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SHUTTLE CONSIDERATIONS FOR THE DESIGN OF LARGE SPACE STRUCTURES

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FOREWORD

This document is a compendium of Shuttle capabilities, constraints and guidelines which have been abstracted from currently available documents generated by NASA, Rockwell International, and by other NASA contractors. The document includes summaries of significant results from Rockwell experience gained in Shuttle integration activities and from a recent, extensive study of Space Construction System Analysis, Contract NAS9-15718. Essentially no new technical data were generated, but an attempt was made to provide updated information concerning orbiter systems, and to discuss potential new Shuttle hardware and procedures concepts currently being studied. At the time of this writing there were no specific plans to have the document updated in the future as results of additional studies concerning design of large space structures are made available and actual flight data are derived from early Shuttle operations. Therefore, the following warning is included herein and several places in the document:

> "Caution: Not a controlled document! See appropriate reference documents for current data."

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1.0 INTRODUCTION - SCOPE

This document provides introductory facilitating information in a form readily accessible to the designer, system analyst, and program planner involved with design of a class of large space structures.[1] The structures considered are those capable of being constructed and tended in space using the remarkable capabilities inherent in the Space Transportation System (Space Shuttle) as a transport method and on-orbit base.

1.1 DEFINITIONS AND ORGANIZATION OF DOCUMENT

Much of the information in this document is presented as specific design constraints/capabilities and guidelines,[2] defined as follows:

- Shuttle Constraints/Capabilities

  Official documented (Ref. Para. 1.1.2) Shuttle system capabilities, requirements or limitations which must be observed by designers in order to assure that Shuttle crews can ascend, rendezvous, dock/berth, perform construction and maintenance/checkout on orbit using the Shuttle Orbiter, and return safely to earth. Examples: power limits, payload bay size, number of crew stations in orbiter cabin, launch weight, range of location of center of gravity of cargo.

- Shuttle Guidelines

  Recommended practice in design and procedures which helps to assure mission success, minimize cost and provide adequate safety, with particular reference to operations involving large space structures, using the Shuttle system. Examples: minimize time on orbit, provide for safe and productive crew work/rest cycles, minimize power demand.

There is an extensive amount of data and variety of types of information which could potentially fit into the category of Shuttle constraints, capabilities and guidelines relative to design of large space structures. Therefore, this document was written to perform the function of a set of keys— a means to identify sources and to gain access to pertinant information. Each major section includes tables which act as checklists of constraints, including capabilities and guidelines, lists of associated references, and suggested implications for space construction. These tables generally are accompanied by narrative and graphic descriptions which further explain what can be done with the available hardware and procedures, provide rationale for guidelines, present example applications (in some cases), and define unusual terms.

NOTE: [1] "Caution: Not a controlled document! See appropriate reference documents for current data."

[2] These guidelines may be of a generally useful, universal nature for space operations, which appear in Shuttle-oriented study reports or NASA reference documents. In many cases the distinction between guidelines, capabilities and constraints is a matter of judgement by the author. In case of doubt, NASA contract monitors should be contacted for clarification.

In the typical subject presentations, checklists are divided into considerations dealing with (1) hardware and (2) procedures. Thus, for each subject there may be as many as four sets of checklists, which are arranged in the order noted in Table 1.1-1.

Table 1.1-1. Matrix of Order of Presentation of Checklists for Each Topic

	CONSTRAINTS	GUIDELINES
HARDWARE	FIRST	SECOND
PROCEDURES	THIRD	FOURTH

This arrangement assumes the majority of the readers are designers who are somewhat familiar with Shuttle systems and are mostly concerned about defined hardware and firm constraints or capabilities. However, system analysts may prefer to first read the guidelines which are usually more introductory in nature and frequently provide rationale for using a given type of Shuttle capability.

Each checklist item is, to the degree practical, a factual statement of a specific constraint/capability or guideline. The statement attempts to provide a broad baseline of information to the reader. However, its more important function may be to alert the reader that a potential capability, constraint or preferred approach exists. Accompanying each statement are references which either provide the authoritative basis for the statement, or provide the reader with discussions and data which are likely to contain further details concerning the Space Shuttle system.

The majority of this document is organized according to Shuttle systems and general payload integration concerns applicable to all types of payloads (e.g., orbiter configurations, attachments, signals, power, thermal, etc.). However, two entry points are offered which emphasize topic headings directly related to large space construction and operations. The first is presented in Section 1.2 in the form of a table of typical large space structures design issues and related discussion material (Sections 1.2.1 through 1.2.7). The second is an alphabetical subject index at the back of the document which includes a great many topic headings dealing with design of large space structures (e.g., construction fixtures, module installations, crew productivity). Both entry points provide references to the appropriate Shuttle constraints/capabilities, guidelines, and associated descriptive material.

> The data provided herein must be considered as advisory in relation to historical time. The Space Shuttle program is a dynamic, changing entity. Therefore, this document cannot hope to be current in all respects for very long. The reader is urged to check for the latest publication or updating supplement of any listed document, and to be cautious about accepting statements in the checklist as final or unequivocal fact.

Also, at this writing, many items of equipment which were conceived as part of the Space Shuttle program had not been contracted for, nor qualified. Therefore, it appeared advisable to qualify such items by a rating number describing related validity or availability, as defined in Table 1.1.2. As funds are made available and experience gained in operations and development tests, such ratings will inevitably change, even if the technical facts do not.

Table 1.1-2. Availability Rating Scale

Rating	Description of Status
R-1	Funded Shuttle Development Program planned operational date (or actual delivered hardware)
R-2	Proposed, well-defined Shuttle hardware concepts, hardware funding deferred
R-3	Study Program in work, supported by NASA or contractor funding of system hardware
R-4	Recommendation from study. No current funding for hardware
R-5	Conceptual only

Note: Application of the ratings is based upon the knowledge and judgment of the author on or about the date of publication, and does not necessarily represent any official NASA or Rockwell viewpoint as to technical quality or public policy.

1.1.1 How To Use This Document

This document is primarily a reference book designed to provide access to facilitating information, cautions and limitations concerning the influence of Shuttle systems on design of the class of large space structures which can be built from the orbiter. Managers and project engineers should first carefully note the categories of information in the Table of Contents and read the introductory narratives at the beginning of each major section.

At the beginning of a large space structure design project, managers and analysts probably will be concerned with understanding and prioritizing for study the issues discussed in Section 1.2. These issues were identified as a result of several years experience of the authors, and by discussions with other contractors. They are considered to be applicable to a large number of possible projects.

Systems analysts and designers should specifically note the organization of material in relation to their respective specialty areas and carefully review the associated checklists of constraints and guidelines.

After these introductory reviews, the major point of entry to the report for most users will likely be the alphabetically arranged subject index at the back. This index lists pages of the report which potentially contain applicable checklists of constraints and guidelines, together with references to source documents and the availability rating number which relates to the status of the associated hardware development.

As a practical aid, each checklist item is preceded by parentheses which provide a convenient spot for a checkmark on working copies or for a reference number to study report paragraphs or charts in design studies. The checklists can be reproduced and used for systems analysis and design planning. Analysts and project leaders may refer to the checklists for design reviews. Requirements analysts should obtain up-to-date copies of the referenced regulatory documents to assure their understanding of the latest changes in details; then they should compile and issue changes to the checklists (where appropriate) for the use of project personnel.

Note that the checklist tables may include callouts which refer to figures or tables which are not mentioned elsewhere in the text. Such figures or tables follow the checklist tables prior to subsequent text material. Acronyms which appear in the checklists and elsewhere in the tables, figures and text are listed in the back of the document. Also included at the back is a brief glossary of technical terms.

In all cases, the referenced regulatory documents should be considered as prime sources of information, since the checklists contained herein are advisory and may not be continually updated. For the convenience of the reader, a short listing of most referenced, prime requirements documents is presented below. These are essential documents for consideration of Shuttle constraints and guidelines for large space structures design.

1.1.2 NASA Documents of General Interest to Shuttle Users

Space Transportation System User Handbook June 1977 and Errata July 1977

Space Shuttle System Payload Accommodations, JSC 07700, Vol. XIV and Attachment 1 (ICD 2-19001)

STS Flight Planning, JSC-11803

Communications and Data Systems Integration (CADSI) End-To-End Configuration Book, JSC-10074

Shuttle Operations Data Book, JSC-08934, Vol. I - Shuttle System Performance & Constraints, February 1980

Payload Interface Verification Requirements, JSC-14046, August 1980

KSC Launch Site Accommodations Handbook for STS Payloads, K-STSM-14.1 (Rev. A, March 1980)

Shuttle EVA Description and Design Criteria JSC-10615

Space Transportation System Reimbursement Guide, JSC-11802, May 1980

Shuttle Payload Integration Activities Plan, JSC-14363, September 1978

Safety Policy and Requirements for
Payloads Using the Space Transportation System
(STS) NHB 1700.7
NASA - Washington, D.C., May 1979

1.2 SHUTTLE INTERFACE ISSUES FOR LARGE SPACE STRUCTURES DESIGN

This section identifies several design issues characteristic of large space structures and relates them to potential capabilities/constraints and guidelines associated with use of the Space Shuttle as a transport vehicle and construction/maintenance base.

Development of the design requirements for proposed structures design/construction concepts must include analysis of Shuttle capabilities/constraints to determine if the mission could be successfully performed. An overall concept for the process of designing a space construction project and integrating it with the Shuttle Orbiter is depicted in Figure 1.2-1. As illustrated in the figure, the Shuttle constraints and guidelines may be grouped and considered approximately in the order in which they are encountered during a mission. However, some aspects of design may require the reverse procedure. That is, it may be necessary to define the desired or required end operational condition, and then to determine the packaging conditions, equipment, supplies, and procedures that must be provided to meet the end needs. For example, a decision concerning whether to add one or two cryogenic fuel kits to the orbiter manifest depends on the estimated total energy demand for the space operations. The latter can only be determined after an extensive end-to-end analysis based on postulated equipment used, duration of orbital operations, and compilation of work performed. Under some conditions, a different equipment or procedure may be selected in order to avoid installing an extra cryogenic kit. During analysis and design, the requirements of a particular design/construction concept are serially compared to the Shuttle constraints and guidelines. Each comparison may result in a determination as to whether there is an over-utilization or acceptable under-utilization of the orbiter capabilities. If the design/construction requirements are greatly different from orbiter capabilities, the process may require revision of the basic design for concept modifications.

A key point of philosophy is implicit in Figure 1.2-1; it is that design of large space structures is inextricably linked to, and driven by, the process of manufacture or construction of the structure in the space environment. In nearly all of the following presentations the concept of design encompasses design of the construction equipment and selection of all construction support equipment as well as that of the actual finished and operational spacecraft. Design is assumed to include considerations of all mission phases from launch of the first piece of construction fixture to the last provision for on-orbit maintenance or refurbishment.

Shuttle capabilities are not necessarily separate and distinct. For example, a larger electrical power supply can be provided for a particular mission, but only at the expense of the amount of mission payload that can be carried. Wherever possible in this report, such trade requirements are pointed out and sources of data for resolving the trade issues are provided or referenced.

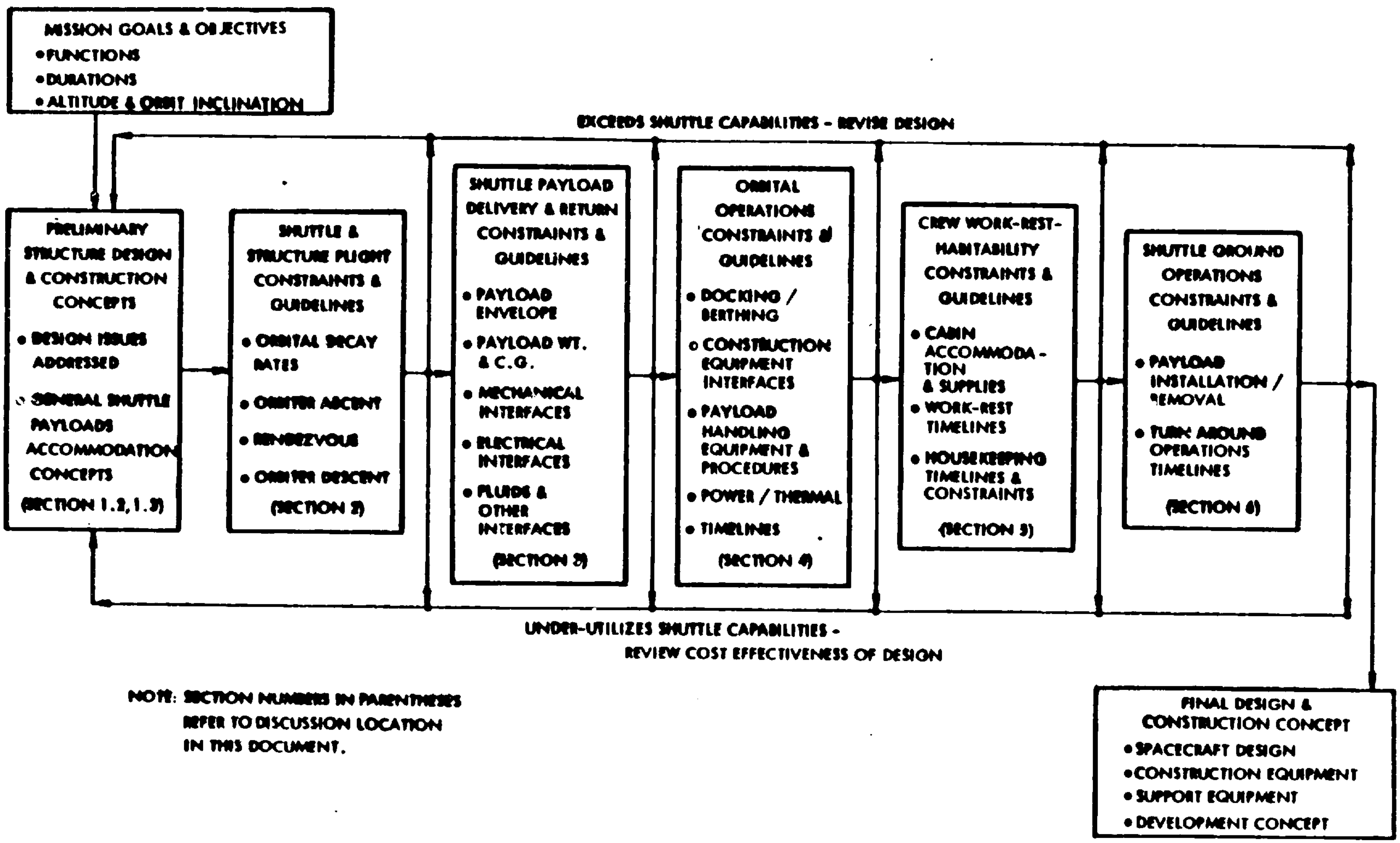


Figure 1.2-1. Suggested Process for Consideration of Shuttle Constraints and Guidelines During Design of Large Space Structures

The wide variety of large space structures projects currently in the stage of conceptualization and preliminary design study include a great many Shuttle payload integration considerations. However, emphasis is given herein to those aspects which have been identified, by previous NASA and contractor studies, as important and serious concerns. Table 1.2-1 lists, in the left column, several general space construction issues involving specific Shuttle constraints and guidelines which are briefly indicated in the middle column. The right column lists sections of this document in which applicable information on the identified Shuttle constraints and guidelines may be found. Thus, Table 1.2-1 is arranged approximately as a table of contents to this document might appear if it were organized according to the viewpoint of a current designer of large space structures. While these issues are of deep concern today, future designers may encounter other design issues, as yet unforeseen as crucial. Therefore, as previously noted, the selected organization of material in this document relates primarily to Shuttle systems and characteristics. The latter are presumed to be relatively stable in overall scope during the next two decades.

