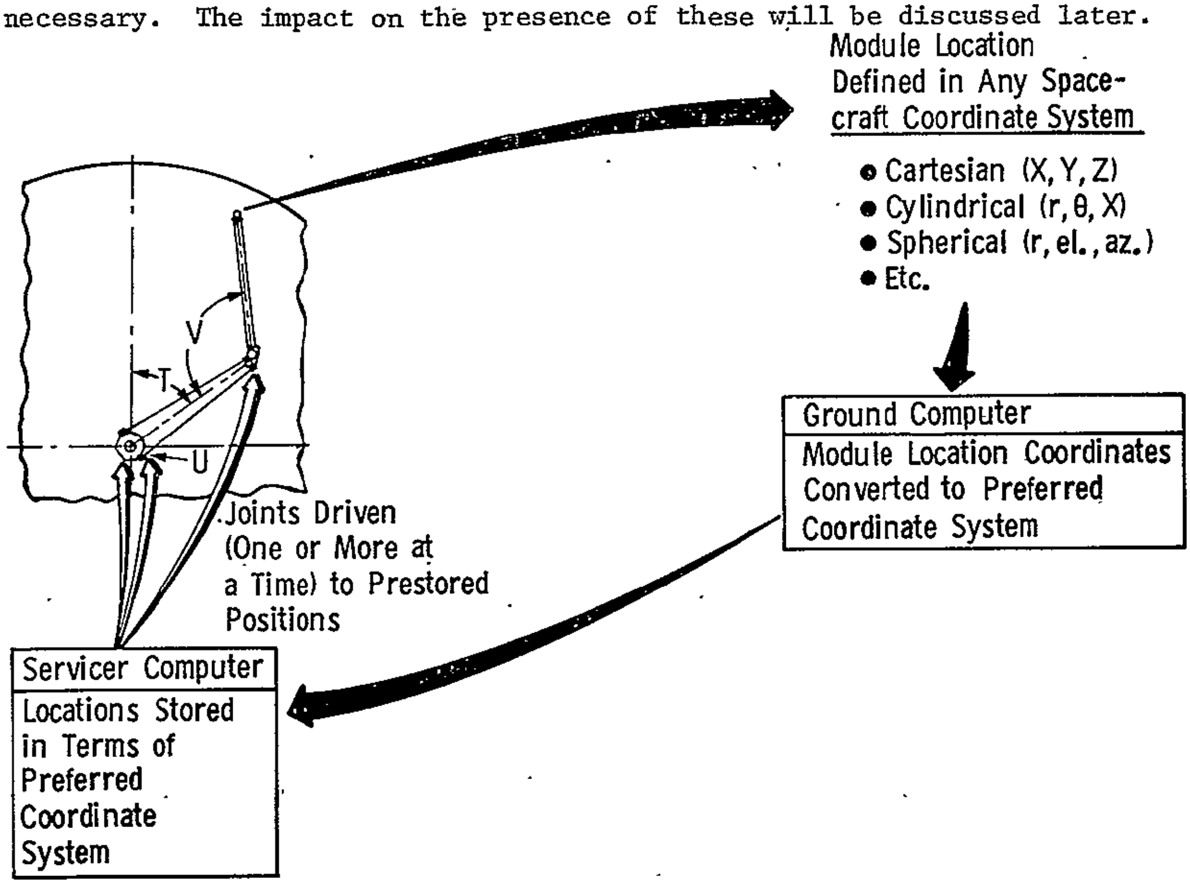

Figure VII-10 Tip Motion Coordinate System Considerations - Automated Trajectory

The module attach point can be defined in virtually any coordinate system that is convenient. A ground computer transformation will accurately derive the final joint angles to achieve that position. The onboard computer then drives the joints to these angles in whatever sequence and at whatever rate is preprogrammed into that computer. There is no concern about the direction of motion of the tip since no one sees it or needs to see it. Any potential hazards can be avoided by the preprogrammed sequence in which the joints are driven. Since the direction of the tip's motion is of no real concern, a coordinate transformation is not necessarily required for visual purposes. In fact, were it not for the arcing motion of the four-bar linkage for $\pm X$ which results in Y or Z motion as well, direct control would be a simple and very satisfactory control mode for automated sequences. The backdriveability feature of the servicer joints may even permit direct joint control with the four-bar linkage. All joints other than U would "float", thereby mechanically compensating for the four-bar linkage effect. This is particularly true for extractions. Whether this is feasible for insertion hinges upon representative hardware testing. Until that time, it is considered best to provide some coordinated joint motion by using either cylindrical or cartesian coordinates. An X, or axial, command in either of these coordinate systems will result in the generation of the proper joint angles so that the arcing motion is compensated for. This is found to be even more desirable when considering the radial case where final movement toward the attach point and during the retraction and insertion actions is preferably along the radius vector. This cannot be achieved with direct joint control alone. Backdriving joints could be effective during the retraction but not for the insertion. It is preferred and recommended that some coordinated joint control be provided for radial operations.

Should a TV be provided for manned monitoring, a single, narrow FOV camera on the end effector is adequate for verification of successful completion of a sequence.

b) Tip Motion Considerations - Manual Operation - The manually operated servicer arm is contrasted to the automated arm in that the arm's tip motion view is of considerable concern. Its movement will be observed directly on a visual monitor and that movement in some way must be correlated to the commands manually generated by hand controls. This is especially true for the manual augmented mode.

The three possible coordinate systems illustrated on Figure VII-9 are implemented manually, by hand control commands, as shown below in Figure VII-11. In direct control, the three directions of the hand control drive each joint directly. This is basically the manual direct mode. Correlation to any spacecraft reference is hard to perceive from a visual scene regardless of its location. To drive multiple joints and accomplish some meaningful tip direction is a very difficult manual task.

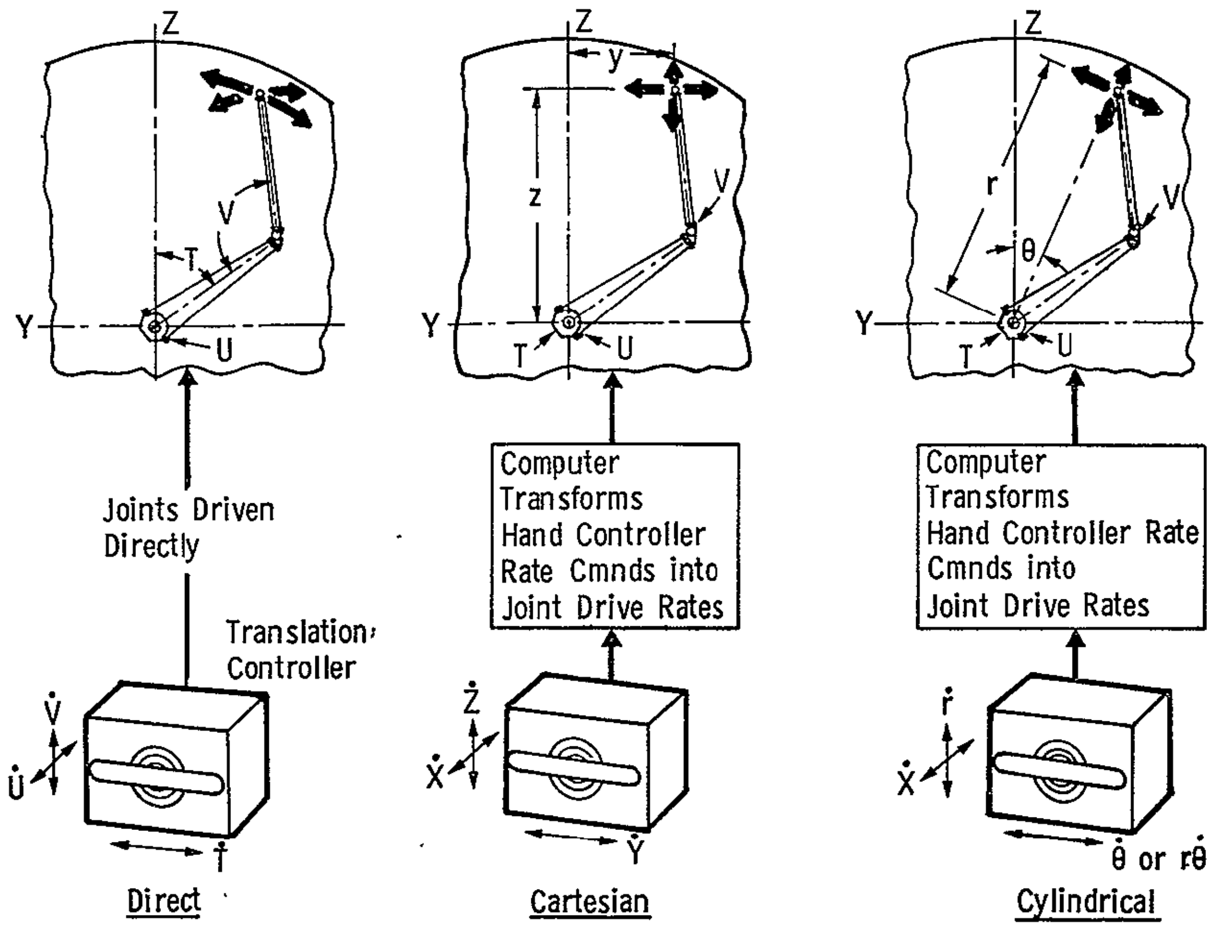

Figure VII-11 Tip Motion Coordinate System Considerations - Manual Operation

The cartesian and cylindrical systems' hand control commands correlate directly to the axes of the coordinate system. A single hand control command, such as $+\dot{z}$ or $\dot{r}$, results in multiple joints being driven from the coordinate transformation matrix.

Note that the hand control commands are rate commands. In any manual system this is preferred since the position loop is really closed by the man and his visual observations. Acceleration commands could also be used but the ability to achieve consistent, accurate motions near the target point is much more difficult. The coordinate transformations for manual thus transform rate commands into desired rates of change of joint angles. The automated system discussed in Figure VII-10 is implemented as a position control loop. The control laws, however, take the position errors and convert them to rate errors. These rate errors correspond to the manual rate commands and must also be transformed to desired rates of change of joint angles. For the selected coordinate system, the same rate transformations are used for both the supervisory and manual augmented modes. In the supervisory mode the control electronics thus do still command and control the arm rate much as the man does until the position and loop error has been driven to zero.

c) Tip Motion Coordinate System Selection - With an understanding now of the utilization of the alternative coordinate references in both automated and manual operations, it is possible to arrive at a preliminary selection of the preferred tip motion coordinate system.

Some of the advantages and disadvantages of the three systems considered are listed in Table VII-4. One fact becomes apparent--there are certain unique advantages to each of the approaches. It is indeed possible that all three could be provided. In a sophisticated system designed for application beyond the time frame this study is considering, all three may very well be provided with a selection switch that permits choosing between one or the other based on the particular step in the servicing activity, the type of spacecraft or module, etc. However, for the current development state, a simple, straightforward mechanization of the controls is highly desirable. Because of the cylindrical geometry of the servicing task and the configuration of the arm selected, it is possible to maintain a relatively simple mechanization yet provide adequately flexible servicer arm control with the cylindrical coordinate system; therefore, it is recommended.

Table VII-4 Coordinate System Selection

SYSTEM	ADVANTAGES	DISADVANTAGES
DIRECT BACKUP	• Easy to implement • Adequate for axial, automated operations	• Difficult visual resolution of motion • Coordination of hand controller/visual system not possible • Radial motion not straightforward • Operator training is more difficult
CARTESIAN	• Natural for any orthogonal motion (visual reference, off-axis removal, etc)	• Not suited to radial module replacement • Difficult to implement
CYLINDRICAL PREFERRED	• Naturally suited to servicing geometry • Ideal for radial exchanges • Straightforward implementation • Operator training is easy	• Additional coordinate transformation for off-axis replacements and visual coordination

The direct system has disadvantages. However, it is a natural backup system since it requires no coordinate transformations at all. With care and enough time, it is considered possible to complete a servicing task manually in this mode. It certainly should be a key objective of any simulation/ demonstration test program to verify this type of control and the crew tasks and timelines associated with it.

Returning to Figure VII-8 for a moment, it can be seen that the selection process has been completed through block 5. The next major area of discussion is a similar examination of the motion of the end effector mounted at the tip of the arm. This motion at the tip will be referred to as "attitude".

2. End Effector Attitude Control Alternatives

For the purposes of explanation, this report has separated discussion of the servicer arm tip motion from the motion of the end effector about that tip. This is a valid assumption when attempting to understand the alternatives available and making a configuration selection. In the mechanization of the control of these two relatively independent elements, however, there is obviously some data transfer necessary between the two control paths, as shown on Figure VII-8. It will be discussed later.

Two alternative methods of end effector control are shown on Figure VII-12. They represent two extremes--one merely leaves the end effector slaved to the arm regardless of arm positioning unless externally commanded as shown by the block on the right. The other internally and automatically slaves the end effector to an independent coordinate reference system. The latter is the control approach that requires data from the arm joints and knowledge of the base reference system (be it cartesian or cylindrical) in order to provide the desired end effector rotation. It is more complex in that additional coordinate transformations are required over and above that provided to accomplish the desired tip translational motion. The advantages are significant for radial operations, visual monitoring, and manual augmented control considerations. As in the case of the first alternative, external commands directly to the joint are required for final end effector positioning.

The subsequent discussion will examine the application of these alternative end effector attitude control modes to the total problem of finding and/or attaching to a replaceable module.

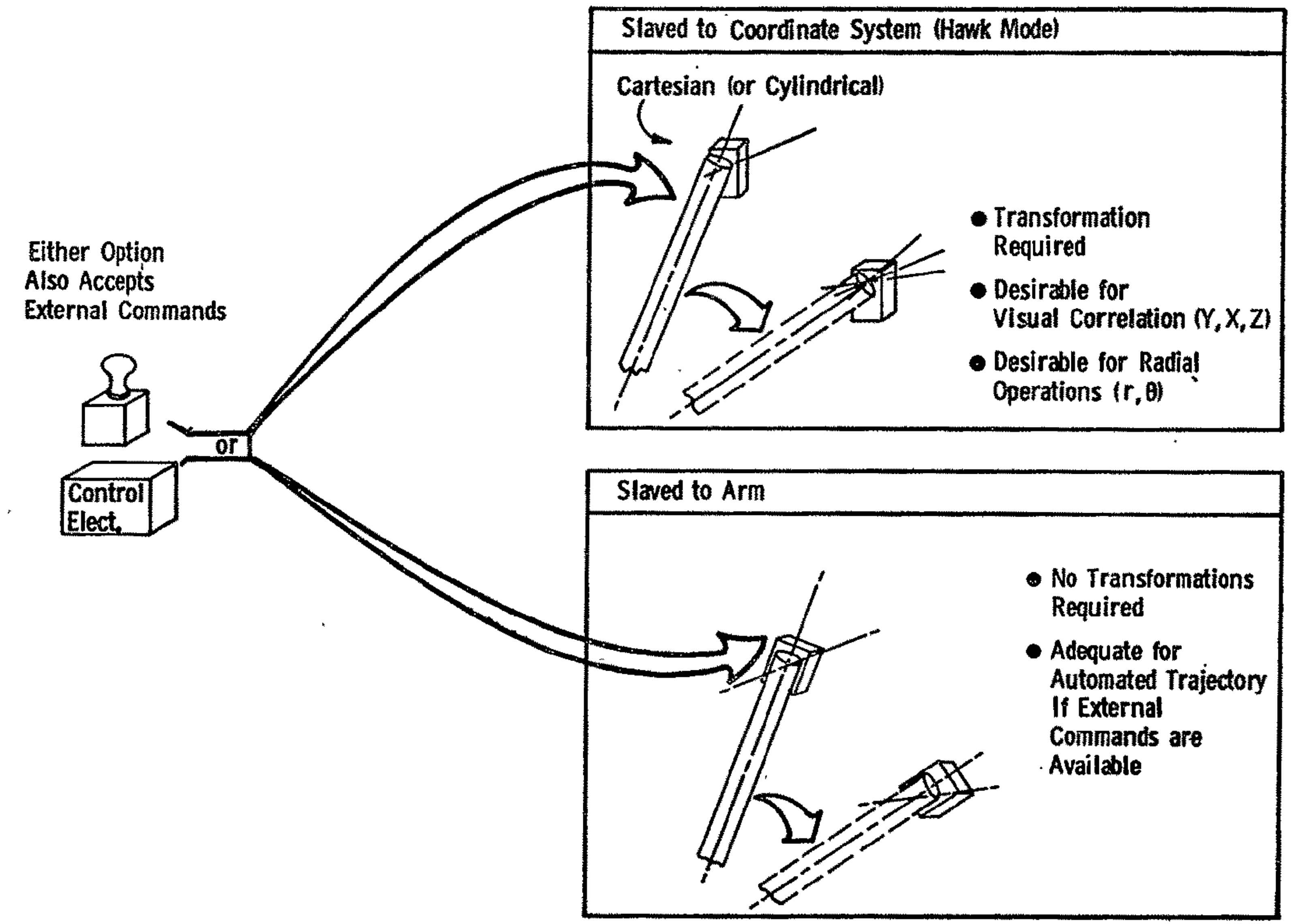

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Figure VII-12 End Effector Attitude Control Alternatives

Again, as was the case earlier for translational tip motion, a proper choice between the alternatives cannot be made until after examining how the end effector control will enter into the automated and manual control mode operations.

a) <u>End Effector Attitude Considerations - Automated Trajectory</u> - The elements of automated end effector control are pictured in Figure VII-13. For totally automated trajectories all physical conditions and locations are known preflight. The positioning of the end effector is a straightforward mathematical computation on the ground using the known geometry and the joint angles computed for the selected module attach point.

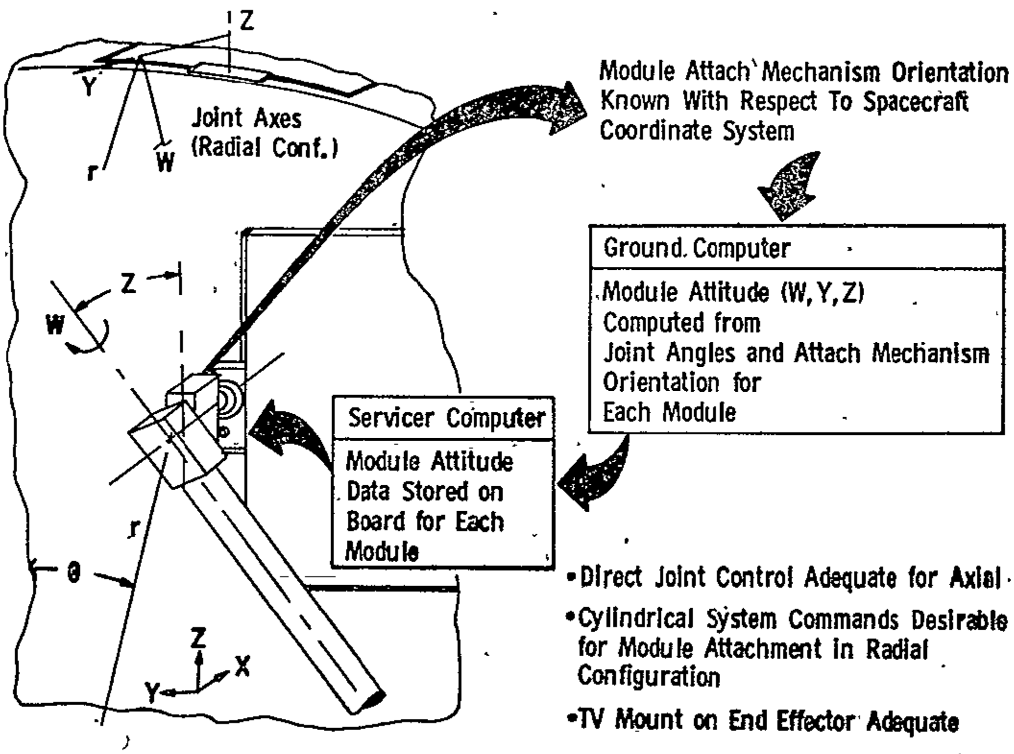

Figure VII-13 End Effector Attitude - Automated Trajectory

All attitude angles will be stored on board and commanded in sequence or in parallel as desired. For axial operations, it is apparent that direct joint control is a feasible alternative and, of course, is most simply implemented. For radial operations, which include a third axis (Y on the figure), the end effector alignment to the module is preferably accomplished in a radial direction. Consequently, the recommended alternative is to be slaved to the coordinates for modules mounted on the periphery of a circular spacecraft or stowage rack. This preferred set of coordinates is shown on the figure. A different geometry could result in preference for a different coordinate system.

A single TV mounted on the end effector appears adequate.

b) <u>End Effector Attitude Considerations - Manual Control</u> - The conditions for manual augmented control, portayed on Figure VII-14, are based on several assumptions. The first is that the end effector and its TV camera were positioned in the vicinity of the module attachment by either an automated sequence but more likely by manual translation hand controller commands. The visual cues to achieve that point were either a second wide-FOV camera, the end effector camera, or joint angle displays. Verification of achieving the target was sighting of the attachment mechanism target in the camera FOV. The next assumed step was alignment of the attachment mechanism into a predefined orientation on the TV screen using the wrist roll drive command on the attitude hand controller. The module interface mechanism shown here is already vertical, the desired orientation on the screen. Had it not been vertical the end effector would have been rotated until it was. If a radial exchange is being conducted a previous command to the Y drive would be necessary to align the Z axes drive vertical. After that the process would be the same.

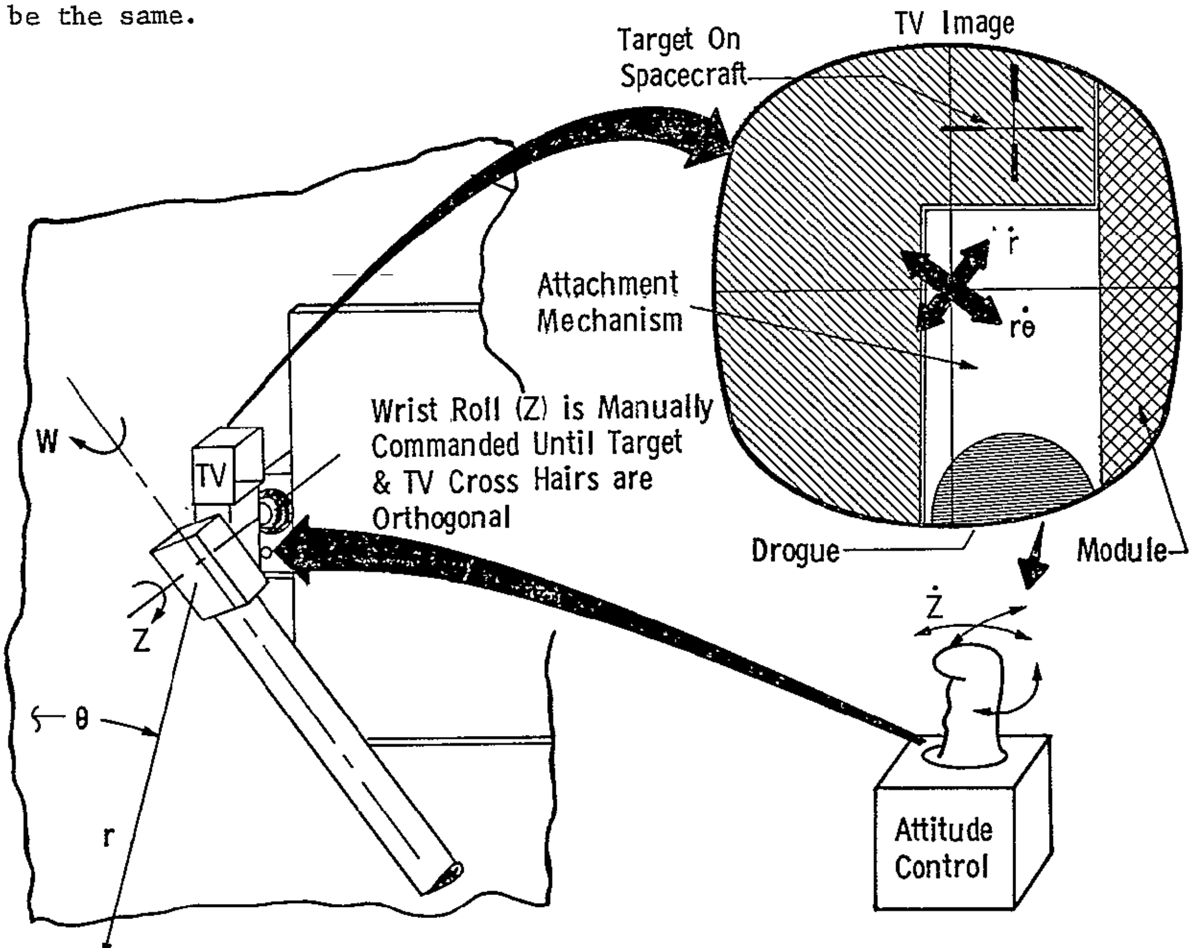

Figure VII-14 End Effector Attitude - Manual Control

At this point the next step is to align the target on the spacecraft so it is under the crosshairs on the TV screen by using the translation controller. In the cylindrical system a commanded motion in r or θ (rotation) will not appear as orthognal motion on the TV. For the angles on the illustration the vehicle, as seen on the TV, will appear to move in the directions of the arrows shown. While it might be possible to coalign the crosshairs with control in those directions it is not very efficient and in fact for some orientations of the arm and attach point that motion could provide a very confusing scene. The fact the motion will appear different for each different module location complicates crew training even more. Consequently, it is apparent that a very desirable feature is to have the translation hand controller command directly correlated to motion of the vehicle on the TV, regardless of attachment point orientation. How this feature can be provided through an additional coordinate transformation is described next.

c) <u>Controller/Display Coordination - Manual Control</u> - Figure VII-14 presented the problem in correlating normal translation hand controller $\dot{r}$ and $\dot{\theta}$ commands to the resultant motion on a visual scene generated from an end effector mounted TV camera.

Figure VII-15 below illustrates how a simple computation of the end effector's attitude with respect to the r vector can be used to seemingly "rotate" the $\dot{r}$, $\dot{\theta}$ commands so as to command the arm in a direction that is correlated with the direction of motions desired (see arrows) on the TV screen. The impact in implementing this capability is not on the end effector joint motion but rather on the tip translation motion discussed earlier. Its effect is reflected as the large "feedback" arrow on the illustrated approach in Figure VII-8.

The figure shows the preferred cylindrical coordinates; however, a similar transformation would be used if cartesian coordinates were preferred.

The transformation to correlate the commands and visual cues will result in a moderate impact on the onboard mechanization, be it electronics or in software, but it is a desirable feature. It is recommended for the servicer controls mechanization. Its performance should be evaluated in simulations and methods of mechanizing it refined.

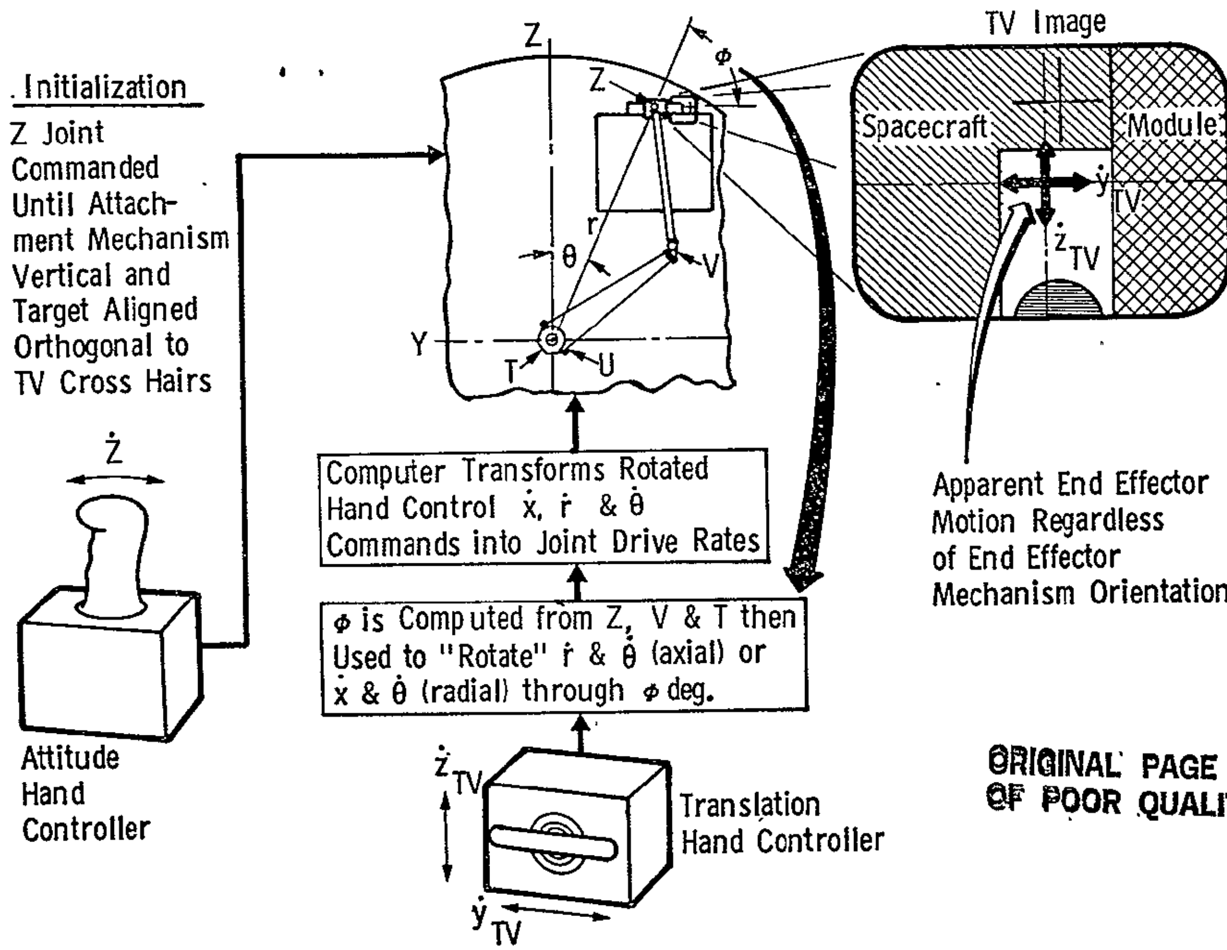

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Figure VII-15 Controller/Display Coordination - Manual Control

Note that the reference frame to which the end effector motion is slaved really does not become a factor in this narrow band of final alignment movements. The only drive of the end effector required is from the attitude hand controller. A slaved end effector is of more value when a broad search is being conducted for the proper attach point, or when specific module orientations must be maintained during a long transfer trajectory.

d) <u>Manual Control with Joint Angle Displays</u> - Discussions of manual control up to this point, either tip translation or end effector attitude, have all presumed a TV is present to provide the necessary visual cues. There is another alternative that should be presented here which is designed to accomplish arm motion manually without the TV. It does not result in any reversals of previous recommendations but is, nevertheless, important in that it is the recommended implementation of the manual direct mode presented in C.

Another impetus regarding this capability is a concern that the carrier vehicles' transmission of data is not at a high enough bit rate to provide anywhere near-real time TV images on the ground for the worst case condition, which is at geosynchronous orbit. The problem is most acute for the manual augmented control mode where picture delays (as much as 15 secs for 50 kbs transmission rates) result in an extremely slow process of manual arm manipulation.

The alternative is to transmit and display, on the ground, the position and rate of each joint. Considerably fewer bits are required (several hundred vs over a million for a TV), consequently the data can be provided virtually real time. The operations are similar to an automated trajectory sequence in that only the end point is of concern for each joint. The joint is manually driven at some rate until the display agrees with a predefined angle.

The operations are conducted without the TV visual cues for success; however, joint position and evidence of motion provide some degree of confidence. Hazard avoidance is, or course, very difficult.

It is recommended that this capability be provided as a backup in any configuration. It is effective and is a straightforward implementation. More visibility regarding its operational application can be obtained in Section H2.