Further discussion of the large structures design issues follows in subsequent sections (1.2.1 through 1.2.7). Such discussion is designed to provide further explanation of the rather cryptic terminology used in Table 1.2-1 and the rationale for considering the listed issues in this document. Another purpose of this discussion material is to familiarize the reader with the vocabulary and historical basis for the conceptual framework which was used in developing the subject index at the back of the document. This is desirable because design approaches and terminology have yet to be standardized in the relatively new field of engineering large space structures. For example, most large space structures will probably require at least one relatively large piece of equipment to stabilize and position the spacecraft with respect to the orbiter during construction. This item might be designated in an index by a descriptive, shape-related name, such as "trapeze" or "claw", while others may use the more general terms of "jig" or "fixture" or "holding-positioning aid." All such titles of devices are cross referenced in the index to the title "construction fixtures." Further assistance is offered by the glossary at the back of this document.

1.2.1 <u>At What Orbit Shall Construction Take Place</u>?

This issue confronts the major Shuttle constraints on maximum altitude versus lifted mass, orbiter turnaround time versus altitude decay of the large space structure spacecraft, and orbiter launch site availability versus desired orbital angle. Additional, less directly related issues include crew protection from ionizing radiation and available crew work time in sunlight.

In general, studies have shown that construction based on use of the Shuttle must take place at relatively low earth altitudes of 150 to 250 nmi and generally would be most economical if launched at 28.5° inclination from Kennedy Space Center. Checklists and discussion material appear in Section 2.0.

Table 1.2-1 Typical Large Space Structures Design Issues Involving Shuttle Constraints & Guidelines

LARGE SPACE STRUCTURE DESIGN ISSUES	SHUTTLE CONSTRAINTS/ IMPLICATIONS	APPLICABLE SECTION NUMBER
• CONSTRUCTION ORBIT • ALTITUDE • INCLINATION	• LAUNCH SITE	2.0, 2.3.1
	• PAYLOAD CAPACITY OF ORBITER (WEIGHT)	2.3.1
	• OMS KIT IMPACT ON VOLUME	2.3.1, 3.11
	• ORBIT DELAY VS SHUTTLE TURNAROUND	2.1, 7.0
	• RADIATION ENVIRONMENT • PARTICLE DENSITY	2.2
• CONSTRUCTION CONCEPTS • DEPLOYABLE STRUCTURE • ERECTABLE STRUCTURE • SPACE FABRICATED STRUCTURE • HYBRID-COMB. OF ABOVE • NEW TECHNOLOGY	• ATTACHMENT POINTS & CONNECTIONS FOR FIXTURES	3.2, 3.8-3.10, 3.11, 3.12, 4.4.2
	• CLEARANCE ENVELOPES	4.10
	• ORBITER CONFIGURATION	3.1, 4.4.1
	• EVA SUPPORT	4.8
	• PAYLOAD HANDLING OPTIONS	4.4.4, 4.5 - 4.7
	• VOLUME & MASS PROPERTY CONSTRAINTS OF ORBITER • CONTINGENCIES	3.1, 3.4
• EQUIPMENT/SYSTEMS TO BE INSTALLED/ DEPLOYED ON STRUCTURE • MODULES • INTERCONNECTS (WIRES & TUBES) • BLANKETS • MOUNTING/CONNECTING DEVICES	• RMS HANDLING & STOWAGE CAPABILITIES	4.4.4, 4.5
	• PIDA, OTHER AIDS	4.6, 4.7
	• COMMUNICATIONS	4.10
	• EVA SUPPORT	4.8
• ASSEMBLY/INSTALLATION SEQUENCE FOR STRUCTURE AND SYSTEMS • ENVIRONMENT - DETERMINED • CONFIGURATION - DETERMINED • FUNCTION - DETERMINED • TIME - DETERMINED o ABORT CONTINGENCY CONSTRAINED (C. G. LOCATION)	• ORBITER PACKAGING CONSTRAINTS	3.0 - 3.11
	• PAYLOAD HANDLING OPTIONS	4.4.4, 4.5-4.7
	• EVA CONSTRAINTS	4.8
	• ATTACH INTERFACES	3.2, 3.8-3.10
	• CREW WORK/REST CYCLE CONSTRAINTS	5.2
	• ILLUMINATION & VISION CONSTRAINTS	4.9
	• THERMAL CONSTRAINTS	4.13

Table 1.2-1 (Cont'd) Typical Large Space Structures Design Issues Involving Shuttle Constraints & Guidelines

LARGE SPACE STRUCTURE DESIGN ISSUES	SHUTTLE CONSTRAINTS/ IMPLICATIONS	APPLICABLE SECTION NUMBERS
• DOCKING/BERTHING HARDWARE PROVISIONS FOR REVISIT • RMS USAGE (BERTHING) • DIRECT DOCKING • OTHER AIDS • STABILIZATION METHODS	• RMS LIMITS • DOCKING SYSTEM CONSTRAINTS (ANGLE, IMPACT) • PLUME EFFECTS • ORBITER CONFIGURATION • ATTITUDE CONTROL OF ORBITER • CONTINGENCIES	4.5 4.1 4.1 4.4.1 4.2
• SIGNALS/COMMANDS BETWEEN ORBITER AND CONSTRUCTION • HARDWIRE • SOFTWARE • VISUAL	• HARDLINES HARNESS • RADIO FREQUENCY • SOFTWARE, C&W RQMTS • VISUAL ANGLES • LIGHTING, TV	3.8 4.8.1.7, 4.11, 4.12 4.9.6
• PACKAGING REQUIREMENTS & NUMBER OF SHUTTLE FLIGHTS • LARGEST PART SIZE • NESTING & INTERLEAVING • CONSTRUCTION METHODS • NO. OF SHUTTLE FLIGHTS PER PROJECT	• ORBITER VOLUME & WEIGHT LIMITS • ORBITER ATTACHMENT PROVISIONS LOCATIONS, LOAD LIMITS • ORBITER UTILITIES CONNECTIONS & KITS	2.3.1, 3.0 3.1, 3.4 3.2, 3.3, 3.5, 3.6 3.8, 3.9 4.3.1, 4.10-4.13
• TIME PACING ITEMS IN CONSTRUCTION (IVA & EVA) • LABOR • MACHINE RATES • ASSEMBLY • TRANSPORT	• NO. OF CREW • ROTATION/TRANSLATION RATES OF HANDLING AIDS • ORBITER RESOURCES • CREW WORK/REST CYCLES & HOUSEKEEPING • EVA CONSTRAINTS	3.1, 5.1 4.5 - 4.7 4.10, 4.12, 4.13 5.2, 5.3 4.8
• POWER/ENERGY REQUIREMENTS & SOURCES • PEAK POWER/DURATION • TOTAL ENERGY	• POWER LIMITS • WEIGHT & VOLUME FOR SUPPLIES OF FUEL OR BATTERIES OR SOLAR CELLS • CRYOGENIC KITS IN ORBITER	4.10 2.3.1, 3.1 4.10.2

Table 1.2-1 (Cont'd) Typical Large Space Structures Design Issues Involving Shuttle Constraints & Guidelines

LARGE SPACE STRUCTURE DESIGN ISSUE	SHUTTLE CONSTRAINTS/ IMPLICATIONS	APPLICABLE SECTION NUMBERS
• DEGREE OF AUTOMATION FOR CONSTRUCTION/MAINTENANCE • MANUAL (EVA) • COMPUTER AUGMENTED REMOTE CONTROL • FULLY AUTOMATIC, MANUAL OVERRIDE, INITIATION	• RMS CAPABILITIES • PAYLOAD HANDLING OPTIONS • EVA CONSTRAINTS • ORBITER RESOURCES • SOFTWARE & DATA INTERFACES	4.5 4.4.4, 4.5- 4.7 4.8 3.11, 3.12, 4.12
• ATTITUDE CONTROL OPTIONS • DOCKING/BERTHING • ORBITER VS STRUCTURE • FREE DRIFT VS CONTROL	• PRIMARY RCS LOADS • VERNIER RCS LOGIC • FREE DRIFT IMPLICATIONS • ENERGY RESOURCES • SOFTWARE, HARDLINES • CONTAMINATION	4.1, 4.2 4.1, 4.2 4.2.2. 4.2, 4.10 4.12
• THERMAL CONTROL & IMPLICATIONS ON SPACE STRUCTURES	• ORBITER SHADOWING • ORBITER HEAT OUTPUT • ORBITER HEAT REJECTION CAPABILITY • SOLAR FOCUSSING OF RADIATORS	4.13 4.13 4.13.1 4.13.2
• COMMUNICATIONS DURING ORBITER TENDED OPERATIONS	• ORBITER ANTENNA LOCATIONS • ORBITER MASKING • ORBITER C&DH CAPABILITY	4.11 4.4.1 4.11, 4.12
• CONFIGURATION CONSTRAINTS • FUNCTIONAL SHAPE • BUILD DIRECTION • CLEARANCES	• ORBITER CONFIGURATION • THERMAL REQUIREMENTS OF ORBITER • SAFE CLEARANCES OF MOVING PARTS	4.4.1 4.13. 4.4

1.2.2 How Shall the Project be Constructed?

The majority of construction processes considered in this report involve three basic structural concepts which have historically evolved during the years prior to 1980. These are defined briefly in Figure 1.2-2.

In fact, a combination and blend of these, and possibly other processes, will probably be involved in most space construction projects. In most cases, some modules or other components will be installed or joined after the basic structure is formed in space. Thus, a process of "assembly" or "erection" is likely to be involved in most space construction projects. Within each of these basic concepts, there is implied the decision of degree of dependence on remotely controlled or automatic machinery versus direct (EVA) processes.

Designers of all processes must eventually come to grips with questions of where they can attach equipment to the orbiter, how they can clear its critical surfaces, and how they can package the system to fit within the payload bay. Such issues are dealt with in Sections 3.1 through 3.10, 4.1 and 4.4. Sections 4.5 through 4.8 provide considerations dealing with handling the piece parts during construction and joining them together by making use of available Shuttle equipment and operational capabilities such as the remote manipulator system and extravehicular activity of the crew.

1.2.3 What Equipment or Systems are to be Installed/Deployed on the Structure?

The general approach taken for this document is to consider the generic shape, dimensions, and mass of items to be handled during construction rather than the function of such items when installed. In its stowed condition in the orbiter, a rather complex, deployable subsystem element may be treated as a simple lump by the orbiter remote manipulator system (RMS) operation. In Sections 4.4.4 and 4.5 through 4.7, this report presents the constraints and guidelines for handling objects with various Shuttle equipment items. When objects are small enough and light enough, they may be transported and installed by EVA within the constraints and guidelines described in Section 4.8.

1.2.4 How Shall the Project Parts be Packaged and Interfaced with the Orbiter for Delivery?

This area of concern includes many of the most specific constraining influences on design of large space structures and associated construction equipment. Among these limits are: the overall maximum size and weight of individual parts which can be delivered to orbit; the number of smaller items available to assemble on any given flight; the sequence of removal from the payload bay; the length and location of electrical umbilicals and fluid lines (if required); the specific locations for mechanical attachments and the loads which they can accept; the overall strength requirements for packaging and packaged material; and the vibration environment to be withstood. Where dual use is made of stowage attachments as on-orbit stabilization attach points, there is a close interaction in detail design. Since everything used in or about the orbital construction site must be lifted and contained within the orbiter, it follows that each item selected (or omitted) for support services and basic vehicle design will be influenced by this set of capabilities to a greater or lesser

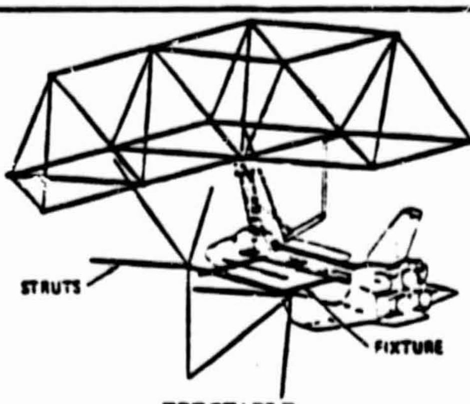
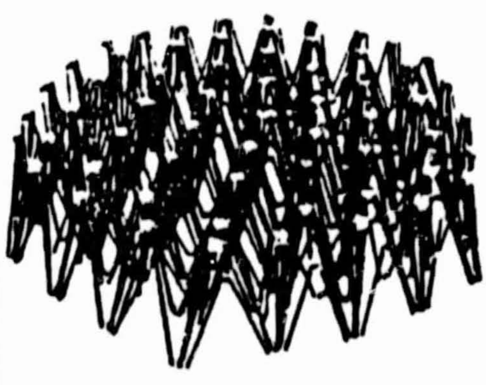
TYPE OF CONSTRUCTION	CHARACTERISTICS	SHUTTLE INTERFACE SPECIAL CONCERNS
 ERECTABLE	• LARGE NUMBER OF PIECE PARTS • ALL PARTS GROUND FABRICATED • SPACE ASSEMBLY OF STRUCTURE AND SYSTEMS TO STRUCTURE • TENDS TO BE TIME CONSUMING: CREW PACED BY MANIPU-LATOR OR EVA METHODS	• RMS OR CRANE REACH, VELOCITY, ACCURACY, LOADS • CARGO BAY VOLUME, ESPECIALLY LENGTH • TIME ON ORBIT VS COSTS, SUPPLIES • MULTI SHIFT OPERATIONS • VISION, LIGHTING • POWER • CLEARANCES
 DEPLOYABLE	• VERY FEW PIECE PARTS • ALL PARTS GROUND FABRICATED • SPACE DEPLOYMENT (UNHINGE, UNFURL STRUCTURE • POSSIBLE JOINING OF DEPLOYED SUB-ASSEMBLIES • ELECTRICAL WIRING SOME SYSTEMS EQUIP. MAY BE INTEGRAL • RAPID, AUTOMATIC	• CARGO VOLUME (LOW DENSITY PKG) • LOADS IMPARTED BY DEPLOYMENT (TBD) • CLEARANCES • REACH, FORCE OF RMS, ETC. FOR SUBASSEMBLY JOINING • VISION, LIGHTING • SUPPORT METHODS IN ORBITER PAYLOAD BAY
 SPACE-FABRICATED	• MODERATE NO. OF PIECE PARTS • SPACE FORMING OF MATERIALS, ASSY FOLLOWS • WIRES, EQUIP. & HOOKUP ON-ORBIT • MODERATE TIME REQUIREMENTS MACHINE PACED, SEMI AUTOMATIC	• POWER & ENERGY CON-STRAINTS VS. RMS, HEATERS, LIGHTS, TRANSLATION DEVICES • TIME ON ORBIT VS. COSTS • MULTI-SHIFT OPERATIONS • RMS REACH, MOBILITY & CLEARANCES • VISION & LIGHTING • CARGO BAY VOLUME

Figure 1.2-2. Typical Methods for Large Space Construction - Characteristics and Shuttle Interface Concerns

degree. Therefore, many of the issues listed in Table 1.2-1 interact with this area of concern. However, the majority of the applicable constraints are discussed in Section 3.0 of this document.

1.2.5 What are the Pacing Construction Concerns?

Typical parameters of concern in large space construction include rates of construction (which may be paced by how fast the orbiter remote manipulator can transport and install items, and how many hours per day the crew can work) and limits of power demand on the orbiter (which can determine how many heaters, motors, and lights can be operating simultaneously). The total time available on orbit (as conditioned by stowage volume, crew cabin supplies, tank kits, etc.) and the sequence of equipment removal from the orbiter may be other related concerns. Such concerns help decide the degree of automation to be built into construction support equipment. The latter can also be affected by support systems for Shuttle crew extravehicular activity (EVA) which determine how many hours per day are actually useful for work. The applicable sections for these issues are listed in Table 1.2-1.

1.2.6 What Attitude Control is Required?

This area of concern has been considered among the most significant by contractors working in the field of large space construction. The major issues in design of large space structures relate to the orbiter capability and its impact on design and procedures. Clearly, the orbiter reaction control systems are needed to null rotation rates for beginning of construction activity, and for the orbiter approach during later revisit operations. The designer of large space structures must determine when and how to provide for stabilization of the spacecraft under construction by evaluation of orbiter constraints related to location of the joint orbiter/large space structure center of mass and the impact on delicate construction and servicing operations, etc. The major constraints are discussed in Section 4.2 of this document.

1.2.7 What are the Configuration Constraints?

Once the orbiter payload bay doors are opened, there are certain major geometric limits affecting which directions one can extend a large space structure. Among these are the orbiter tail, wing, payload bay doors/radiators and cabin, as described in Section 4.4.1. Less obvious but real concerns include blockage of orbiter antennas, blocking of the radiators, obscuring of windows, TV cameras and lamps (noted in Sections 4.9, 4.11, and 4.13). Other specific details are related to such items as attach points on the payload bay longerons, crew safety slidewire and handholds, OMS kits, airlocks, and docking ports (described in Sections 3.0 and 4.4.2). The limits of reach and articulation modes of the remote manipulator system (RMS, Section 4.5) may be a major influence on design of large space construction projects. For example, in a recent Rockwell study [Roebuck (1980)], a space-fabricated structure concept devised for a large communications platform was configured as a long, narrow shape which permitted RMS reach to all points on the structure as it was constructed by translating it axially back and forth across the orbiter payload bay, above the RMS shoulder joint (Figure 1.2-3).

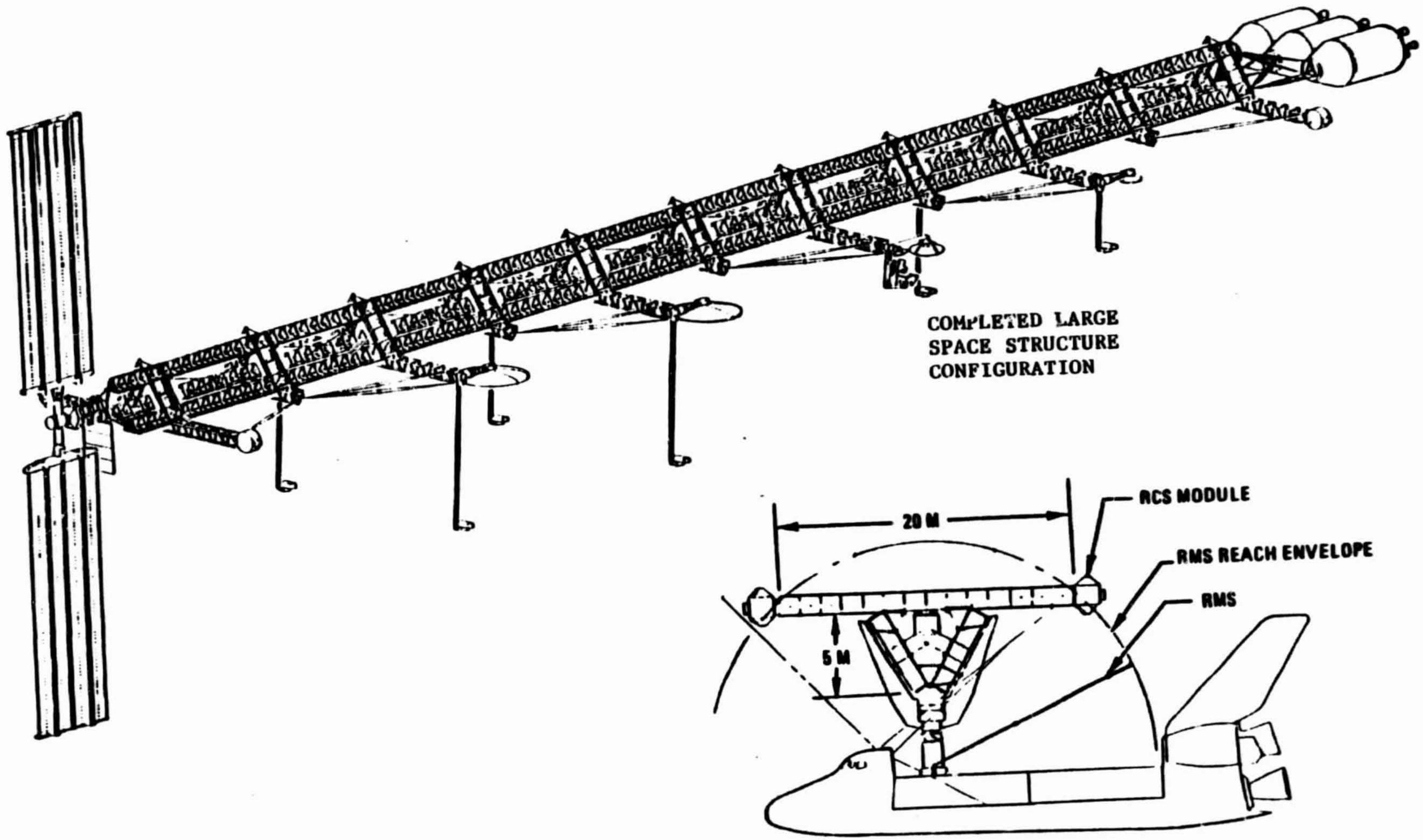


Figure 1.2-3. Example of Possible Influence of RMS Reach on Shape of Large Space Structure

The foregoing discussion concludes the introduction of subject matter which is arranged according to issues viewed by the designer of large space structures. The remainder of the document is organized in terms of orbiter systems and emphasizes kinds of accommodations which are available in the orbiter for payloads, but with selected attention to those of special concern to designers of large space structures.

1.3 SPACE SHUTTLE PAYLOAD ACCOMMODATIONS

The Space Shuttle and its standard payload carriers are designed to accommodate a wide range of payload missions. Figure 1.3-1 identifies those orbiter subsystems which may be used to support, directly or via carrier systems, space construction equipment, modules, and material stowage devices to be flown.

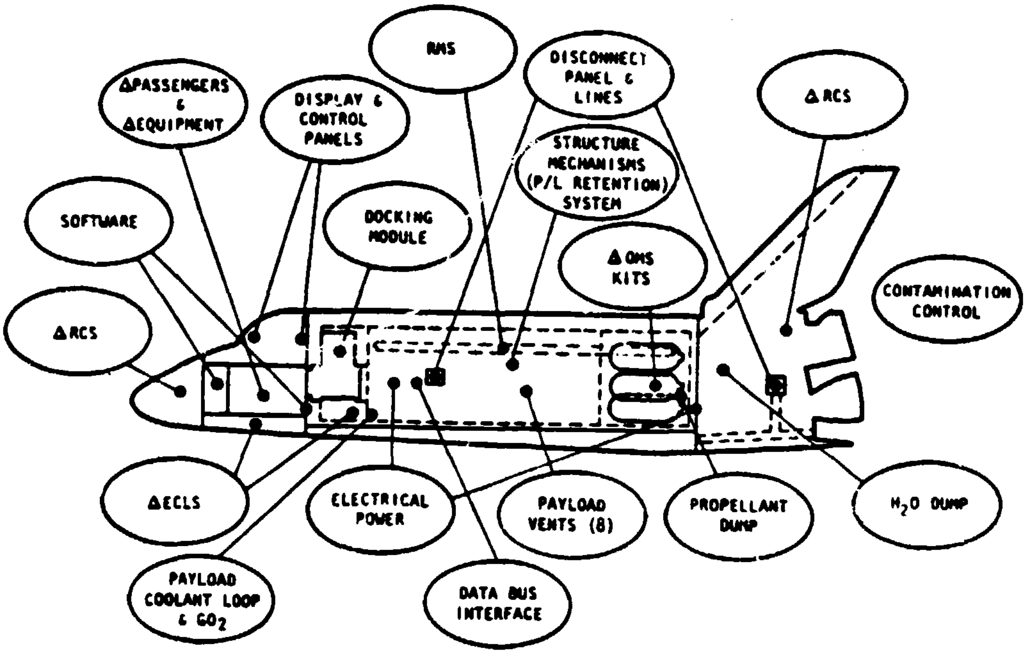


Figure 1.3-1. Summary of Orbiter Payload Accommodations

Table 1.3-1 provides a checklist of types of payload accommodations and cross-references to applicable paragraphs in this document where further information on constraints and guidelines may be found.

The remote manipulator system, docking module, and payload retention system are 100% dedicated to payloads. Additional flexibility is gained with EVA capabilities and mission extension kits. Kits can augment crew support, orbiter propulsion, primary power supply energy, and heat rejection capability. Limits on power availability, heat rejection capability, c.g. requirements, etc., presented here apply to the cargo, i.e., the total complement of payloads and associated hardware carried on a given flight. Provisions will be provided for trimming cargo c.g. and for managing and isolating services provided to the payloads in order to minimize these concerns for payload design.

Table 1.3-1. Checklist: Orbiter Payload Accommodations Pertinent to Large Space Structures Design

ACCOMMODATIONS AND SERVICES	SECTION NO.	TYPICAL IMPLICATIONS FOR LARGE SPACE STRUCTURES DESIGN AND SUPPORT EQUIP.
( ) Payload bay volume	3.1	Limits package volume
( ) Control/display volume in crew cabin aft flight deck	3.1	Constrains area and types of controls and displays for cabin crew control of construction
( ) Stowage volume in crew cabin for supplies, equipment	3.1, 5.1	Constrains crew on-orbit time or number of crew
( ) Seats and volume for additional passengers in crew cabin	3.1, 5.1	Potential for added construction crew specialists
( ) Sleeping provisions for four crew members	3.1, 5.1	Constrains number sleeping simultaneously, may dictate multi-shift operations for large crew
( ) OMS kit for added altitude, on-orbit maneuvering	3.1.1	Constrains payload bay volume usage, increases altitude and payload weight capability
( ) Additional N_2 supply tanks for multiple repressurizations of airlock	3.1	Reduces payload, may require added development costs
( ) Payload attachment bridge fittings and trunnion supports	3.2	Recommended specific attach points for all mission phases
( ) Payload attachment keel fittings and supports	3.2	Possible attach points for all mission phases, primarily for ascent and descent, limited locations
( ) Electrical power/connection points	3.8.1	Affects electrical harness design
( ) Standard mixed cargo harness (SMCH)	3.8.1	Standard payload electrical power & signal attach method

Table 1.3-1. Checklist: Orbiter Payload Accommodations Pertinent to Large Space Structures Design (Cont.)

ACCOMMODATIONS AND SERVICES	SECTION NO.	TYPICAL IMPLICATIONS FOR LARGE SPACE STRUCTURES DESIGN AND SUPPORT EQUIP.
( ) Hard points for attaching electrical and fluid lines in payload bay	3.8	Affects routing of electrical harness, fluid lines
( ) Electrical signal and data/line connection points	3.8.2	Affects control capability of space construction
( ) Thermal control/fluid connections in payload bay	3.9	Potential means to cool/heat construction equipment
( ) Payload vents (8) in payload bay	2.6, 4.14.2	No identified design constraints
( ) Propellant dump	3.10	Operational safety provisions
( ) Water dump	4.14	Contamination source
( ) Contamination control	4.14	Facilitates mission success
( ) Docking and berthing capability	4.1	Design of berthing/docking port, structure clearance
( ) Attitude control	4.2	Impacts space structure loads, lighting and vision
( ) On-orbit equipment supports	4.4.2	Affects design of construction fixtures
( ) Payload handling equipment and methods	4.4.4 4.5. to 4.8	Impacts design of structure (reach, clearances, attach points
( ) Remote manipulator system (RMS)	4.5 4.8.2.1	Location of structure within reach of RMS; major payload handling device

Table 1.3-1. Checklist: Orbiter Payload Accommodations Pertinent to Large Space Structures Design (Cont.)

ACCOMMODATIONS AND SERVICES	SECTION NO.	TYPICAL IMPLICATIONS FOR LARGE SPACE STRUCTURES DESIGN AND SUPPORT EQUIP.
( ) Manned maneuvering unit (MMU) stowage and support provisions	4.4.4 4.8.2.3	Defines capability for EVA inspection, transport of small objects, manual assembly operations
( ) Extravehicular maneuvering unit (EMU—space suit) and stowage/support provisions; normally two units for EVA	4.4.4 4.8.1.6	Defines capabilities for manual operations in EVA mode
( ) Communications for EVA	4.8.1.7	Affects operations planning and electronics support
( ) Airlock (two-man capacity), tunnel adapter, docking module	4.8.1.1 4.8.1.2 4.8.1.3	Provides for EVA Defines impact on breathing gas consumables
( ) Handholds, safety tethers in payload bay for EVA	4.8.1.5	Assists EVA in area of payload bay; safety feature
( ) TV cameras in payload bay, RMS	4.9.3	Useful for monitoring space construction, deployment and retrieval of payloads
( ) Lights in payload bay ( ) Docking light	4.9.2	Assists construction on dark side of orbit
( ) Crew optical alignment sight (COAS)	4.9.4	Assist berthing/docking
( ) Payload viewing windows—aft flight deck, overhead	4.9.4	Provides crew view of some construction operations
( ) Electrical power and energy storage for payloads	4.10	Defines capability for power usage for machinery and total energy needed for construction, lighting, etc., may affect payload bay volume (extra cryo tanks)

Table 1.3-1. Checklist: Orbiter Payload Accommodations Pertinent to Large Space Structures Design (Cont.)