3. Coordinate System Conclusions

The conclusions of this section can be summarized as follows:

Tip Translation

- Cylindrical coordinates for normal automated or manual augmented control
- Direct joint coordinates drive as backup (manual direct mode)
- Controller/display coordination when manually fine tuning to a target.
- Joint position data used for manual cues as alternative to TV

End Effector Attitude

- Slaved to a preferred coordinate system for manual or automated control
- Joint position data used for manual cues as alternative to TV

F. CONTROLS RELATED HARDWARE

In previous sections of this chapter, a number of hardware elements of the servicer system that are related to control of the arm have been introduced and/or evaluated. Some of these are straightforward--sensors, motors, etc. They will be discussed first. There are some other more subtle hardware areas, such as overall arm accuracy (tip capture volume), module motion in the guides and strain relief, which also interrelate directly with the control subsystem. They will also be discussed.

1. Controls Hardware Components

The controls hardware elements were introduced in Chapter VI, and are listed in Table VI-8. This section will bring out some rationale for the selection of these components.

a) Motors - DC torquer motors are preferred. They have reduced gearing, high accuracy and good response. The same type of motor was selected for a similar application--the protoflight manipulator arm (PFMA) delivered to MSFC by MMC on MSFC Contract NAS8-30266. The specific sizes, torques, current ratings and travel limits for each joint were introduced earlier in Chapter VI.

b) Hand Controllers - Discussion of the hand controls in the earlier sections on coordinate systems and trajectories showed that the attitude of the end effector and the translational motion of the arm's tip were relatively independent actions. This has resulted in a recommendation for use of two hand controllers, rather than a single 6 DOF controller. Crew familiarization and training is easier with independent controllers.

c) Position Feedback - Position feedback is required for the supervisory mode. In order to achieve the desired tip accuracy, discussed in detail later on in this section, a position accuracy of less than 0.1° (1σ) is required at the shoulder and elbow joints. A potentiometer, which is a 0.5° sensor at best is not acceptable for these joints, though it can be used at the end effector joints. An encoder is a highly accurate device but is bulky and requires more wiring. Its high accuracy is not really required. Other error sources dominate for accuracies much beneath 0.1°. The resolver accuracy of 0.1° can be easily achieved, therefore appears to be an ideal match for the shoulder and elbow while potentiometers can be used for cost savings at the end effector.

d) Rate Feedback - Rate feedback is essential for all modes. The low arm rates require sensitive and accurate rate sensing. Direct sensing rather than any derived rate determination is essential. A tachometer is recommended as it is small, light and accurate. Permanent magnet direct current tachometer generators, mounted directly to the motor shaft are preferred so the potential uncertainty at very slow tachometer rotations is avoided.

e) Brakes - Brakes are provided in each of the drives. They have not been discussed prior to this. Since many joints are backdriveable, brakes are essential to maintain arm position when not driving. Brake slippage is provided for strain relief at the joints in the event of abnormal torque conditions. Strain relief is discussed in more detail in 4. below.

f) Electronics - The core of the control system is, of course, the electronics (and/or computer) that schedules the trajectory sequence and generates the joint drive commands. It also is characterized by considerable flexibility in how it is implemented. Many of the decisions on the form of the electronics can and should be delayed until the next development phase. Some thoughts, derived from the simulation demonstration activities and the ETU electronics design task, however, are appropriate and meaningful at this point. A recommended low-risk approach is to close the rate loop around each of the joints in an analog fashion. Capability of driving the joints from the ground with just these circuits, should be provided on a joint-by-joint basis. This is essentially the manual direct mode as introduced earlier. Simply mechanized, fail-safe, backup operation is achieved very cost effectively. The remainder of the controls and control modes are recommended for implementation in a dedicated microprocessor. The functions in software could include:

- Trajectory sequences
- Mode control
- Coordinate transformations
- Target position library
- Calibration curves
- Parameter and constants storage
- Failure detection
- Hazard avoidance
- Status monitor

The memory required for all the above functions is highly dependent on the sophistication of some of them, such as the failure detection and hazard avoidance routines. The impact of these will be detailed later in Section F1. On the conservative side, where these functions are left to the operator's monitoring capability as much as possible, it is feasible for the remaining functions to be incorporated in less than a 4 K word memory. On the high side, the hazard avoidance alone could require that much memory, resulting in high side estimates upwards of 10 K words. An unsubstantiated engineering judgment has placed ultimate operational memory size between 3 K and 6 K words. No estimates on timing have been made but the rate loop being analog and the generally low arm rates indicate no real timing problems are foreseen. In demonstration/simulation described in Chapter VIII the computation repitition rate was marginal. Twenty-five computations per second are recommended.

A dedicated computer is recommended over sharing the computer role with the host vehicle. The potential for using the servicer with more than one host vehicle leads to mechanizing as simple an interface as possible, including software. This also certainly facilitates more complete checkout and preflight operations at a lower test level.

2. Capture Volume

One of the secondary hardware related issues is the error allowed between arm end effector and module capture drogue. This is referred to as capture volume and is influenced considerably by the control system's accuracy.

The capture volume at the servicer arm end effector directly impacts the mecahnical sizing of the attachment mechanism as well as the margin of volume between the module and the cavity into which the module must be positioned. It is an important factor as it results in unusable volume in the spacecraft.

To arrive at credible estimates of capture volume, an in-house funded IRAD task was completed that performed a detailed definition of the inaccuracies that impact the positioning of a servicing arm. Autonomous positioning was assumed as it represents the most stringent requirements. A number of different configurations of servicer arms of varying capability and complexity were analyzed in order to gain confidence in the results. At one extreme are the simple Integrated Orbital Servicer (IOS) arms that remove and replace modules in either the axial or radial direction only. Arm motions are straightforward and can be programmed in sequence for the most part. A sophisticated general purpose manipulator (GPM) was evaluated at the other extreme. It can reach and remove modules mounted in the second tier of spacecraft periphery in radial, axial, and off-axis directions with ease. Multiple coordinated joint driving is required.

All possible error sources for each of these configurations were identified. They generally fall into the following categories--attachment point target locations errors, joint drive errors, mechanical deformations, and vehicle docking misalignments. The approach was to quantify all errors in each of the above categories, then adjust--through changes to design--the large "drivers" until the errors in the four categories were approximately equal.

The resulting error envelopes for the arm tip and accompanying module misalignments, when attached, is shown in Table VII-5. The IOS near-radial configuration was selected and its design is discussed in Chapter VI.

Table VII-5 Error Analysis Summary

Error	Pivoting Arm	IOS		GPM Two-Tier Radial
		Axial	Near-Radial	
Pointing Accuracy (3σ)	±0.76"	±0.78"	±1.04"*	±1.32"*
End Effector (or module) Angular Misalignment (3σ)	±0.81 deg	±0.81 deg	±0.60 deg*	±1.0 deg*

*Assumes comprehensive assembly alignment and calibration.

In order to achieve these accuracies, several hardware characteristics were found to be dominant and special attention to requirements in these areas will be necessary. These, along with required accuracies, are listed below. All values shown are three sigma values. There were no new technology requirements necessary to achieve the accuracies above.

- Joint drive error < 0.3°
- Shoulder installed on stowage rack to 0.1° and 0.06"
- Elbow-to-member twist error < 0.15°
- Arm temperature differential < 10°F

- Module locations known to 0.06"
- Docking probe stiffness > 2500 lb/ft
- Docking accuracy ≤ 0.3° in all axes
- ACS jet firing moment (carrier vehicle or spacecraft) < 1500 ft-lbs
- Comprehensive assembly alignment/calibration performed preflight

Many of the above requirements are interrelated. Also, some have been derived based on the specific servicer design assumed for the analysis. For example, a 60" docking probe with a diameter of 9" was used. A change to any other shape could affect docking probe stiffness and acceptable ACS jet firing moment.

3. Module Motion Control in Guides

Two alternative approaches were considered for the removal and replacement of modules from their respective cavities using the module attachment mechanism and guides. They are backdriving vs coordinated joint drive. The simplest of these is backdriving or powering just one drive joint to extract or insert a module. The joint will provide a tip force that forces the module down the guides. The guides maintain the orientation of the module. If the module translation motion requires other joints to move because of the geometry of the arm or trajectory, the capability can be easily accomplished by releasing the brakes and letting the joints backdrive. This capability is currently provided in the joint design. The same backdriving can provide all secondary joint motion during insertion as well, again requiring just one joint for transmitting force. A concern for the backdriving method, and it becomes more apparent for the insertion, arises at the point where the module is nearly out of, or just starting into, the cavity. It is uncertain as to whether the guides can force the necessary backdriving with the very short moment arm available. The module could even tend to cock or become jammed.

Therefore, it is considered the safest and most reliable approach to select the second alternative--coordinated multiple joint drive--as the primary control during this phase. The backdriving joint capability should still be provided but as a backup only. It is really essential for any failure that loses the coordinate transformation provided for coordinated joint control.

The impact of providing the necessary coordinate transformation is minimal. At the most, it is only the implementation of several equations in software. The transformations will very likely be provided anyway for effective manual control and coordinated visual displays.

4. Strain Relief

Strain is defined here as resulting when an external torque is placed on any element of the servicer arm. Normal torques necessary to place a 20 lb force on the module for retraction or insertion are not considered a strain. One possible source is the side forces resulting from the wedging action during normal insertion of a slightly misaligned module. The wedge effect can create forces much greater than the tip force of 20 lb in a direction orthognal to the tip force being applied. These lateral forces can be absorbed by brake slippage or motor backdriving when in a plane perpendicular to joint rotation. Depending on the module location and orientation of the arm some of these lateral forces are reflected in a direction where there are no joint drives to absorb the misalignments. For these forces the arm flexibility itself must and can absorb these torques. However, it has been elected to leave the arm flexibility independent of this disturbance and permit its design to be governed only by control system natural frequency requirements, arm strength and weight criteria. Instead a special preloaded spring joint has been added in the arm that breaks away at excessive torques and affords the desired strain relief through the springs. See Chapter VI for more details.

G. ADDITIONAL CONSIDERATIONS

Before going to the final discussion of just how the control subsystem accomplishes the servicing task in the three proposed control modes, there are two or three controls-related considerations that, for lack of a better term, have been called additional considerations, yet are very important. One is hazard avoidance, another is failure detection/correction and finally, servo loop stability.

1. Hazard Avoidance

This feature is an obvious requirement in some form, because of the multitude and variety of arm motions, the large number of module shapes and sizes and the close proximity of mechanical structure on either side of the servicing arm's theater of activities.

Table VII-6 lists some of the options available for performing hazard avoidance during servicer arm trajectories.

Options #1 and #2 in the table are straightforward. Both predefined trajectories and onboard TV are likely servicer capabilities for more overriding reasons. They are certainly effective and obvious candidates for hazard avoidance.

Option #3, the computer augmented visual scene, uses joint angle data as a direct display for more current knowledge of joint position. In the application referred to here, however, it would be made more useful by processing the joint position and rate data and incorporating it into a simulated visual scene that presents near-real time, or even predictive, computer-simulated TV scenes. The scene would be updated periodically from the true TV scene whenever downlinked. This would certainly be useful for more effective manual control but would also make hazard avoidance monitoring more current. There is, of course, a software impact but it would be restricted to ground software only.

Option #4 is an extrapolation of #3. It assumes or requires no TV whatsoever, using only engineering data from the joint and known measurements of the servicer's physical elements to create a totally simulated visual scene that is very near real time. Engineering data would be updated on less than 0.1 sec intervals. The options does lack a little of the warm feeling a live TV picture can convey, particularly when the concern is for hazard avoidance, but it does drastically reduce downlink data.

Table VII-6 Hazard Avoidance Options

No.	OPTION	ADVANTAGES	DISADVANTAGES
1	PREDEFINED TRAJECTORY -- NO TV ● Automated (Supv. mode) ● Manual	● Simple onboard implementation ● Nothing to implement onboard	● Known hazard avoidance only ● Subject to human error ● Less effective than automated due to inaccuracies ● Vulnerable to failure-induced hazards
2	VISUAL (TV)	● Simple ● Capability for avoiding unknown hazards	● Limited capability for remote stations (≈15 secs between pictures) ● Some trajectory definition still required
3	VISUAL (TV) WITH COMPUTER AUGMENTATION	● Permit slowed TV rate ● No onboard implementation ● More current visual scene	● Increased ground software ● Increased engineering data required
4	GROUND COMPUTER SCENE GENERATION -- NO TV ● Manual or automated	● No onboard implementation ● Some unknown (failure) hazards avoided ● Better visual perspectives	● Considerable ground software impact ● More engineering data downlinked
5	COMPUTER STORED SOFTWARE HAZARD BOUNDARIES ● Onboard ● Ground	● Arbitrary manual commands permitted ● Automated trajectories less constrained ● No onboard software impact	● Considerable onboard software impact ● Known hazard avoidance only ● Useful for manual only
6	PROXIMITY SENSOR (Measured Signal Reflections)	● Detect unknown (failure) or known hazards or both	● Hardware cost and development unknown ● Trajectories still must be defined
7	COLLISION DETECTOR (Joint Motor Current Monitors)	● Simple and straightforward hardware implementation	● Detect unknown (failure) hazards only ● Trajectory definition still required ● Damage may occur

Option #5 has a number of possible variations which have not been evaluated in much detail yet. It is basically a software program that defines all the physical geometry of the servicing elements much as the two previous options. One of the differences is that it can automate the hazard avoidance function. By keeping track of the arm position, it can signal an alert when a leading edge has been computed to approach an obstacle. It even can, in the onboard implementation, modify the trajectory via software to circumvent a known hazard. This could permit more generalized onboard, automated trajectories. The same routine could be used on the ground, however, it would probably just provide a warning when approaching a hazard. A program similar to this was written at Martin Marietta for a proposed Mars roving vehicle. It was programmed and tested in a simulation, showing good feasibility. It can have a large software impact (1000's of words) which could prove undesirable for onboard implementation. It does not really alleviate any of the existing requirements for visual monitoring.

The proximity sensor, Option #6, is the first option that would detect unknown as well as known hazards. It presents some real technical challenges particularly in the hardware area. Possible methods are: sensing reflected light, reflected rf, laser schemes, sensed IR sources and others. None of these have been examined closely or have had any development work completed that relates directly to the servicer task.

The last option, collision detector, is straightforward. It really must fall in the category of a backup device since it implies the possibility of damage to the structure or the module. There are, of course, preventive measures that could be provided such as metal or teflon bumpers on the module or arm.

The recommendations, by mode, resulting from these many options are:

- Supervisory Mode: Automated predefined trajectories with hazard monitoring via use of TV image
- Manual Mode: Use predefined manual sequences where possible; monitor joint angles and computer augmented visual scene

These recommendations have maximized usage of capabilities already present and provided in the basic servicer system. The trajectories presented earlier were purposely designed to maximize clearances between module and hazard and to simplify the trajectories as much as possible. One reason for this is to minimize the hazard avoidance problem, thereby permitting adequate avoidance without costly or complex mechanizations in both hardware or software.

As the capabilities of the arm increase along with an expanded servicing role in the future, the hazard avoidance schemes will undoubtedly be expanded as well. To prepare for this point in development, effort should continue to be directed at some of the hazard avoidance techniques other than just those recommended. Development of a hardware proximity sensor should be continued as well as the computer augmented and computer generated visual scene generation schemes. Much work has already been done on the latter and should be studied for application to the servicing system.

2. Failure Detection

A brief failure modes and effects analysis was performed on the servicer system hardware in order to determine the performance of the servicer in the presence of failures and to scope the magnitude of the hardware/software needed to detect the more critical of these failures.

One obvious solution, should a critical failure occur is to at least disable further arm motion so damage does not occur to nonfailed equipment. It is a more likely alternative that a reliable, reusable, flight operational servicing system of the future will employ a significant amount of redundancy. For that system, equally reliable as well as effective, failure detection is necessary in order to make proper use of that redundancy.

Several straightforward failure detection methods were derived from the list of basic failures anticipated. These are listed on Table VII-7 together with the failures they will detect directly.

It can be seen that most of those shown can be performed on the ground by man, provided the proper displays have been furnished. Some delays must be tolerated in this mechanization but it is a good starting point in the development. It is desirable that eventually most of these methods be mechanized onboard. There are no problems foreseen in onboard mechanization, either hardware or software.

Table VII-7 Failure Detection Methods

METHOD	FAILURES DETECTED
• MONITOR FOR EXCESSIVE RATES	• FEEDBACK DEVICE
• MONITOR POSITION--Desired vs Actual	• MOTOR OR MOTOR DRIVE • POSITION FEEDBACK DEVICE
• JOINT MOTOR CURRENT--High or Zero	• MOTOR • ANY IMPROPER TRAJECTORY THAT RESULTS IN COLLISION - Software - Feedback - Human - Manual Command or Uplink Error • MECHANICAL DEFECT OR JAM
• VISUAL (TV)	• IMPROPER TRAJECTORIES
• EXCESSIVE TIME FOR COMPLETION	• INCOMPLETE STEP OR TARDY PERFORMANCE

If redundancy is provided the selection of the backup elements will be conducted from the ground, at least initially. Eventually, higher levels of autonomy will drive toward correction techniques onboard. Current development efforts and simulation/demonstration test programs should consider failure detection and correction in defining test objectives. One parameter that has a direct influence on both the failure detection and correction function is the timing. That is, how quickly must it be detected, how quickly can it be detected, and how quickly must it be corrected. This kind of information can come directly from trajectory and timeline evaluations in a simulation test.

3. Stability Analysis and Design

Finally, the last but not the least important is an assessment of the stability of the joint rate control loops.

In order to evaluate the stability of the proposed servicer control system, an analytical block diagram was defined that incorporated the basic cylindrical coordinate system, the control modes and hardware elements introduced earlier. The open loop transfer function for a single joint drive was defined and its stability evaluated on a bode plot. The loop was unstable, as expected, however the addition of a straightforward compensation term resulted in a phase margin over 30° and gain margin over 6 dB.

A reasonably high amplifier drive gain--4000 V/V--was assumed to maintain accurate, linear rate control in the presence of motor and bearing friction at rates below the expected minimums to be commanded (≈10% of the nominal maximums of 5 to 12°/sec).

As hardware design proceeds, the preliminary work completed to date will be repeated and refined and expanded to each joint. This effort is necessary to ensure the loops are indeed stable. Also the electronics designer must know the compensation terms in order to mechanize them in the electronics. The mechanical designer must know what constraints on characteristics such as stiffness, inertias and friction levels are placed on his mechanical design.

H. CONTROL MODES IMPLEMENTATION

A block diagram of the key elements in the servicer controls system and their interfaces is shown on Figure VII-16. The figure shows the typical hardware devices in one of up to seven different joints--electronics, the motor, and a rate and position feedback. The three different modes are shown in a functional form on the figure. How each mode accomplishes control of the arm will be discussed in greater detail in the subsections to follow.

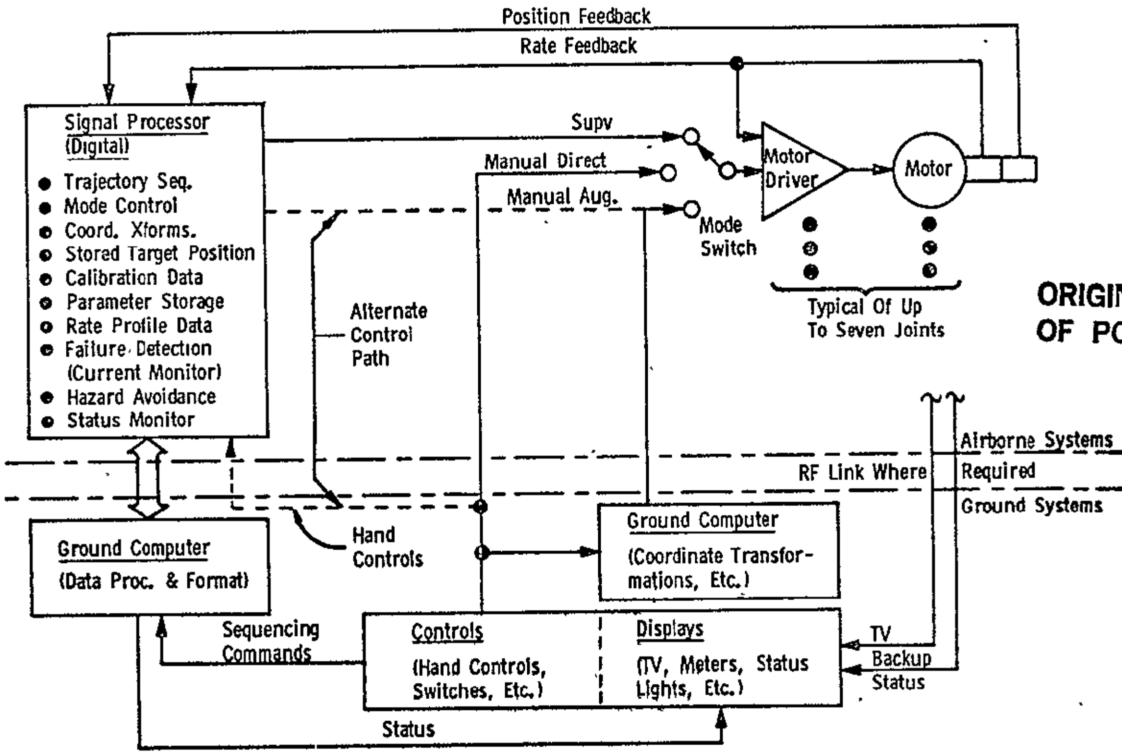

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Figure VII-16 On-orbit Servicer Control System Block Diagram

The onboard data computation and command generation for the normal supervisory mode is assumed to be a digital computer. See section F1(f) above. Its size will vary from a small minicomputer to a larger general purpose machine depending upon the number and sophistication of the features shown in the box that eventually are implemented. Only a digital machine has the flexibility and capacity to accommodate all those features cost effectively.

The ground support required is in the form of controls and displays for the purpose of obtaining status and other visual information necessary to maintain control of the servicing task remotely from a station on the ground.

A TV is assumed, as is a sophisticated ground computational facility for engineering and TV data processing and display formatting.

When the control station is remote from the servicer, data will be transmitted through existing ground and carrier vehicle RF communication systems.

The remainder of this section will discuss the three control modes individually. Wherever applicable the simulation demonstration (Chapter VIII) configuration will be used to more clearly show the intended operational scenario for each mode.

1. Supervisory Mode

All the servicer arm motions and trajectories are preprogrammed onboard in this mode. The heart of the system is in the signal processor, envisioned as a microprocessor or minicomputer in an operational system. The failure detection, hazard avoidance, and status monitor routines are relatively elementary as opposed to the complex routines required of a totally autonomous system. They merely will augment the manned monitoring via TV on the ground and provide for reactions onboard that are time critical. The link with the man who may be on the ground or in the Shuttle Orbiter, or wherever, is through an RF link if it is a remote station. The data transmitted are status indications and TV. The uplink data is the sequencing commands from the ground that are initiated on successful completion of a phase of the removal/replacement sequence.

The digital implementation provides considerable flexibility in that trajectories and/or module coordinates can be easily modified via digital uplink command. Calibration data can be modified from the ground or via an onboard calibration routine using a known spacecraft target. Selection from more than one coordinate system, programmed onboard, is possible.

Man's principal interface with the servicer system in the supervisory mode is envisioned as a computer terminal display/keyboard. This plus most of the other elements of a servicer system control station were realistically incorporated in the simulation demonstration of this contract. The simulation control station setup is shown in Figure VII-17.

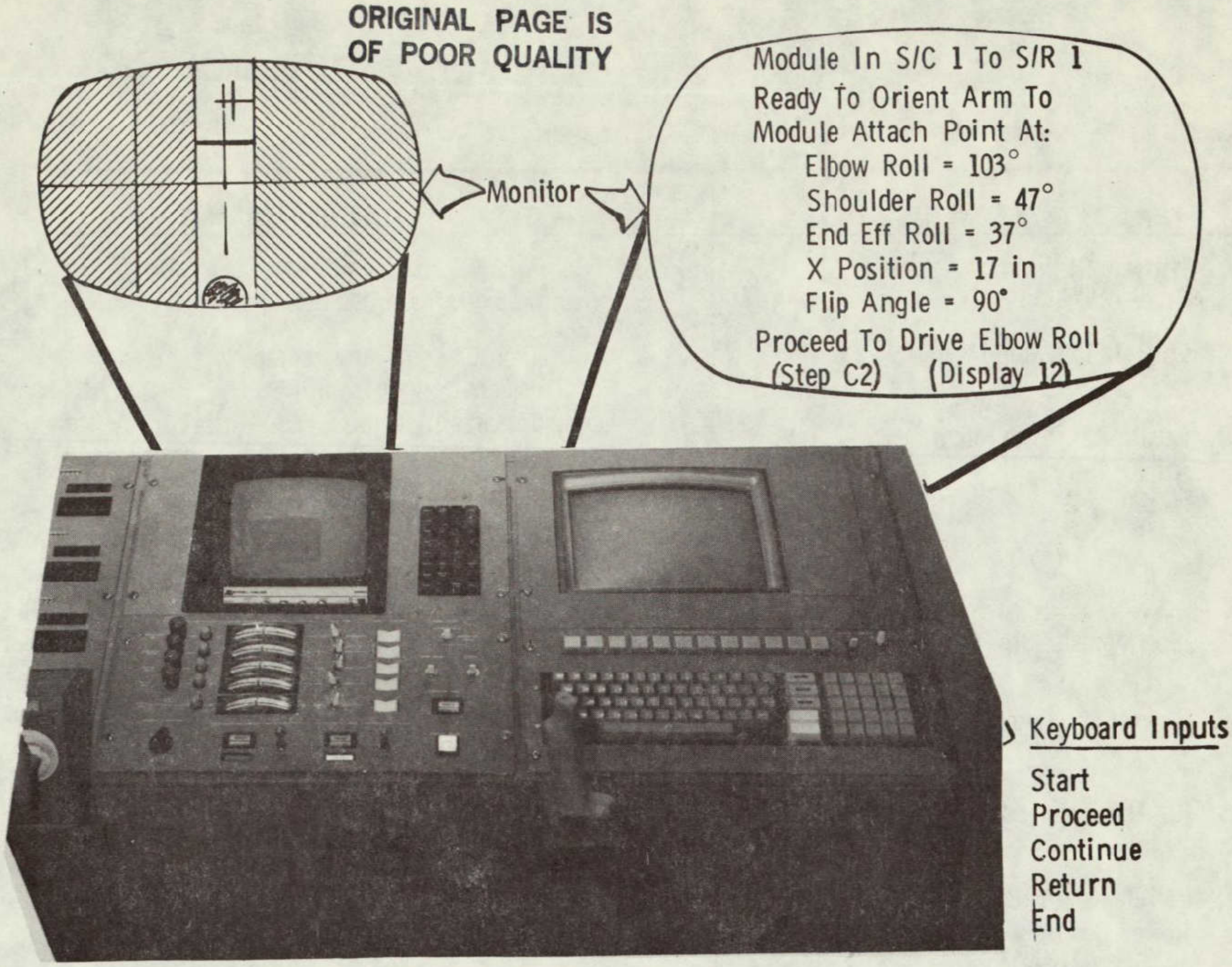

Figure VII-17 Simulated Servicer Control Station

The setup shown is reasonably representative of a ground based or Orbiter based control station.

A particular step in progress and its status is displayed on the CRT at the right of the figure. The display is typical of a number of displays appearing throughout a complete trajectory sequence.

The trajectory sequence used in the demonstration is summarized in Table VII-8, and tracks very closely that derived earlier in Section D for an axial-to-axial exchange.

This sequence is implemented in computer software as a series of automated steps, each step driving just one joint through a given angular travel at a time. These steps have been grouped into functional segments as shown on the left of the table for a single module exchange. A listing of a typical set of steps in one of those segments (C) is shown on the right of the table. A unique display is associated with each step. An example is the display for

Table VII-8 Simulation Trajectory Sequence

TRAJECTORY SEGMENTS	NO. OF STEPS	TYPICAL SEGMENT SEQUENCE
A. Initiate Simulation	5	1. "CONTINUE" provides display requesting go-ahead of segment.
B. Initiate Exchange	5	2. "CONTINUE" commands computer to extract desired joint angle from memory, initializes comparator and shows error signal and error light ON on panel.
C. Position Arm to Module Attach Point	9	
D. Engage and Extract Module	6	3. "PROCEED" initiates drive of elbow joint until error is zero. Error light goes OFF and display changes to request a CONTINUE TO NEXT STEP.
E. 180° Module Flip	10	4 and 5. Repeat of Steps 2 and 3 for shoulder roll joint.
F. Module Insertion	9	6 and 7. Repeat of Steps 2 and 3 for end effector roll joint.
G. Return Arm to Rest Position	8	8 and 9. Repeat of Steps 2 and 3 for shoulder translation joint
H. Initiate Next Exchange or Power Down	2	
TOTAL	54	

step C2 shown on the CRT on Figure VII-17. Note that the display tells the operator what word to type in next, e.g., "PROCEED" in the last line of the CRT display on Figure III-17 is the next word for the operator to type. After typing in the word, the operator also strikes the carriage return button to execute the command. To simplify the typing and minimize errors, only the first two letters of each word need be typed correctly, e.g., the computer would read PROCEDE as PROCEED. While a total of 54 such steps and displays were required for one exchange in the simulation the operator need not memorize them; rather the computer leads him sequentially through the exchange process.

Any module exchange requires basically the same steps and segments. Only the beginning and end points pulled from memory are different.

The meters, lights and TV screen on the left of the control station in Figure VII-17 are for monitoring arm motion during the automated segments of the trajectory sequence. A typical view of the interface mechanism prior to mating is shown on the TV screen. The end effector jaw closure and opening and interface mechanism latch/unlatch functions can be manual operations from the control panel on the left, as was the case for the simulation; or automated in the computer sequence.

2. Manual Direct Mode

The Manual Direct Mode is provided as a totally unsophisticated means of backup control. It sends commands directly to the joints themselves. Commands are one joint at a time. Motion is with respect to each joint's mounting base rather than with respect to the display coordinate system, making the control task somewhat awkward for some configurations. Its uses are: 1) as a possible normal control mode for certain simple arm configurations that lend themselves to direct joint control; 2) as a backup in the event of a failure in the ground computations or downlink used in the augmented manual mode; or 3) in the event a joint failure has occurred that can be worked around but the normal coordinate transformations either onboard or on the ground are not valid.

The portion of the servicer control station used in the simulation/demonstration, containing the servicer controls and display panel for the manual direct mode, is shown on Figure VII-18. Accomplishing a module exchange, or a segment of it, is purely a manual operation with an explicit procedure, or checklist, that is followed.

The general order of the steps necessary to complete a single joint motion is shown in the figure with the respective control or display associated with that step. First the desired segment end condition is set in on an angle set pot. This results in a non-zero reading on the error meter and the error light coming on. The operator then presses a direct mode control rocker switch in the direction which causes the error meter reading to return to zero. Note that the switch motions are coordinated with meter motions. The meter indications are nonlinear so more sensitivity is obtained near null. When the error signal reaches zero, the error light will go off. As the appropriate joint drive rates cannot be easily determined and may vary for different parts of a trajectory, any of three different levels of drive rate may be selected.