ACCOMMODATIONS AND SERVICES	SECTION NO.	TYPICAL IMPLICATIONS FOR LARGE SPACE STRUCTURES DESIGN AND SUPPORT EQUIP.
( ) Communications, tracking, and data management	4.11	Impacts antenna locations, equipment requirements
( ) Data processing and software	4.12	Impacts design of data handling equipment, software
( ) Heat rejection from payloads	4.13	Defines capability for payloads to reject heat through orbiter, physical interfaces for design
( ) Crew cabin accommodation (general) and supplies	5.1	Provides habitability for crew and work station in shirtsleeve cabin for remote control equipment
( ) STS ground handling accommodations and equipment	6.0	Provides for stowage on ground, ground transport and payload installation into orbiter; affects design of hardpoints for handling payload, turn-around time.

Payloads may be designed to interface directly with orbiter hardware or with an STS carrier. The environments seen, services available, and STS fees for carrier-mounted payloads depend upon the carrier as well as the Shuttle.

Sections 2.0 through 5.0 describe selected payload environments and support capabilities, constraints, and guidelines where payloads are taken to interface directly with the Shuttle. Therefore, the information applies most directly to carriers with their integrated payloads and directly mounted payloads.

1.3.1 Standard Shuttle Services

The following list briefly outlines the standard Shuttle services available for the basic flight cost. (NASA 1977 (c)), Attachment B (NMI 8610.8))

- Two standard mission destinations:
  1. 160 nmi altitude; 28.5° inclination
  2. 160 nmi altitude; 56.0° inclination
- One-day mission operations
- Orbiter flight planning services
- Transmission of payload data to compatible receiving stations
- A three-man flight crew
- On-orbit payload handling
- Deployment of a free flyer
- NASA support of payload design reviews
- Prelaunch payload installation, verification and orbiter compatibility testing
- NASA payload safety review

1.3.2 Optional Shuttle Services

The following list indicates additional services available to Shuttle users at extra cost. (NASA 1977 (c)), Attachment B (NMI 8610.8))

- Revisit and retrieval
- Use of Spacelab or other special equipment
- Use of Mission Kits to extend basic orbiter capability
- Use of Upper Stages
- EVA services
- Unique payload/orbiter integration and test
- Payload mission planning services, other than for launch, deployment and entry phases
- Additional time on-orbit

Payload data processing

Launch from Western Test Range

- Two standard mission destinations are available from the Western Test Range site:
  1. 160 nmi altitude; 90.0° inclination
  2. 160 nmi altitude; 104.0° inclination

1.4 DESIGN IN RELATION TO PAYLOAD INTEGRATION PROCESS

The Space Shuttle Orbiter is sometimes referred to as a "space truck", which implies that its major function is simply transportation of a relatively inert, separate cargo. However, the realities of space travel complexity are such that a great many payloads carried in the orbiter become intimately linked to the orbiter payload accommodation system in a symbiotic fashion in order to function effectively and safely. The process of assuring payload compatibility with orbiter systems and checking that the necessary links between orbiter and payload will match and function properly is called "Payload Integration."

1.4.1 Planning for Variety of Orbiter Roles

Payload Integration is a major consideration in space construction. Design of a large space construction project envisages use of the orbiter in many roles. For example, it may act as a stabilizing device, as construction crew living quarters, as a power plant, as a thermal conditioning system, as a communications station, as a portable illumination stand, and as a space "crane" for handling large equipment. Therefore, there are many Shuttle payload integration issues involved in space construction, just as there are in most sortie experiments or satellite launch and retrieval missions. "Dedicated" construction flights will have (normally) inert cargos which should enable factory-to-pad processing without intermediate payload testing. Mixed cargos will, of course, require integration and test as dictated by other cargo elements.

1.4.2 Process

NASA has established procedures and reviews for the Shuttle payload integration process. These are indicated in Figure 1.4-1. Further discussion with NASA concerning administrative procedures and related engineering support efforts is encouraged before extensive design efforts are begun, since the results could affect project schedules, costs and ultimately, the constructed spacecraft design.

1.4.3 General System Safety Considerations

Many of the payload integration reviews involve the subject of safety for all phases of STS operations. The basic policy and requirements for Shuttle safety are contained in NASA document NHB 1700.7 and its subsidiary documents (see NASA 1979 in Bibliography). Such generally applicable requirements are not included as special constraints or guidelines in this document unless there are major new, unusual applications or key issues specifically involved in large space structures design. Such concerns appear in Sections 4.3, 4.8 and 5.0.

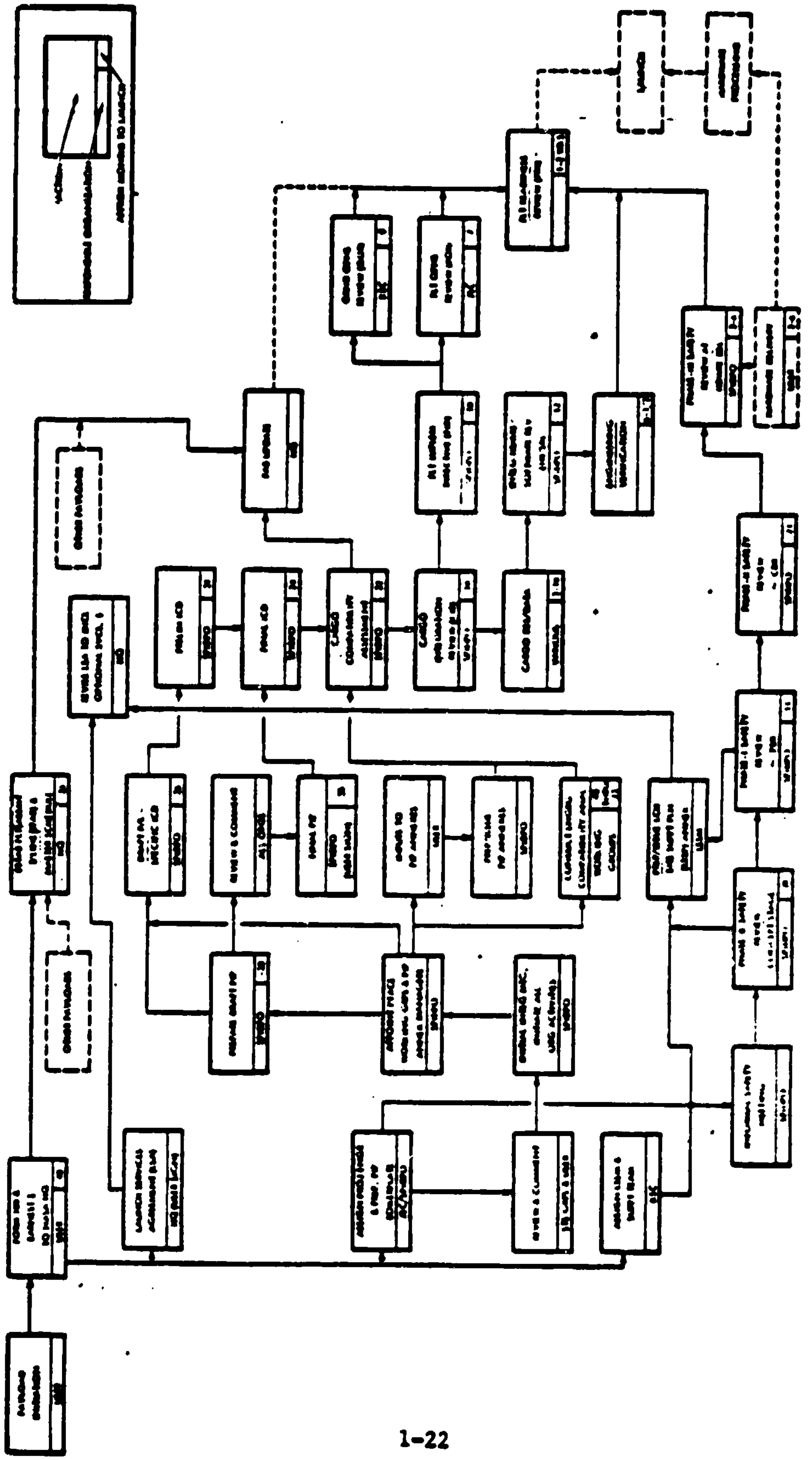


Figure 1.4-1. Payload Integration Process

2.0 SHUTTLE FLIGHT CONSTRAINTS ON LARGE SPACE STRUCTURES DESIGN

Table 2.0-1 lists significant Shuttle flight-related considerations affecting selection of construction orbit altitude and inclination, mission timelines, pressures and loads on Shuttle cargoes. Unless otherwise noted, essentially circular orbits are assumed in order to minimize on-orbit drag effects. These constraints/capabilities and consequent guidelines are considered in Sections 2.1 through 2.7.[1]

Table 2.0-1. Potential Shuttle-Related Effects on Construction Orbit Selection

ORBIT SELECTION CONSIDERATION	TYPICAL SHUTTLE CONSTRAINTS ON ORBIT
ORBIT DECAY DUE TO AERODYNAMIC DRAG	• DRAG OF ORBITER • COMBINED DRAG AND WEIGHT FOR ORBITER AND CONSTRUCTION PROJECT • ORBITER TURNAROUND VS. CONSTRUCTION ORBIT DECAY TIME AND MAKEUP REQUIREMENTS
IONIZING RADIATION ENVIRONMENT REQUIREMENTS • CREW SELECTION • SHIELDING • EVA DURATION	• CREW DOSAGE WHILE WITHIN CABIN IS FUNCTION OF ALTITUDE, INCLINATION, AND DURATION • CREW DOSAGE DURING EVA: FUNCTION OF ALTITUDE, INCLINATION, DURATION, AND SUIT MATERIALS
SHUTTLE PERFORMANCE (INCLUDES BOOSTERS, EXTERNAL TANKS, ORBITER ENGINE)	• ALLOWABLE LIFTOFF WEIGHT • ALTITUDE VS. WEIGHT WITH AND WITHOUT OMS KITS* • CENTER OF MASS OF CARGO • VOLUME LIMITATION OF OMS KIT(S), UNDERLINER STOWAGE SPACE • TIME TO ATTAIN ORBITAL CONDITIONS, OPEN DOORS, AND BEGIN CONSTRUCTION
*OMS = ORBITAL MANEUVERING SYSTEM	

The Shuttle ascent also imposes physical, environmental constraints on the crew and on design of construction equipment, materials, and their support systems (e.g., cradles, pallets) within the orbiter. Although related to Shuttle performance, these constraints are relatively independent of construction orbit altitude. Examples include acceleration loads, vibration loads, pressure changes, and thermal changes. Such constraints are discussed under

[1] "Caution: Not a control document! See appropriate reference documents for current data."

Sections 2.3.2 and 3.0 (Payload Accommodations). Prelaunch operations are discussed in conjunction with Shuttle ground operations constraints (Section 6.0). Deorbit and descent phases of orbiter operation also impose constraints and guidelines which are included in Section 2.4 (and 3.0, as regards attachment loads).

2.1 ORBIT DECAY

One of the main concerns in selecting the orbit altitude for space construction is orbit decay. Large area space systems now made possible by construction in space using the Shuttle need only to be sized for very small forces and loads and, hence, are typically very low in density. Therefore, they tend to have a very low ballistic coefficient (W/C_DA) which can result in relatively high rates of orbit decay. This high decay rate is primarily a function of the space construction project design, which is only indirectly influenced by specific Shuttle constraints. However, there is a small number of Shuttle-related concerns, which are reflected in Tables 2.1-1 and 2.1-2.

To estimate the combined effects of construction project and orbiter on decay time, ballistic coefficients are calculated for different area/weight features through the construction process. These are combined to form a "drag profile" representing the overall construction process, including periods with and without the orbiter. Representative values for the orbiter are shown in Figure 2.1-1. However, each project will have specific weight and drag considerations according to the supplies carried, type of construction, range of attitudes on orbit, etc.

2.2 RADIATION ENVIRONMENT IMPACTS

The natural radiation environment is another factor which could potentially affect orbit altitudes for space construction. Of particular concern is the increased exposure during EVA activity and the possible need for designing automated construction techniques and processes which minimize the use of manned EVA participation. The Shuttle orbiter cabin and the standard Shuttle pressure suit both enter into the calculation of crew dosage rates by virtue of their respective mass distribution, which provides a degree of radiation shielding. Table 2.2-1 lists pertinent Shuttle-related constraints and guidelines. Further explanation of the radiation hazard and results of analytical studies which dictate limits to orbital altitude for various EVA cases are presented in the subsequent paragraphs.

For 28° inclination circular orbits, only Van Allen belt electrons and protons are significant. Solar flare particles are excluded by the geometric field (cutoff energies $\geq$3 GeV) and galactic (cosmic ray) particles contribute $\leq 10^{-2}$ rad/day independent of shielding (for $\leq$100 gm/cm). In the absence of man-made nuclear radiation, therefore, only the Van Allen belts need be considered.

The Space Shuttle orbiter has an effective shield thickness for the crew of ~3 gm/cm^2. Thus, the cutoff energies are ~50 MeV for protons and ~5 MeV for electrons. For EVA operation the typical space suit provides ~0.2 gm/cm^2 effective shielding, which has cutoff energies of ~11 MeV (protons) and

Table 2.1-1. Checklist: Shuttle Constraints on Orbit Altitude Selection Related to Orbit Decay

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) Altitude during non-tended mode is high enough to maintain viable orbit during estimated orbiter turnaround time (one month minimum). (R-4) • Consider launch site • Consider inclination • Average weight/drag ratio? • Does untended construction project have orbit makeup, guidance, and control capability? • Consider additional orbiter to reduce turnaround time.	Rockwell, 1979 (a): SSD 79-0123 • Potential impact on orientation of large space structure to control drag and on weight liftable to orbit selected. • Dedicated orbiter is assumed.
( ) Altitude during orbiter-tended construction phases is high enough to avoid excessive makeup thrust fuel requirement in orbiter. (R-4)	Rockwell, 1979 (a): SSD 79-0123 • Potential impact on orientation of space structure, weight liftable to orbit and system complexity
( ) Trades of orientation to control drag vs. orbit makeup and high construction orbits are required to optimize the construction altitude for each project system. (R-4)	Rockwell, 1979 (a): SSD 79-0123

Table 2.1-2. Checklist: Guidelines for Selecting Orbit Altitude/Inclination for Space Construction

( ) GUIDELINES/• QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) Within constraints of the requirements for Orbit decay prevention, altitudes for space construction should be as low as feasible to maximize weight which can be lifted to orbit. (R-4)	Rockwell, 1979 (a): SSD 79-0123
( ) Large projects involving multiple Shuttle flights will likely require OMS kits to meet the minimum altitude limits for orbit decay. • Attitude control to minimize drag effects can significantly reduce minimum altitude limits for space construction, possibly to the point where OMS kits are not required. (R-4)	Rockwell, 1979 (a): SSD 79-0123
( ) Avoid deployment of solar arrays and other large-area devices which can create high drag until required for orbital operations. (R-4)	Rockwell, 1979 (a): SSD 79-0123
( ) An altitude range of 450-500 km (250-275 nmi) appears satisfactory for use in initial analyses of most projects. (R-4)	Rockwell, 1979 (a): SSD 79-0123

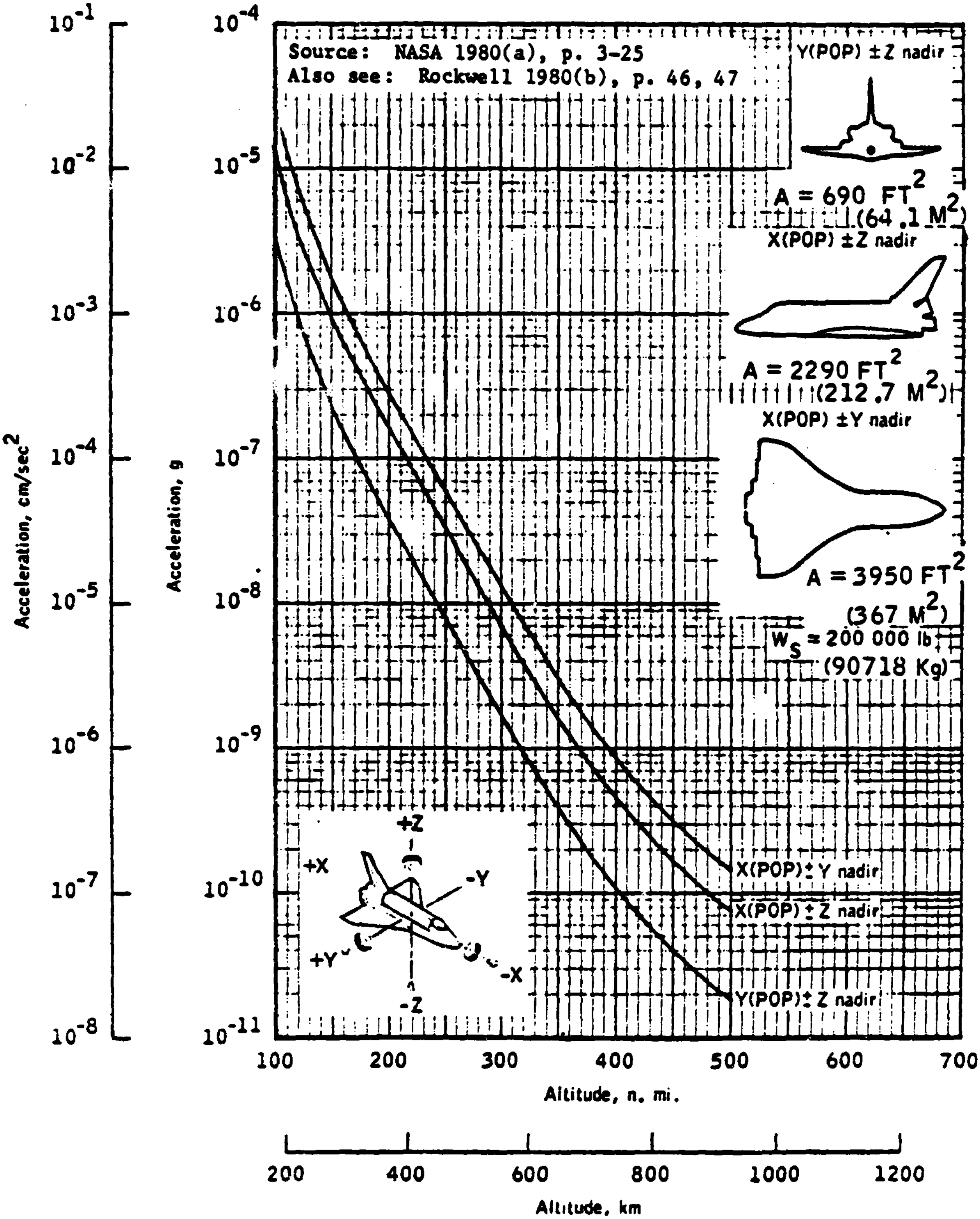


Figure 2.1-1. Basic Data for Orbiter Ballistic Coefficients (and associated effects of atmospheric drag on orbiter)

Table 2.2-1. Checklist: Shuttle Constraints and Guidelines Involving Ionizing Radiation Effects on Orbit Selection

( ) CONSTRAINTS	REFERENCES / IMPLICATIONS
( ) The Space Shuttle orbiter cabin has an effective shield thickness of approximately 3 gm/cm^2 for ionizing radiation. (R-1)	Rockwell, 1979 (a): SSD 79-0123 • Orbit altitude and duration • Crew selection
( ) The estimated effective radiation shielding for the EMU (space suit) is approximately 0.2 gm/cm^2. (R-3)	Rockwell, 1979 (a): SSD 79-0123 • Orbit altitude • EVA duration
( ) GUIDELINES	
( ) Ionizing radiation is unlikely to be a serious limit on orbit altitude or inclination for space construction using the orbiter as a base. For construction orbit altitudes below 500 km (275 nmi), more than 100 EVA construction missions could be flown within career dosage limits. Radiation altitude limits are significantly above the limits due to orbit decay. (R-3)	Ionizing radiation is generally not a constraint

~0.45 MeV (electrons). The particle fluxes of interest are those with energies above these cutoff energies.

Calculations have been carried out to obtain the Van Allen particle fluxes and tissue dose rates as a function of altitude for 28° inclination circular orbits. The SREP computer code (Hamilton, 1974) was used to calculate the daily electron and proton fluxes, with the flux-to-tissue dose conversions being accomplished by hand calculations. The results obtained were tissue doses for 0.2 and 3 gm/cm^2 shielding as functions of altitude.

The Van Allen belt skin dose rates as a function of altitude are shown in Figure 2.2-1 for two shielding thicknesses—0.2 gm/cm^2 and 3 gm/cm^2. These are daily averages for circular orbits with an inclination of 28°. At the altitudes of interest (≤1000 km) most of these doses will be received in the South Atlantic anomaly. Since the spacecraft passes through this anomaly only 3 to 7 orbits per day, depending upon altitude, it may be possible to schedule short-term (≤6 hr) EVA during the orbits when the South Atlantic anomaly will not be encountered. No account of this effect (which is not important above ~1000 km altitude) was taken in this analysis.

While there are no "official" radiation dose limits for astronaut, the National Academy of Sciences recommendations are often used for mission analysis studies. These recommendations, listed in Table 2.2-2 were used on this study. For small shield thicknesses (e.g., an EVA suit) the skin dose limits are the overriding factor, but for large shield thicknesses usually the bone marrow dose limits determine the mission limit (duration or altitude). The tissue dose rates for the skin, eyes, and bone marrow are shown in Figure 2.2-2.

In order to estimate the maximum altitude for the 7-, 10-, and 30-day missions, the 30-day dose limits of Table 2.2-2 were used. These are 75 rad (skin), 37 rad (eyes), and 25 rad (bone marrow). The numbers used to estimate maximum altitude for astronauts in the Shuttle orbiter cabin are listed in Table 2.2-3. The first part of the table lists the allowable dose rates (rad/day) permitted for the bone marrow, skin, and eyes. These numbers are merely the 25, 75, and 37 rad limits from Table 2.2-2 divided by the mission durations. The second part of the table lists the altitudes from Figure 2.2-2 for the dose rates in the top part of the table. It is seen that bone marrow is the limiting organ for all three mission durations.

The corresponding numbers for EVA are listed in Table 2.2-4, except only the skin was used since it will be the limiting factor inside the 0.2 gm/cm^2 EVA suit. If continuous (24 hr/day) EVA were necessary, the maximum altitudes (read from the top curve of Figure 2.2-1) vary from 800 km (7 days) to 430 km (30 days). However, by limiting EVA to 6 hr/day, the maximum altitudes can be increased to 1030 km (7-day missions) to 680 km (30-day missions). The numbers will be decreased somewhat to allow for the radiation doses received from the 18 hr/day when the astronauts are inside the cabin.

For example, on a 10-day mission, the astronaut will spend 2.5 days of EVA at a skin dose rate of ~10X and 7.5 days inside the cabin at a bone marrow dose rate of ~X. The total dose will be ~32.5X to the skin and ~10X to the bone

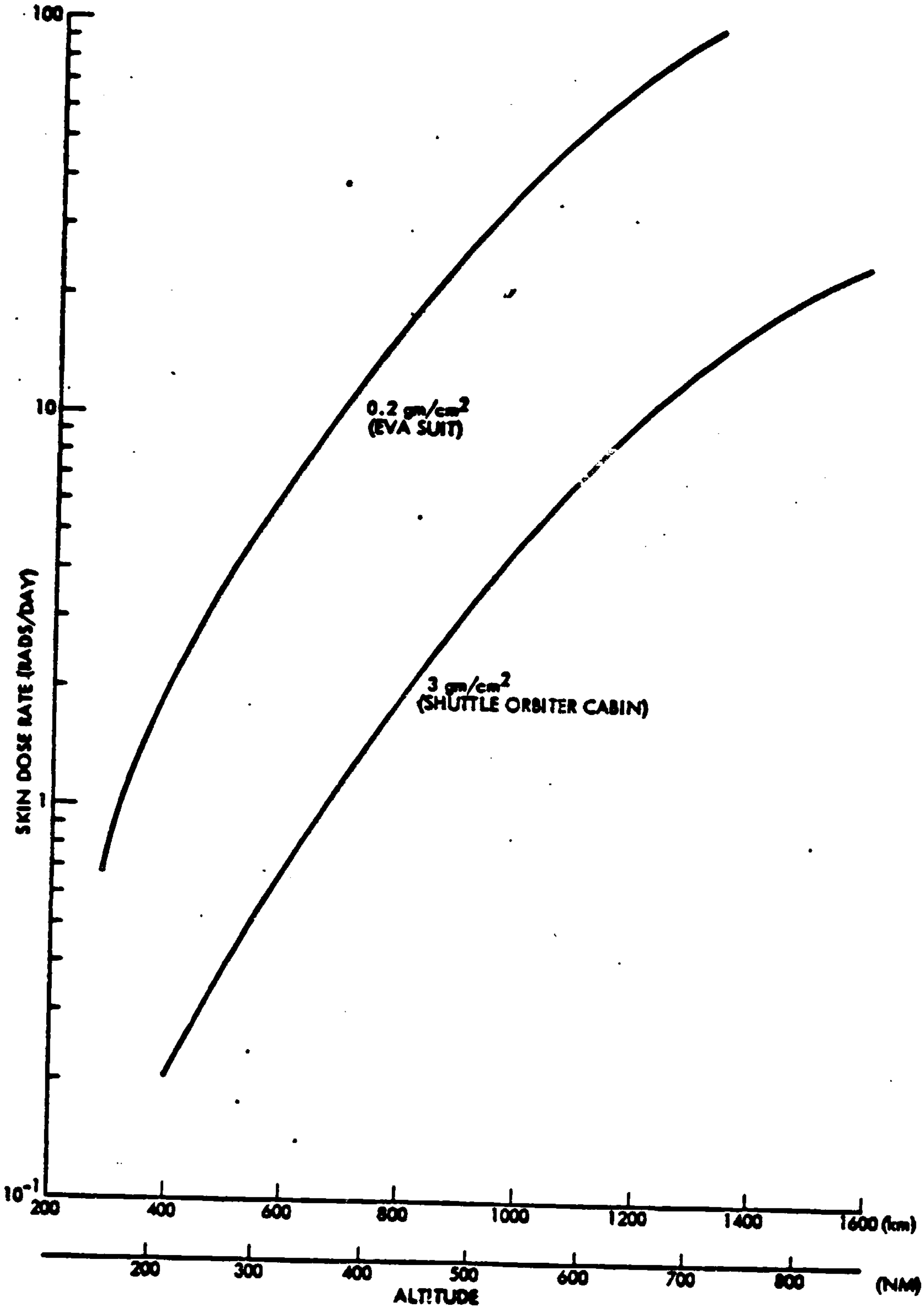


Figure 2.2-1. Skin Dose Rates in the Van Allen Belts

Table 2.2-2. **Recommended Astronaut Dose Limits**

Mission Duration	Dose Limit (rad) Bone Marrow (5 cm depth)	Skin (0.1 mm depth)	Eyes (3 mm depth)
30 days	25	75	37
90 days	35	105	52
180 days	70	210	104
1 year	75	225	112
Career limit	400	1200	600

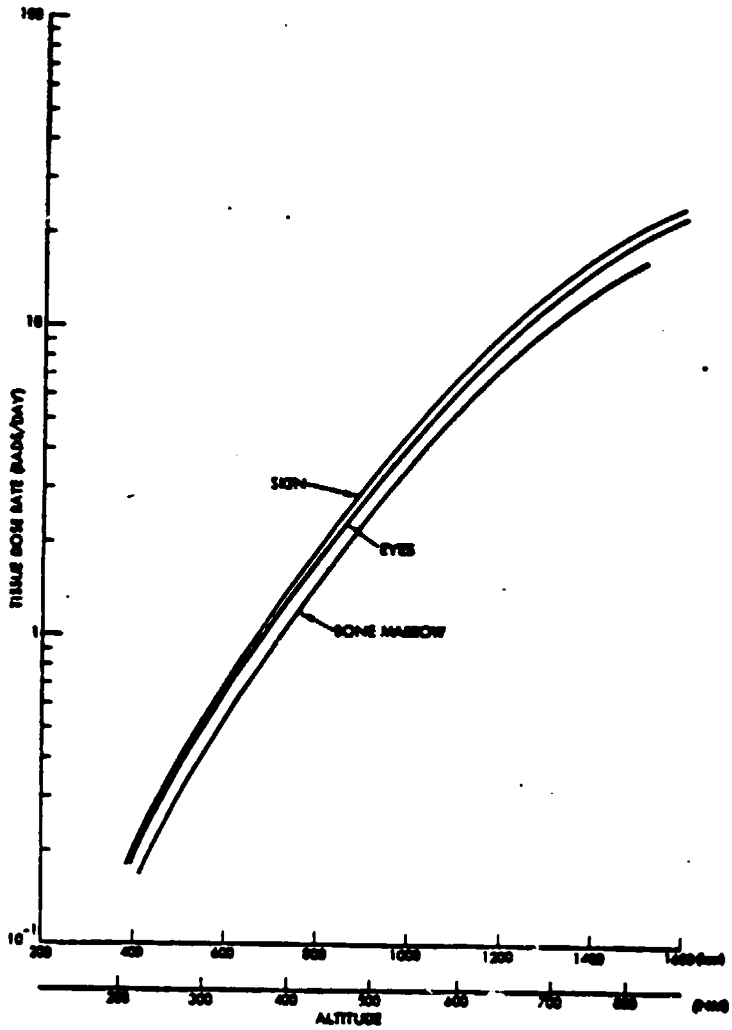


Figure 2.2-2. Tissue Dosz Rates in the Van Allen Belts (Behind 3 gm/cm^2 Shielding)

Table 2.2-3. Maximum Altitude for Astronauts Inside the Orbiter

Mission Duration (days)	Bone Marrow	Skin	Eyes
7	3.57	10.7	5.29
10	2.50	7.5	3.7
30	0.833	2.5	1.23

Mission Duration (days)	Maximum Altitude inside Orbiter (km)		
	Bone Marrow	Skin	Eyes
7	1000	1230	1060
10	920	1130	970
30	680	860	730

Table 2.2-4. Maximum Altitudes for EVA Operation

Mission Duration (days)	Allowable Skin Dose Rate (rad/day)	Maximum Continuous EVA Altitude (km)
7	10.7	710
10	7.5	620
30	2.5	430

Mission Duration (days)	EVA Duration (days)	Allowable Skin Dose Rate (rad/day)	Maximum EVA Altitude (km)
7	1.75	43	940
10	2.5	30	860
30	7.5	10	620

marrow. (The bone marrow dose rate is approximately the same for EVA and cabin occupancy.) Therefore the value of X for the skin is 75/32.5 = ~2.3, so that the EVA skin dose rate should be ~23 rad/day and the in-cabin dose rate to the bone marrow should be ~2.3 rad/day. These numbers lead to an EVA altitude of ~860 km (from Figure 2.2-1) and an in-cabin altitude of 900 km (from Figure 2.2-2). To check, if the orbit altitude is the smaller of the two numbers (860 km), the skin dose rate will be 23 rad/day × 2.5 days = 57.5 rad during EVA and 2.5 rad/day × 7.5 days = 18.7 rad during cabin occupancy (total 76.2 rad, slightly above the 75 rad allowed). The bone marrow dose will be ~2 rad/day × 10 days = 20 rad, less than the 25 rad allowed. This 860-km altitude is less than the 930 km allowed (on the basis of the EVA alone) or the 920 km allowed on the basis of cabin occupancy alone.