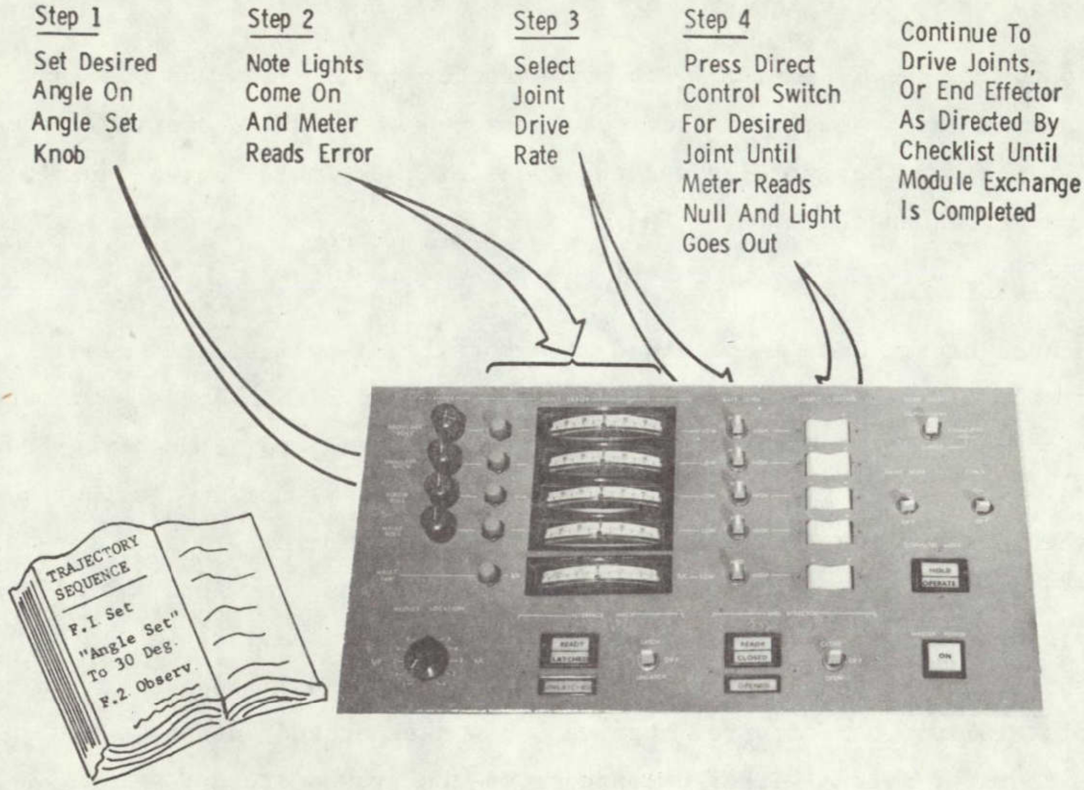

Figure VII-18 Simulation of Manual Direct Operations

3. Manual Augmented Mode

The manually augmented mode has man doing most of the arm control as in the Direct Mode above only using hand controllers and the TV instead of panel switches. Also, the computer, whether it is on the ground, Shuttle or on-board, is still in the loop to facilitate the direction of motion of the arm and provide optimization of its motion with respect to the displays provided. The operator controls all the remaining module exchange activities such as the trajectory, hazard avoidance, sequencing, and fail-safe aspects based on TV images. The computer can be programmed to provide visual aids in the area of trajectories and hazard avoidance. The most useful role for the manual augmented mode is to perform unscheduled motions to previously unidentified targets of opportunity.

For distant, remote control stations such as ground to geosynchronous orbit, the arm motions would probably be slowed from the rates used during automated control in order to be compatible with the slow TV refresh rate caused by RF transmission bit rate constraints. Two hand controllers--one for arm translational motion and another for controlling the attitude of the module at the tip--is the preferable method of implementing Manual Augmented control.

The second feature of this mode is that arm motion will not be simply a single joint at a time but rather, will be such that the end effector moves in some visually resolvable coordinate system. This is accomplished with equations in software.

The manual augmented modes are thought of as a man controlled alternate to the supervisory mode and use a visual reference, such as a TV, for guiding the arm or module to the target. Under these conditions, the more recognizable and consistent the visual cues are the more effective the control. To that end the manual augmented mode features a number of software coordinate transformations. The visual display hand controller coordinate system is defined such that the up/down motion of the hand controller is up/down on the TV and similarly for right/left on the TV. These visual display coordinates are transformed into the cylindrical coordinate system discussed earlier in E. The effect of this is that the TV display motion is essentially in cylindrical coordinates only the commands are rotated so as to align with the image on the TV.

Another augmentation or image enhancement that is incorporated is to slave the arm end effector to the cylindrical coordinate radius vector. This is often referred to as the hawk mode. It is most useful for an arm that is being moved through complex or arbitrary motions yet the end of the arm or a TV camera mounted at the end is preferably pointed in the same direction throughout. This is so that clear and continuous monitoring of a target can be performed while the arm is moving. This mode gets its name from the claws of a hawk, or it could be a man's hand as well; both remaining in a given orientation with respect to a defined reference, such as an object being grasped, regardless of the arm's, or hawk's, movements during the process. Two useful references to which a servicer end effector could be slaved are shown in Figure VII-19 below. One is the vehicles references axes (x, y and z), while a potentially more useful reference to the servicer is the cylindrical coordinate system. This is the preferred approach for manual control. In this case, the TV image always appears as though the axis of arm rotation was in the same place, certainly improving the operators ability to orient himself.

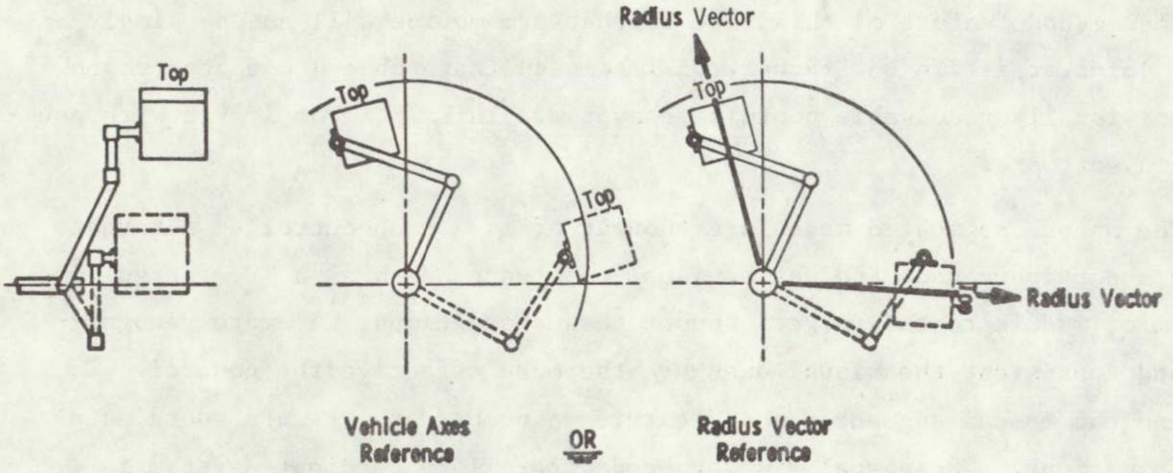

Figure VII-19 Alternative Hawk Mode Characteristics

In the satellite servicing arm configuration either reference is easily implemented. The only axis that must be driven to accomplish either is the end effector roll joint. The change in this angle necessary to maintain the module in the desired attitude while motion about the shoulder (or elbow) roll joint is induced can be seen in the two sketches on the right of the figure.

The operator's operations for all the manual augmentation modes is essentially the same. A typical control station is that shown in Figure VII-20 for the simulation. The operator's principal activity is controlling the arm's position and the end effector's attitude using the two controls shown. The joint angle meters and lights can be monitored, if desired, for an independent evaluation of the arm joint movement. This, however, is not essential or even useful, except for those cases where the desired joint angles are known ahead of time for a given target and the time is taken to dial them in on the attitude set knobs. In the general application of the manual augmented mode, this data is not always available nor is it essential if a good TV presentation is available.

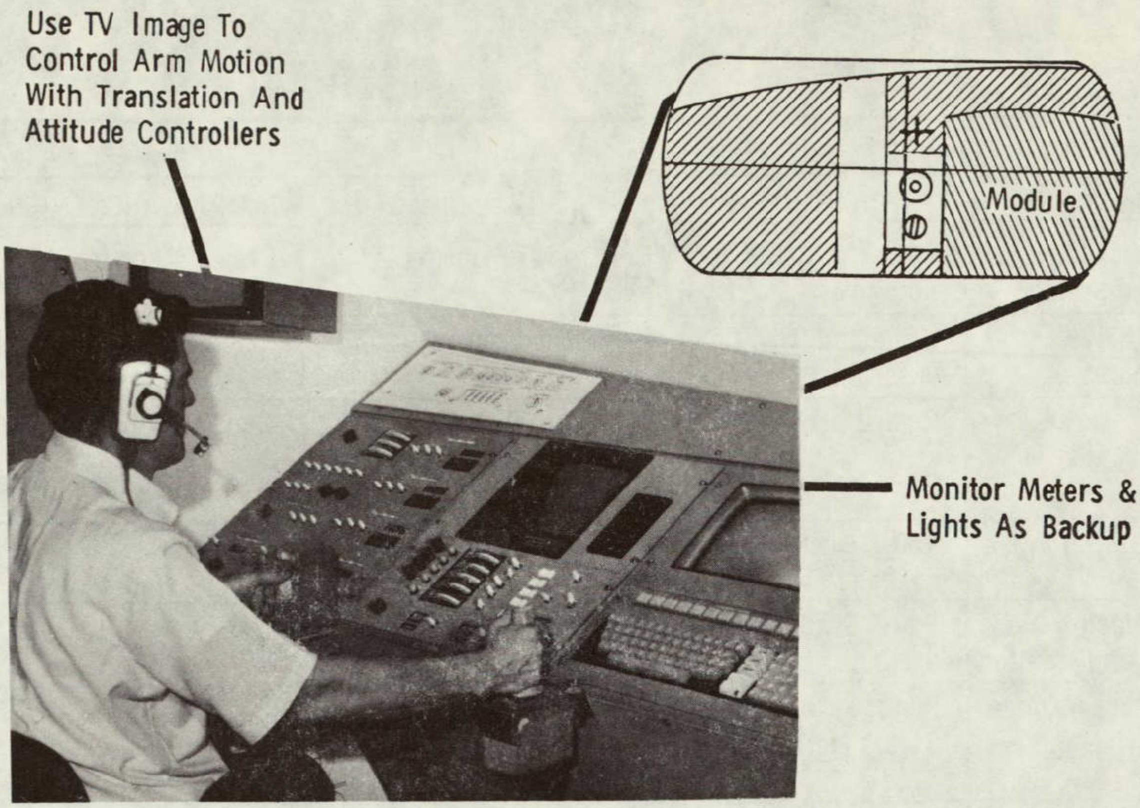

Figure VII-20 Manual Augmented Mode Operations

4. Control Mode Implementation

A summary of how each of the three control modes were implemented in the simulation is provided in Figure VII-21. It is considered quite applicable to the proposed flight design. The basic functions that must be performed in accomplishing a trajectory are essentially the same for each of the modes. Only the manner in which they are implemented may vary. Considerable commonality is evident in that a number of functions are accomplished similarly regardless of mode.

FUNCTION	CONTROL MODE		
	SUPERVISORY	MANUAL DIRECT	MANUAL AUGMENTED
Trajectory Sequence Storage	Stored in Computer Software	Crew Procedure	Crew Procedure
Joint Drive Command	Command from Computer Comparator	Direct Control Switches	Hand Controller
Determine Desired Angle Achieved	Computer Comparator	Manual Monitor of Meters and Lights	Manual Monitor of TV
Rate Selection	Rate Select Switch on Panel		
End Effector Drive and Monitor	Panel Switch and Status Lights		
Interface Mechanism Drive and Monitor	Panel Switch and Status Lights		
Overall Status Monitor	Man Monitors TV, Lights, and Meters	Man Monitors Lights and Meters	Man Monitors TV

Figure VII-21 Summary - Control Mode Characteristics

It is appropriate at this time to again stress that the three modes described are not to be construed as a final selection for an ultimate servicer design. They are purposely selected at this time to span as broad a spectrum of control sophistication as possible. Whether one, two or all three of the types of control represented by these modes will ultimately be implemented will depend on: the evaluations currently initiated, on those planned for the remainder of this year, and on evaluations that will be conducted at MSFC after delivery of the tools developed here. Regardless of which or how many modes are planned, the exact form of each will undoubtedly change as these evaluations continue and firm designs begin to evolve.

VIII. SIMULATION/DEMONSTRATION

A simulation/demonstration of the servicing module exchange operation was conducted at Martin Marietta, Denver. It represents a preliminary to the design and fabrication of the Engineering Test Unit and its subsequent use in an Orbital Servicing Demonstration Facility at MSFC. It is anticipated that the MSFC facility will repeat, extend, and expand the demonstration activity conducted in Denver. The Denver demonstrations used existing general purpose motion generation systems and computers. The emphasis in the simulation was to identify good module transfer trajectories and control systems that can be developed to flight equipment.

The objective was to conduct a demonstration of the module exchange activity of on-orbit servicing in the Supervisory and Remotely Manned Backup modes to confirm and define the control approaches which have been proposed. This was accomplished and the conclusions are summarized in this chapter. It was important at this point in the development of the servicer design to investigate the servicer control approaches in a functioning setup. The understanding of the control problem gained from the simulation has provided a cost-effective, sound basis for determining what control investigations should be performed in the future. The conduct of this controls simulation provided a focus which further evolved and confirmed the control approaches, including definition of the important man/machine interfaces, through a process of discovery, refinement, and expansion.

The simulation was the first level of integration of the servicer mechanism design into an operating servicer system. It has increased our confidence in the eventual utility of the space design.

The control system simulated was the Supervisory with Remotely Manned Backup control concept identified in the first IOSS. It was studied in three separate, but related, parts. These are a) Supervisory, b) Manual-Direct, and c) Manual-Augmented. Each of these control modes is described in detail in Chapter VII. Time did not permit evaluation of the Manual Augmented Mode. The axial configuration of the servicer mechanism was represented with module motion during installation and removal in the axial direction.

The results from the simulation include:

a) verification of the utility of the two modes investigated;
b) definition of control and display scaling factors;
c) identification of TV system parameters (lens focal lengths, need for focus adjustment, location, gimballing, etc);
d) identification of visual aids;
e) suitability of the Payload Specialist Station as a control station;
f) adequacy of joint rates and torques;
g) need to redesign interface mechanism guide shapes;
h) adequacy of attach capture volumes;
i) a set of recommended module transfer trajectories; and
j) verification that the selected timelines are suitable.

The approach to the physical simulations involved use of an existing Martin Marietta physical motion generator, control logic systems, and control stations in conjunction with a partial full-scale mockup of a serviceable spacecraft and module stowage rack. Figure VIII-1 shows the interconnection of the major elements. The selected motion generator is the Space Operations

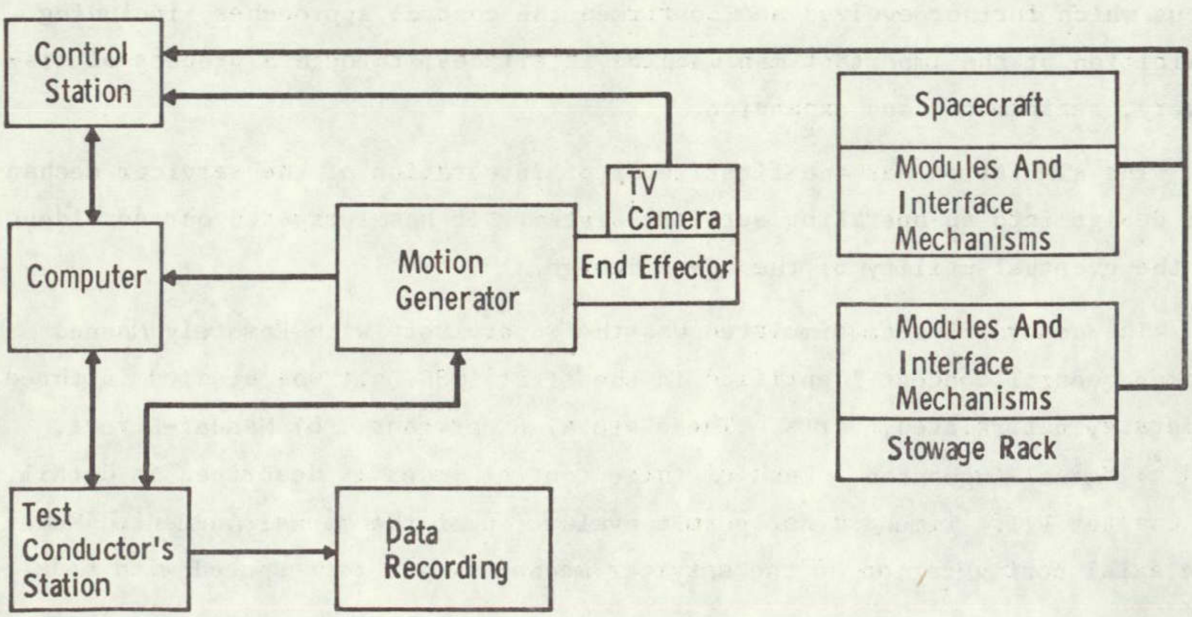

Figure VIII-1 Simulation Elements

Simulator (SOS) which operates in cartesian coordinates and has a large weight capacity. Thus it was not necessary to get involved in counterbalance aspects during the demonstrations. The motion travel of the Space Operations Simulator is such that the activities could be conducted at full scale.

The selected control station was the Payload Specialist Station (PSS) which is located in a mockup of the Orbiter aft flight deck. The PSS is being designed to handle a variety of orbital activities such as on-orbit servicing. The spacecraft and stowage rack partial mockups were made of foam-core over a metal/wood structure at full scale. The metal structure supports the interface mechanisms and the replaceable modules. The side mounting interface mechanism was used. The end-effector and interface mechanisms delivered to MSFC under the first IOSS were used, and a second interface mechanism baseplate receptacle was fabricated so a module could be moved back and forth between the spacecraft and stowage rack. A second removable module location in both the spacecraft and stowage rack was provided so the effect of different module locations could be studied. The test set-up is described in detail in Section A. Design evaluations and conclusions are covered in Section B. Further details on the simulation/demonstration have been covered in a memo, *On-Orbit Servicer Demonstration/Simulation*, March 25, 1977. The memo should be considered as a basic reference for this entire chapter on the demonstration/simulation.

A. TEST DESCRIPTION

The servicer configuration for the simulation/demonstration is illustrated in Figure VIII-2. The simulation is located in the large high-bay Space Operations Simulator (SOS) located in MMC-Denver's General Purpose Laboratory.

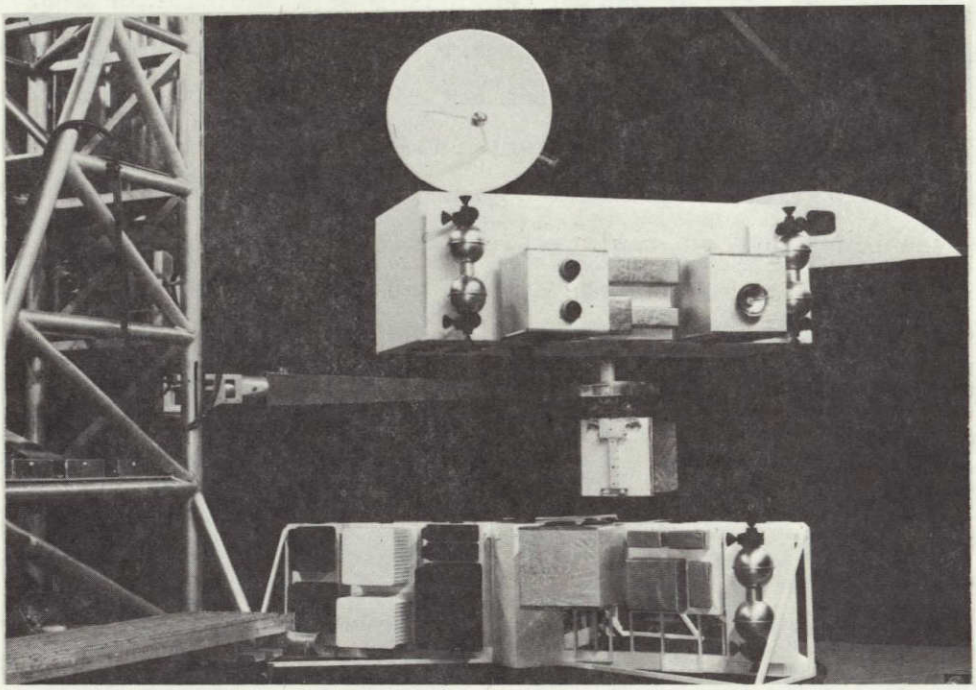

Figure VIII-2 Selected Motion Generator/Mockup Configuration

The servicer's mechanical arm motion is provided in this simulation by an existing multipurpose motion generator capable of three degrees of translational freedom in cartesian coordinates which can be operated independently of the additional three rotational degrees of freedom associated with the servicer forearm representation. The full-scale mockups of the spacecraft and storage racks, which are integral elements of the simulation, are shown at the end of the arm. Of the modules shown on these mockups, one is fully operational for removal and replacement, but it can be moved from either of two locations in the spacecraft to or from two locations in the storage rack.

The mockup of the Shuttle Orbiter Payload Specialist Station can be seen at the far end of the room. It served as the control station for the servicer operation. This is a realistic approach since a potential role for the servicer system is as a fixture in the bay of the Orbiter for low earth orbit payload maintenance. For orientation of the viewer, the Shuttle Orbiter can be thought of as pointing nose forward out of the far end of the room. The servicing mockup then is located in the general vicinity of the center of the Orbiter bay and appears, from the aft facing window of the Payload Specialist Station, much as it would in an operational Orbiter application.

The test setup consists of the elements shown in Figure VIII-3. The

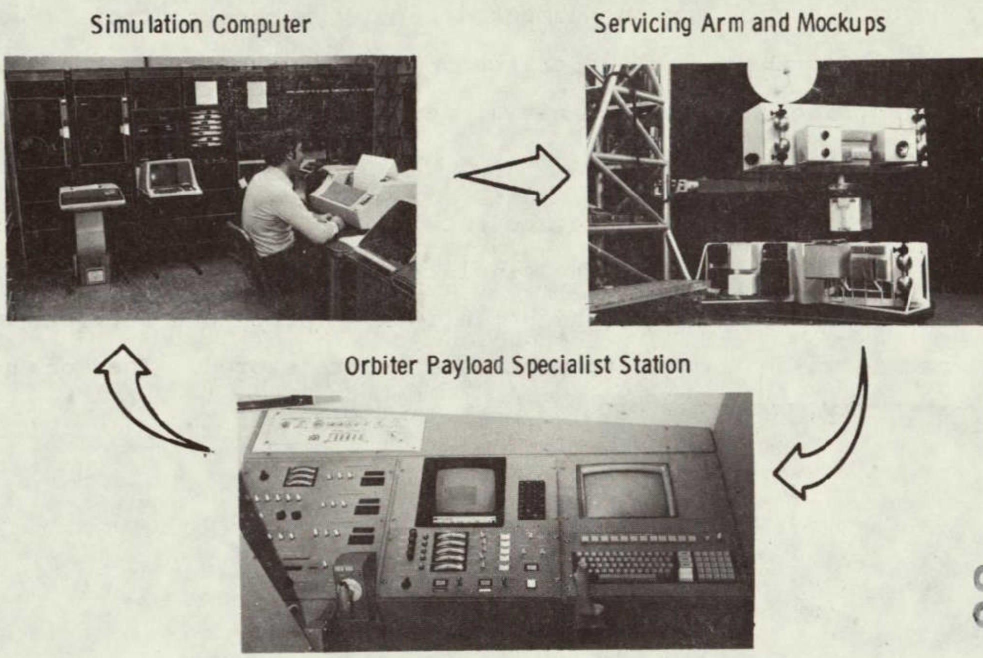

Figure VIII-3 Demonstration/Simulation Signal Flow

servicer operator conducts operations from the control console where his control information is displayed to him via TV, lights, and meters. The control station to be used is a functional mockup of the Orbiter Payload Specialist Station which is described in detail later. The operator initiates control commands just as he would on an operational servicer mission. The control commands are routed to a computer containing the servicer system math model. The control commands cause servicer system motion in the math model. The servicer system motion is transformed into simulator commands via simulation equations. This transformation is necessary to put the servicer motion into the coordinates of the motion generator. The simulator commands cause the motion generator to move the servicer mechanism realistically. The end of arm moves the same as the forearm of the actual mechanism would move. The operator observes the servicer mechanism moving between the spacecraft and stowage rack as it would in a servicing mission.

Three control modes were planned to be run in the MMC Denver simulation -- Supervisory, Manual Direct, and an Augmented form of backup control. Each is described in detail in Chapter VII, Servicer Control System. The Manual Augmented mode was not included in the initial demonstration in February, 1977. Axial exchange only was incorporated.

The full scale mockup of the storage rack and spacecraft used in the simulation is shown in Figure VIII-4. The simulated servicer arm is also shown. The storage rack is the element to which the docking probe and servicer arm are attached and in which the replacement modules are stored. The mockup is

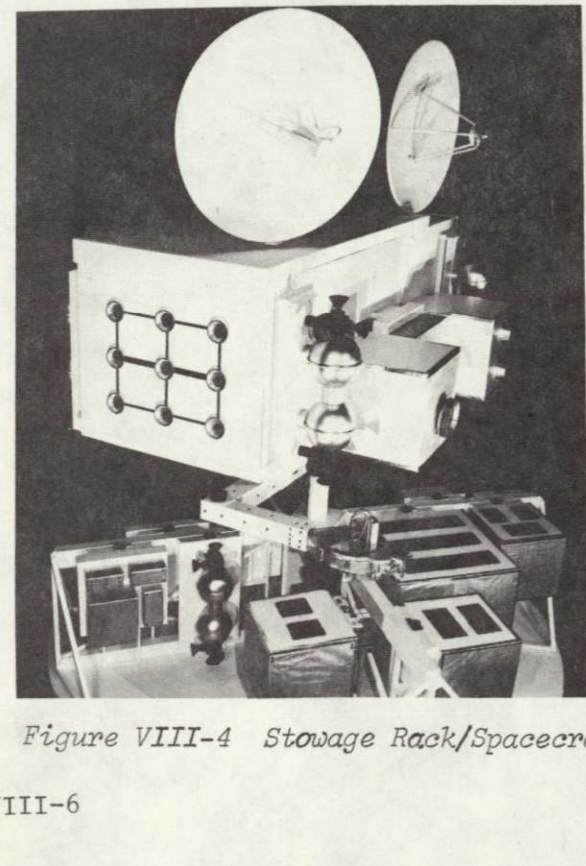

Figure VIII-4 Stowage Rack/Spacecraft Mockup shown with the docking axis vertical. This is the orientation when mounted in the Orbiter bay. The storage rack modules illustrate the variety of types of modules which might occur. They are not intended to represent any specific spacecraft servicing mission. In fact, most missions will have fewer and generally smaller modules. A representation of the largest (40 in. cube) module is shown in the right hand quadrant. The other cubical modules are 24 inches in dimension. The modules which were exchanged in the demonstration are located in the left hand quadrant.

The basic truss structure shown in white is representative of the flight unit stowage rack configuration. However, only three of the normal four trusses have been included. The fourth truss would have inhibited motion generator operations.

The spacecraft mockup is shown at full scale in the docked configuration with the storage rack. Separation of spacecraft to storage rack is the 60 in. recommended in this study. The spacecraft has been configured as a generalized serviceable spacecraft in order to better reflect the spectrum of potentially replaceable modules on a serviceable spacecraft. A variety of shapes and sizes of replaceable modules are shown mounted around and within a core structure. A self-contained propulsion module, incorporating both thrusters and tanks, is shown. In the mockup, only the two module locations in the center of the spacecraft core structure are operationally usable by the servicing arm. The two locations selected represent the minimum radius (closest to the docking probe) and the maximum radius (farthest from the docking probe) that will be encountered in any spacecraft for axial module replacement.

The servicer mechanism in the flight unit or the engineering test unit mounts on the docking probe half way between the stowage rack and spacecraft mockups. The moving base motion generator used in the simulation to generate servicer mechanism motion is an existing item. Basically no changes were required to it. Some modifications were required to the arm, however, to provide the proper degrees of freedom. Figure VIII-5 shows the areas in which the modifications were necessary. The joint at the base of the arm (simulator yaw) was reconfigured to provide movement in a plane parallel to the floor. The arm itself was fabricated for the servicer simulation and is unique to this application. The simulator yaw drive along with the moving base horizontal drives (X and Y) provide the motion equivalent to the servicer mechanism shoulder roll and elbow roll. Thus, the orientation of the arm in the simulation is representative of the actual servicer mechanism forearm. This

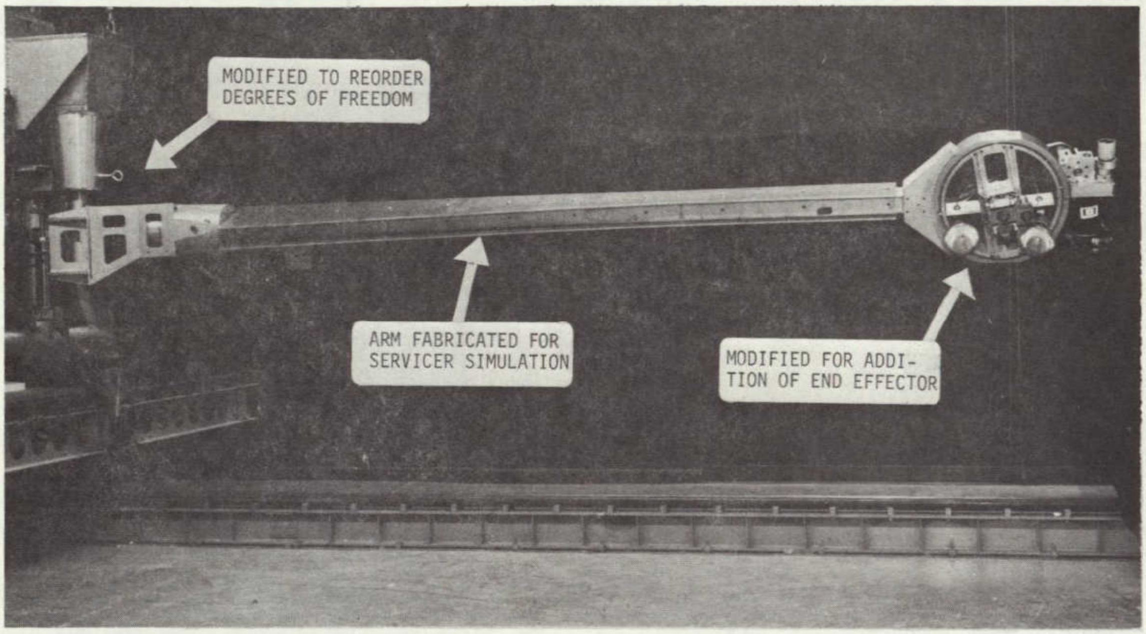

Figure VIII-5 Motion Generator Modifications

can be seen better in Figure VIII-4. The yoke at the end of the arm (Figure VIII-5) was existing but was modified to accommodate the servicer end effector and to permit installation of lights and TV camera. The simulator end effector has two degrees of freedom in the servicer end effector. The simulator arm rotates about itself providing a wrist yaw. The yoke rotates about an axis normal to the face of the yoke providing a wrist roll.

The end of the simulated servicing arm and end effector mounted within it is shown in the closeup photo in Figure VIII-6. The circular ring which supports the end effector and provides the rotations of the end effector is larger than the eventual ETU or a flight design. It is being used to take advantage of the existing mechanism, structure and motor which was designed initially for much heavier loads than this application. It also provides more flexibility for investigation of alternative TV and lighting system locations than can be provided easily with the engineering test unit.

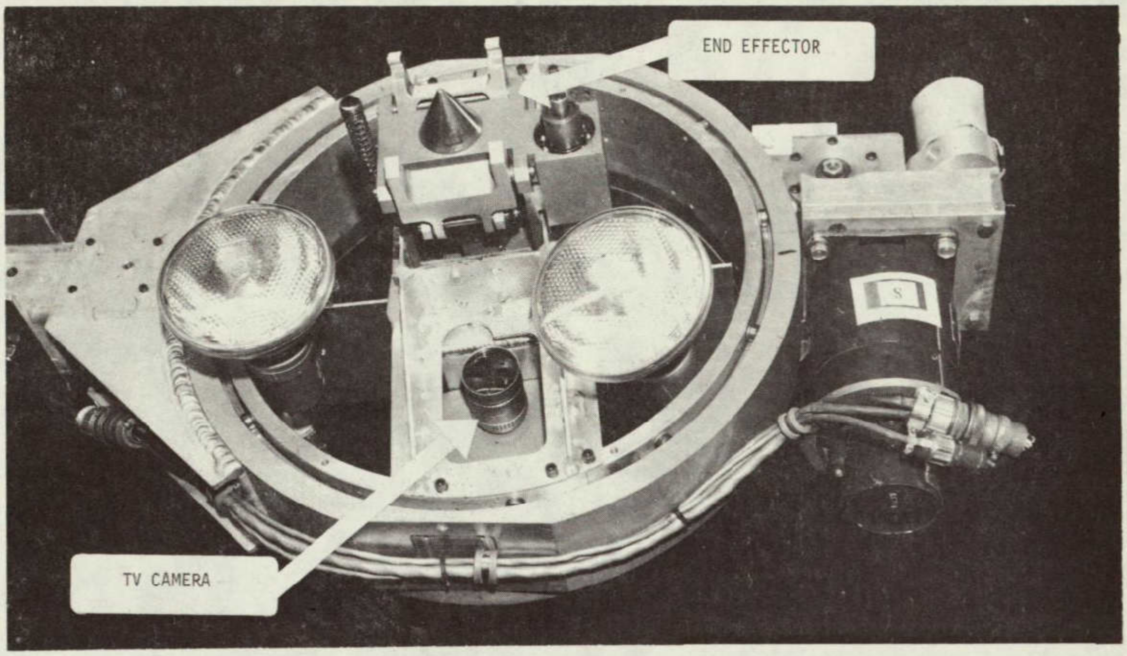

Figure VIII-6 Simulator End Effector

The cone shaped probe and movable jaws on each side accomplish attachment with the module. The slotted drive next to the probe turns a screw drive after engagement to perform latch and unlatch of the module from the interface mechanism receptacle or guide. The receptacles are permanently mounted in the spacecraft and stowage rack.