In this way (by iteration), the maximum altitudes for 7-, 10-, and 30-day missions with 25% EVA and 75% cabin (Shuttle orbiter) occupancy were calculated to be 940, 860, and 620 km, respectively. These are high enough that atmospheric drag will not unduly limit orbit lifetime. It is possible to increase the orbit altitudes somewhat by using a heavier space suit, but the difficulty of working in a heavier suit outweighs the slight orbit altitude increase. For example, increasing the EVA suit to 3 gm/cm^2 (the same shielding as the orbiter provides) would only increase the orbit altitude permitted by ~100 km.

If the orbit inclination were decreased to 0°, the environment decreases for orbit altitudes $\leq$900 km but increases for altitudes of $\geq$900 km (Figure 2.2-3). The effect of other orbit inclinations can also be seen. In the low altitude region of interest for space construction, the 30- to 60-degree orbit inclination band has the most severe radiation environment (due to the South Atlantic anomaly). Thus, the EVA altitude limits defined in the preceding tables are applicable to all orbit inclinations and, in fact, offer higher dose margins in the equatorial and solar inclination regions.

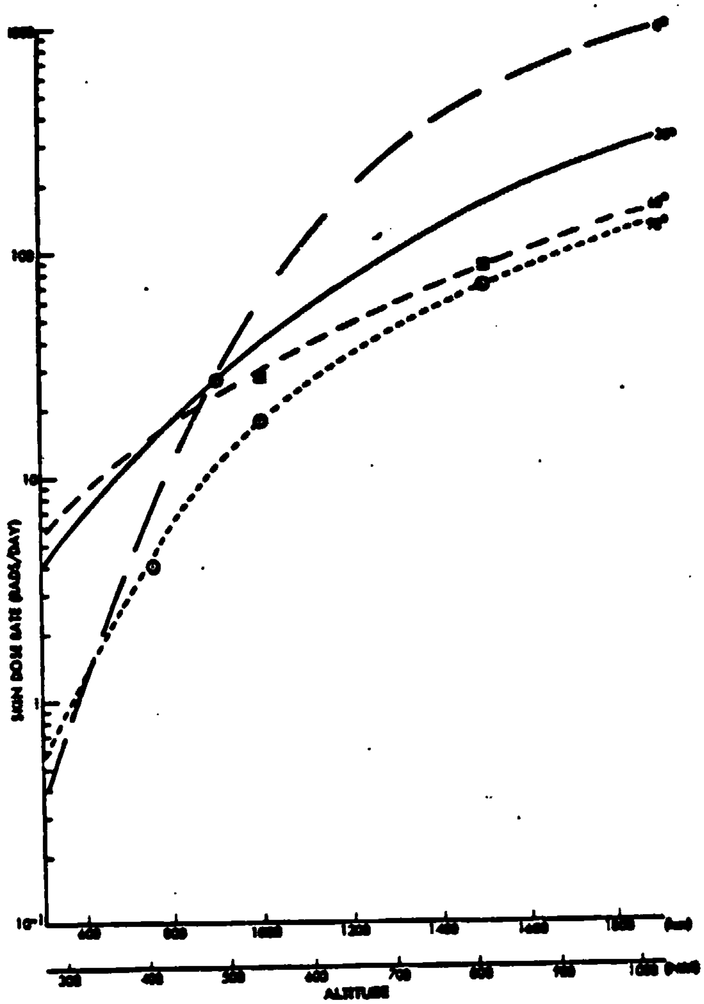


Figure 2.2-3. Effect of Orbit Inclination on EVA Skin Dose (0.2 gm/cm^2 Shielding)

2.3 SHUTTLE PERFORMANCE

In addition to orbit decay and radiation hazard, Shuttle delivery performance is a major constraint which must be considered in selecting suitable orbit altitudes for space construction and in planning mission schedules and consumables. The Shuttle weight lifting capabilities are considered first.

2.3.1 Shuttle Payload Weight Constraints

Shuttle cargo weight limits to circular orbits are defined in Section 3.1.1 of Volume XIV, JSC 07700 (NASA 1980(a)). They are described by means of a series of graphs relating cargo weight to launch site and desired orbit altitude at selected orbit inclinations. At the time of this writing a set of revisions to these charts was being processed (Murrah, 1980). These revisions show that the maximum weight which the Shuttle can lift to orbit will vary according to planned changes in orbiter capability between 1982 and 1985. Figures 2.3-1 through 2.3-4 are the performance graphs relating orbiter weight lifting capability to launch site, circular orbit altitude and number of OMS kits provided, as described in the foregoing reference (Murrah, 1980).

However, Figure 2.3-4 also includes superimposed graphical representations of upper and lower limits of radiation hazards and drag effects which were previously described. Radiation hazard bounds are shown for both EVA and the orbiter crew cabin. Two limits for minimum altitude due to orbit decay are also shown. One corresponds to the higher decay rates associated with random unconstrained orientations and the other (lower limit 370 km) reflects low decay rates associated with orientation continuously constrained to maintain minimum drag valles. The region between the "radiation" and "orbit decay" limits wolld be suitable for space construction.

This region tends to be centered on that portion of the Shuttle performance envelope which requires the inclusion of a single OMS kit. Thus, many construction payloads, depending upon packaging characteristics, would require the use of OMS. This would be particularly true for high drag configurations. The intrusion of the OMS kit into the available cargo bay volume must, therefore, be considered in planning construction cargo manifests. Lower drag configurations might possibly be constructed at low enough altitudes that some of the construction flights, those with cargoes that are "volume limited", could be performed without the need for OMS. (Experience has indicated construction mission payloads tend to be volume-limited rather than weight-limited.)

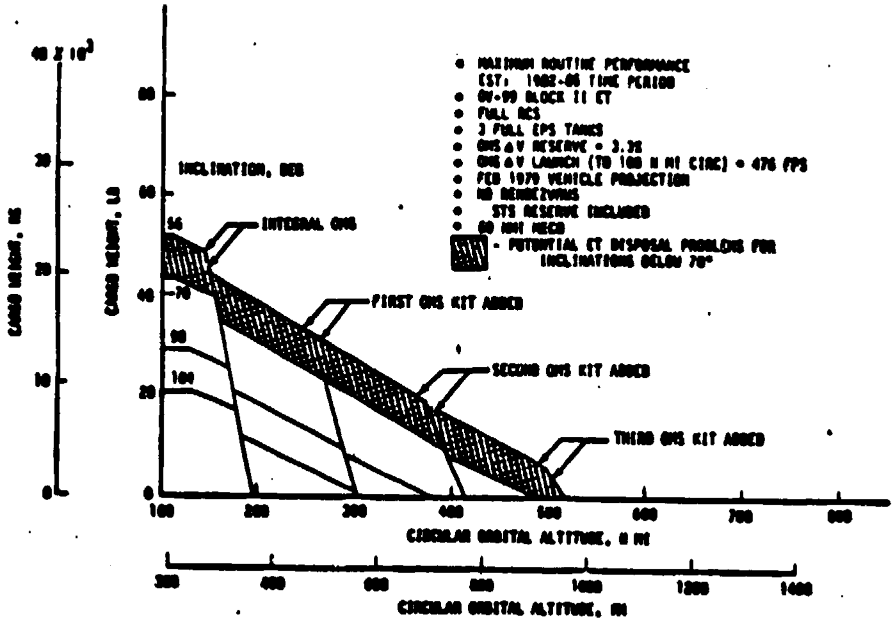


Figure 2.3-1. Near Term (Pre 1985) Cargo Weight Versus Circular Orbital Altitude - VAFB Launch, Delivery Only

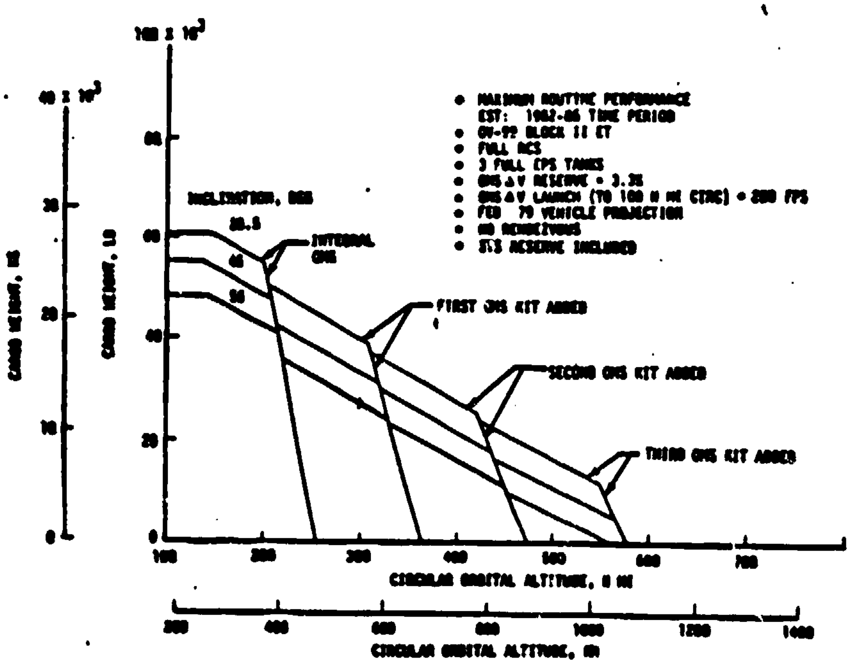


Figure 2.3-2. Near Term (Pre 1985) Cargo Weight Versus Circular Orbital Altitude - KSC Launch, Delivery Only

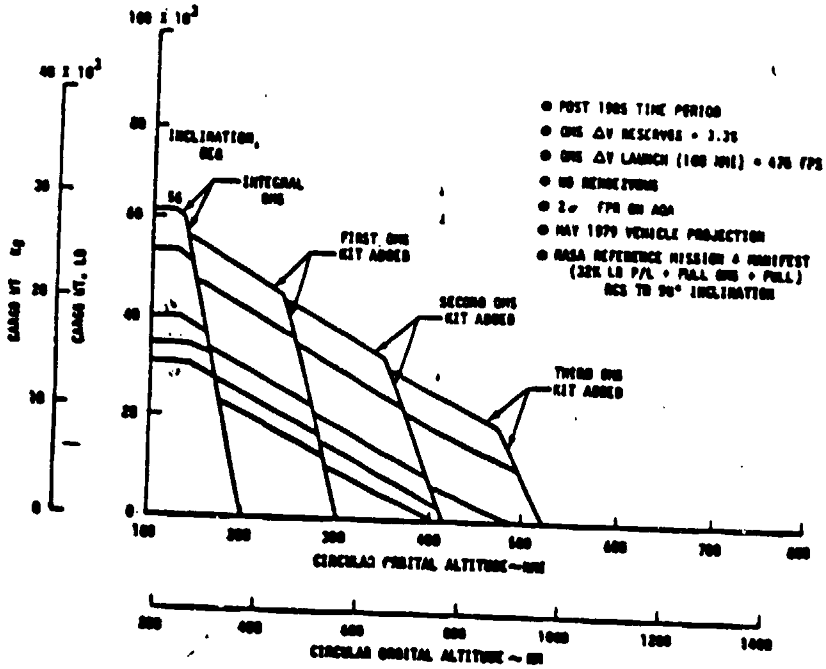


Figure 2.3-3. Maximum Routine Performance of the Augmented Shuttle - VAFB Launch

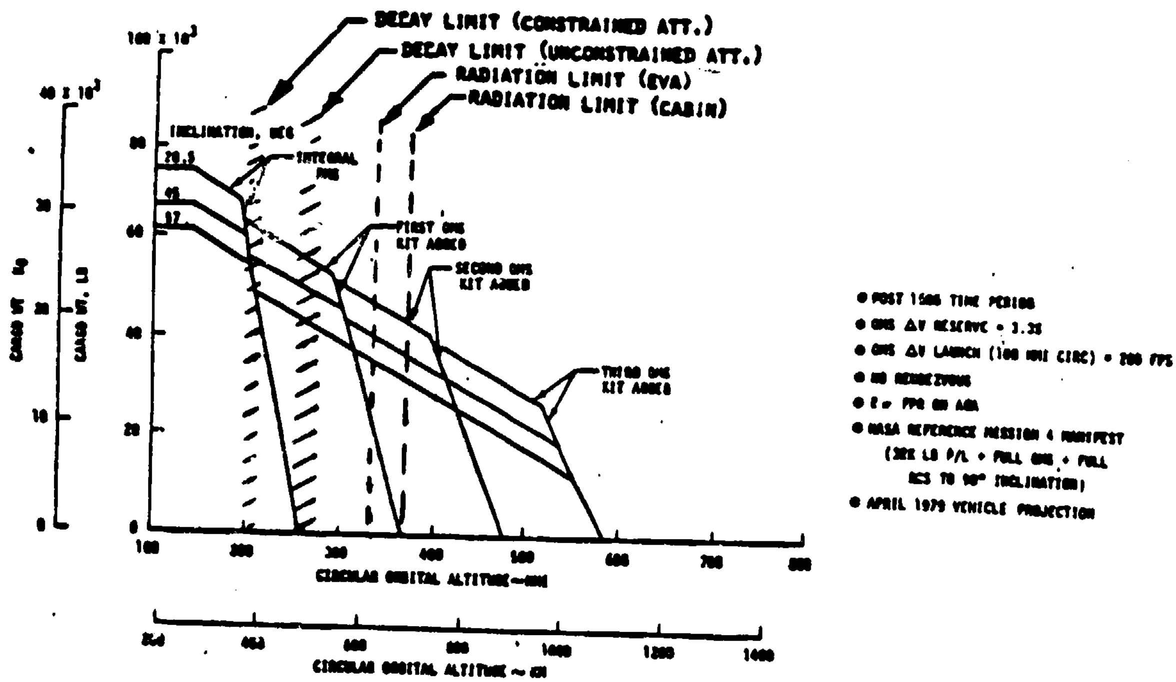


Figure 2.3-4. Maximum Routine Performance of the Augmented Shuttle - KSC Launch with Superimposed Construction Constraints

Detailed analyses are required on the integrated construction process to adequately determine the actual drag history and orbiter bay packaging of the individual construction flights to more accurately determine the construction orbit altitude requirements for a given project system. These could be further refined by inclusion of solar cycle effects on atmospheric density for the projected project system schedules. However, the preliminary analysis reported here serves to identify the key factors affecting construction orbit altitude and highlights their significance to the specific project systems contained in the study. Table 2.3-1 summarizes the above conclusions.

Table 2.3-1. Checklist: Guidelines for Selecting Orbit Altitude and Inclination for Space Construction considering Orbit Decay, Ionizing Radiation, and Shuttle Performance

( ) GUIDELINES/• QUALIFICATIONS	REFERENCES IMPLICATIONS
( ) Large projects involving multiple Shuttle flights will likely require at least one OMS kit to meet the minimum altitude limits for orbit decay. (R-4) • Attitude control to minimize drag effects can significantly reduce minimum altitude limits for space construction, possibly to the point where OMS kits are not required. • Projects with specific large-volume modules/packages may be limited to lower altitude because of OMS kit volume impacts on the orbiter payload bay.	Rockwell, 1979 (a): SSD 79-0123 Requires trade of cargo weight and volume versus altitude achievable.
( ) An altitude range of 450-500 km (250-275 nmi) appears satisfactory for use in initial analyses of most projects. (R-4)	Rockwell, 1979 (a): SSD 79-0123

2.3.2 Shuttle Ascent Acceleration History

Another aspect of Shuttle ascent performance is the acceleration load history experienced by the crew and orbiter payload (space construction equipment and materials). Figure 2.3-3 shows parameters of a typical mission ascent trajectory, which leads to the constraints listed in Table 2.3-2.

2.3.3 Shuttle Ascent/Rendezvous Timelines

For purposes of mission planning in terms of timelines and expended consumables, the complete duration of a mission must be known. The ascent phase

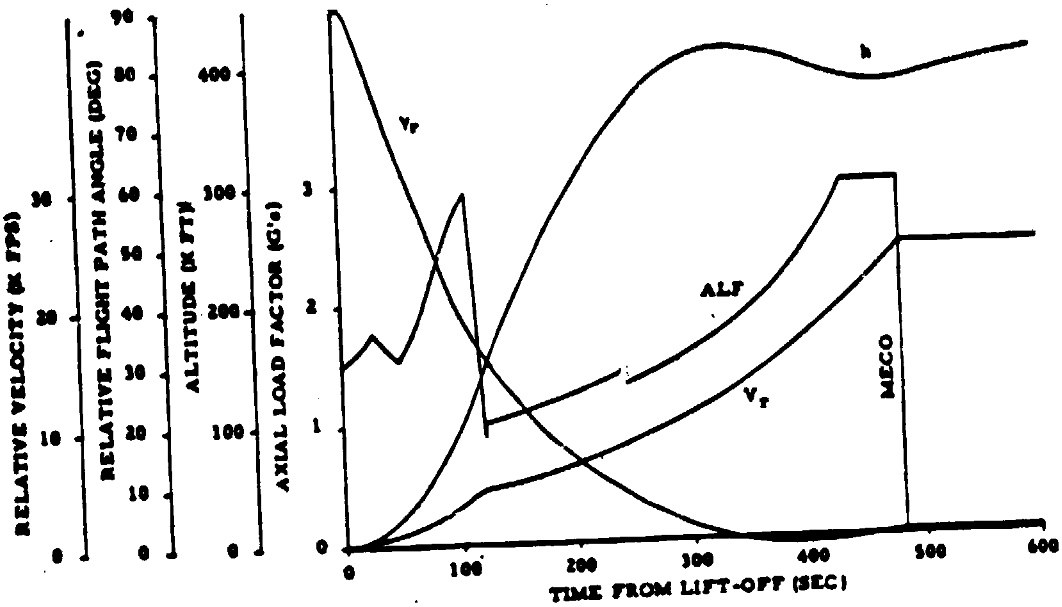


Figure 2.3-5. Typical Mission Ascent Trajectory Data

Table 2.3-2. Checklist: Shuttle Constraints Related to Ascent Acceleration History

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES IMPLICATIONS
( ) All space construction equipment and materials shall be supported and packaged or inherently strong enough to withstand ascent acceleration loads of at least 3 g plus combined loads of vibration and control forces. (R-1) • See also, Sections 3.5 and 3.6 for further information on loads.	NASA, 1980 (a): JSC 07700, Vol. XIV Affects strength of materials, packaging, support equipment design
( ) Orbiter crew and passengers, including construction crews for large space projects, shall be subjected to ascent acceleration loads not in excess of 3 g while in a semi-supine position. (R-1) • For most healthy individuals, this is an acceptable acceleration stress level. However, crew selection screening should consider the biomedical implications of the total acceleration history and the directions in which it acts on the human body.	NASA, 1980 (a): JSC 07700, Vol. XIV Affects crew selection for space construction.

(with phasing) may be a significant portion of this duration. Another concern is the duration which certain kinds of equipment are placed in a storage mode. Such items as batteries, cryogenic fuel tanks, and certain thermally sensitive equipment may fall into this category.

Two types of ascent conditions may be necessary for space construction: direct and with rendezvous. The direct mode is associated with the first of a series. Subsequent flights will be required to rendezvous with the construction project. Each launch and ascent is a unique event with durations determined by payload, time of launch, and orbit parameters. However, some specific limits and probable boundaries can be defined for purposes of this document. Such constraints and guidelines are presented in Table 2.3-3.

2.3.3.1 Reference Ascent Profile with Rendezvous

A typical profile timeline for a rendezvous mission with an arbitrary 12-hour phasing is presented in Table 2.3-4. A detailed description of the activity in each phase is presented below.

	PHASE START G.E.T.			PHASE DURATION		
	HR	MIN.	SEC	HR	MIN.	SEC
1. ASCENT TO INSERTION	00	00	00	00	09	55

The Shuttle launch is assumed to occur from KSC on an azimuth of 90°. The orbiter is launched into an initial orbit of 93×182 km with an inclination of 28.5 degrees. Solid rocket booster (SRB) staging is performed at 00:02:05 G.E.T. at an altitude of approximately 140,000 feet. The SRB is jettisoned to impact into the Atlantic Ocean approximately 117 nmi down range. Ascent of the mated orbiter/ET continues to main engine cutoff (MECO) at 00:08:04 G.E.T. The ET separation sequence requires 45 seconds from MECO to OMS ignition. The sequence consists of opening the forward RCS doors, activation of all thrusters, structural/plumbing/electrical orbiter/ET tie releases, translation of the orbiter away from the ET by means of an 8-fps RCS -Z, ΔV, a delta pitch maneuver, and OMS ignition. The dual OMS burn is of 44 seconds duration, sufficient to insert the orbiter into a 93×182 km orbit at 00:09:55 G.E.T.

	PHASE START G.E.T.			PHASE DURATION		
	HR	MIN.	SEC	HR	MIN.	SEC
2. COAST TO APOGEE	00	09	55	00	14	23

Main engine propellants trapped in the orbiter lines after ET separation are dumped—LOX first and then LH_2—commencing shortly after OMS insertion burn ignition. Dumping, purging, and vacuum inerting the lines require 300 seconds. After the main valves have been closed, the APU's are shut down at 00:13:24 G.E.T. The star tracker doors are opened immediately after completing clearing of the main propulsion system (MPS) lines.

Table 2.3-3. Checklist: Shuttle Constraints on Ascent Duration (With Rendezvous)

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) The Space Shuttle trajectory is optimized for intact abort capability. This provides an abort-once-around (AOA) capability at the time the return-to-launch-site (RTLS) capability is lost. (R-1)	Rockwell, 1980 (a): SSD 80-0038 (See also definitions in Section 2.5)
( ) As part of the nominal mission profile mission, the orbiter is always initially injected into a 93×182-km elliptical orbit. At the apogee of this orbit, an OMS circularization burn is performed to place the orbiter in a more stable 182-km circular orbit. Any necessary phasing and orbiter checkout with ground stations are performed at this altitude. (R-1)	Rockwell, 1980 (a): SSD 80-0038 Affects timeline planning
( ) Phasing requirements depend on mission geometry at launch and on the target orbit altitude. (R-1) • See Figure 2.3-4 for the worst-case geometry conditions (maximum phasing time) Considerably shorter phasing times are likely for average conditions.	Rockwell, 1980 (a): SSD 80-0038 Affects timeline planning
( ) After completion of necessary phasing and/or checkout of the orbiter, a go-ahead is given to proceed to the mission altitude. (R-1) • This ascent scenario does not preclude missions where direct ascent-to-mission altitude is performed without going through the intermediate 182-km circular parking orbit. Such missions obviously shorten the time from liftoff to insertion in the mission orbit, but affect the mission planning analysis only in a superficial manner.	Rockwell, 1980 (a): SSD 80-0038 Affects mission planning

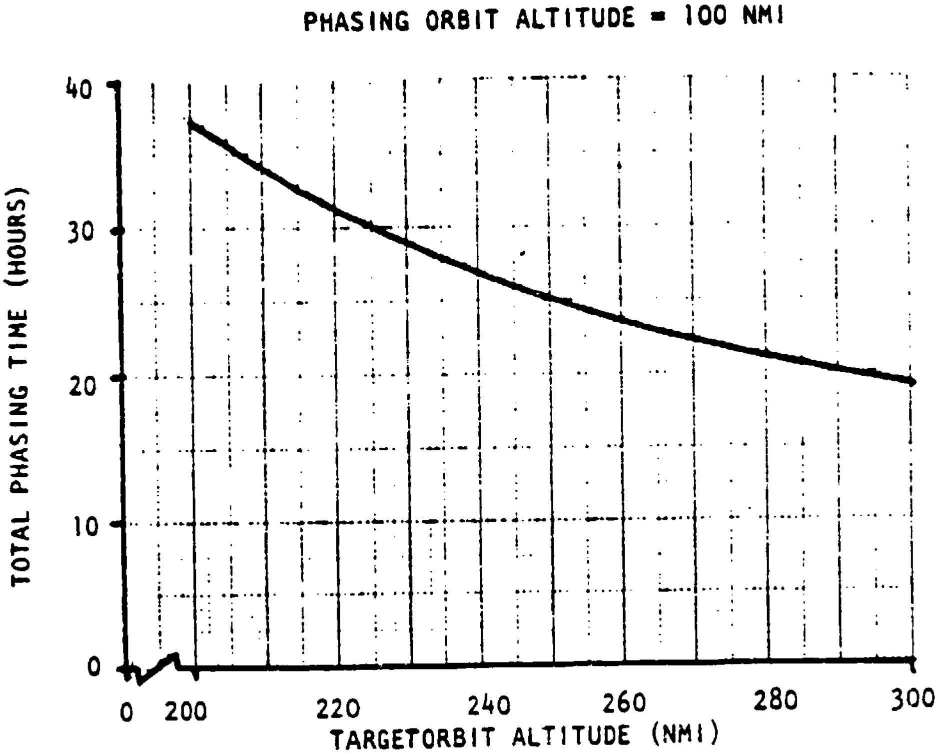


Figure 2.3-6. Total Maximum Phasing Time

Table 2.3-4. Mission Ascent Timeline
(Rendezvous with 12-hr Phasing)

MISSION PHASE SUBPHASE EVENT/OPERATION	MISSION G.E.T.	DURATION	REMARKS & COMMENTS
● POWERED ASCENT			
MATED ASCENT			
LIFTOFF	00 00 00	00 00 00	
CLEAR LAUNCH TOWER	00 00 05	00 00 00	
PERFORM PITCH PROGRAM (TRAJ. SHAPING)	00 00 05	00 00 10	
MAXIMUM DYNAMIC PRESSURE	00 00 54	00 00 00	
WATER BOILER ACTIVATION	00 01 45	00 00 00	30 km ALTITUDE- UP TO THIS POINT NO ACTIVE COOLANT LOOP
SRB TAILOFF	00 01 53	00 00 05	
SRB-ORBITER/ET SEPARATION			
BEGIN ORBITER TVC	00 02 00	00 00 00	
INITIATE SRB STAGING SEQUENCE	00 02 03	00 00 00	
SRB JETTISON	00 02 05	00 00 00	
ORBITER/ET ASCENT			
VERIFY ENGINE THRUST	00 02 06	00 00 04	
REACH 3G	00 07 13	00 00 00	
ORBITER MAIN ENGINE CUTOFF (MECO)	00 08 04	00 00 00	
ORBITER-ET SEPARATION			
OPEN FORWARD RCS COVERS	00 08 04	00 00 22	
ET STRUCTURAL RELEASE	00 08 26	00 00 00	
PERFORM RCS SEPARATION ELEV	00 08 26	00 00 08	
ORBITER MANEUVER (RCS) TO OMS BURN ATTITUDE AND COAST	00 08 34	00 00 15	
● INITIAL ORBIT INSERTION			
PERFORM OMS BURN	00 08 49	00 01 06	
PERFORM MPS LO2 DUMP	00 08 59	00 02 30	
ORBIT INSERTION-OMS SHUTDOWN	00 09 55	00 00 00	INSERTION ORBIT IS 50 x 100 N.MI 93 x 182 km
● COAST TO APOGEE			
DISARM OMS-VERIFY ORBIT INSERTION PARAMETERS	00 09 55	00 00 15	
SELECT ORBITAL RATE MODE	00 10 10	00 00 10	
PURGE MPS ENGINE	00 11 39	00 00 10	
PERFORM MPS LH2 DUMP	00 11 49	00 00 40	
VERIFY MPS DUMP COMPLETE	00 12 29	00 00 10	
SECURE MPS ENGINE	00 12 39	00 00 15	
POSITION MPS ENGINE FOR DEORBIT	00 12 54	00 00 30	
DEACTIVATE APU'S	00 13 24	00 00 15	
ACTIVATE PAYLOAD MONITORING AND CONTROL FUNCTION	00 14 01	00 00 30	

Table 2.3-4. Mission Ascent Timeline (Cont.)
(Rendezvous with 12-hr Phasing)

MISSION PHASE SUBPHASE EVENT/OPERATION	MISSION G.E.T.	DURATION	REMARKS & COMMENTS
INITIALIZE STAR TRACKERS	00 14:30	00 00 10	
SELECT WIDE DEADBAND	00 14:40	00 00 10	3.0 DEG. DEADBAND
ORIENT TO TARGET STAR FIELD	00 15:00	00 03 00	
PERFORM IMU ALIGNMENT	00 18:00	00 15 00	
• CIRCULARIZATION			
ORIENT TO OMS BURN ATTITUDE	00 24 18	00 03 00	
SELECT INERTIAL ATTITUDE HOLD	00 27 18	00 00 10	1.0 DEG. DEADBAND
PERFORM PRETHRUST FUNCTION	00 30 00	00 02 00	
SELECT NARROW DEADBAND	00 33 38	00 00 10	0.5 DEG. DEADBAND
ENABLE ENGINE IGNITI^N CIRCUIT	00 33 48	00 00 10	
PERFORM OMS BURN ΔV = 91 fps = 28 m/s	00 34 18	00 00 56	RESULTANT ORBIT IS 182 x 182 km. 100 x 100 N.MI (Start Phasing) 12 HRS
POST THRUST FUNCTIONS	00 36 00	00 00 30	
NULL RESIDUAL VELOCITY	00 38 00	00 00 10	
DISABLE ENGINE IGNITION CIRCUIT	00 39 00	00 00 15	
SELECT WIDE DEADBAND	00 40 00	00 00 10	
• ORBIT AND P.L. CHECKOUT WITH GROUND STATIONS	00 43 00	03 45 00	
PERFORM IMU ALIGNMENT	03 00 00	03:03:00	
PLANE CHANGE (IF REQUIRED)	03 10 00		
PERFORM IMU ALIGNMENT	10:21:38		
PERFORM PHASING CORRECTION (IF REQUIRED)	11:06:08		
OMS burn attitude maneuver	12:24:k8	00:03:00	