The end effector discussed previously is shown in Figure VIII-7 attached to the interface mechanism baseplate of a typical module. The simulated servicer mechanism is shown inserting the module (baseplate) into the mating part (baseplate receptacle) in the stowage rack. A 24-in. x 24-in. x 24-in. module is shown.

A functional mockup of the Shuttle Aft Flight Deck has been fabricated (as shown in the Figure VIII-8) at MMC. Its location in the facility was shown previously in Figure VIII-2. A servicer system operator's station has been incorporated.

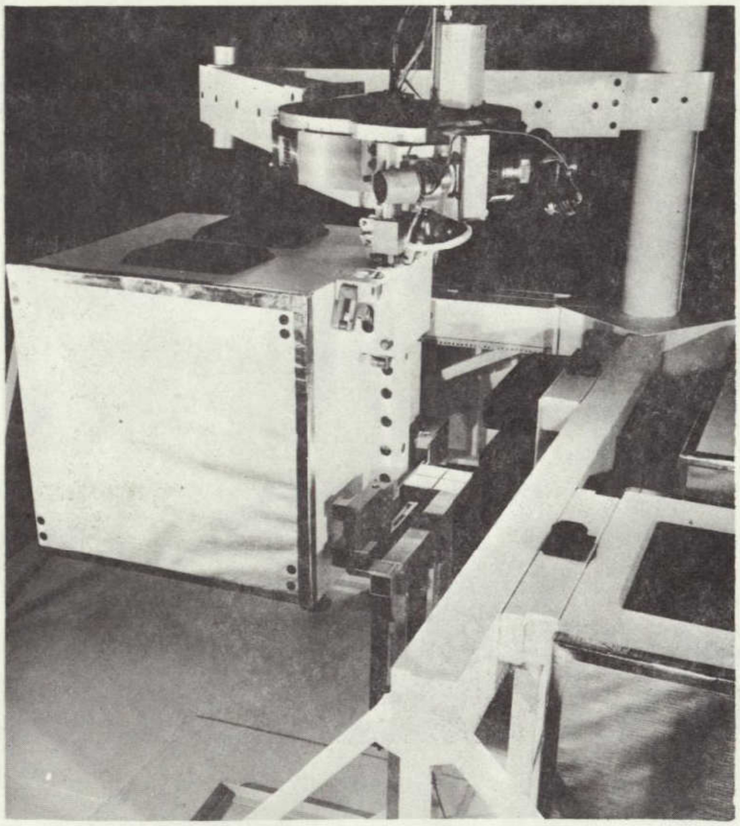

Figure VIII-7 End Effector with Module Attached

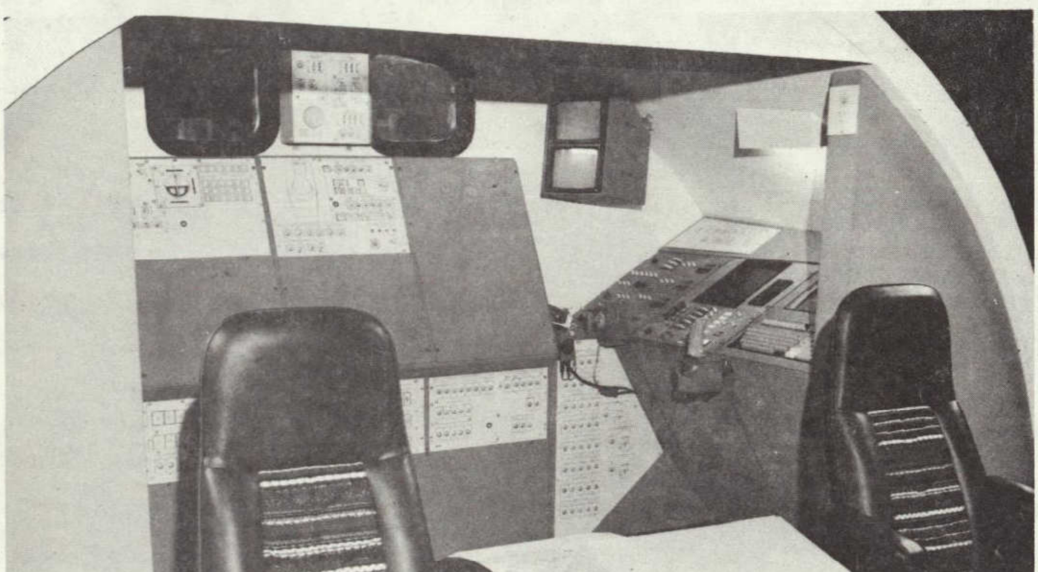

Figure VIII-8 Orbiter Aft Flight Deck Mockup

The servicer control station shown on Figure VIII-9 is located on the right of the aft flight deck mockup shown in Figure VIII-8. This corresponds to the left hand side of the orbiter aft flight deck.

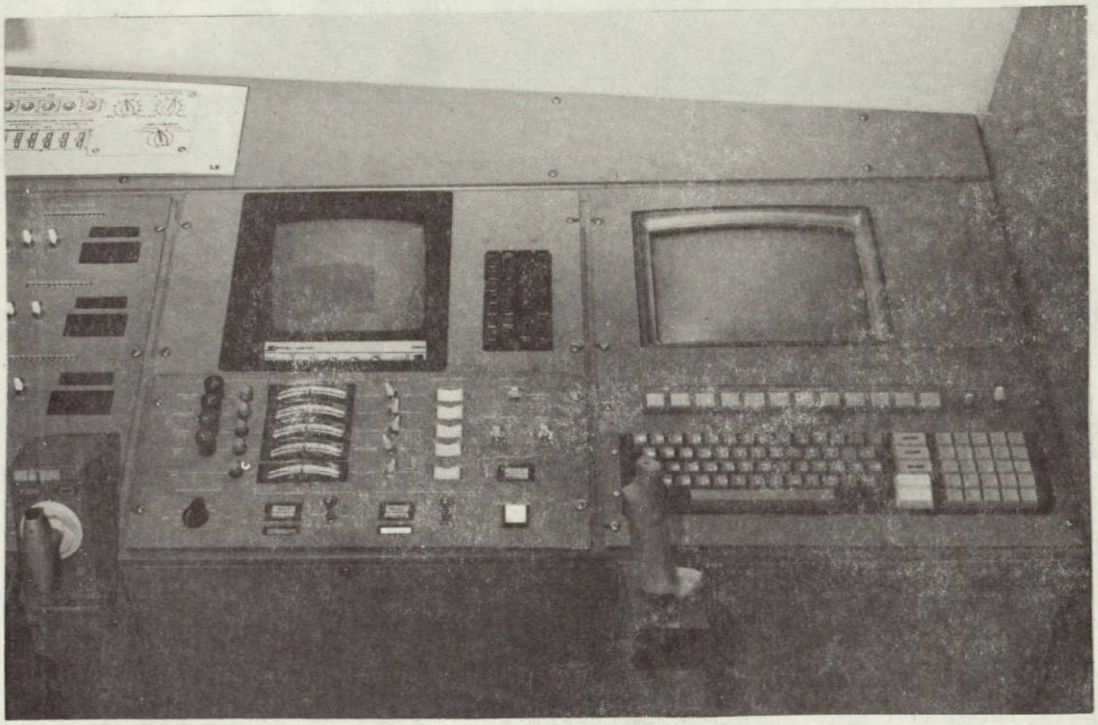

Figure VIII-9 On-orbit Servicer Control Station

The servicer control and display panel in the lower center and the monitors are not mockups but are in fact operational and are integral to the servicer demonstration. The function of the various elements shown at the center and right hand stations are described later as the operations required for each of the three control modes are described in detail. The right-hand CRT screen and keyboard is solely for interface with the computer.

The panel at the left is not a part of the servicer controls or display. The TV above the control panel normally will show the view from the end effector mounted camera. The TV screens at the upper left of Figure VIII-8 are intended to monitor the Orbiter bay with the Orbiter cameras.

The panel shown below the TV monitor on Figure VIII-9 is illustrated in greater detail in Figure VIII-10. The panel contains some controls for basic

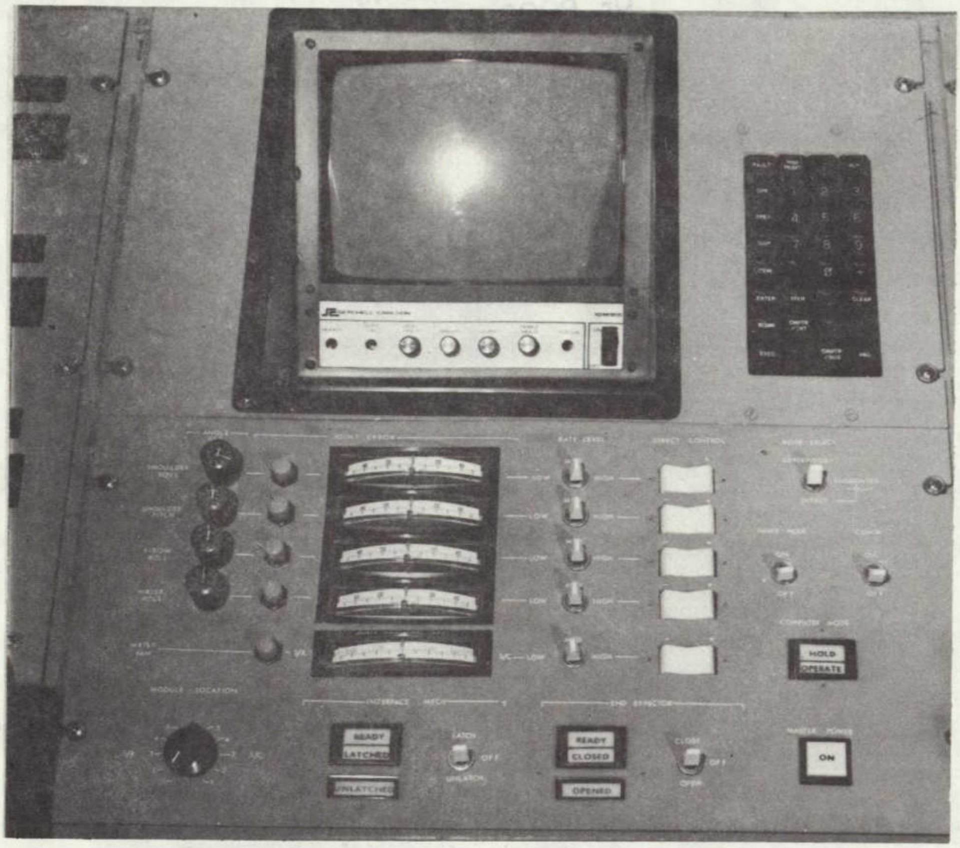

Figure VIII-10 Servicer Controls and Displays Panel

initialization. These include such items as: power on/off, communications on/off, Hawk mode on/off and mode select. The lower part of the panel has one region for end effector control and another region for interface mechanism control. The end effector control consists of status lights (Ready, Closed and Opened), and a momentary switch to drive the jaws closed or opened. The interface mechanism control also consists of status lights (Ready, Latched and Unlatched), and a momentary switch to drive the latching mechanism. There is a module location selection switch in the lower left hand corner. It is used for selecting the module number, and whether the module is located in the spacecraft or stowage rack. The upper part of the panel is used for the Manual Direct control mode. A servicer mechanism joint by joint control is provided by a combination of control inputs and displays which is paralleled for each mechanism joint drive. A row represents the controls and displays for a given drive like the shoulder roll. A desired angle on the joint is set in with the angle set pot on the left. The light and null meter indication represents the joint error which is the difference between the angle set pot value and the actual position of the joint. A joint is driven to the angle set pot value with the white rocker switches. Three levels of drive rate are available.

The alphanumeric display and keyboard as shown in the right hand portion of Figure VIII-9 is typical of most computer interface terminals. It has been incorporated for the Payload Specialist Station design studies being conducted for MSFC. It will eventually be operational with the PDP 1145 computer, however, the initial simulations utilized an existing 1145 terminal--INFOTON--stationed directly beside the unit shown.

The CRT screen and keyboard is the prime control element in the Supervisory control mode. Status and requests for commands are displayed on the screen. Commands to perform actions are inserted in the form of words typed in on the keyboard. These commands have been designed to be a minimum and simple in form to enhance operator understanding and simplify training.

The PDP 1145 computer being used for the servicer simulations is pictured in Figure VIII-11. The machine is a general purpose facility tool. The PDP-1145 is a 16-bit computer representing the large computer end of the PDP-11 family of computers. It is designed for high-speed real-time applications and for large multi-user, multi-task applications requiring up to 124K words of addressable memory space. Among its major features are a fast central processor with choices of semi-conductor and core memory, an advanced Floating Point Processor, and a sophisticated memory management scheme.

Some of the peripheral equipment included in the PDP 1145 installation above are:

2 RK05 Disk Drives
2 Seven Track Tape Drives
2 Nine Track Tape Drives
Paper Tape Reader
INFOTON (Keyboard and CRT)

Line Printer
Card Reader
RT-11, RSX-11D and FORTRAN IV Plus Operating Systems

Figure VIII-11 Simulation Computer (PDP-1145) Facility

The block diagram in Figure VIII-12 depicts the interconnection of the basic elements of the MMC Denver simulation, most of which have been discussed previously. The computer interface electronics performs the Input/Output function between the computer, controls, displays and the motion generator. It should be noted that this configuration includes basically the same elements as the proposed flight configuration. The only difference is that the motion generator and its control panel would be replaced with the flight servicer and its electronics. Subsequent simulation/demonstrations will incorporate the ETU and its electronics, bringing the configuration that much closer to a flight design.

A PDP 1145 is the computer being employed and is a laboratory facility item, resulting in a low cost operation. This computer has been used for a simulation of the Orbiter Payload Specialist Station configured earlier for a study with MSFC.

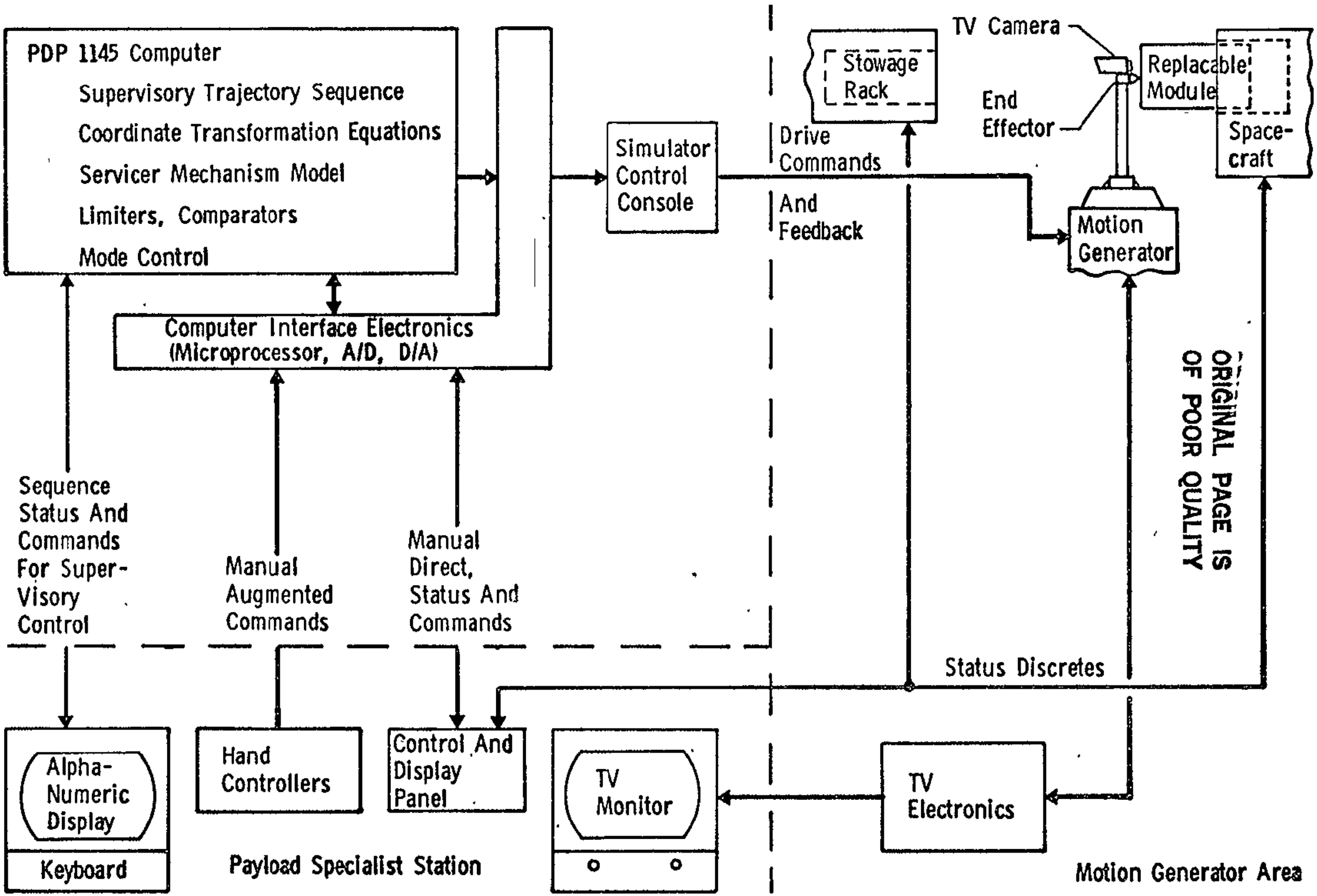

Figure VIII-12 MMC - Denver Simulation Block Diagram

B. DESIGN EVALUATIONS AND CONCLUSIONS

This section contains the design evaluations performed during the simulation and the resultant conclusions. The areas covered are:

1) End Effector Attachment Status Signals;
2) Interface Mechanism Latch/Unlatch Status Signals;
3) End Effector TV Camera Location;
4) Latch/Unlatch Mechanism;
5) Visual Aids;
6) Capture and Alignment Conditions of End Effector to Interface Mechanism;
7) Connector Alignment and Mating;
8) Module Transfer Trajectory;
9) Timelines and Velocity and Acceleration Levels; and
10) Control Modes.

The conclusions are presented in the above order because it was the order in which they were investigated in the simulation. The order represents a very logical build-up of information for each succeeding conclusion. The simulation represents the first level of integration of the servicer mechanism design into an operating servicer system. The simulation provided a focus for effective study of the above listed areas and the total servicing system at a very logical time in the development of the servicer system. The value gained and to be gained is not easily assessed, but it certainly has proven to be very significant already.

1. End Effector Attachment Status Signals

Three status indications were mechanized on the end effector: jaws opened, jaws closed and ready for jaw actuation. These status signals aided in the attachment of the end effector to the interface mechanism.

The end effector jaws opened and jaws closed status signals were generated by microswitches actuated by cam surfaces attached to the jaws. This configuration has proved to be entirely satisfactory during the simulation testing. The setting of the microswitches to actuate at the end of the jaw travel is critical. However, the microswitch positions were set-up only once and did not have to be reset during the evaluation period. The jaw mechanical motion and available surfaces make the mounting and operating of micro-switches very easy.

The end effector ready status sensor is not as readily configured as the jaws opened and closed sensors. An acceptable configuration for lab operation was evolved, but different approaches will have to be considered for the space application. The configuration used is shown in Part A of Figure VIII-13. A push type button switch was mounted on the end of a coil spring. The switch was set to be actuated just before the capture cone on the end effector was bottomed out in it's mating part on the interface mechanism.

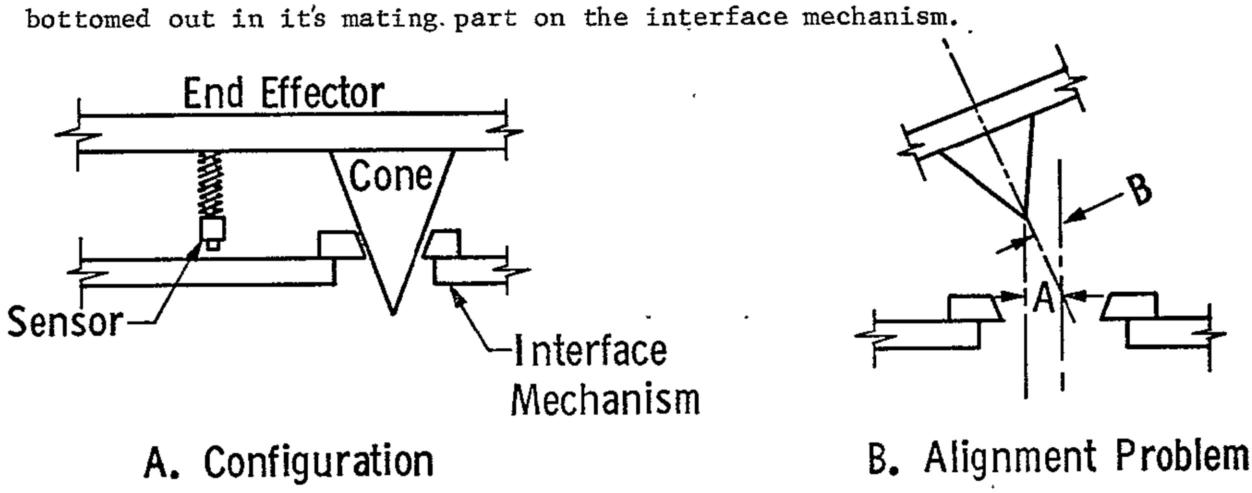

Figure VIII-13 End Effector Ready Status Sensor

The reasons for mounting the switch on a spring are obvious once the relative alignment errors between the end effector cone and the mating part on the interface mechanism are reviewed. The alignment conditions are shown in Part B of Figure VIII-13. The cone can be misaligned both in displacement and angle. In the figure the displacement error is labeled A and the angular error is labeled B. The present capture capability is ±3/4 inches for the displacement error. The capture capability for the angular error has not been evaluated. The spring mounting of the ready switch requires a mounting support that could give when angular misalignment existed prior to the cone bottoming out. As was stated this worked satisfactorily for the simulation tests.

The manner in which the status indications should be used in each mode is another consideration. The status signals were used as control indicators for the manual direct mode. After the operator aligned the end effector cone properly according to the known pot settings, he drove the end effector cone into contact with the mating part on the interface mechanism using the shoulder pitch drive (U) only. He drove it until the ready light came on. Then he actuated the jaw closed control switch on the C&D panel.

However, for this first physical demonstration it was decided that in the Supervisory mode, the driving in motion would not be terminated based on getting the ready indication. Instead, it was decided to determine the desired inward travel as a calibration value and store it in the computer. This was done, and it worked satisfactorily. The ready light always came on during the runs in Supervisory mode evidencing that it could be calibrated for the simulation set up. The Supervisory mode steps were not interlocked with the jaw actuation. The alphanumerics display called for performing and checking for completion of the jaw actuation. The manner in which the status signals should be incorporated in the Supervisory mode for the space case is still under consideration.

2. Interface Mechanism Latch/Unlatch Status Signals

Three status indications were mechanized on the interface mechanism: interface mechanism latched, interface mechanism unlatched, and ready for latching actuation. These status signals aided in the fastening of the interface mechanism baseplate to the baseplate receptacle.

The interface mechanism unlatched and ready for latching status signals were generated by microswitches mounted to the track of the baseplate receptacle. The second set of guide rollers on the baseplate receptacle were the cam surfaces which actuated the microswitches. The configuration was adequate for the demonstration. However, the location of the ready for latching microswitch is very critical. It is desirable to have the microswitch mounted normal to the direction of travel instead of along it. This could be done if the microswitch were located in the region of one of the latching/unlatching drive arms. The interface mechanism unlatch status microswitch when mounted on the track has the same problems. It also should be located by one of the drive arms. The microswitches were not mounted where they could be actuated by the latch/unlatch drive arm because it would have required considerable machining. Also, it was concluded later in the simulation that when using the tracks for mounting the microswitches, they should be mounted further along the tracks so the first set of guide rollers are the actuators.

The interface mechanism latched status signal was obtained from the homing of the electrical connector mounted at the end of the interface mechanism. Continuity through the connector was used to turn on a light on the control panel. This is an indication that latch up has been completed. Also an indication of a functioning connector will probably be required as a basic post servicing check. However, this type of latch signal will not be adequate in the space design for motor turn-off as it occurs at first pin contact and not at the end of the connector travel.

The status signals were used as control indicators for the manual direct mode. The ready to latch light was used to indicate to the operator when to stop driving shoulder pitch and actuate the latch drive switch. The latch status signal was used to turn on a light which indicated to the operator to stop the latching operation. The unlatch status signal was used to turn on a light which indicated to the operator when to shut off the unlatch motor drive. He then could initiate a shoulder pitch to remove the module. In all modes a compensation term for the shoulder pitch drive (U) was initiated in the computer whenever the latch or unlatch switches were actuated. This term compensated for the 1 and 3/4 inches of travel that the baseplate goes through during the latch and unlatch operation.

However, for this first physical demonstration it was decided that in the Supervisory mode, the driving in motion would not be terminated based on getting the ready indication. Instead, it was decided to determine the desired inward travel as a calibration value and store it in the computer. This was done, and it worked satisfactorily. The ready light did not come on consistently indicating that it is difficult to actuate the microswitch for a given inward travel of the baseplate. The Supervisory mode steps were interlocked partially to the latch/unlatch operation. The program could not be advanced unless the latch/unlatch action was initiated. The manner in which the status signals should be incorporated in the Supervisory mode for the space case is still under consideration.

3. End Effector TV Camera Location

During the simulation the end effector TV camera scene was displayed on a monitor in the servicer control panel. For the Manual Augmented control mode the TV scene is the prime source of control cues for the servicer operator. However, in the Supervisory and Manual Direct control modes the TV visual scene is used as monitor type information. By observing the TV scene as steps in the module exchange are being performed, the operator can gain assurance that the steps were completed correctly.

After reviewing the existing end effector TV camera location requirements and the additional thoughts generated while setting up the simulation, the following list of requirements was defined.

End Effector TV Camera Location Requirements

1) Same visual aid for the capture and attachment of both the end effector and the interface mechanism
2) Keep camera as close to the center line of the end effector cone as possible to minimize error differences between the sensor (TV camera) and the parts being aligned (end effector and interface mechanism)
3) Keep camera as close to wrist roll axis as possible to minimize apparent translational motion on TV monitor when wrist roll moves
4) Have end effector jaws in opened and closed positions in the camera field of view
5) Have end of baseplate in view as it enters baseplate receptacle
6) Perform #5 for interface mechanism lengths up to 40 inches
7) Use same type of visual aid(s) for stowage rack and spacecraft
8) Have latch/unlatch arm linkage in latched and unlatched positions in the camera field of view

The preceding servicer system requirements are desirable. They do not all drive the TV camera location to a single location. In addition, the following servicer system mechanization conditions were considered.

<u>Servicer System Mechanization Conditions</u>

1) Physical size of wrist roll drive
2) Physical size of camera
3) Minimum focal length of TV camera lens
4) Integration with the TV camera lights

There are three regions to be considered for locating the end effector TV camera. These regions surround the area taken up by the end effector itself as shown in Figure VIII-14. Region A is where the module is located.

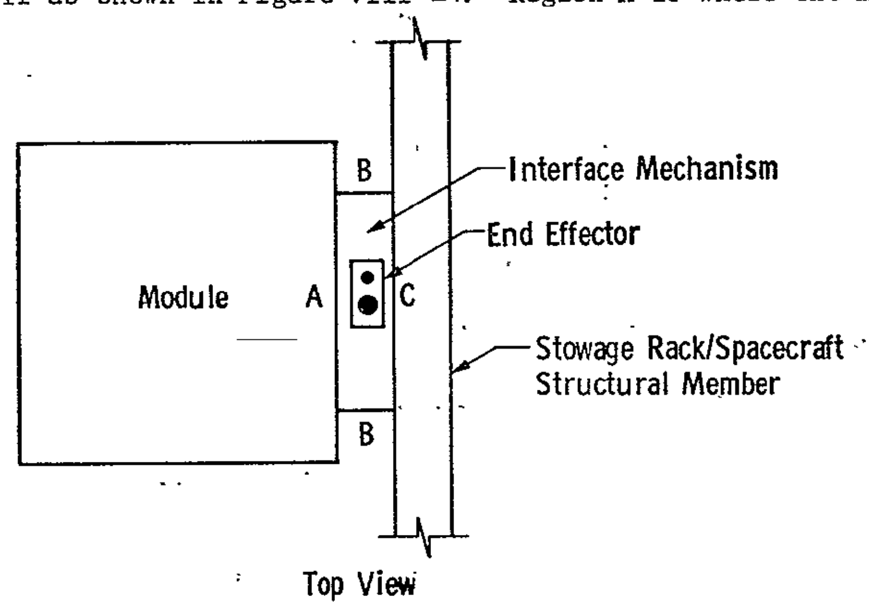

Figure VIII-14 End Effector TV Camera Location Options

Region B is on either side of the interface mechanism, and Region C is the spacecraft/stowage rack structural member supporting the interface mechanism. Table VIII-1 summarizes the results of the end effector TV camera location trade-off. The ability of each of three regions to satisfy the previously listed requirements was evaluated.

Requirement #1 says that the same visual aid should be used for capture and attachment of the end effector and the interface mechanism. In Region A this requirement would necessitate a slot down through the module because the visual aid would have to be mounted on the baseplate receptacle. This would place the visual aid a long distance away from the camera (up to 40 inches). The requirement can easily be met for Regions B & C. However, in Region C there would be a separation limit (camera lens to visual aid) in order that holes would not have to be cut in the structural member.

Table VIII-1 End Effector TV Camera Location Tradeoff Results

Requirements	Regions		
	A	B	C
1	No	Yes	Yes
2	Good	Poor	Good
3	Good	Poor	Good
4	Yes	No	Yes
5	No	No	Yes
6	No	No	Yes
7	Yes	Yes	Yes
8	No	No	No
(See Page VIII - 20 For Definition Of Each Requirement)			

The TV camera was located 4.2 inches to the side of the end effector center line as shown in Figure VIII-15. The separation distance between the lens and the visual aid was 5.5 inches. A TV camera available in the lab facility was used. The TV camera characteristics are shown in Table VIII-2. Obviously, the physical dimensions are not representative of available space qualified TV cameras.

The TV camera location in the simulation did prove to satisfy all the requirements except #8. "Feeler" or "flag" type visual aids which the linkages operate at the end positions could help. Also, the opened and closed positions for the end effector jaws could not be identified as positively as might be desired. However, jaw motion could be readily observed.

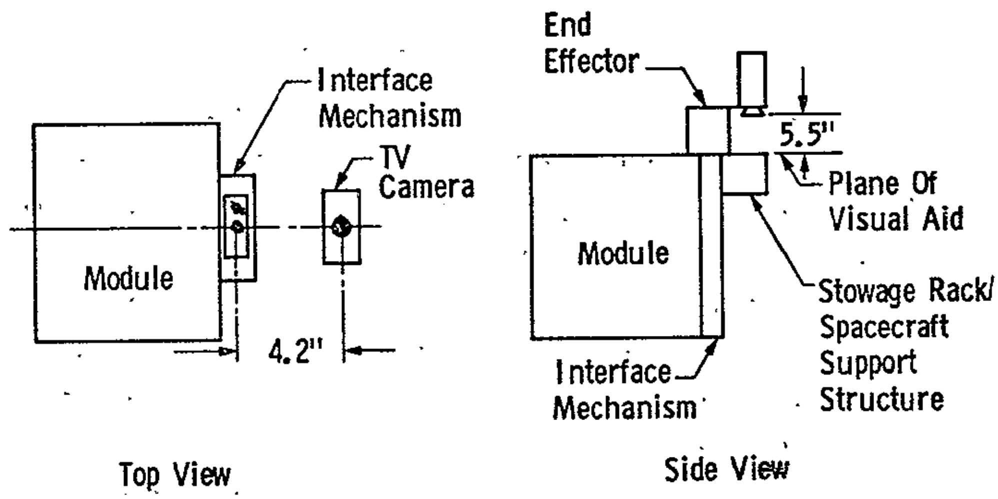

Figure VIII-15 End Effector TV Camera Location Geometry

Table VIII-2 Simulation TV Camera Characteristics

Camera	Lens
Manufacturer: RCA Size: 9.5" X 4.5" X 7.7" Weight: 3-1/2 Lb. Power: 14 Watts Resolution: 500 Lines	Manufacturer: Angenioux Focal Length: 10 mm F/1.8

4. Latch/Unlatch Mechanism

Early in the simulation setup some difficulty was encountered in engagement of the latch drive. The condition was cured by cutting a chamfer on the top edge of the hole in the interface mechanism. This provides a greater capture capability of the drive shaft under alignment errors. Also, the slot in the drive shaft was opened up somewhat. After these two minor changes the engagement of the latch drive was performed consistently for all runs.

A latch/unlatch compensation had to be added to the computer math model to accommodate the 1 and 3/4 inch motion of the interface mechanism baseplate as a latch/unlatch is performed. The compensation consisted of a term added to the shoulder pitch (U) drive command. An integration was initiated based on actuating the latch/unlatch drive switch. This integration was limited to the 1 and 3/4 inch maximum value. The compensation term was needed in the simulation because the motion generator is not back driveable. In the ETU and space design the compensation will be implemented because a back drive capability is not planned for axial module exchange. However, the ETU stiffness may be such that the compensation can be deleted later.

5. Visual Aids

The Supervisory and Manual Direct control modes do not require visual cues from the end-effector TV camera. The Manual Augmented mode is baselined to use the video scene as its prime source of control cues. However, under current thinking, the servicer operator can use the video scene as supplementary cues for the Supervisory and Manual Direct control modes. Because of this a preliminary investigation of visual aids was performed.

The type of visual aid to be used is obviously dependent upon the control cues (errors) to be determined from the visual aid. The first major separation of control cues is between translational ones and rotational ones. The translation and rotational effects are not easily separated if each has three components to control. The design approach used for the 5 DOF servicer mechanism has considered this factor. The servicer mechanism design requires only one final alignment in rotation to be made. This is wrist roll. There is no wrist pitch in the 5 DOF servicer condiguration. The wrist yaw which is used to flip the module will have very accurately calibrated end points. Thus, the need for correcting wrist yaw based on visual cues is eliminated.

The control cues required reduce to three degrees of translation and one rotation. The rotation is about an axis normal to the end surfaces of the stowage rack and spacecraft.

Since the control cues were reduced to three translations and one rotation, a very simple visual aid can be used. All that is needed as a target is a cross within a circle (Figure VIII-16). The reticle on the TV monitor would consist of a cross with ranging circles. Having only one rotation to contend with decouples rotation from translation. For translation the operator flies or evaluates errors based on the relative separation between the center of the reticle cross hairs and the target's cross as shown in Figure VIII-16, Part C. In the other degree of translational freedom the ranging circles are used. A rotational error would result in the view shown in Part D of Figure VIII-16. This error can be corrected by driving the wrist roll.

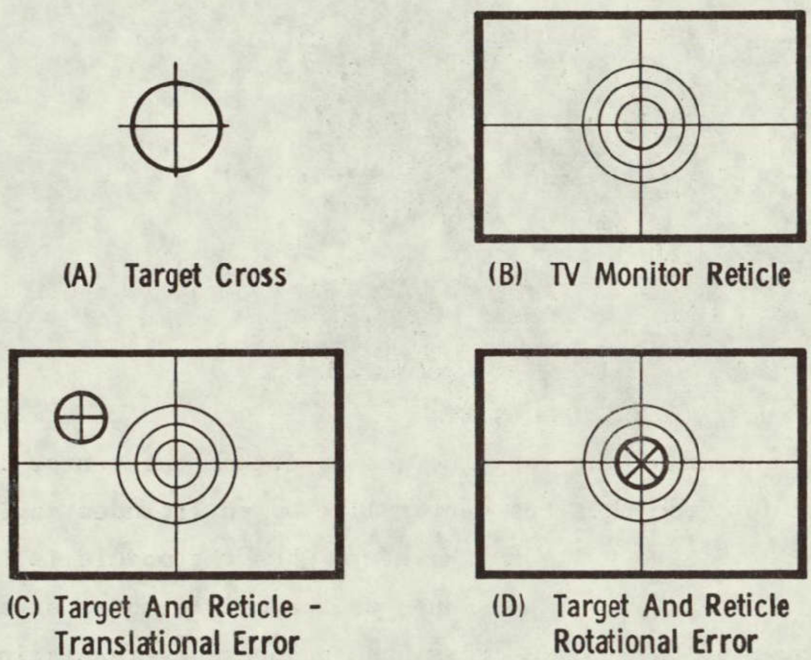

Figure VIII-16 End Effector/Latch Visual Aid

The visual aid set up used in the simulation is shown in Figure VIII-17. The visual aid is not mounted on the stowage support member because of the manner in which the baseplate receptacle had to be mounted for the simulation. A cross with no range circle was used. This visual aid proved satisfactory for monitoring purposes in the Supervisory and Manual Direct control modes for attaching the end effector to the interface mechanism. A problem arose using the same visual aid during the module insertion operation. For module insertion the TV camera is used to view the target cross shortly before (2 inch standoff) the baseplate enters the baseplate receptacle. The view distance

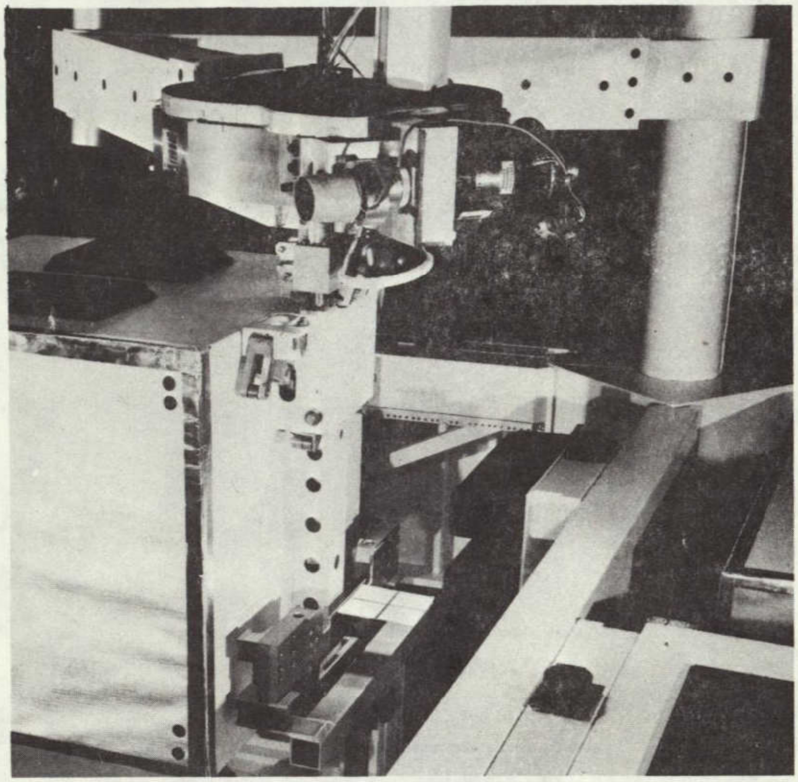

Figure VIII-17 Visual Aid in Simulation

is 30 inches. The center of the target cross is not on the center line of the camera lens, it is off from the lens center line by an included angle (α_1) as shown in Case 1 of Figure VIII-18. However, when the module is inserted as in Case 2 the included angle is α_2. Thus, as the module moves straight in as it should, the center of the target cross has an apparent translational motion on the monitor. This can be handled by marking the start and end points on the monitor's reticle or aligning the lens center line with the target cross. These possibilities are being considered.

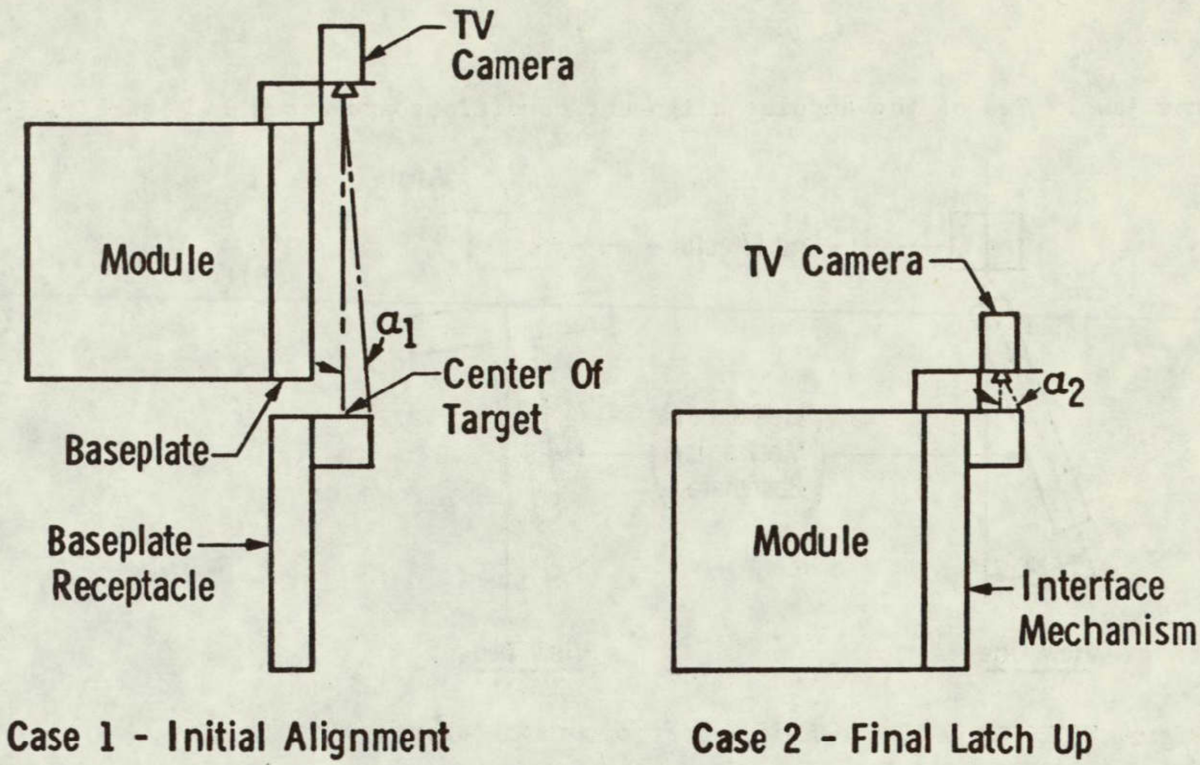

Figure VIII-18 TV Camera - Visual Aid Geometry

6. <u>Capture and Alignment Conditions of End Effector to Interface Mechanism</u>

The ability of the end effector to capture the interface mechanism was repeatedly tested during the simulation. The accuracy of the motion generator was such that the end effector cone was positioned across the total $\pm 3/4$ inch translational capture distance of the cone's mating part on the interface mechanism. The accuracy to which the attitude of the end effector was positioned was sufficient for capture but very difficult to measure. Once the interface mechanism was mounted on rubber pads, capture including clamping of the jaws to the jaw plate on the interface mechanism was consistently accomplished.

One additional alignment problem arose. With jaws fully clamped the angular error between the end effector and the interface mechanism baseplate was too large to allow the baseplate to be captured by the baseplate receptacle during a module insertion. Tests were run to ascertain the angular accuracy to which the end effector and interface mechanism were aligned after clamping of the jaws. Two of the angular alignment conditions are shown in Figure VIII-19.

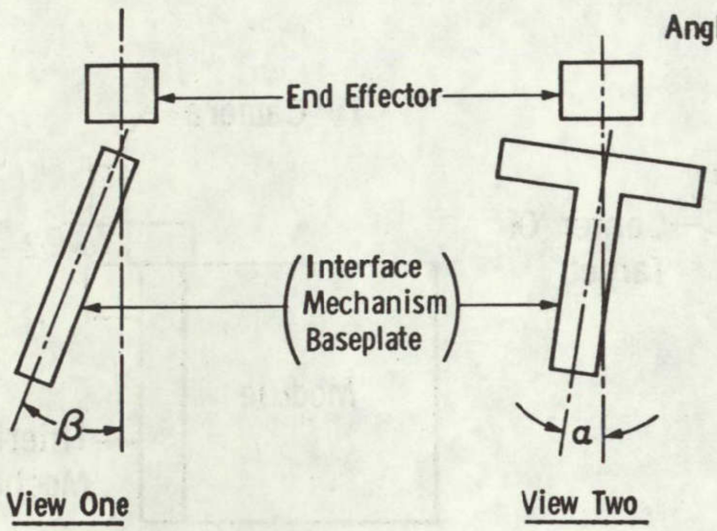

Figure VIII-19 Alignment of End Effector to Interface Mechanism

View one shows the angular error (β) that proved to be the problem. The alignment accuracy for β was ±1.8 degrees which results in a ±3/4 inch displacement error at 26 inches. The α angle error was always acceptable. The two wrist angular errors were under ±1/2 degree and appeared acceptable for the capturing capability.

The homing of the jaws at the end of the clamping action was reviewed and reevaluated. It was determined that the jaw mechanism could be allowed to drive closer to its over center point. It was felt that this should provide better alignment. This was accomplished by adjusting shims under the jaw mechanism. The β angular error was reduced to ±0.5 degrees (±1/4 inch at the tip) which was acceptable for the simulation. After this one change was made a module could be inserted consistently.

7. Connector Alignment and Mating

The engaging/disengaging of a space qualified connector (Type 40M39569) was evaluated during the simulation runs. The connector engaged consistently during all the runs. Mating of the connector was verified by a continuity check which turned on the interface mechanism latched light on the controls and displays panel.

The only modification made to the connector was to remove the locking ring which requires a rotation for locking. Figure VIII-20 (arrow) shows how the connector was mounted on the interface mechanism baseplate and baseplate receptacle. Both halves were hard mounted with the male part on the baseplate receptacle.

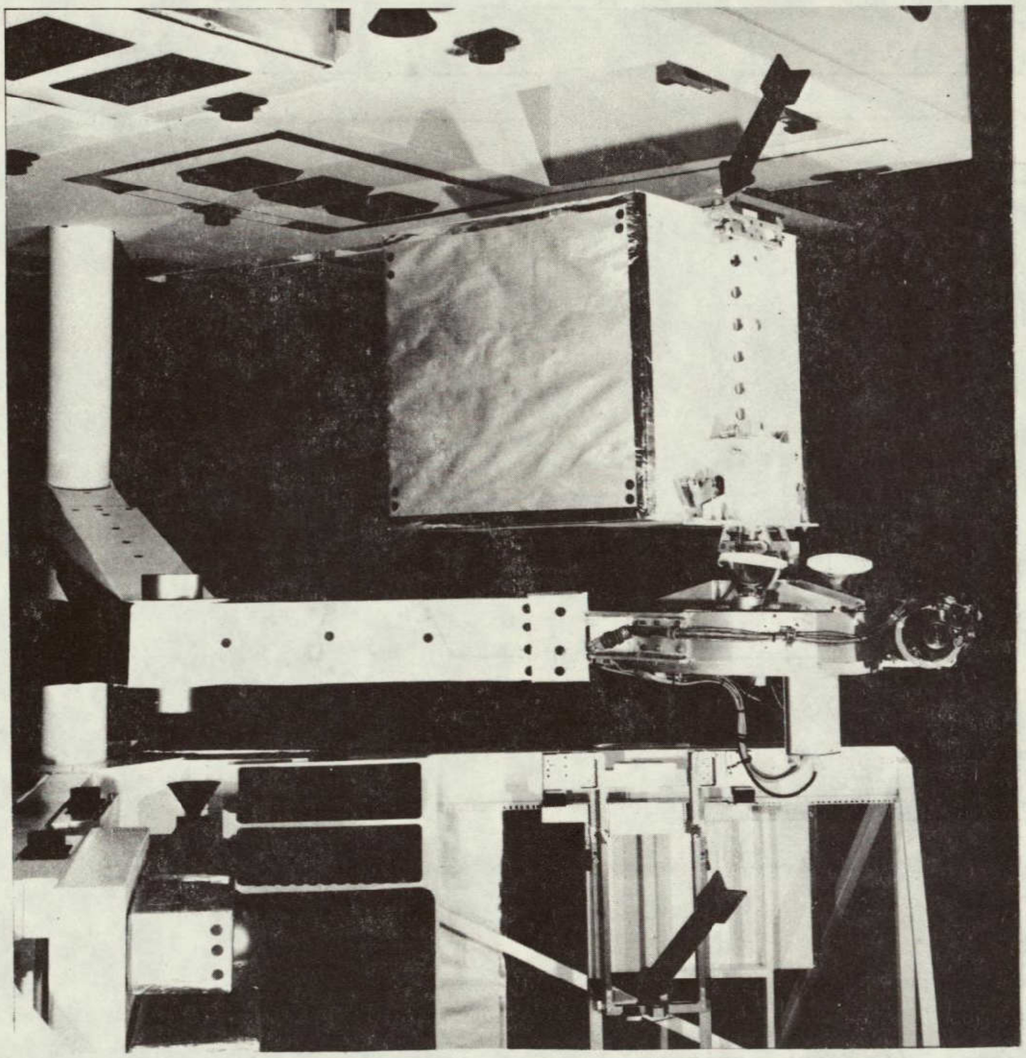

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Figure VIII-20 Interface Mechanism Connector Mounting

The interface mechanism design provides a fine alignment over the last 1.5 inches of travel prior to alignment pin engagement (Figure VIII-21). The alignment pins have a 0.5 inch travel. The connector is mated and homed over the last 1/4 inch of travel. The final alignment of the interface mechanism realized from the 3 alignment pins is sufficiently accurate to mate the connector. The connector does have some small amount of capture/alignment capability. The tips of the pins are rounded off, and the holes in the female connector are chamfered. This along with mounting the connector pins in an elastomer does aid in the mating of the connector.

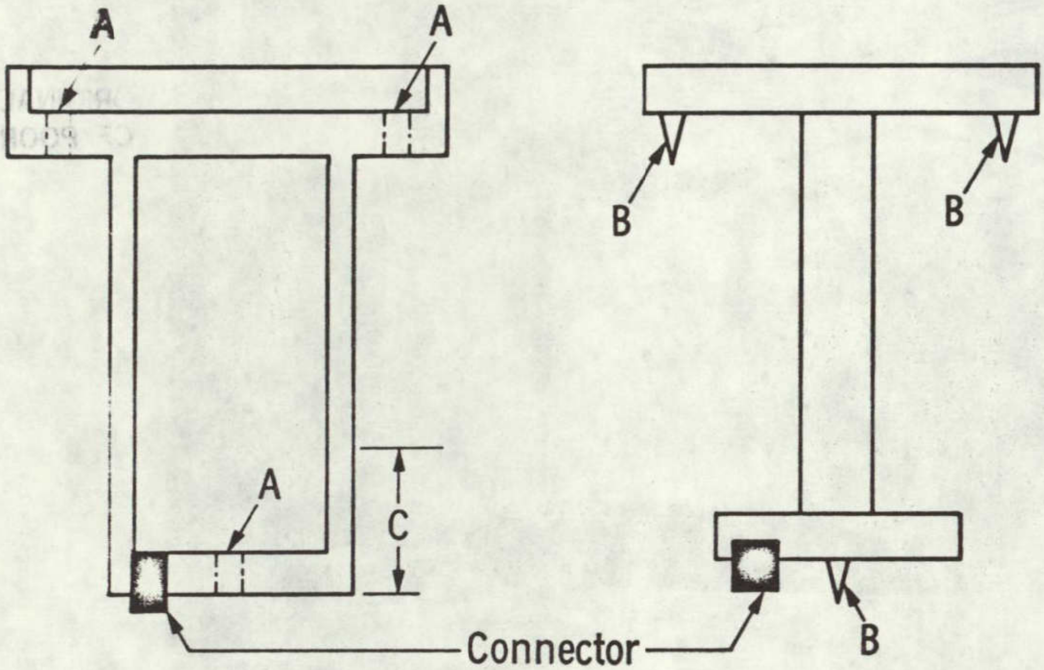

Figure VIII-21 Connector Alignment

8. Module Transfer Trajectory

Selection of the module transfer trajectory was influenced by three factors:

- sequential or compound control
- module flip region
- utilization of a rest position

Each of these factors will be defined as they are discussed.

Sequential control versus compound control was considered. Sequential control means control of one degree of freedom at a time and compound control implies motion of more than one degree of freedom at a time. Simplicity is the important driver in the sequential versus compound control trade-off. A variable that must be considered is the control mode: Supervisory, Manual Direct and Manual Augmented. Simplicity can result in different effects in each of the control modes. In the Supervisory mode it was concluded that simplicity dictates performing the module transfer using the sequential control approach. Simplicity has several factors. Programming of the transfer trajectory considering hazard avoidance is much simpler. Operator training is simpler. Ability of the operator to monitor the controls and display panel for assurance of correct servicer mechanism functioning is much simpler. Sequential control of the module transfer trajectory was verified in the simulation runs as a logical, easily performed and monitored approach. The Manual Direct mode by definition controls one joint at a time. Thus, it directly used a sequential approach for a module transfer trajectory. The Manual Augmented control mode has a different implication based on simplicity. The ability of the servicer operator to observe visual cues on a TV monitor and operate hand controllers in a coordinated manner is the key to simplicity. Thus, this dictates compound control for some of the transfer trajectory segment motions.

The region for the module flip has several factors associated with it. The major consideration is to simplify the hazard avoidance problem. This directly indicates the module flip should be performed outside the spacecraft/stowage rack envelope and not in the region between them. Operator training and monitoring of the flip are also simplified. One operational need requires the flip be done for a few spacecraft in the region between the spacecraft and stowage rack. This requirement is the desire for servicing some spacecraft down in the Orbiter bay, but even in this case there are regions above the bay where the flip could be performed outside the spacecraft/stowage rack envelope.

Utilization of a rest position between module transfer trajectories can affect the number of steps and the time somewhat. The rest position is a reference position that the servicer mechanism is taken back to after completing a trajectory from the stowage rack to the spacecraft or vice versa. As such it also becomes the starting position for the next trajectory. Two rest positions are used which are identical for all degrees of freedom except for wrist yaw (flip angle). For one rest position, the end effector faces the spacecraft, and for the other, it faces the stowage rack. Programming is simplified by this approach as well as operator training. The rest position approach was used in the simulation and proved to be an effective manner of operating.

Another module transfer trajectory step was introduced in the simulation runs. A 2 inch stand-off before end effector attachment and module insertion was used to aid in evaluating the next critical step. This turned out to be very advantageous and effective.

9. Timelines and Velocity and Acceleration Levels

During the simulation runs it was established that an average of 18 minutes (supervisory mode) was required to remove a failed module from the spacecraft and replace it with a new module from the stowage rack. This is considerably longer than the 9.2 minutes (see *Functional Hard Mockup Demonstration Plan*, August 1976, page III-3) estimated previously for several reasons. It was decided prior to the simulation that initially the module transfer should be performed at angular joint rates which were approximately one half the values for the space design. It was felt that this was safer during the "get acquainted" phase. However, time never permitted making runs at the space design values. This would not make a 2 to 1 difference because of the time between steps. Other factors affecting the module exchange time are those discussed previously in the module transfer trajectory section. The earlier timeline estimates were based on compound joint control and no rest position. These two factors increase the number of steps and the required time. A logical estimate of the module exchange time at space design rates based on the simulation runs and related time factors would be 12 to 14 minutes per complete module change out. This would be performed in the Supervisory mode.

Three rate (velocity) levels were available for control in the Manual Direct mode.

Low - 1/2 degrees/sec, Medium - 2 degrees/sec, High - 4 degrees/sec. During the simulation runs the three rate levels appeared satisfactory for most steps in the module transfer. The detailed sequence of displays and commands representing a complete module transfer (which is half of a total exchange) is contained in Appendix C of *On-Orbit Servicer Demonstration/Simulation Report*, March 25, 1977.

10. <u>Control Mode Evaluation</u>

In the control systems description of Chapter VII, three modes of control were proposed for the servicer system. Two of these were implemented in the simulation demonstration.

Some observations and conclusions that were apparent during the servicing demonstrations are discussed below for each of the modes evaluated.

a) <u>Supervisory Mode</u> - This mode, being the normal mode of control, was utilized extensively not only by the servicer study team members but also by a number of relatively "uninitiated" visitors to the demonstration facility. Also, complete trajectories were accomplished in this mode. Consequently, operator interfaces, actions and reactions with the Supervisory mode were extensively observed. The observations and conclusions follow:

- The CRT displays provided a very effective substitute for a procedure or checklist.
- Familiarity with the sequence and the CRT display/keyboard method of controlling it was acquired very quickly. Operators completely unfamiliar with the tasks were able to pick up the flow within 5 or 6 steps and complete a total exchange quickly and easily.
- The fail safe approach of locking out the next command until completion of the current step was very effective and permitted an untrained operator to take responsibility for performing the operations with little risk.

- The CRT terminal software would respond only to the proper command at the proper time. This proved to minimize damage due to operator error. Any "pilot error" representing a deviation from the desired was flashed as such to the operator along with an audio signal. The original display and corresponding requested command was then returned to the screen and the pilot given an opportunity to enter the right command on the keyboard. This type of failure, which most often was a keyboard input error or a failure to read the command request carefully, occurred on the order of once each trajectory for "unfamiliar" pilots.
- The software control law that reduced joint rate as the desired position drew near resulted in very smooth motion both on the TV monitor and on the error meters.
- The step, initiated by a CONTINUE on the keyboard, that was used to set up the servicer mechanism joint position errors and to display the errors on the control panel null meters seemed to be a useful step. However, some operators eventually tended to ignore the meters and often proceeded without the desired meter-monitor check.
- The CRT displays which were all much the same and meant to be as explanatory as possible may have been a little too much so. The first step in a segment which provided the joint values to be gained may not have had to been repeated for each subsequent step in that segment. The unfamiliar operator often felt obligated to read the whole display only to find most of it the same as the last one. The real change in the display, which was at the bottom, was overlooked on occasion. The final lines of the display, which change from step to step could be a sufficient display until the next segment is initiated.
- The initial command format of typing in a word such as PROCEED, CONTINUE, or START was found too cumbersome by most operators. Since the computer only interrogated the first two characters, PR, CO and ST were quickly accepted and substituted. It was felt even this was a nuisance of sorts since the two letters had to be followed by "carriage return" as well to execute the step. A single letter, clearly marked in an easily seen and accessible location could be sufficient

- The trajectory sequence offered very little flexibility in changing the sequences or any of the values. Subsequent simulation software is planned to offer considerably more flexibility in this area. Far more data read-out capability was found desirable and will also be provided.

b) Manual-Direct Mode - Three different members of the servicing team utilized this mode to control the servicing arm during the set-up and checkout of the simulation/demonstration. Various steps of the module exchange sequence were performed. Some conclusions drawn are listed below. Before discussing each in detail, it should be pointed out again that the expected utilization of this mode is on a joint by joint basis that follows a detailed checklist using only the angle set knobs, meters and lights for error display and the direct control switch for driving each joint. As a backup mode it must be assumed the TV may have failed and trajectory completion must be accomplished without it. In reality the TV could be available. In this case the normal response is to use it as was the case in the simulation. The fifth conclusion below relates to that configuration with some suggestions for enhancing its use.

- The one major conclusion regarding this mode is that a Manual Direct trajectory exchange is possible. With a good checklist and the error displays any operator can be trained to accomplish the joint by joint sequences in a trajectory.
- A significant factor in accomplishing the Manual Direct operations was the sensitivity of the error meter, particularly near null. Initial sensitivity was selected to correlate meter displacement somewhat to the joint's span of travel. At null it was difficult to fine position the joint accurately. The meter sensitivity was increased such that near null a single division represented less than a degree, greatly enhancing the positioning capability. The more frequent "meter pegged" conditions were found to present no problem whatsoever. The only real function required of the meter when not near null was to provide polarity information which it did very effectively.

- An observation similar to the previous conclusion related to the error lights. For manual direct the threshold for extinguishing the light as null was approached was $\pm 0.5^{\circ}$. It is recommended this be decreased to $\pm 0.1^{\circ}$. In the supervisory mode $\pm 0.1^{\circ}$ was used and found to be more meaningful. No noise or hunting effect was ever evidenced.
- The fixed-rate commands that result from the direct control switches could be selected at three levels. The high levels were fine for gross joint motions. The slowest rates were compatible with the error light thresholds in that it was very easy to drive the joints to where the lights went off and stayed off. It is anticipated that the slowest rates will be adequate with lower error light thresholds.
- A number of observations were made on use of the TV in the Manual Direct mode. While the mode was not initially designed to be efficient in this configuration, there is a finite probability that a failure could occur, requiring the manual direct mode, that leaves the TV operable. Certain features could be included to enhance control using the TV.
  - It was initially difficult to correlate the joint commands entered on the DIRECT CONTROL switches to the motion of the end effector as viewed on the TV monitor.
  - It was found that a number of visual displays or aids at the control station were helpful to acquire the desired correlation between joints and TV.
  - It was difficult to complete a trajectory segment in the same fashion from run to run using the TV. It should be pointed out that in the supervisory mode where each segment is repeated in an identical sequence each time this problem did not show up. All biases and errors were treated the same each time and were essentially biased out automatically. Approaches to solving these problems have been identified and are detailed in *On-Orbit Servicer Demonstration/Simulation Report*, March 25, 1977.

IX. OPERATIONS ANALYSIS

A. GEOSYNCHRONOUS SERVICING ANALYSIS

The Geosynchronous Servicing Analysis was to further evaluate the profitability of high earth orbit satellite maintenance/service functions. This profitability was evaluated in the initial IOSS. However, profitability was only addressed for one maintenance concept even though several additional concepts were briefly considered. Therefore, this study evaluated the following alternative methods of on-orbit servicing, as well as baseline expendable satellites launched by Tug or IUS.

a) Tug/servicer launched on demand;
b) IUS/servicer launched on demand;
c) Fully-spared rover warehouse with chemical propulsion stage (CPS)/servicer - one servicing circuit per year;
d) Fully-spared rover warehouse with solar electric propulsion stage (SEPS)/servicer - one servicing circuit per year;
e) Partially-spared rover warehouse with CPS/servicer - one servicing circuit per year;
f) Partially-spared rover warehouse with CPS/servicer - two servicing circuits per year;
g) Partially-spared rover warehouse with SEPS/servicer - one servicing circuit per year;
h) Partially-spared rover warehouse with SEPS/servicer - two servicing circuits per year;
i) Fixed-location fully-spared warehouse with CPS/servicer - two servicing circuits per year;
j) Fixed-location fully-spared warehouse with CPS/servicer - incremental servicings of two satellites at a time;
k) Fixed-location fully-spared warehouse with SEPS/servicer - two servicing circuits per year;
ℓ) Fixed-location fully-spared warehouse with SEPS/servicer - incremental servicings of two satellites at a time.

The following major conclusions resulted from this study:

- On-orbit servicing of high earth orbit satellites is less costly than the use of expendable satellites.

- The preferred on-orbit servicing methods are:
  - Tug/servicer demand launch with dual satellite servicing. Program costs of \$5.7B are 19% lower than \$7.0B costs for Tug launch of expendable satellites.
  - Fully-spared warehouse at fixed longitude, with SEPS/servicer servicing two satellites and return to warehouse. Program costs of \$6.0B are 15% lower than Tug launch of expendable satellites.

The Tug-demand-launch option will have several months response time before a servicing mission is initiated. If satellite redundancies make satellite availability not a problem, then the first option is preferred. If satellite availability is desired to the extent the additional \$300M program costs are justified, then the fixed-warehouse with SEPS incremental servicing option is preferred. These two recommended options are compatible with the baseline servicer system.

The following additional results were generated:

- All servicing options are within ±3% (\$185M) of the average; therefore, other selection criteria become important.
- Major factors for selection of servicing methods are:
  - Program costs;
  - Servicing response time (minimize satellite downtime);
  - Compatibility to baseline servicer system.
- Lower system costs and shorter servicing response times make SEPS propulsion preferable over a chemical propulsion system.
- Rover warehouse systems require more than the single baseline servicer spares stowage rack and would make on-orbit servicing difficult.
- Minimum program costs, for systems compatible with the baseline servicer, occur with the Tug-demand-launch servicing method.

- Minimum servicing response time is achieved with the fixed-location warehouse and incremental dual servicing of satellites using the SEPS for propulsion.
- It is cost effective to expend servicers and spares stowage racks in orbit. Costs to return these systems exceed procurement costs.
- Servicing system investments (servicer, warehouse, and propulsion vehicle) cost about 2% to 6% of the program costs for the various servicing options.
- Servicing system investments will cost from 6% to 26% of the potential program cost savings when compared to the use of expendable satellites.
- Maintaining the spacecraft programs at the same level as the peak in the mission model used or increasing the missions will if anything result in greater savings from on-orbit servicings.
- Dual deployment of expendable satellites by Tug is not cost effective because of maneuvering propellants that must be taken to geosynchronous orbit.
- Tug program costs for servicing missions are greater than determined in the first IOSS because of greater launch costs and escalation to 1977 dollars.
- IUS program costs are greater than the use of Tug for launch of expendable satellites and for high earth orbit demand servicing missions.

The material in this section was abstracted from *High Orbit Service/Maintenance Analysis*, Martin Marietta Corporation, Denver, Colorado, May 1977. It should be referred to for complete details and the specifics of the analysis.

1. Introduction

This effort was initiated as a part of Task 5 - Operations Analysis; and in particular, is *Subtask 1--High Orbit Service/Maintenance Analysis.* The subtask objectives were to reevaluate the profitability of high earth orbit satellite servicing and to compare various alternatives to achieve servicing. Considerations to be included were:

- Sensitivity of full capability Tug;
- Sensitivity/need for an alternate Tug;
- Expendable Tugs and servicer;
- Multiple service;
- Servicer and spares stationed on orbit.

The study was initiated with a definition of the geosynchronous-satellite mission model to be used in the analyses. Performance data and properties of support systems (e.g., Orbiter, Tug, IUS, SEPS, etc) were then established and defined. Twelve alternative methods for geosynchronous-satellite servicing were then developed and analyzed in sufficient detail to define capabilities, limitations and other information needed for cost analyses and tradeoffs. To enable complete comparison of program costs between servicing options and use of expendable replacement satellites, the requirements for all satellite launches were then determined. With all the foregoing data, total program costs were developed for the 12 servicing options and two methods (Tug and IUS boost) of launching expendable satellites. Subjective evaluations and cost comparisons were then conducted and recommended servicing concepts were identified.

As an aid in the study and to focus the analyses on certain objectives, the assumptions and guidelines in Table IX-1 were used.

2. Geosynchronous Satellite Mission Model

This paragraph establishes the mission model that was used as the baseline for the analysis. The geosynchronous satellite data listed in Table IX-2 was extracted from the IOSS Final Report, Volume II, September 1975. Minor modifications were made to the IOSS launch schedule data to maintain consistency in the fleet size (n_f) and average operating time (AOT) data. For purposes of establishing a servicing-requirements baseline, it was assumed that, on the average, each satellite will be serviced at the

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Table IX-1 Assumptions and Guidelines

- Satellite system redundancies and on-orbit spare satellites are such that replacement of failed subsystem modules and depleted commodity modules can be effected at the average operating time (AOT) on the average, to extend the satellite operating life through the subsequent AOT period.
- Failure analysis telemetry data will enable the scheduling of specific servicing missions with the required replacement modules known prior to the mission.
- Replaced satellite modules are jettisoned to space and not recovered.
- The current NASA launch cost reimbursement policy (LCRP) for the Orbiter will be used.
- The LCRP will apply to the Tug and IUS.
- All costs are in 1977 dollars.
- The following STS support systems are applicable to the study and costs per flight are:

Orbiter	\$23.4M
Tug	\$ 2.4M
IUS	\$ 5.7M (expendable)

- The SEPS from the PLUS studies is an applicable support system candidate.
- Data from IOSS (September 1975) are used where applicable and still valid.
- The spinning solid upper stage (SSUS) does not have performance capabilities required for servicing operations and will not be considered as an applicable servicing support system.
- All launches of geosynchronous satellites will be from KSC.
- Long-term storage of replacement modules in space is acceptable.

Table IX-2 Geosynchronous On-orbit Maintainable Satellite Data

NO	SATELLITE	P/L NO.	P/L MODEL CODE	MASS (LBS)	ORBIT LONGITUDE	n	n_f	AOT	pf	LF	LF_3
1	Advanced Radio Astronomy Explorer (RAE)	AS-05-A	AST-1C	3971	80°W	3	1	3	.22	.06	.16
2	Synchronous Earth Observatory Satellite (SEOS)	EO-09-A	EO-4	4035	100°W, 110°W	8	2	2	.15	.16	.54
3	Intelsat (Ist)	CN-51-A	NN/D-1	2526	24°W, 30°W, 36°W, 42°W, 168°W, 174°W, 180°W, 80°E, 86°E	18	9	6	.28	.36	.81
4	Domsat B (Dom B)	CN-53-A	NN/D-2B	3905	88°W, 94°W, 100°W, 106°W, 112°W, 118°W, 125°W	14	7	6	.28	.28	.63
5	Domsat C (Dom C)	CN-58-A	NN/D-2C	2576	41°W, 171°W, 104°W (spare)	6	3	5	.31	.12	.27
6	Disaster Warning Satellite (DWS)	CN-54-A	NN/D-3	1949	94°W, 124°W	4	2	5	.24	.08	.18
7	Traffic Management Satellite (TM)	CN-55-A	NN/D-4	1322	29°W, 52°W, 140°W, 162°W, 176°W, 60°E, 80°E	14	7	5	.29	.28	.63
8	Foreign Communications Satellite - A (FC-A)	CN-56-A	NN/D-5A	1371	60°W, 50°W, 40°W, 30°W, 20°W, 10°W, 0°, 84°W, 104°W, 124°W, 135°W, 155°W	24	12	6	.24	.40	1.08
9	Communications R&D Prototype (C-R&D)	CN-59-A	NN/D-6	2772	100°W	3	1	4	.25	.06	.18
10	Foreign Synchronous Meteorological Satellite (FSM)	EO-57-A	NN/D-9	1181	60°W, 75°E	6	2	4	.37	.12	.32
11	Geosynchronous Operational Meteorological Satellite (GOMS)	EO-58-A	NN/D-10	1181	138°E, 142°E	8	2	3	.38	.16	.48
12	Geosynchronous Earth Resources Satellite (GER)	EO-59-A	NN/D-12	4035	98°W, 120°W	10	2	2	.15	.20	.72
13	Foreign Synchronous Earth Observation Satellite (FSEO)	EO-62-A	NN/D-13	4035	60°W, 75°E	9	2	2	.14	.20	.72

<u>LEGEND:</u> n - Number of operating cycles; n_f - On-orbit fleet size, AOT - Average operating time; pf - Parts factor - portion of satellite replaced; LF - Emplacement loss factor, LF_3 - Satellite loss factor

end of each AOT (except the last). When the costs of expendable satellites are determined, it was assumed a new satellite is launched at the end of each AOT.

The baseline servicing schedule was identified by year. There will be from two to 12 servicings a year. It was seen that this mission model has an increase in activities until 1990 and then a decrease. Actually the servicing schedule (or expendable replacements) would probably hold at some level similar to that in 1990 or might even increase some. Potential effects of different distributions are discussed in paragraph 7.

The servicing schedule and orbit locations are presented pictorially in Figure IX-1. The years of servicing are represented by the bands. The orbital location of each type of satellite is shown within the band for the applicable servicing year (coded to the Table IX-2 listing). Much of these data were combined to present the annual servicing needs. The weight of the replaced modules at each servicing was determined by the product of parts factor (pf) and the satellite weight. The total annual servicing replacement weight and total longitudinal distribution of the satellites requiring servicing in a given year were summarized to aid in subsequent selections of servicing options.

3. Support Systems Definitions

This paragraph defines and summarizes capabilities of those systems applicable to supporting high earth orbit servicing operations. The systems discussed include:

- On-orbit Servicer System;
- Shuttle Orbiter;
- Tug (full-capability);
- Interim Upper Stage (IUS);
- Solar Electric Propulsion Stage (SEPS)
- Chemical Propulsion Stage (CPS).

a) Servicer Systems - The servicer mechanism and stowage rack is that described in Chapter VI. The weight of the servicer mechanism is 184 lbs. The length of the system with a single stowage rack is approximately five feet in the launch configuration. The stowage rack weighs 253 lbs plus

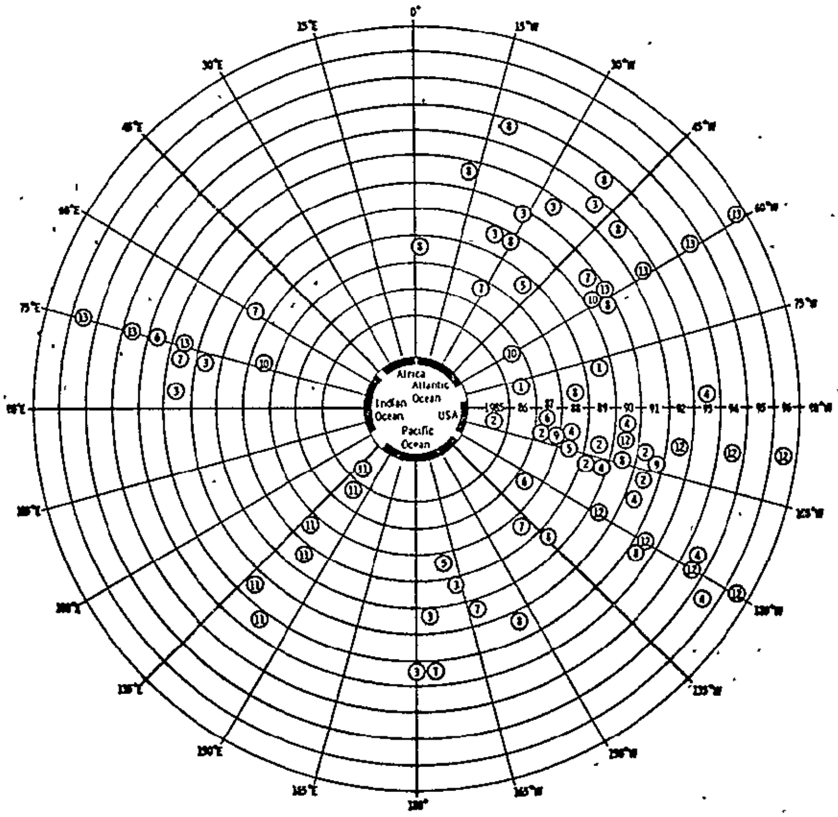

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Figure IX-1 Baseline Servicing Schedule and Locations

4.7 lbs for each set of module interface mechanism guides. Each stowage rack can carry replacement modules for one or two satellites with a total module capacity of approximately 1800 lbs.

b) Shuttle Orbiter - The Orbiter provides a 15-foot diameter by 60-foot long payload bay. A 65,000 lb payload can be delivered to a 160 n mi circular orbit at 28.5 deg inclination from a KSC launch. The nominal on-orbit duration is seven days.

c) Tug - The full capability tug provides a third stage capability to the STS for boosting satellites from the Shuttle orbit to higher orbits and/or retrieving satellites. The reference data were extracted from *Baseline Space Tug Configuration Definition*, MSFC 68M00039-2, July 15, 1974. The Tug is capable of boosting a 7000 lb payload from Shuttle orbit to a geosynchronous equatorial orbit. The full Orbiter-to-payload interface accommodations systems weight of 1900 lbs was assumed when launching a servicer, CPS, or SEPS. With passive spares only, fewer accommodations are needed and a variable margin for support systems was provided in these cases.

d) Interim Upper Stage (IUS) - The IUS (shown in Figure IX-2) provides an early third stage capability to the STS for boosting payloads from the Shuttle orbit to higher orbits. Data from the IUS Technical Interchange at MSFC, April 4, 1977, indicates the two-stage IUS weighs 32,643 lbs and is capable of boosting 5218 lbs of payload from Shuttle orbit to geosynchronous equatorial orbit. The total IUS length is 16.4 feet.

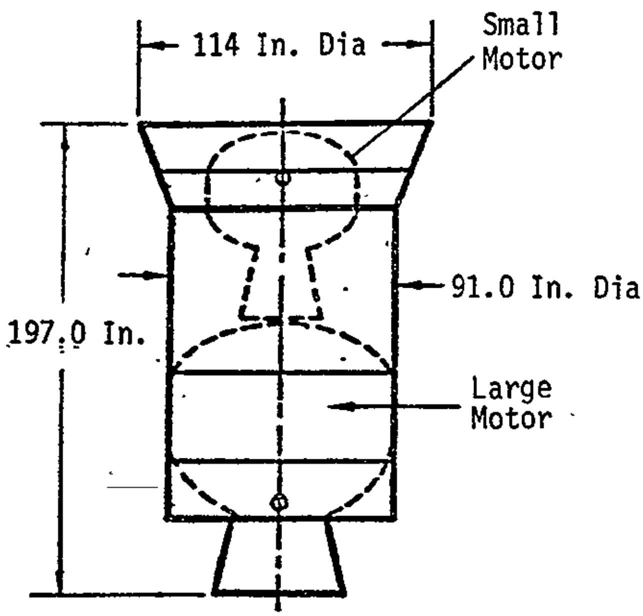

Figure IX-2 Interim Upper Stage

e) Solar Electric Propulsion Stage (SEPS) - The SEPS (shown in Figure IX-3) is a long-duration propulsive device for transferring payloads between higher orbits. The data used were based on the earth orbital (EO) version from Boeing studies (the planetary version had approximately twice the propellant capacity). The avionics systems have a five-year life and thrusting can be maintained for 20,000 hours. It was assumed that the SEPS can be refueled on orbit (as assumed in Boeing and Rockwell studies). The Tug can deliver a mercury module, along with other spares, and the servicer system could replace the module on the SEPS. The tank and other structure is assumed to weigh 50 lbs.

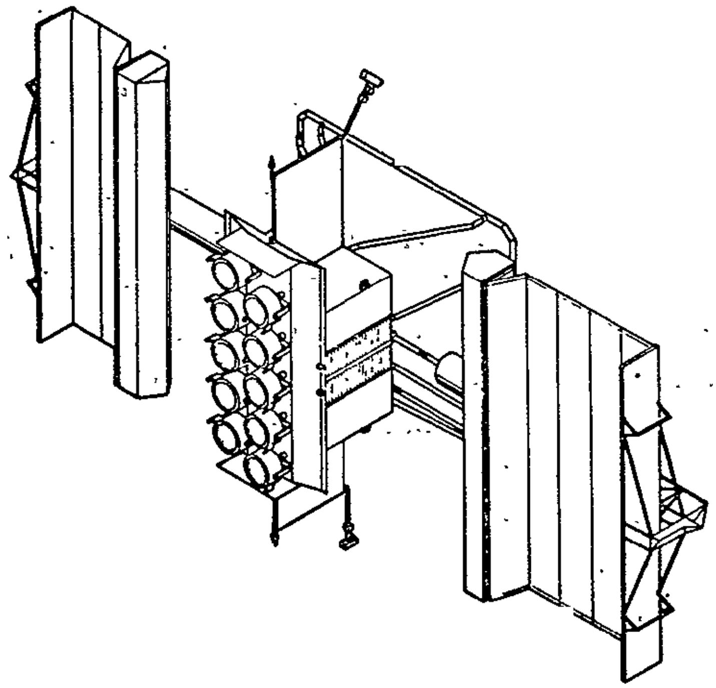

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Figure IX-3 Solar Electric Propulsion Stage

f) Chemical Propulsion Stage (CPS) - A chemical-propulsion vehicle was conceived to evaluate this type of on-orbit servicing. The maximum total weight was assumed to be 6321 lbs to be compatible with Tug boost to geosynchronous orbit and to remain within 65,000 lbs for the Tug, CPS, and Orbiter interface accommodations. Subtracting 184 lbs for the servicer mechanisms and assuming 0.85 propellant mass fraction results in a propellant load of 5200 lbs. An I_{sp} of 300 sec was assumed for a storable hypergolic propulsion system. The CPS lifetime was assumed to be three years.

4. Evaluations of On-orbit Servicing Options

The 12 on-orbit servicing options listed above were evaluated and are discussed briefly here.

In the demand launch from earth options, a Tug or IUS with a servicer system and applicable spares would be boosted to the failed or degraded satellite(s) after the need and type of servicing have been identified. A full complement of unique spares would be maintained on earth.

a) Tug/Servicer Launched on Demand - The servicing requirements were evaluated for possible accomplishment by Tug/servicer missions. It appeared that a Tug/servicer would be capable of servicing two satellites and it is logical that conditions will exist where much of the time two satellites will need servicing. Therefore, a worst-case condition of servicing two satellites was identified and evaluated. This condition exists in 1995 where the required replacement weight is 1658 lbs for two satellites 160° apart in orbit. Figure IX-4 presents the performance analysis for this servicing mission. As can be seen, acceptable margins of inerts, auxiliary propulsion propellants, and main propulsion propellants remained. From Tug separation from the Orbiter until return to the Orbiter took about 159 hours. This would leave a marginal nine hours for the Shuttle to be launched to orbit and to return from orbit. However, since the mission examined was a worst case, it can be generally accepted that the Tug/servicer is capable of servicing two satellites on one mission. Table IX-3 presents the breakdown of Tug servicings to accomplish the baseline mission model servicing schedule. In most cases dual servicings were assumed. Single servicings were assumed where an odd number of servicings is required in a year. This schedule requires 40 Tug/servicer missions in the 12 years, with up to six missions per year.

Table IX-3 Tug Servicings

YEAR	SERVICINGS		REMARKS	YEAR	SERVICINGS		REMARKS
	REPLACEMENT WEIGHT (LBS)	Δ LONGITUDES			REPLACEMENT WEIGHT (LBS)	Δ LONGITUDES	
1985	1054	122°	*	1991	1036	26°	
					1298	0°	
1986	1311	20°			1698	2°	
	449		← Single		329		← Single
					1156	52°	
1987	851	65°			948	5°	
	1298	0°					
	468		← Single	1992	1036	8°	
					1170	38°	
1988	1128	41°			934	4°	
	1422	16°			1090	4°	
	1182	36°			886	67°	
	1248	51°					
	437		← Single	1993	894	95°	
					1422	48°	
1989	1036	6°					
	1479	20°		1994	1170	38°	
	1698	4°			1698	2°	
	1036	33°					
	832	82°		1995	1658	160°	Worst case - analyzed for Tug capability.
1990	1090	22°					
	1002	0°		1996	565		← Single
	1422	34°			1210	22°	
	934	6°					
	988	42°					
	1414	106°					

*All are dual servicings except those identified as "single".

The Tug and Shuttle Orbiter payload requirements for the 40 servicing missions are identified in Table IX-4. The Tug payload weight was determined by adding the weights of the spares, servicer, and rack. The Shuttle payload weight was determined by adding the 1900 lbs of Orbiter-payload interface accommodations equipment to the Tug system weight.

EVENT	EVENT DURATION (HRS)	TOTAL TIME (HRS)	INERTS/ LOSSES (LBS)	APS (LBS)	MPS ΔV (FT/SEC)	INITIAL WEIGHT (LBS)	MAIN PROPELLANT USED (LBS)
1. Tug separation from Orbiter	2.0	2.0	10.0	8.6	-	58,935	-
2. Phase in Shuttle orbit	11.0	13.0	44.0	21.4	-	58,916	-
3. Burn into phasing orbit	.13	13.13	-	-	4494	58,851	15,513
4. Coast in phasing orbit, 1 rev	3.0	16.13	9.0	17.5	-	43,338	-
5. Inject into geo transfer (includes 2.2° plane change)	.11	16.24	-	-	3672	43,311	9,581
6. Coast to midcourse	1.5	17.74	5.0	13.8	-	33,730	-
7. Midcourse correction	.03	17.77	-	-	50	33,711	122
8. Coast to geosynchronous	3.96	21.73	12.0	14.0	-	33,589	-
9. Circularize at geosynchronous (includes 26.3° plane change)	.12	21.85	-	-	5826	33,563	10,990
10. Rendezvous and dock at 125°W	12.0	33.85	48.0	30.0	50	22,573	74
11. Changeout Domsat B - jettison 1093 lbs	3.0	36.85	12.0	12.0	-	22,421	-
12. Inject into phasing orbit - 160° exterior	.05	36.9	-	-	436	21,304	655
13. Coast in phasing orbit - 3 revs	82.7	119.6	249.0	20.0	-	20,649	-
14. Circularize at geosynchronous	.05	119.65	-	-	436	20,380	627
15. Rendezvous and dock at 75°E	12.0	131.65	48.0	27.0	50	19,753	65
16. Changeout FSEO - jettison 565 lbs	3.0	134.65	12.0	12.0	-	19,613	-
17. Phase at geosynchronous for nodal crossing	12.0	146.65	36.0	12.0	-	19,024	-
18. Deboost burn	.08	146.73	-	-	5840	18,976	6,226
19. Coast to midcourse correction	1.0	147.73	3.0	7.5	-	12,750	-
20. Midcourse correction	.01	147.74	-	-	13	12,739	14
21. Coast to 170 n.mi perigee	4.2	151.94	17.0	8.1	-	12,725	-
22. Inject into return phasing	.05	151.99	-	-	3791	12,700	2,889
23. Coast 1 rev in phasing	3.0	154.99	12.0	7.8	-	9,811	-
24. Circularize at 170 n.mi	.05	155.04	-	-	4243	9,791	2,457
25. Rendezvous and dock with Orbiter	4.0	159.04	-	32.4	-	7,334 (7,302)	-
			517.0	244.1			49,213
		Margins →	30.0	72.9			976

INITIAL SYSTEM:

Dry Weight	5150
Unusable Residuals	576
APS Reserves	29
Expendables	547
Propellant Reserves	300
Usable Propellants	49889
Usable APS	288
Replaceable Modules	1658
Servicer	184
Rack	314
	58,935 lbs

RETURN SYSTEM:

Dry Weight	5150
Unusable residuals	576
APS	73
Expendables	30
Propellant	976
Servicer	184
Rack	314
	7303 lbs

Figure IX-4 Worst Case Tug Servicing Mission (1995 Servicing of Domsat B and Foreign Synchronous Earth Observation Satellites)

Table IX-4 Option a--Tug/Shuttle Launch

Vehicle	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991	1992	1993	1994	1995	1996
Payload Weights:															
Spares				1054	1311/449	851/1298 468	1128/1422 1182/1248 437	1036/1479 1698/1036 832	1090/1002 1422/934 988/1414	1036/1298 1698/329 1156/948	1036/1170 934/1090 886	894/1422	1170/1698	1658	565/1210
Rack Servicer	314 184 } each launch →														
Tug P/L Weights:															
1				1552	1809	1349	1626	1534	1588	1534	1534	1392	1668	2156	1063-S
2					947-S*	1796	1920	1977	1500	1796	1668	1920	2196		1708
3						966-S	1680	2197	1920	2196	1432				
4							1746	1534	1432	827-S	1588				
5							935-S	1330	1486	1654	1384				
6									1912	1446					
Shuttle P/L ** Weights:															
1				60231	60488	60028	60305	60213	60267	60213	60213	60071	60347	60835	59742
2					59626	60475	60599	60656	60179	60475	60347	60599	60875		60387
3						59645	60359	60876	60599	60875	60111				
4							60425	60213	60111	59506	60267				
5							59614	60009	60165	60333	60063				
6									60591	60125					

* S signifies single servicings. These payload weights will be used directly to determine b (cost sharing factor). All others are dual servicings. To determine b, 800 lbs will be added for portion of payload needed for propellant to make average maneuver to reach second satellite.

** Payload interface accommodations ot 1900 lbs included.

In the cost analyses, the Tug and Shuttle costs will be shared with other users since the Tug has greater payload capability than those shown. For the dual servicing missions, 800 lbs of the MPS propellant will be considered as additional servicing system weight. This is the propellant needed for translation and docking to accomplish the second servicing operation, assuming an average longitude difference of 80°.

b) <u>IUS/Servicer Launched on Demand</u> - Since the IUS motors are expended on reaching geosynchronous orbit, only one satellite per mission can logically be serviced by this option. In fact, additional propulsion system capability would be needed to enable rendezvous and docking with the satellite. This added feature is feasible for some maneuvers but not to enable large phase changes and additional maneuvers.

The IUS and Orbiter payload weights for the 75 required missions were calculated. In each of the missions, the IUS, servicer, rack, and replaced spares are expended.

In rover warehouse-fully spared options, it was assumed that a fully-spared warehouse with servicer is maintained in geosynchronous orbit. As satellite failures occur, the entire warehouse system would be moved by a CPS or SEPS to the applicable satellite. Since the warehouse mass and propellant needs would be very large, only one annual orbital circuit to service applicable satellites was assumed.

The fully-spared warehouse will contain a full complement of unique spares for each type satellite in orbit. That is, as each new type satellite is launched, a full complement of spares is launched and placed on the warehouse. As subsequent satellites of that type are launched, one set of supplemental spares are stocked in the warehouse to the quantity of pf x W_{sat}. As modules are replaced, the warehouse will be restocked with the mass of modules removed.

The mass of initial unique spares required was determined by first summing the unique modules listed for the applicable satellites in *Operations Analysis (Study 2.1), Payload Designs for Space Servicing,* Aerospace Reports ATR-74(7341)-3, June 30, 1974 and Addendum, September 30, 1974. The ratio (RR) of unique replaceable modules to the Aerospace determined satellite weight was applied to the IOSS satellite weights to determine the replaceable module weights for the mission model satellites. It was assumed that no common modules exist between different satellite types. This analysis is summarized in Table IX-5. Lower RR values occur where there are more redundant modules within the satellite.

Table IX-5 Initial Unique Replacement Spares Determination

SATELLITE	TOTAL WEIGHT FROM IOSS (lbs)	AEROSPACE DATA TOTAL (kg)	AEROSPACE DATA REPLACEABLE (kg) *	AEROSPACE DATA REPLACEMENT RATIO (RR)	REPLACEMENT SPARES FOR IOSS WEIGHTS (lbs)
Advanced Radio Astronomy Explorer	3971	1175	728	.62	2460
Synchronous Earth Observatory Satellite	4035	1923	974	.51	2044
Intelsat	2526	2727	1040	.38	963
Domsat B	3905	2705	1018	.37	1470
Domsat C	2576	1315	778	.59	1524
Disaster Warning Satellite	1949	1420	683	.48	937
Traffic Management Satellite	1322	1145	606	.53	700
Foreign Communications Sat-A	1371	975	564	.58	793
Communications R&D Prototype	2722	3138	1098	.35	952
Foreign Synchronous Meteorological Satellite †	1181	1582	1015	.64	758
Geosynchronous Operational Meteorological Satellite †	1181	1582	1015	.64	758
Geosynchronous Earth Resources Satellite	4035	2201	1352	.61	2479
Foreign Synchronous Earth Observation Satellite	4035	2344	1346	.57	2317

*Replaceable mass assumes one of each unique module and no common modules between satellites.
†Used EO-7 data.

A schedule for stocking the warehouse with the initial complement of spares (I), supplementary spares (S) when an additional satellite is launched, and replacement spares (R) when AOT servicing occurs was prepared as was a summary of total weight (including racks) to be delivered to the warehouse each year.

It was seen that the warehouse will become very large, with 52 spares racks or with any other configuration that packs spares more efficiently. Some mechanism would be needed to move the appropriate modules to the servicer or the servicer would have to undock from the warehouse to extract the modules from the stowage locations. The latter method would require a stabilization system for the warehouse. An additional warehouse systems weight of 500 lbs was assumed for providing these additional systems. A four-year life time was assumed for the warehouse systems.

c) <u>Fully-spared Rover Warehouse, CPS</u> - Once each year, starting in 1985, the CPS will move the entire warehouse around the orbital circuit and will service each satellite requiring service (see Figure IX-1). The CPS has a three-year life. The first CPS/servicer will be on-orbit in 1982, even though the first servicing is predicted for 1985. The next CPS/servicer will be on-orbit from 1985 until 1988, etc. The three-year servicing period, 1989-1990-1991, would be the worst case for this option and it was analyzed. A 7000 lb CPS/servicer was assumed at this point in the study. Taking 186, 197, and 178 days each year for the servicing circuits required 2942 lbs of the 5200 lbs available propellant. Therefore, one circuit per year is about the best possible with this option. The Tug and Shuttle flights to orbit the CPS/servicer and to stock the warehouse were summarized. A total of 26 launch missions are needed in 15 years, with a maximum of three launches per year.

d) <u>Fully-spared Rover Warehouse, SEPS</u> - The SEPS has a predicted life of five years. Therefore, three SEPS/servicers will be required for the 15-year mission model: 1982 to 1987; 1987 to 1992; and 1992 to 1997. The second servicing period (1988 to 1992) is the worst case. This period was analyzed to evaluate SEPS capabilities. It was seen that the SEPS will require refueling twice in the five years. A larger propellant capacity, as in the planetary SEPS, could reduce refueling needs to once in the five years. The other two SEPS on-orbit servicing periods (1985 to 1987 and 1993 to 1996) will not require refueling since there are only 9 and 13 servicing activities respectively and the rate of propellant usage in the mission analyzed averaged less than 45 lbs per servicing. The Tug and Shuttle launches required to orbit the SEPS/servicers, spares and racks, SEPS refueling modules, and new warehouse systems were summarized. A total of 23 launch missions are required in the 15 years, with a maximum of two launches per year.

In rover warehouse-partially spared options, it was assumed that once or twice a year, after specific satellite failures have been determined, replacement spares are orbited and joined to an on-orbit warehouse and transfer vehicle. This partial warehouse is then moved around the orbital circuit to service the applicable satellites. The same spares will be required as in Options c. and d, but most spares will be maintained on the ground. The on-orbit warehouse will therefore be much smaller. However, there will still be up to three (two circuits) or six (one circuit) racks accumulated at a time on the partial warehouse. Therefore, additional warehouse systems would again be needed.

e) and f) Partially-spared Rover Warehouse, CPS - Two suboptions were investigated: 1) one orbital servicing circuit per year, and 2) two orbital servicing circuits per year. The second suboption reduces the average satellite downtime to three months instead of six months but requires approximately twice as many Tug/Shuttle launches. The worst case period was analyzed for two service circuits per year. Only about 1500 lbs of propellants were used in this method. The suboption of one circuit per year would require even less propellant. Because of the reduced need for propellants, the total CPS weight was assumed to be 3000 lbs in the summaries of the requirements for Tug and Shuttle launches. Note that a CPS must be available for launch and servicing all the time that maintainable satellites are on orbit. With periodic maintenance, checkouts, etc, it was assumed the three-year lifetime expires in 1985 even though only one year of actual on-orbit use is achieved. A new CPS/servicer is launched in 1986. Similar assumptions were made in subsequent SEPS analyses. A total of 16 launches are required for the two-circuits per year option.

g) and h) Partially-spared Rover Warehouse, SEPS - The same suboptions were investigated for the SEPS as for the CPS for the partial warehouse method. An analysis of the SEPS worst-case 1988 to 1992 period for two circuits per year was made. The SEPS would have to be refueled twice in this period. The Tug/Shuttle launch requirements for the one-circuit and two-circuit cases were calculated. A total of 18 launches are required for the two-circuits per year option.

In the fixed-location fully-spared on-orbit warehouse options, it was assumed that a fully-spared warehouse is maintained at a fixed location (100°W longitude) in geosynchronous orbit. Sparing requirements are as determined in options c and d. Resupply of used spares is accomplished after the servicing activities. Periodically, the servicer system will load the required spares/ racks and service the applicable satellites. Two suboptions were analyzed: 1) two orbital circuits per year, and 2) incremental transits where two satellites are serviced on each trip. The first suboption again has the problem that up to three racks may be required, imposing additional complications in the servicer mechanisms. The second suboption would require only one rack, allowing use of the baseline servicer system. In all cases, a stabilization system (500 lbs assumed) will be required to maintain the warehouse while the servicer system is away. A four-year life was assumed for this system.

i) and j) Fixed-location, Fully-spared Warehouse with CPS - Based on previous analyses in option f, it was known that the CPS has the capability to perform the two-circuits per year case with under 1500 lbs propellants. The second suboption was analyzed for the 1989-1991 worst case. A 7000 lb CPS/servicer was again assumed initially. About 1300 lbs propellant was used for the three-year period. Therefore, a 3000 lb CPS/servicer should again be adequate for these options. A summary of the Tug/Shuttle launch requirements was prepared. No significant differences exist between the two suboptions as far as launch requirements are concerned. A total of 23 launches are required.

k) and l) Fixed-location, Fully-spared Warehouse with SEPS - The same suboptions were investigated as for the CPS. SEPS capabilities to service satellites in the 1988 to 1992 period by doing two circuits per year were analyzed. The SEPS would need to be reserviced once (in 1990). A partial analysis of this period for the incremental (two satellites and return) servicing suboption was made. This analysis was carried far enough to determine that again only one SEPS refueling would be required (in 1990). The Tug/Shuttle launch requirements were summarized. A total of 23 launches are required. No significant launch differences exist between the suboptions.

5. Satellite Launch Requirements

To enable comparing costs of the on-orbit servicing options to the costs of using expendable satellites, the satellites and their launch costs must also be included in all cost analyses. The IOSS satellite costs were reevaluated to assure comparisons on a common basis.

The launch requirements for establishing the on-orbit fleet of serviceable satellites were determined. They included the satellite weight, the actual weight boosted by the Tug or IUS (assuming 500 lbs payload interface accommodations for the Tug or IUS), and the Orbiter payload weights for both a Tug and the IUS upper stage. With the Tug, 1,900 lbs of payload support equipment was assumed. The IUS will not require as much support equipment, therefore support equipment was estimated as 1,000 lbs.

The same data for launching expendable satellites was calculated. The lower satellite weights from the first IOSS and additional replacement satellites were launched at the end of the AOTs to maintain the on-orbit fleet size.

6. Cost Analysis

This paragraph discusses the methods, considerations, and results of analyses to determine the costs of the on-orbit servicing options. Costs for the expendable satellite program were reevaluated to enable comparisons on a common basis and to determine the profitability of on-orbit servicing of high earth orbit satellites. Data and methods from the initial IOSS and from the follow-on Task 5.2 report, April 1977, were used where applicable and are still valid. Changes and updates are discussed.

a) Launch Cost Reimbursement Policy - The new launch cost reimbursement policy (LCRP) for the Orbiter was used. In general, this policy makes the Orbiter launch costs proportional to the Orbiter load factor (λ), up to 0.75. Over 0.75 load factor, the entire cost is assessed to the payload. In all analyses in this study, since a Tug or IUS is always launched, the weight ratio determines the load factor. As the Orbiter down capabilities are not used in these ratios, the result tends to be even higher costs for return of equipment from geosynchronous orbit.

The Orbiter LCRP was assumed to also apply for Tug and IUS. The payload capabilities to geosynchronous orbit are 7,000 lbs for Tug and 5,000 lbs for IUS. In the case of the recoverable Tug, when a servicer system is also returned, the payload capability to orbit is reduced. For a servicer system weight of 500 lbs (round trip) the total up payload capability is 6,250 lbs. The 750 lbs reduction in payload capability is charged as additional servicing program payload.

Additional costs are incurred through premium payments charged for late scheduling. Replacing failed expendable satellites or servicing failed satellites cannot be accurately scheduled. Ten or 17 percent premiums were charged for late scheduling and they are discussed as used.

As the Shuttle/Orbiter launch cost was taken as 12.0 million 1975 dollars in the first IOSS, it can be seen that a significant change in the launch costs in 1977 dollars may be anticipated. However, the cost sensitivity analysis indicated that increases in launch costs increased savings but not very significantly for geosynchronous satellite programs.

b) Expendable Spacecraft Program Costs - The expendable program cost equations were defined. The spacecraft cost items are as used in the initial IOSS. All terms were recalculated to assure conformance to the assumed mission model and to determine costs in 1977 dollars.

The Orbiter and Tug/IUS launch costs are comprised of two terms--1) emplacement of the initial fleet size with no premium payments, and 2) emplacement of replacement satellites after failures with premium payments due to late scheduling. A 17% premium was used. This relates to the assumption that the replacement launches can be scheduled six months in advance. Shorter response times would be very difficult to achieve. Longer periods would result in excessive satellite downtime.

A portion of each satellite would be maintained on the ground to be used in operations analyses, troubleshooting, and verifications of new commands/operations. This spares allocation is identified as the S term. This term represents the ratio of unique modules to the spacecraft weight and is the same as RR, derived in Table IX-5.

The resulting program costs were summarized for the two options of launching with Shuttle/Tug or Shuttle/IUS. The $7.03B costs for the Tug-boost option is about $1.8B or 33% higher than previously calculated in the IOSS for the geosynchronous satellites. This increase is primarily due to the increase in Orbiter/Tug launch cost estimates (97% increase) and 17% escalation to 1977 dollars in all other items. The spares allocation is a new contributor.

A suboption to the Tug launch of replacement satellites was analyzed wherein two satellites were launched where the total weight did not cause the Orbiter capacity to be exceeded. This resulted in 44 missions instead of 75. However, because of extra payload allowance for Tug propellant to place the second satellite at the desired longitude, the program costs for this suboption were actually about $90M greater. Therefore, this suboption was no longer carried in the evaluations.

The cost of launching the satellites using the Shuttle/IUS is about $420M higher than using the Shuttle/Tug. This is primarily due to the fact that the IUS is expendable and costs about $5.7M per basic mission, compared to $2.4M for the recoverable Tug.

c) <u>On-Orbit Maintainable Spacecraft Program Costs</u>

1) <u>Demand Launch from Earth</u> - The cost equations for the demand-launch-from-earth options were established. The first seven items were as in the initial IOSS except the refurbishment operations. The term R was originally (1 + R) to account for both the launch checkout costs (R = .09) for the replacement spares and the cost of the spares. The replacement spares costs have now been moved to the last term and combined with other spares costs. With all on-orbit servicing options there are three spares categories costed:

$RR \times pf \times (1 + sf) \times C_{S/C}$	-	full complement of unique replacement modules for each type satellite
$(n - n_f) \times pf \times (1 + sf) \times C_{S/C}$	-	spares to replace failed modules;
$pf \times (1 + sf) \times C_{S/C}$	-	partial complement of spares for eac satellite program where n_f is greate than one.

In the demand-launch options, only the replacement spares category is launched. The others are purchased and maintained on the ground to be available for a servicing mission if the basic set of spare modules should fail.

The servicer system costs were determined using the results of the Task 5.2 memorandum, with modifications as applicable. The share of the DDT&E costs for the geosynchronous mission was taken to be:

C_{SN} = \$12.54M.

Production costs (C_{S1}) were initially based on a need for two flight units and two spares for the Tug-demand-launch option a). As will be noted below, it was later decided to expend all servicers going to geostationary orbit because of the high costs of returning equipment. Seventy-five flight units, with two spares, were provided for the IUS-demand-launch option b). A 90% learning curve was used in all analyses to determine multiple production costs. Examples of the total servicer production costs are shown in Tables IX-6 and -7.

Table IX-6 Option a--Tug Demand Launch - Servicer Production Costs (\$M)

WBS Element	Unit Cost	Production (2)
Project Management (at 6%)	0.12	0.44
Project Engineering and Integration (at 11%)	0.20	0.80
Structure and Thermal	0.98	
Mechanisms	0.35	3.24
Control Electronics	0.30	
Assembly and Checkout	0.16	
Airborne Spares		4.02
TOTALS	2.11	8.50

Since the geosynchronous missions are a subset of the total maintenance program evaluated in the Task 5.2 memorandum, the servicer operations costs (C_{S2}) in this study were derived by ratios of the Task 5.2 results. The ratios were determined by considering the type of activities.

The Orbiter and Tug/IUS launch costs come from two sources--1) emplacement of satellite fleet size,

$$\Sigma\ (n_f + LF)\ [a\ b\ C_o + b\ C_{T/I}]$$

where LF is from the initial IOSS and no premium is paid for the STS mission,

Table IX-7 Option b--IUS Demand Launch - Servicer Production Costs ($M)

WBS Element	Unit Cost	Production (75)
Project Management	0.12	4.43
Project Engineering and Integration	0.20	8.11
Structure and Thermal	0.98	69.75
Mechanisms	0.35	
Control Electronics	0.30	
Assembly and Checkout	0.16	
Airborne Spares		4.02
TOTALS	2.11	86.31

and 2) launch of servicing missions,

$$\Sigma \ (1 + LF_4) \ [a \ b \ C_{op} + b \ C_{TP/IP}]$$

where LF_4 is a new loss factor term and launch premiums are paid (17%).

The LF_4 term is derived from summing unreliabilities for the operations involved. For single servicing missions, LF_4 = 0.07 and for dual servicings LF_4 = 0.13.

Payload weights (e.g., Table IX-4) were used to calculate the servicing launch costs. Initially, the Tug servicing option included the return of the servicing system. However, additional calculations revealed a cost savings by expending the servicer system in orbit. Because of the 750 lb payload penalty to return the 500 lb servicer system, program launch costs were $198M greater to return the servicers. The 40 expendable servicers cost an additional $45M resulting in a net savings of $153M. For the Tug servicing option, 800 lbs were charged to the servicing payload for dual servicings to account for maneuvering propellants. A 17% premium was charged because of a need for about six months' scheduling time.

Program cost analyses for the Tug and IUS demand-launch options (servicer expended) were made. Program costs were about \$162M higher using the IUS. One would expect a greater difference because more IUS flights are made and costs are greater. However, the Orbiter costs are less with the smaller IUS because of lower Orbiter load factors. The Tug-servicing program costs of \$5.69B were about \$1.66B higher than those previously calculated in the IOSS for the geosynchronous satellites. This increase is due to about \$1.3B increase in STS launch costs and escalation to 1977 dollars and to about \$0.3B for the spares included in this analysis.

2) Fully-spared Rover Warehouse - The cost equations for the orbital warehouse options (fully-spared rover, partially-spared rover, and fixed) were identified. These equations are as in previous IOSS work except for differences previously discussed (spares costing and new launch cost equations) and the addition of cost terms for the propulsion stage and warehouse systems. Servicer production and operations costs were calculated as previously discussed. For option c, five CPS/servicer flight units and two spares are needed. For option d, three SEPS/servicer flight units and two spares are needed.

The DDT&E and production costs for the CPS were based on similarity to the Common Support Module cost estimates in the *Geosynchronous Platform Definition Study*, NAS9-12909, Rockwell SD 73-SA-0036-6, June 1973. Operations costs for the CPS were based on ratios of the applicable activities to the servicer operation costs. DDT&E costs for the SEPS were assumed absorbed in the total STS program since this vehicle would be another STS element like the IUS or Tug. SEPS mission cost estimates were given in Boeing studies. The SEPS operations costs were also determined by ratio of activities to the servicer operations costs.

The warehouse was assumed made up of spares racks and either a spares-retrieval mechanism or stabilization system. The structure and thermal system costs previously included in the servicer system cost analyses were used to estimate or calculate the warehouse costs. The options c and d warehouse will eventually become 52 racks in size. With two spares, rack production costs will be about \$29M. The racks are assumed accumulated at the warehouse rather than rearranging spares stowage locations and returning the empty racks to earth. The cost of retrieving payloads from geosynchronous orbit is about $2.6M per rack. The production cost of each rack is less than $1M. Therefore, it is cost effective to not recover racks. Four mechanism or support systems will be needed. With one spare, these were estimated at $10M production costs. Warehouse operations are included to some extent in the servicer operations. However, an additional 25% of servicer operations was included for handling warehouse systems, spares, etc.

Flights could be scheduled on the order of about a year beforehand based on servicing system reliabilities and less urgency to replace the warehouse spares used. Therefore, a 10% premium was charged on launch costs.

The results of the program cost analyses for the fully-spared rover warehouse servicing options using either the CPS or SEPS as the orbital transfer vehicles were determined. The costs are only about $174M different with the CPS at $5.98B, and would be closer if some of the SEPS DDT&E costs were charged to the servicing programs.

3) <u>Partially-spared Rover Warehouse</u> - Cost considerations for these options are the same as for the fully-spared warehouse. However, since only the spares actually needed for replacement in orbit are launched, the launch costs and warehouse costs are less. There will be 40 racks of spares launched for the one-circuit servicing options (e for CPS and g for SEPS), and 43 racks of spares for the two servicing circuits per year options (f for CPS and h for SEPS).

Since the spares are launched after satellite failures occur and spares needs are determined, less time is available for getting the spares into orbit. Therefore, six months scheduling and a 17% premium on launch cost is assumed.

The results of the program cost analyses for the four suboptions for the partially-spared rover warehouse were determined. These options are about $90M to $240M lower than the fully-spared warehouse options, primarily because of fewer spares launches.

4) <u>Fully-spared Fixed-location Warehouse</u> - Cost considerations for these options are the same as for the rover warehouse with the exception of

added DDT&E and production costs for a spacecraft bus to maintain the warehouse at a fixed longitude in orbit while the propulsion stage is on a servicing mission. The spacecraft bus is assumed to have DDT&E ($64.05M) and production ($56.5M for five units) costs similar to the CPS, which was based on the estimates for the geosynchronous platform support module studies. Since there is less urgency in replacing the spares on-orbit, scheduling of one year with a premium of 10% on launch costs was assumed.

The results of program cost analyses for the four suboptions to the fully-spared fixed-location warehouse method were determined. These costs are close to those for the fully-spared rover warehouse. Less propellant is required for orbit phasings in this option than when the full warehouse is moved. However, this saving in launch costs for propellants is offset by the increased warehouse costs.

7. Tradeoff Evaluations

The operations, program costs, and principal advantages/disadvantages of each of the 12 servicing options and baseline launches of expendable satellites are presented in Table IX-8. The total range of servicing options analyzed have program costs between $5.67B and $6.03B, for only a ±3% variation from the mean. However, the servicing options are significantly lower than program costs from the use of expendable satellites by $1.0B to $1.8B (14 to 24%).

From the advantages/disadvantages lists there are three other major comparison features:

- Compatibility with baseline servicer system;
- Servicing response to satellite failures;
- Unique benefits or undesirable features.

The baseline servicer is designed to handle spares stowed in one integral rack. More than a one-rack configuration would require other methods for translating the spares to the servicer or to move the servicer about the stowage areas to access the spares.

The value of on-orbit satellite availability varies with the programs. Availability time is generally less important to the observatory satellites

Table IX-8 Evaluation of Servicing Options

OPTIONS	OPERATIONS SUMMARY	PROGRAM COSTS, $B	ADVANTAGES	DISADVANTAGES
1.a - Tug/servicer launched on demand	Servicer mission flown after satellite failure. STS missions scheduled based on reliability predictions. Spares maintained on ground.	5.688	• Fair response to failures - limited by STS availability. • Compatible with baseline servicer.	• Up to 6 STS missions per year. • STS scheduled based on satellite reliability predictions.
1.b - IUS/servicer launched on demand	Servicer mission flown after satellite failure. STS missions scheduled based on reliability predictions. Spares maintained on ground.	5.850	• Fair response to failures - limited by STS availability. • Compatible with baseline servicer.	• Up to 12 STS missions per year. • STS scheduled based on satellite reliability predictions. • Needs IUS rendezvous and docking capabilities.
2.a - Fully-spared rover warehouse with CPS/servicer - one servicing circuit per year	Large warehouse carried along on servicing trips. After servicings, spares are replaced from ground stores. Full spares complement maintained in warehouse.	5.983	• Fair response to failures - 1/2 year average downtime since failure. • Up to 3 STS missions per year. • Can service unexpected or recent failures. • More compatible to STS scheduling.	• Several months for servicing trip. • Large warehouse not compatible to baseline servicer - needs access mechanism or stabilization system.
2.b - Fully-spared rover warehouse with SEPS/servicer - one servicing circuit per year	Large warehouse carried along on servicing trips. After servicings, spares are replaced from ground stores. Full spares complement maintained in warehouse.	5.809	• Fair response to failures - 1/2 year average downtime since failure. • Up to 3 STS missions per year. • Can service unexpected or recent failures. • More compatible to STS scheduling.	• Up to 2 months for servicing trip. • Large warehouse not compatible to baseline servicer - needs access mechanism or stabilization system. • SEPS requires refueling or planetary version.
3.a(1) - Partially-spared rover warehouse with CPS/servicer - one servicing circuit per year	Spares launched and servicing trip performed after satellite failures. Full spares complement maintained on ground	5.784	• Fair response to failures - 1/2 year average downtime plus STS scheduling. • Up to 2 STS missions per year.	• Several months for servicing trip. • Requires late STS scheduling. • Up to 6 racks carried along - not compatible.
3.a(2) - Partially-spared rover warehouse with CPS/servicer - two servicing circuits per year	Spares launched and servicing trips performed after satellite failures. Full spares complement maintained on ground.	5.812	• Fair response to failures - 1/4 year average downtime plus STS scheduling. • Up to 2 STS missions per year.	• Several moths for servicing trips. • Up to 3 racks carried along - not compatible. • Requires late STS scheduling.
3.b(1) - Partially-spared rover warehouse with SEPS/servicer - one servicing circuit per year	Same as 3.a(1)	5.674	• Fair response to failures - 1/2 year average downtime plus STS scheduling. • Up to 2 STS missions per year.	• Up to 2 months for servicing trip. • Requires late STS scheduling. • Up to 6 racks carried along - not compatible. • Requires SEPS refueling or planetary version.
3.b(2) - Partially-spared rover warehouse with SEPS/servicer - two servicing circuits per year	Same as 3.a(2)	5.764	• Fair response to failures - 1/4 year average downtime plus STS scheduling. • Up to 3 STS missions per year.	• Up to 2 months for servicing trip. • Requires late STS scheduling. • Up to 3 racks carried along - not compatible. • Requires SEPS refueling or planetary version.
4.a(1) - Fixed-location fully-spared warehouse with CPS/servicer - two servicing circuits per year	Large warehouse with full spares complement maintained at fixed longitude. After satellite failures, required spares carried on servicing circuit.	5.994	• Good response to failures - 1/4 year average downtime. • Fair compatibility to STS scheduling. • Up to 3 STS missions per year.	• Up to 3 racks carried along - not compatible. • Requires warehouse bus.
4.a(2) - Fixed-location fully-spared warehouse with CPS/servicer - incremental servicings	Same as 4.a(1) except servicing trip services two satellites and returns to warehouse. Up to six trips per year are planned.	6.033	• Compatible with baseline servicer. • Good response to failures - 1 month max. • Most compatible to STS scheduling. • Up to 3 STS missions per year.	• Requires warehouse bus.
4.b(1) - Fixed-loaction fully-spared warehouse with SEPS/servicer - two servicing circuits per year	Same as 4.a(1)	5.928	• Fair response to failures - 1/4 year average downtime. • Fair compatibility to STS scheduling. • Up to 3 STS missions per year.	• Up to 3 racks carried along - not compatible. • Requires SEPS refueling or planetary version. • Requires warehouse bus.
4.b(2) - Fixed-location fully-spared warehouse with SEPS/servicer - incremental servicings	Same as 4.a(2)	5.990	• Compatible with baseline servicer. • Good response to failures - 1 week max. • Most compatible to STS scheduling. • Up to 3 STS missions per year.	• Requires SEPS refueling or planetary version. • Requires warehouse bus.
Baseline 1 - Launch expendable satellites by Tug	Replace failed satellites with new satellites.	7.033	• Allows updated satellite technology.	• Most expensive Tug option. • Up to 12 STS missions per year.
Baseline 2 - Launch expendable satellites by IUS	Replace failed satellites with new satellites.	7.451	• Allows updated satellite technology.	• Most expensive option. • Up to 12 STS missions per year.

than to the communications satellites. However, the larger communications programs provide redundant subsystems and even redundant (standby) satellites. For the mission model analyzed, 581 years of on-orbit time is provided for about \$6B. Assuming 10% standby redundancy is included in this operating time, the general assumption can be made that on-orbit operating time costs about \$1M per month. This value was used in comparing servicing response by the various options.

a) <u>Demand-launch Options</u> - The Tug/servicer (a) and IUS/servicer (b) demand-launch suboptions are basically compatible with the baseline servicer since only one rack is required. The greatest deficiency in these options is the delayed response to satellite failures, since a servicing mission is flown after satellite failures occur and the specific servicing needs are identified. However, as will be seen later, satellite downtime may not be a large disadvantage if examined on a cost basis.

Since the Tug option is less costly than the IUS option and since the present IUS configuration would not have the capability to rendezvous and dock with failed satellites, the Tug/servicer demand-launch option is favored.

b) <u>Fully-spared Rover Warehouse Options</u> - Because of the large warehouse required with these options, additional access mechanisms will be needed to translate spares to the servicer or a stabilization system would be required to permit the servicer/propulsion system to disengage and move about the warehouse to acquire the required spares.

Since the warehouse contains a full complement of spares and used spares are replaced later, the STS scheduling considerations discussed for options a and b are not as critical. However, since only one servicing circuit per year is conducted, there would still be an average delay of six months prior to starting the servicing circuit and an additional several months for the CPS and up to two months for the SEPS to make the circuit. The present earth operations SEPS would require refueling during its lifetime or the planetary SEPS with a larger propellant tank would be required. Neither appears to present a significant problem.

These options do provide the unique benefit that many satellite failures detected after the servicing mission starts can also be corrected since more spares are carried along. Other options only carry the spares known to be needed.

Of the two suboptions, the SEPS method is preferred because of lower costs and less time required. These advantages of SEPS over the CPS are consistent through all options analyzed.

c) Partially-spared Rover Warehouse Options - These options have the disadvantages of warehouse incompatibility with the baseline servicer and slower response times. The one-circuit per year options result in average satellite down times of six months during the servicing trips and the two-circuit options result in average downtimes of three months. Since the spares are launched from the ground after servicing needs are identified, the STS delays discussed in options a and b are also imposed.

d) Fixed-location Warehouse Options - These options have several advantages but are generally more costly. Since fewer spares are carried along, the servicing missions can be achieved faster. Incremental servicing of two satellites per trip can be achieved in a month using the CPS and in a week using the SEPS. This method should be feasible since satellite failures would be random and dispersed such that the several incremental trips should accommodate servicing needs and maintain good satellite availability. The incremental servicing method requires only one spares rack and is therefore compatible with the baseline servicer.

The two-circuits per year suboptions result in an average satellite downtime of three months during the servicing missions. Also, these suboptions require up to three spares racks and are therefore not compatible with the baseline servicer. These suboptions require a complete warehouse bus to maintain orbital positioning and housekeeping functions. However, no technology problems are foreseen for this. STS scheduling is not a critical factor since the spares are replaced after the servicing missions.

The SEPS incremental servicing option (ℓ) is preferred among these options because of the lower costs, compatibility with baseline servicer, and rapid response to servicing.

e) Expendable Satellites - Program costs and STS scheduling requirements are worst for the expendable-satellites methods. However, expendables have the advantage of permitting the update of satellite technology and capabilities through ground modifications prior to launch of later satellites. Although these features could be achieved through on-orbit spares replacements, they can be obtained more completely on the ground.

f) General - A decision tree which summarizes these discussions was prepared. The suboptions within each option category were first compared based on program costs and servicing response time. Then the options were compared. Where no clear-cut preferences exist, the options were compared on effective costs when considering servicing response times. For example, considering the $1M per month "satellite-availability-rate" previously developed, the Tug-demand launch option will cost abour $240M in loss of satellite availability, relative to the SEPS-incremental-servicing option. This is based on a six month preparation time for each of the 40 Tug-demand missions (most were dual missions). The actual servicing times are comparable because the SEPS option also services two satellites in a week. Therefore, the equivalent program costs for the Tug option would be $5.928B compared to $5.990B for the SEPS option. STS preparation times of seven or eight months for the Tug-demand-launch option would make the two options about equal in equivalent costs.

Based on these results, the Tug-demand-launch option would be preferred. The fixed warehouse with SEPS incremental servicing option would be the recommendation if fast servicing response is desired and the increased program costs are acceptable. Another advantage of these two recommended options is that they are both compatible with the baseline servicer, requiring only one stowage rack.

Servicing system investments for the servicer, warehouse, and propulsion vehicle will run $84M to $374M, or about 2% to 6% of the program costs, for the various servicing options. In terms of investments versus potential savings over the use of expendable satellites, the extremes are about 6% and 26% for options a and ℓ. That is, the servicing investment for the Tug-demand option is about $84M to achieve a savings of $1.35B. For the fixed warehouse with incremental SEPS servicing, the investment is \$273M for a potential savings of \$1.04B.

The mission model used in these analyses had peak servicing years about 1991, with the spacecraft programs dropping off later. If the programs continued at the peak rate, no significant changes in the results of this study would occur since servicing capabilities were investigated for the worst case years. If the programs should expand requiring more servicings, one would expect an increase in net savings from servicing since more efficient use of the servicing systems would occur and launch load factors would be greater, resulting in more payload capabilities with disproportionally less additional launch costs; e.g., any payload weight over the 75% load factor on the Tug would not increase the costs.

B. SERVICER SYSTEM OPERATIONS SUPPORT

The analysis objective was to reevaluate all elements of the servicer system operations costs and support activities to determine if the costs were properly stated and if all support activities had been identified. Two aspects were addressed in significant detail. First is the communications links between the various elements of the on-orbit servicing operation for both low and high orbits. Each link using the STDN, TDRSS, and NASCOM was identified and the time of usage for representative servicing missions was determined. Second is a bottoms-up identification of the servicer system operations costs as opposed to the similarity approach of the first IOSS.

Servicer system operations costs were addressed in the first IOSS. In the final report, MCR-75-310, *Integrated Orbital Servicing Study for Low Cost Payload Programs*, Volume II, *Technical and Cost Analysis*, September 1975, the supporting data was introduced in different chapters and thus was difficult to follow. All the necessary material was collected and is presented along with the details of the current task in a coherent fashion in a reference memorandum, *Servicer System Operations Support--Task 5.2*, April, 1977. This section of the final report is a summary of the approach and results. For further details the reader is referred to the referenced memorandum.

The approach used in this analysis consists of the steps listed in Table IX-9 and are as follows. First, the ground rules for the analysis were established. These included consideration of both high and low earth orbits, use of the Chapter VI on-orbit servicer design, and the mission model used. The mission phases of Table III-15 of the first IOSS were reviewed, updated, and defined for the on-orbit maintenance mode to embrace each and every mission phase. The functional requirements and equipment required to support the orbital service and maintenance were then identified. Next was the definition of a series of cost allocation elements such as servicer, Orbiter, Tug, and mission operations center. The potential communications system links were identified and expressed in terms of planned STDN, TDRSS, and NASCOM links.

The heart of the analysis is the allocation of each of the specific functional and cost generating requirements to a cost allocation element. This

Table IX-9 Analysis Approach

1. Identify Ground Rules
2. Identify Mission Phases
3. Identify Functional Requirements
4. Define Cost Allocation Elements
5. Define Communications System Links
6. Allocate Functional and Cost Generating Requirements
7. Isolate Orbiter/Tug/Spacecraft Extra Cost Functions
8. Identify Communications Link Usage
9. Define Mission Models
10. Define Ground Processing Operations
11. Identify Servicer Cost Effects
12. Introduce Work Breakdown Structure
13. Determine Operational Site Services Costs
14. Summarize Servicer Operations Costs
15. Summarize Servicer System Costs
16. Present Example Spacecraft Program Costs

allocation was done in a manner (coding) that clearly identified those functions which had been included in the basic system (e.g., Orbiter launch costs) and those which are extra cost items. The servicer system operational cost generating requirements were also explicitly identified. The extra cost functions relating to Orbiter, Tug, and serviceable spacecraft were next isolated and discussed. Usage of the various communications links were identified and summarized on a per-mission basis.

Two mission models were defined based on reductions to the large mission model of the first IOSS. The ground processing operations from the first IOSS and the Multi-mission Support Equipment studies were next reviewed to form a basis for evaluating operations costs. The ramifications of mission model sizes on servicer production units required and operations requirements for both ETR and WTR were then identified. The basis of cost collecting, namely the work breakdown structure (WBS), was introduced next. The basic information from the allocation of functional requirements to servicer operations was reviewed in terms of the ground processing flow and mission models, converted to costs, and collected under the WBS as operational site services costs and then as servicer operations costs. Finally, the costs were extended to a total cost of operations analysis for the servicer system and for two example spacecraft programs--the SEOS and LXRT.

The effect of this analysis is to include all ground and flight equipment to support on-orbit maintenance operations. It also addresses directly, through the function allocation step, compatibility of orbital servicing with the Shuttle Orbiter and Tug. Those areas where compatibility does not clearly exist were identified as extra cost functions.

The Tug definition used was the same as that used in the first IOSS. It is a full capability tug, is reusable, can rendezvous and dock, and can take 7,000 lbs to geosynchronous orbit or return 3,000 lbs from geosynchronous orbit. Note that NASA has no firm plans for an upper stage with the required capabilities. The Interim Upper Stage (IUS) current definition does not include rendezvous and docking. The Solid Spinning Upper Stage (SSUS) does not even have three-axis, body-fixed attitude control. The need for an upper stage that will have the necessary capabilities must be addressed if orbital maintenance in other than low earth orbits is to be realistically addressed.

1. Ground Rules

The selected ground rules are shown in Table IX-10. These resulted from the contract statement of work and a review and updating of the costing ground rules used in the first IOSS. The Orbiter launch costs were taken from recent documentation as $20 million in 1975 dollars. To simplify the analysis, the launch cost sharing ratios used in the first IOSS were retained. The servicer stowage rack is lighter in weight now so the Tug costs shown are slightly high. The Orbiter launch costs are the same because they were length-critical rather than weight-critical. The servicer stowed length is similar to the first IOSS value.

2. Mission Phases

A set of servicing mission phases was identified and is presented in Table IX-11. These mission phases were adapted from Table III-15 of the first IOSS

C-5

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Table IX-10 Cost Estimation Ground Rules

Base on costing approach of the first IOSS On-orbit Maintenance using the 6 DOF servicer mechanism. All costs are 1977 dollars. STS Costs per Flight - Orbiter - $23.4 Million - Tug - $2.4 Million Shuttle IOC from ETR - 1980 Shuttle IOC from WTR - 1983	Full-capability Tug IOC - 1984 Launch Cost Reimbursement Policy Sharing Ratios per first IOSS One 1975 Dollar inflates to 1.17 1977 Dollars Two Mission Models 75% of first IOSS 50% of first IOSS

Table IX-11 Servicing Mission Phases

MISSION PHASES	APPLICABILITY		MISSION PHASES	APPLICABILITY	
	ORBITER	TUG		ORBITER	TUG
Pre-mission	X	X	Docking (Attachment)	X	X
Mission Preparation	X	X	On-Orbit Servicing Operations	X	X
Prelaunch	X	X	Undocking (Release) of Spacecraft	X	X
Orbiter Launch/Ascent	X	X	Servicer Stowage	X	X
Orbiter Orbital Operations	X	X	Spacecraft Preparation for Normal Operation	X	X
Tug/Servicer Checkout		X	Spacecraft Checkout	X	X
Tug Deployment		X			
Tug Orbit Transfer		X	Tug Return to Orbiter		X
Tug Orbital Operations		X	Tug/Orbiter Rendezvous, Retrieval and Stowage		X
Rendezvous	X	X	Orbiter Preparation, Reentry, and Landing	X	X
Servicer Checkout and Deployment	X	X			
Spacecraft Preparation for Servicing	X	X	Post-mission	X	X

final report, Volume II. A new phase--Servicer Stowage--was added. There was some reordering of phases to show spacecraft checkout being performed from the ground as is conventional practice with current spacecraft programs. There are 22 Tug (medium or high earth orbit) mission phases and 16 Orbiter (low earth orbit) mission phases. Each of the servicing mission phases is defined in Table III-B-2 of the Servicer System Operations Support memorandum to clearly identify what functions belong in which mission phase.

Table IX-11 indicates overlap in mission phases between Orbiter and Tug. To simplify the analysis, both Tug and Orbiter missions were considered together. To help the reader reconstruct a straight Orbiter or a straight Tug mission, a coding system was adopted. It is: Orbiter mission only - O; Tug mission only - T; and either mission - E.

3. Functional Requirements

Functional requirements for each mission phase were identified and are detailed in Tables III-C-1 through -22 of the Servicer Systems Operations memorandum. These tables were adapted from work done on the first IOSS, but were not totally included in the final report. The requirements have been reviewed carefully and updated as appropriate. A draft of the first IOSS set of requirements was reviewed by the NASA. Their comments were carefully considered and included in the revision process. The approach was to definitely include all items which are part of on-orbit servicing. The objective in considering a requirement for inclusion was whether it might lead to a cost that had been overlooked, underestimated, or overestimated in the first IOSS costing process. The 22 mission phases included a total of 128 requirements for an average of six per mission phase. The number of requirements vary from one (Spacecraft Checkout) up to 32 (On-orbit Servicing). These requirements were used to generate the specific requirements and equipment associated with each cost allocation element.

4. Cost Allocation Elements

For an on-orbit servicing mission, the costs are generated from a multitude of requirements. It was deciced to collect costs in the same general method as used for the first IOSS. The top level breakdown then becomes--1) Orbiter, 2) Tug, 3) Spacecraft, and 4) Servicer System. Within these elements it was also necessary to consider whether a cost was included as part of a standard service or whether it was an extra cost requirement. The reference used for the Orbiter costs was *Buying a Shuttle Ticket,* W. F. Moore and C. Forsythe, Astronautics and Aeronautics, January 1977. Moore and Forsythe give the basic Orbiter cost as $19 to $20.9 million in 1975 dollars. They identified the standard and optional Shuttle services as shown in Table IX-12. Table IX-12 was used to identify functions that would be included as part of the standard cost.

Moore and Forsythe present the launch cost sharing rules and these have a minor variation for small payloads that does not impact the final IOSS cost sharing and a major difference for higher load factors. The Moore and Forsythe

Table IX-12 Standard and Optional Shuttle Services

STANDARD	OPTIONAL
Payload design review	Payload mission planning
Orbiter flight planning	Nonstandard-orbit flight and planning
Payload safety review	Launch-window constraints
Payload installation, verification, and compatibility	Special integration and testing
Three-man flight crew	Special crew training
One day of mission operations	Short-term call-up
On-orbit payload handling	Upper stages and services
Transmission of payload data	Mission kits
Deployment of free flyer	Spacelab, LDEF, and special equipment
Standard mission destinations--	Revisit and retrieval
• altitude, 160 n mi	EVA
• inclination, 28.5 or 56 deg	Additional time on orbit
	Payload data processing
	VAFB launch at 90 or 104 deg, inclination standard mission available

curve goes to 100% cost at 75% load factor and then is constant at 100%. The first IOSS data continued on up to 141% cost at 100% load factor. This resulted because of the use of an average load factor of 70% in the first IOSS which implied the 141% cost at 100% load factor. A review of the Orbiter load factors used in the first IOSS showed only two cases (AST-8 and PHY-3B) for expendable or on-orbit maintainable when the load factor was greater than 0.75 (0.78 and 0.76, respectively). Thus the first IOSS launch cost sharing factors will continue to be used.

A new feature has been added to the LCRP which may increase on-orbit maintenance cost. This feature is a premium for launch scheduling less than three years before launch. The premium is 17% for six months before launch and 21% for three months before launch. There is also a feature that permits rescheduling flights for a smaller premium. As there is no convenient way to determine how far before launch a maintenance flight will be scheduled, a 17% premium was used on all Orbiter and Tug flights. This premium was applied to maintenance and to replacement flights, but not to those which establish the on-orbit fleet. Note that only space-available flights can be scheduled on less than a one-year-ahead basis. What this means in terms of a tug maintenance or replacement mission is not clear. As no information is available on what would be included as standard on a Tug flight, the same guidelines were used as for the Orbiter flights.

The servicer system functions were allocated into two groups. Those associated with DDT&E and production activities are the first group, while the second group is the servicer system operation costs. All the functions relating to servicer operations costs were identified as a class. This is so the total servicer operations costs could be costed from the bottom up. Four additional major elements were identifed to ensure all areas are properly considered. These are--1) ground support equipment, 2) communications links, 3) mission operations control center (MOCC), and 4) other. Standard and extra-cost codes were generated using the above rules and are presented in Table IX-13. These codes are used to indicate the allocation of costs in later paragraphs.

Table IX-13 Cost Allocation Element Codes

CODE	DEFINITION
A	Orbiter standard service
B	Orbiter extra-cost service
C	Tug standard service
D	Tug extra-cost service
F	Serviceable spacecraft function
G	Not previously identified as a serviceable spacecraft function
H	Servicer system DDT&E or production function
I	Servicer operations function

5. Communications System Links

The communications system links were addressed separately because of the number of possibilities and the desire to be able to cost this activity separately. The complexity involves the spacecraft locations, Tug and Orbiter, space-to-ground links, two-way aspects of each link, ground links, and the number of activity centers involved. The Tracking and Data Relay Satellite System (TDRSS) subnet services spacecraft in low earth orbit. It consists of a ground terminal in the continental United States, and two active and one standby TDR geosynchronous satellites. The Spaceflight Tracking and Data Network (STDN) will be used during launch and for communicating with spacecraft at high earth orbits and in particular at geosynchronous orbit. The six basic sites provide full coverage for geosynchronous orbit spacecraft.

The TDRSS ground station and all the STDN sites are connected together to GSFC, JSC, MSFC, the launch facilities, and many other sites by the worldwide NASA Communications Network (NASCOM). The Network Operations Control Center (NOCC) is located at GSFC. The primary Mission Operations Control Center (MOCC) for manned operations is at JSC. Each unmanned spacecraft program will generally have a Project Operations Control Center (POCC). The POCCs are often at GSFC, although they may be any place that can be connected to NASCOM. To make things more clear, we will identify a user POCC and a servicer POCC. The user POCC is the one associated with the unmanned spacecraft which is to be repaired. The servicer POCC is a center set up to control servicer operations during a maintenance mission. The servicer POCC could be at MSFC or at JSC. The advantages/disadvantages of such a choice have not been addressed. To maintain flexibility a NASCOM link is shown. The activity centers and communications links involved in either high earth orbit (STDN) or low earth orbit (TDRSS) servicing missions are shown on Figure IX-5. The specific links and subnet elements that might be

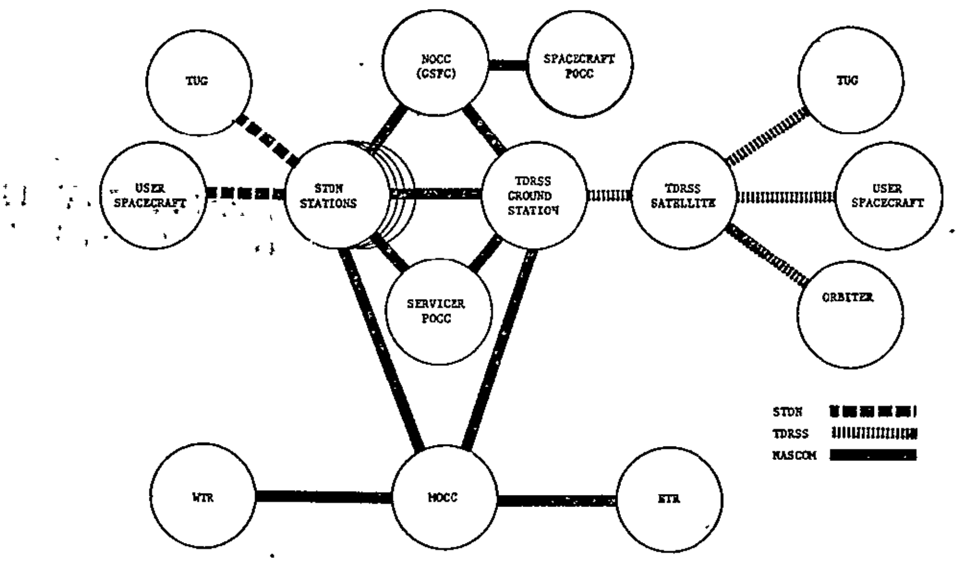

Figure IX-5 Communication Links for On-Orbit Servicing

involved in any of the servicing missions were addressed and are incorporated in the cost allocations.

6. Allocation of Functional and Cost Generating Requirements

The functional requirements for each mission phase were examined and expanded to identify cost generating requirements, hardware, or software for each of the major elements. These elements are Servicer, Orbiter, Tug, Spacecraft, Ground Support Equipment, Communications System Links, Mission Operations Center, and other. Next the specific cost generating requirements, hardware, or software were allocated to one of the cost allocation elements (Orbiter, Tug, Spacecraft, or Servicer System). The resulting data are presented in detail in Tables III-F-1 through -22 in the referenced memorandum, Servicer System Operations Support, April 1977. Each table covers a different mission phase. A sample is shown as Table IX-14.

During the first IOSS, tables similar to those generated under the current task were sent to NASA for review. Two sets of comments were received. These were reviewed carefully to ensure comprehension of the intent of the comment. Where the intent of the comments was still applicable they were included.

7. Orbiter/Tug/Spacecraft Extra Cost Functions

The costs allocated to the Orbiter, Tug, and serviceable spacecraft that were judged to lie within the standard charges were included in the previous cost allocations. The Orbiter and Tug extra cost functional requirements are listed in Table IX-15. Each of the extra cost functions associated with the Orbiter or Tug were considered nonstandard and thus were not included in the standard costs except for the PSS. When a PSS charging policy becomes available, its effect should be included. No serviceable spacecraft extra cost functions were identified.

Table IX-15 Extra Cost Functions

Provision of Fluids if Necessary (Thirteen Requirements)--Orbiter and Tug
Provision of Additional Orbit Maintenance System (OMS) Kits (One Requirement)--Orbiter
Provision of an On-orbit Servicer Controls and Displays Capability as part of the Payload Specialist Station (32 Requirements)--Orbiter
Equipment for Contamination Monitoring (Four Requirements)--Orbiter and Tug

8. Communications Link Usage

The number of usages of each communications link was determined, the extra-cost usages were designated, and the time intervals for each use were

Table IX-14 Functional and Cost Generating Requirements

Functional Requirements	SERVICER CHECKOUT AND DEPLOYMENT FUNCTIONAL REQUIREMENTS							
	Major Elements							
	Servicer	Orbiter	Tug	Spacecraft	GSE	Communications Links	Mission Operations Center	Other
1) Power servicer system up from standby* E	Power system H	Provide power-A Provide PSS - B	Provide power, Command system C			3D - I 4C, 4D - I	Data console A,C Command Console - C Use - I	
2) Monitor servicer system and replacement module status.* E	Sensors, C&W system H	Data console A	Data transfer system C	Sensors, C&W system H		3D - I 4D - I	Data Console-A,C Use - I	
3) Activate and test video systems on servicer.* E	Video system H	Data transfer system A	Data transfer system C			3D - I 4D - I	TV Console - A Use - I	
4) Operate and monitor video systems in Orbiter cargo bay O		Video system TV console A				3D - I	TV Console - A Use - I	
5) Monitor video systems during servicer deployment E	Video system H	Video system TV console A	Data transfer system C			3D - I 4D - I	TV Console - A,C Use - I	
6) Activate servicer mechanism.* E	Command system. Servicer system. H	Provide PSS B	Command system Data transfer system C			3D - I 4C, 4D - I	Command system,C Data Console-A,C Use - I	
7) Deploy servicer mechanism from launch stowage configuration to operating configuration.* E	Command system. Servicer system. H	Provide PSS B	Command system Data transfer system C			3D - I 4C, 4D - I	Command System,C Data Console-A,C Use - I	
8) Verify that servicer system is functional. E	Servicer system H	Provide PSS B	Command system. Data transfer system C			3D - I 4C, 4D - I	Data Console A, C Command system C TV Console A, C Use - I	

PSS - Payload Specialists Station
*From Orbiter to ground

developed. The number of link usages related to servicing operations is 168 and involves 14 of the 20 links hypothesized. The servicing operations link usage is broken down by mission phase in Table IX-16. The highest level of use is for the actual servicing operation. However, the next highest usage is for spacecraft preparation for servicing and for returning the spacecraft to its configuration for normal operation after servicing. To provide part of the

Table IX-16 Maintenance Associated Communications Link Usages

MISSION PHASES	COMMUNICATIONS LINK CODES														TOTAL	TIME/SPACECRAFT (HRS)		REMARKS
	1C	1D	2C	2D	3C	3D	4C	4D	7C	7D	9C	9D	10C	10D		ORBITER	TUG	
Premission -1															0	---	---	
Mission Prep -2															0	---	---	
Prelaunch -3	1	1	1	1											4	2	2	
Orb. Launch/A -4															0	---	---	
Orb. Orb. Opn. -5															0	---	---	
Tug/Serv. C/O -6					1	3									4	0	1	
Tug Deploy -7															0	---	---	
Tug Orbit Trans -8															0	---	---	
Tug Orb Opn -9															0	---	---	
Rendezvous -10															0	---	---	
Serv C/O & Deploy -11						8	4	7							19	3.2	3	
S/C Prep -12									2	2	6	6	6	6	28	1.5	1.5	Spacecraft despin is optional.
Docking -13						1		1							2	0	0	Contamination Monitor is optional
OOS -14						24	19	24		1		1		1	70	2.8	2.8	
Undocking -15						1		1							2	0	0	Contamination Monitor is optional
Serv. Stow -16						4	3	4							11	2	2	
S/C Norm -17						1	0	1	3	3	4	4	4	4	24	2.5	2.5	Spacecraft spin-up is optional
S/C C/O -16											1	1	1	1	4	1	1	
Tug Return -19															0	---	---	
Tug Retrieval -20															0	---	---	
Orb Reentry -21															0	---	---	
Post Mission -22															0	---	---	
TOTAL	1	1	1	1	1	42	26	38	5	6	11	12	11	12	168	15.0	15.8	

basis for communications system costing, time estimates for link usage during a servicing operation were made. Time to exchange one module was taken as 10 minutes and six modules per spacecraft were exchanged. The checkout times were related to these design module exchange times. Servicer checkout and deployment takes longer in the Orbiter because the Orbiter TV system will be used to verify operations. The total time estimates are similar for Orbiter and for Tug missions--15 and 15.8 hours respectively, as shown in Table IX-16.

For Orbiter missions, all the communications links are through the TDRSS and NASCOM. Table IX-17 shows the subnet usage for a representative Tug mission. The data of Table IX-17 and the corresponding data for the Orbiter (TDRSS--13 hours, NASCOM--15 hours) provides a basis for determining communications link charges to on-orbit servicing missions. An attempt was made to

Table IX-17 Communications Subnet Usage in Hours for a Tug Mission

MISSION PHASE	STDN	TDRSS	NASCOM
Prelaunch	----	-----	2
Tug/Servicer Checkout	----	1	1
Servicer Checkout and Deploy	3	-----	3
Spacecraft Preparation	1.5	-----	1.5
Orbital Servicing	2.8	-----	2.8
Servicer Stowage	2	-----	2
Spacecraft Normalization	2.5	-----	2.5
Spacecraft Checkout	1	-----	1
TOTAL	12.8	1	15.8

convert this data to dollars per mission. However, the basis for the rates have not yet been established. When such data becomes available, it should be included in the overall cost numbers.

9. Mission Models

The mission model used for the first IOSS was patterned after that used for justification of the Shuttle program and was somewhat larger than present considerations would project. The generation of a new model is an involved and time-consuming task. For the subject analysis, the effect of a mission model is to provide a basis for servicer production quantities, spares required, need for WTR operations, and level of servicer operations. It was thus decided to use two mission models, each somewhat smaller than the first IOSS model.

The first IOSS included a cost sensitivity analysis. That analysis postulated a 75% mission model and a 50% mission model. These models were used in the analysis. The specifics with regard to expendable spacecraft flights, or operating cycles (n), and on-orbit fleet size (n_f) are given in Table III-I-1 of the reference memorandum, Servicer System Operations Support, April 15, 1977. The reduced models were obtained by reducing the number of operating cycles. No programs were eliminated. The large reductions came in programs with a lot of operating cycles. The number of maintenance activities was then determined as $n-n_f$. These were allocated to ETR and WTR according to the orbital inclination of the spacecraft as given in the SSPD. Certain adjustments to the model were made to delete those years in which there was little activity. The results are given in Table IX-18. The table illustrates the tendency of servicing activity to go down as mission model size is reduced. Note also how the number of flights at WTR decreases even though the number of years has been reduced. The specific levels and number of years were used to calculate servicer system costs.

Table IX-18 Mission Models for Servicing Operations

MODEL	LAUNCH SITE	YEAR												TOTAL	NUMBER OF YEARS	AVERAGE MISSIONS
		80	81	82	83	84	85	86	87	88	89	90	91			
100%	ETR	2	5	6	7	10	11	14	17	21	23	24	25	165	12	14
	WTR	1	2	2	5	4	3	8	6	6	5	11	5	58	12	5
	Total	3	7	8	12	14	14	22	23	27	28	35	30	223	12	19
75%	ETR		3	4	3	9	5	9	12	16	21	18	21	121	11	11
	WTR			3	4	3	2	5	5	4	4	8	3	41	10	4
	Total		3	7	7	12	7	14	17	20	25	26	24	162	11	15
50%	ETR				2	5	6	7	9	11	11	13	11	75	9	8
	WTR					3	2	3	3	2	3	5	2	23	8	3
	Total				2	8	8	10	12	13	14	18	13	98	9	11

10. Ground Processing Operations

When it is realized that the expenses of maintenance occur on the ground during development, production and operations, it is obvious that the ground processing operations should be examined more carefully. Thirty-two operations are involved in the overall cycle of maintenance with fifteen of these occurring on the ground. The first IOSS included in-depth assessments of ground and flight operational requirements as applicable to the STS elements and servicing hardware end items (spacecraft, servicers, replacement modules, etc). Those assessments were made for the purpose of identifying those operational and support requirements which were common or unique to specific maintenance modes and/or servicing concepts. The assessments involved the use of ground processing operations flow charts. They were found sufficiently helpful that they were used in this analysis.

Details of the ground processing operations analysis are contained in the reference memorandum. The on-orbit servicer system was estimated to correspond to a medium level of complexity and a corresponding offline turnaround time would be 5.8 weeks. This gives a total turnaround time for a seven-day mission of 7.5 weeks. The first IOSS used a three week span. The effect of longer spans is to require more equipment at the launch site to maintain the launch schedule.

11. Servicer DDT&E and Production Cost Effects

The servicer DDT&E and production costs from the first IOSS were reviewed and updated. Where no changes are indicated, the data from the first IOSS final report can be used. The costs are expressed here in 1977 dollars. The applicable costing considerations are given in Table IX-19. The DDT&E costs were taken to be independent of mission model size. There is an effect due to operations at both ETR and WTR which has been included. The resulting costs are presented in Table IX-20.

Table IX-19 Costing Considerations

WBS ELEMENT	BASELINE ON-ORBIT SERVICER
Project Management	6% of subtotal
Project Engr. and Integration	11% of subtotal
Structures and Thermal	309 lb storage rack Rack/Tug adapter not required
Mechanism	140 lb Manipulator Arm
Control Electronics	45 lb--six 7.5-lb units
Assembly and Checkout	DDT&E 5% of hardware Production 10% of hardware
Airborne Spares	4 flight articles for 75% mission model 3 flight articles for 50% mission model Subsystems--partial
Airborne Support Equipment	482 lb Cradle-rack
Logistics	Logistics Management Inventory Control O&M Manuals Trainers (I ETR, 1 WTR) Training
Ground Support Equipment	Mechanical--47 units Electrical--15 units
Facilities	Rearrangement
Operational Site Services	Launch Operations Flight Operations Maintenance

The total DDT&E cost is $37.3 million dollars compared to $33.8 million 1977 dollars from the first IOSS. The difference is due to the greater capability (axial and radial module removal) of the current servicer design.

Table IX-20 On-Orbit Servicer DDT&E Costs

WBS ELEMENT	BASIS	COST ($M)
Project Management	6% of subtotal	2.1
Project Engr. and Integ.	11% of subtotal	3.5
Structures and Thermal	SAMSO Data	5.5
Mechanisms	Analogous to PDRM Data	2.2
Control Electronics	Analogous to PDRM Data	8.2
Assembly and Checkout	5% of hardware cost	0.8
Airborne Spares		N/A
Airborne Support Equipment		1.6
Logistics	Analogous to PDRM and Tug Data	6.6
Ground Support Equipment	Analogous to Tug	4.3
Facilities	Analogous to Tug	0.7
Operational Site Services	Analogous to Tug	1.8
TOTAL		37.3
PDRM = Payload Deployment and Retrieval Mechanism		

Unit and production costs for the on-orbit servicer system are given in Table IX-21. The production costs for the 75% mission model involve four units, and for the 50% mission model involve three units.

The airborne spares costs were based on four units for the 75% mission model and three units for the 50% mission model. The airborne support equipment was based on the number of Orbiter-only missions and required 5.8 equivalent units for the 75% mission model and 4.1 equivalent units for the 50% mission model.

The on-orbit servicer unit cost is 2.11 million dollars as compared to the first IOSS estimate of 2.11 million 1977 dollars. The increased servicer complexity was offset by the lighter stowage rack. Production costs for the 75% mission model are greater than for the 100% mission model of the first IOSS when expressed in 1977 dollars. This is due to the larger quantities of units required due to the longer turnaround times.

Table IX-21 On-Orbit Servicer Unit and Production Costs

WBS ELEMENT	COST ($M) Unit	75% Mission Model Production	50% Mission Model Production
Project Management	0.12	1.22	0.90
Project Engr. and Integ.	0.20	2.01	1.50
Structures and Thermal	0.98	3.92	2.94
Mechanisms	0.35	1.40	1.05
Control Electronics	0.30	1.20	0.90
Assembly and Checkout	0.16	0.65	0.49
Airborne Spares	-	8.04	6.03
Airborne Support Equipment	-	3.07	2.17
Logistics	-	-	-
Ground Support Equipment	-	-	-
Facilities	-	-	-
Operational Site Services	-	-	-
TOTAL	2.11	21.5	16.0

12. Operational Site Services Costs

The operational site services costs are addressed first as they are the largest element of the operations costs and because they are needed to help generate the logistics costs. The largest number of functional requirements by mission phase are for Mission Preparation, Servicer Checkout and Deployment, On-orbit Servicing, and Post-Mission. The operational costs were generated for each WBS element by identification of specific tasks, manloading and time. Minimum headcount levels were established as necessary. The headcount rules were applied to the mission models of paragraph 9 and totaled to give man-years for each WBS heading at each launch site for each of the two mission models. A cost per man-year was generated using an average across the type of individuals used and updated to 1977 dollars. The result is given in Table IX-22. The scheduled maintenance, unscheduled maintenance, and depot maintenance costs were obtained by ratioing against the first IOSS costs for these elements.

Table IX-22 Operational Site Services Costs, Millions of 1977 Dollars

WBS ELEMENT	75% Mission Model			50% Mission Model		
	ETR	WTR	TOTAL	ETR	WTR	TOTAL
Years	11	10	-	9	8	-
Site Services and Support	2.51	1.31	3.82	1.77	1.05	2.82
Mission Planning	9.30	3.18	12.48	5.83	1.89	7.72
Flight Control	1.44	--	1.44	1.18	--	1.18
Flight Evaluation	1.38	0.67	2.05	0.85	0.52	1.37
Scheduled Maintenance	1.92	0.91	2.83	1.26	0.64	1.90
Unscheduled Maintenance	2.87	0.91	3.78	1.90	0.64	2.54
Postflight Checkout	1.72	1.31	3.03	1.19	1.05	2.24
Tug Mating and Checkout	0.88	0.35	1.23	0.55	0.26	0.81
Depot Maintenance	1.92	1.82	3.74	1.26	1.27	2.53
TOTAL	23.94	10.46	34.40	15.79	7.32	23.10

The 75% mission model total cost is about 70% of the comparable number for the first IOSS when expressed in 1977 dollars. This is partly due to the fewer missions being addressed. The operational site services cost per mission ($0.24M) for the 50% mission model is 110% of that ($0.21M) for the 75% mission model. This implies that the skills retention aspects (minimum headcounts at each launch site) are not as serious as had been expected. Mission planning is the dominant cost source.

13. Servicer Operations Costs

The operational site servicer costs were combined with the elements of the work breakdown structure to obtain the total servicer operations costs. Five logistics-type requirements were identified. Four of these can be grouped together and handled by three people at ETR and two at WTR. The fifth activity is training. Training was estimated at ten percent of the operational site services costs for all activities except scheduled maintenance, unscheduled maintenance, and depot maintenance where the training was estimated at five percent of their costs. Ground support equipment operations costs (software and equipment maintenance) was taken the same as for the first IOSS updated to 1977 dollars. The total on-orbit servicer operations costs are presented in Table IX-23 for the two mission models by WBS element. The operations

Table IX-23 On-Orbit Servicer Operations Costs (Millions of 1977 Dollars)

WBS ELEMENT	BASIS	Mission Model 75%	Mission Model 50%
Project Management	6% of subtotal	2.7	1.9
Project Engr. and Integ.	11% of subtotal	4.6	3.1
Structures and Thermal	SAMSO Data	-	-
Mechanisms	Analogous to PDRM Data	-	-
Control Electronics	Analogous to PDRM Data	-	-
Assembly and Checkout	Percent of Hardware Cost	-	-
Airborne Spares		-	-
Airborne Support Equipment		-	-
Logistics	Bottoms up	6.4	4.8
Ground Support Equipment	Analogous to Tug	0.6	0.6
Facilities	Analogous to Tug	-	-
Operational Site Services	Bottoms up	34.4	23.1
TOTAL		48.7	33.5

PDRM = Payload Deployment and Retrieval Mechanism

category summarizes the cost of launch operations, flight operations, maintenance (scheduled and unscheduled), refurbishment, management and supporting functions.

The operations costs for the 75% mission model are 74% of those for the first IOSS when expressed in 1977 dollars. This is partly due to the fewer missions being addressed and partly to the differences in estimating approaches. The operations costs per mission in thousands of 1977 dollars are--1) first IOSS, 270; 2) 75% mission model, 300; and 3) 50% mission model, 340. These data show a consistent expected trend with mission model size even though they have been reached by quite different approaches.

14. Servicer System Costs

The total servicer system costs were compiled by combining the servicer DDT&E and production costs with the operations costs. The results for the two mission models are shown in Table IX-24. The total on-orbit servicer system cost for the 75% mission model is 90% of the comparable first IOSS cost when expressed in 1977 dollars. This is well within the estimating tolerances, especially when the effect of a reduced mission model size is included. The 75% mission model costs are a little higher than the mission model size change might lead one to expect. Other factors include higher capability servicer mechanism, lower stowage rack weight, larger number of spares, and different operating periods at the launch sites.

Table IX-24 On-Orbit Servicer System Costs (millions of 1977 dollars)

WBS ELEMENT	BASIS	Mission Model	
		75%	50%
Project Management	6% of subtotal	6.0	4.9
Project Engr. & Integ.	11% of subtotal	10.1	8.1
Structures and Thermal	SAMSO Data	9.4	8.4
Mechanisms	Analogous to PDRM Data	3.6	3.3
Control Electronics	Analogous to PDRM Data	9.4	9.1
Assembly and Checkout	% of hardware cost	1.5	1.3
Airborne Spares		8.0	6.0
Airborne Support Equipment		4.7	3.8
Logistics	Analogous to PDRM and Tug Data, bottoms up	13.0	11.4
Ground Support Equipment	Analogous to Tug	4.9	4.9
Facilities	Analogous to Tug	0.7	0.7
Operational Site Services	Bottoms up	36.2	24.9
TOTAL		107.5	86.8
PDRM = Payload Deployment and Retrieval Mechanism			

The servicer system costs are summarized in a different way in Table IX-25 to show the comprative sizes of the three major WBS level 3 cost elements. The operations costs remain the most significant item except for the 50% mission model where the DDT&E costs are comparable. The other important servicer system cost is the $2.11 million 1977 dollar cost for each production servicer system.

15. Spacecraft Program Costs

To illustrate the effect of servicer operations costs, an analysis of the total costs of operations for two spacecraft programs were derived. These are

Table IX-25 Servicer System Costs at WBS Level 3

Mission Model	COSTS (millions of 1977 dollars)			
	DDT&E	Production	Operations	Total
First IOSS	33.8	20.0	66.1	119.9
75%	37.3	21.5	48.7	107.5
50%	37.3	16.0	33.5	86.8

the Synchronous Earth Observations Satellite (SEOS) and the Large X-ray Telescope (LXRT). These two spacecraft were selected as they are representative of two major spacecraft categories--high earth orbit and low earth orbit. They also represent two of the three spacecraft analyzed by TRW in the serviceable spacecraft design activity.

The approach used was that of the first IOSS with appropriate updating. The results of this costing analysis are shown in Table IX-26. The LXRT mission

Table IX-26 Spacecraft Program Costs

SPACECRAFT PROGRAM	Mission Model	Expendable Mode*	On-Orbit Maintainable Costs*			Savings	
			Basic	Servicer	Total	$*	%
LXRT	75%	582.0	500.1	0.3	500.4	81.6	14
SEOS	75%	532.4	387.4	1.2	388.6	143.8	27
	50%	407.5	355.7	0.7	356.4	51.1	13

*Millions of 1977 dollars

definition is the same for both the 75% and 50% mission models. The dollar and percentage savings are significantly lower than for the first IOSS. This appears to be due to the smaller mission models. In the 50% mission models, the number of servicing activities equals the on-orbit fleet size. They are thus at the minimum level and may not be economic. The 75% mission model for SEOS involves two servicings for each spacecraft in the on-orbit fleet size and thus more significant savings (dollars and percentage) result.

As can be seen from the table, the servicer operations costs are a small part of the total costs or savings. The largest ratio is for the SEOS 50% mission model use where the servicer operations costs are 1.5% of the savings. The ratio for the other two cases is lower.

16. Conclusions

All the factors entering into an estimate of servicer system life cycle costs have been reviewed and reevaluated. The important operational costs have been generated from the bottom up, as compared to the similarity approach of the first IOSS. The result is no fundamental change in servicer system costs, the total servicer system cost being within 10% of the first IOSS data when expressed in 1977 dollars. As the mission model size is reduced the servicer system life cycle costs become smaller, but the cost per service action becomes larger. The servicer per unit production cost in 1977 dollars remained constant from the first IOSS to this evaluation at $2.11 million.

Spacecraft program costs for the Large X-ray Telescope and the Synchronous Earth Observation Satellite were developed for 75% and 50% mission model sizes. The LXRT savings (as compared to the expendable mode) were 81.6 million 1977 dollars and 14% for both mission models. The SEOS savings were 144 million 1977 dollars and 27% for the 75% mission model and 51 million 1977 dollars and 13% for the 50% mission model. These savings are smaller than for the 100% mission model of the first IOSS as was expected. The smaller mission models correspond to only one servicing per unit of on-orbit fleet size. This situation usually resulted in a 10% to 15% savings. The on-orbit servicer operations cost on a per-mission basis continues to be a small part of the savings.

The functional requirements approach used in this analysis, in addition to the cost data, examined many potential space transportation system impacts. No such impacts were identified. All interfaces identified lie within the bounds of the planned STS capabilities.

The communication links required for each mission phase were identified from end point to end point. Each link was also presented in terms of the STDN, TDRSS, and NASCOM. The servicer system and all its operations are being designed to operate in the Multiple Access mode. The NASA communications system for the STS era is to have all the capability that on-orbit servicing will require.

The current version of the NASA Launch Cost Reimbursement Policy has been directed toward encouraging--1) deployment of simple payloads, 2) short term on-orbit operations, and 3) an early transition to the STS. An effect has been to put many of the activities associated with on-orbit servicing into the extra cost category. Thus the potential savings from on-orbit servicing will be reduced. The extent of the reductions is not clear at this time. The new LCRP was applied to the LXRT in the 75% mission model, the SEOS in the 75% mission model, and in the 50% mission model. These three cases resulted in a 38% savings of launch costs of on-orbit maintenance as compared to the expendable mode. A similar comparison from the first IOSS data (100% mission model) for the LXRT and the SEOS showed a 49% savings. Several effects were different in the two calculations. These include higher Orbiter and Tug launch costs and 1977 vs 1975 dollars which have no effect on the comparison. Also included were a 17% premium for a six-month launch schedule which favors on-orbit maintenance and a smaller mission model which favors the expendable mode.

Information was not available with regard to certain aspects of the LCRP and thus the cost effects were not included. The factors are:

1) Special integration and testing;
2) Special crew training;
3) Retrieval;
4) VAFB (WTR) launch at 90 or 104 deg inclination;
5) Payload specialists station and related software;
6) Communications system.

Future analyses should attempt to size and cost these factors.

The operational site services costs were found to be an important cost element in the first IOSS. They were generated in this analysis from the bottom up. For the 75% mission model the operational site services costs were 70% of the comparable number from the first IOSS expressed in 1977 dollars. This is partly due to the smaller mission model. The analysis involved use of minimum headcounts so skills would be retained at each launch site. However, this was not a significant cost effect. There are some differences in relative contributions of the different WBS elements from the first IOSS to the current analysis. None of these were major. Mission planning continues as the largest factor.