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Table 2.3-4. Mission Ascent Timeline (Cont.)
(Rendezvous with 12-hr Phasing)

MISSION PHASE SUBPHASE EVENT/OPERATION	MISSION G.E.T.	DURATION	REMARKS & COMMENTS
OMS BURN HEIGHT MANEUVER 1 ΔV = 246 fps = 75m/sec	12:34:18	00:05:00*	RESULTANT ORBIT is 100 x 240 N.MI
OMS BURN ATTITUDE MANEUVER	10:09:35	00:03:00	182 x 444 km
1st COELLIPTIC BURN ΔV = 244 fps = 74 m/s	13:19:35	00:05:00*	RESULTANT ORBIT IS 240 x 240 N.MI 444 x 440 km
OMS BURN ATTITUDE MANEUVER	15:31:20	00:03:00	
CORRECTIVE COMBINATION BURN ΔV = 21 fps 6.4 m/sec	15:41:20	00:00:26*	
OMS BURN ATTITUDE MANEUVER	16:08:20	00:03:00	
2nd COELLIPTIC BURN ΔV = 21 fps 6.4 m/sec	16:18:20	00:00:26*	
START RENDEZVOUS PHASE	16:18:50	03:02:45	
TPI ATTITUDE MANEUVER TPI BURN	17:09:50 17:19:50	00:03:00	
1st BRAKING BURN	17:51:20		SIX BREAKING MANEUVERS ΔV ≈ 1.5 m/sec each
LAST BRAKING GATE	17:56:05		30 m from target
HARD DOCK	18:21:35		ORBIT ALTITUDE IS 250 x 250 N.MI 463 x 463 km
START OPERATIONS	19:21:35		

* SINGLE OMS ENGINE BURN

3. IMU ALIGNMENT

An IMU alignment is performed next by maneuvering the orbiter sufficiently to acquire three star sightings. It is estimated that the alignment can be performed in three minutes. A navigation update is performed at any convenient time during this period using the one-way doppler technique and any available STDN station.

	PHASE START G.E.T.			PHASE DURATION		
	HR	MIN.	SEC	HR	MIN.	SEC
4. ORBIT CIRCULARIZATION	00	24	18	00	10	10

At 00:24:18 G.E.T., the orbiter maneuvers to OMS burn attitude. OMS ignition occurs at 00:34:18. The burn is a horizontal in-plane posigrade of 28 m/s, lasting 56 seconds, which circularizes the orbit at 182·182 km.

	PHASE START G.E.T.			PHASE DURATION		
	HR	MIN.	SEC	HR	MIN.	SEC
5. PHASING	00	34	18	11	50	00

After circularization, the orbiter commences a phasing period to reach the proper position for rendezvous initiation. No specific attitude constraints are imposed other than those necessary for thermal conditioning. The payload bay doors are opened 60 minutes after launch, following checkout of orbiter venicle functions and assurance of good ground contact. The ECLSS water boiler is then deactivated and cooling is subsequently provided by radiators mounted on the payload bay doors. An IMU alignment using the star trackers is performed preceding each course correction, and navigation updates are acquired by one-way doppler on a station-available basis. A plane change, if required to correct insertion errors, would be performed at 03:10:00 G.E.T.

An IMU alignment is made and, if necessary, a phasing correction maneuver is performed at 11:06:08 G.E.T.

6. HEIGHT ADJUSTMENT

The orbiter commences to maneuver to the required OMS burn attitude for the first of four height adjustments at 12:29:18 G.E.T., ten minutes prior to OMS ignition. All of these four OMS burns are posigrade, performed heads down with the thrust/sector essentially in plane. A tabulation of these burns follows. All of these burns are made with a single OMS engine.

Identification	Delta-V	Begin Attitude Maneuver	Ignition	OMS Duration
Height	75 m/s	12:24:18	12:34:18	5:00
Coelliptic	74 m/s	13:09:35	13:19:35	5:00
Corrective	6.9 m/s	15:31:20	15:41:20	00:26
Coelliptic	6.4 m/s	16:08:20	16:18:20	00:26

On completion of the listed burns, the orbiter will have transferred from the 182-km phasing orbit to a 463-km circular orbit 18 km less than that of the target satellite. At the last OMS cutoff, the orbiter will be both behind and below the target at a line-of-sight distance of about 152 km.

	PHASE START G.E.T.			PHASE DURATION		
	HR	MIN.	SEC	HR	MIN.	SEC
7. RENDEZVOUS	16	18	20	01	37	45

Immediately subsequent to the last coelliptic OMS burn, the orbiter is maneuvered to an attitude with the payload bay doors pointed forward in the general direction of the target. The rendezvous sensor is deployed, activated, and lock-on the target obtained as soon as possible. The estimated lock-on time is less than two minutes. The target is tracked until the range has decreased to about 50 km and a terminal-phase-initiate (TPI) OMS burn solution derived. At 17:09:50 G.E.T., the orbiter commences to maneuver to the TPI OMS burn attitude. The TPI burn attitude is such that the thrust vector is along the line of sight to the target and is timed to occur when the line of sight is 27 degrees above local horizontal. Subsequent to the TPI OMS burn at 17:19:50, the orbiter is maneuvered to bring the braking axis and optical sight into line with the target inertial attitude hold and narrow deadband is commanded. The theoretical braking delta-V is 9 m/s. Braking is accomplished in six increments of about 1.5 m/s each, with the first at 17:51:20 G.E.T. Cross-axis corrections are applied at the astronaut's discretion and the total delta-V is estimated at 17 m/s. The last braking gate occurs at 17:56:05 G.E.T. and on completion the orbiter will have achieved a stationkeeping position about 30 m from the target.

	PHASE START G.E.T.			PHASE DURATION		
	HR	MIN.	SEC	HR	MIN.	SEC
8. BERTHING	17	56	05	00	25	30

On achieving stationkeeping, the construction project is given a visual inspection to assure that berthing may proceed. The target attitude is such that the berthing port axis is about 45° away from the sun. Berthing aids, such as closed-circuit TV cameras and the manipulator arms, are activated. With the manipulator arm extended, the orbiter next approaches close enough to grapple the target. The fixture/platform assembly will have been stabilized by a system which is initiated by ground command at an appropriate time prior to launch of the orbiter. The attitude control systems of both the target and the orbiter are then commanded free and the manipulator arm is employed to assist in achieving a hard berth at 18:21:35 G.E.T. A special energy absorbing system may be required in the RMS mechanism or the end effector, unless very slow relative velocities can be assured at the time of grapple. However, at this time it is assumed that use of the standard RMS end effector will be feasible. After berthing, the orbiter switches to wide deadband inertial attitude hold. Berthing aids are deactivated and stowed. Berthing has been timed to occur in sunlight, although this does not negate use of floodlights to fill shadows.

The RMS is assumed capable of drawing the orbiter and fixture/platform together in order to perform a hard mechanical latching and electrical connection at the base of the construction fixture.

2.3.3.2 Reference Ascent Profile Without Rendezvous

The ascent timeline without rendezvous is essentially the same (see Table 2.3-5). It is assumed that the orbiter and any payload checkout with ground stations will still be performed in the 182-km circular orbit before the transfer maneuver to the mission orbit is initiated. The orbiter achieves the 463-km circular mission orbit 5 hours, 45 minutes, and 20 seconds after launch.

Table 2.3-5. Mission Ascent Timeline Without Rendezvous

MISSION PHASE SUBPHSE EVENT/OPERATION	MISSION G.E.T	DURATION	REMARKS & COMMENTS
● POWERED ASCENT			
MATED ASCENT			
LIFTOFF	00 00 00	00 00 00	
CLEAR LAUNCH TOWER	00 00 05	00 00 00	LAUNCH TOWER IS ASSUMED TO BE 90 METERS IN HEIGHT
PERFORM PITCH PROGRAM (TRAJ.SHAPING)	00 00 05	00 00 10	
MAXIMUM DYNAMIC PRESSURE	00 00 54	00 00 00	
WATER BOILER ACTIVATION	00 00 45	00 00 00	30 km ALTITUDE UP TO THIS POINT NO ACTIVE COOLANT LOOP
SRB TAILOFF	00 01 53	00 00 05	
SRB-ORBITER/ET SEPARATION			
BEGIN ORBITER TVC	00 02 00	00 00 00	
INITIATE SRB STAGING SEQUENCE	00 02 03	00 00 00	
SRB JETTISON	00 02 05	00 00 00	
ORBITER/ET ASCENT			
VERIFY ENGINE THRUST	00 02 06	00 00 04	
REACH 3G	00 07 13	00 00 00	
ORBITER MAIN ENGINE CUTOFF (MECO)	00 08 04	00 00 00	
ORBITER-ET SEPARATION			
OPEN FORWARD RCS COVERS	00 08 04	00 00 22	
ET STRUCTURAL RELEASE	00 08 26	00 00 00	
PERFORM RCS SEPARATION ELEV	00 08 26	00 00 08	
ORBITER MANEUVER (RCS) TO OMS BURN ATTITUDE AND COAST	00 08 34	00 00 15	
● INITIAL ORBIT INSERTION			
PERFORM OMS BURN	00 08 49	00 01 06	
PERFORM MPS LO2 DUMP	00 08 59	00 02 30	
ORBIT INSERTION-OMS SHUTDOWN	00 09 55	00 00 00	INSERTION ORBIT 50 x 100 N.MI 93 x 182 km.
● COAST TO APOGEE			
DISARM OMS-VERIFY ORBIT INSERTION PARAMETERS	00 09 55	00 00 15	
SELECT ORBITAL RATE MODE	00 10 10	00 00 10	
PURGE MPS ENGINE	00 11 39	00 00 10	
PERFORM MPS LH2 DUMP	00 11 49	00 00 40	
VERIFY MPS DUMP COMPLETE	00 12 29	00 00 10	
SECURE MPS ENGINE	00 12 39	00 00 15	
POSITION MPS ENGINE FOR DEORBIT	00 12 54	00 00 30	
DEACTIVATE APU'S	00 13 24	00 00 15	
ACTIVATE PAYLOAD MONITORING AND CONTROL FUNCTION	00 14 01	00 00 30	

Table 2.3-5. Mission Ascent Timeline Without Rendezvous (Cont.)

MISSION PHASE SUBPHASE EVENT/OPERATION	MISSION G.E.T	DURATION	REMARKS & COMMENTS
INITIALIZE STAR TRACKERS	00:14:30	00 00 10	
SELECT WIDE DEADBAND	00:14:40	00 00 10	3.0 DEG. DEADBAND
ORIENT TO TARGET STAR FIELD	00:15:00	00 03 00	
PERFORM IMU ALIGNMENT	00:18:00	00 15 00	
ACTIVATE RADIATOR COOLING	00:19:10	00 00 00	
● CIRCULARIZATION			
ORIENT TO OMS BURN ATTITUDE	00:24:18	00 03 00	
SELECT INERTIAL ATTITUDE HOLD	00:27:18	00 00 10	1.0 DEG. DEADBAND
PERFORM PRETHRUST FUNCTION	00:30:00	00 02 00	
SELECT NARROW DEADBAND	00:33:38	00 00 10	0.5 DEG. DEADBAND
ENABLE ENGINE IGNITION CIRCUIT	00:33:48	00 00 10	
PERFORM OMS BURN ΔV=91 fps 28 m/s	00:34:18	00 00	RESULTANT ORBIT IS 100 x 100 N.MI 182 x 182 km
POST THRUST FUNCTIONS	00 36:00	00 00 30	
NULL RESIDUAL VELOCITY	00 38:00	00 00 10	
DISABLE ENGINE IGNITION CIRCUIT	00 39:00	00 00 15	
SELECT WIDE DEADBAND	00 40:00	00 00 10	
● ORBIT AND P.L. CHECKOUT WITH GROUND STATIONS	00 43 00	03 45 00	
● ORBITER READY TO CONTINUE	04:28:00		
IMU ALIGNMENT	04:30:00	00:15:00	
OMS BURN ATTITUDE	04:45:00	00:03:00*	
OMS BURN ΔV=264 fps 80 m/s	04:55:00	00:05:20*	RESULTANT ORBIT IS 100 x 250 n.mi 182 x 463 km
OMS BURN ATTITUDE	05:30:20	00:03:00*	
OMS BURN ΔV= 261 fps 80 m/sec	05:40:20	00:05:20*	RESULTANT ORBIT IS 250 x 250 n. mi 463 x 463 km
● ORBITER IN MISSION ORBIT	05:45:20		

*SINGLE OMS ENGINE BURN

2.3.4 Other Rendezvous Concerns

The Shuttle orbiter has the capability to rendezvous with orbiting payloads that are either cooperative or passive. In most cases it will use a multi-orbit and multi-impulse maneuver sequence associated with a parking orbit rendezvous mode, but is is also capable of performing a rendezvous and retrieval in one revolution. The rendezvous limits for cooperative and passive targets are given in Table 2.3-6. RCS propellant estimates for rendezvous and payload retrieval are presented in Table 2.3-7.

Table 2.3-6. Representative Propellant Usage Summary for a 90,718-kg (200-Klb) Orbiter[1]

Operation	Propellant Usage
Worst-case translation, kg/mps (lb/fps)	52 (35)
Three-axis rotational attitude maneuver, kg (lb)	
High rate (1°/sec)	31 (69)
Low rate (1/2°/sec)	15 (35)
Passive thermal control, kg (lb)	6 (13)
Rendezvous terminal phase braking, kg (lb)	717 (1580)
Payload retrieval, kg (lb)	163 (360)
OMS	
Translational, kg/mps (lb/fps)	30 (20)

[1]Usage values for other orbiter weights are directly proportional to orbiter weight ratio, orbiter/90,718 kg (200 Klb).

Table 2.3-7. Rendezvous Radar Limits

Parameter	Target Types: Cooperative[1]	Target Types: Passive[2]
Range limit	560 km (300 nmi) to 30 m (100 ft)	19 km (10.3 nmi) to 30 m (100 ft)
Range rate limit	TBD	TBD
LOS angle limit	±40° (function of range)	±40° (function of range)
LOS angle rate limit		
(1) Acquisition	±4 mr/sec	±4 mr/sec
(2) Tracking	±5°/sec	±5°/sec

[1]Requires transponder on target compatible with radar
[2]Target has an average radar cross section of 1.0 m^2 (10.8 ft^2)

2.4 ORBITER ENTRY AND LANDING CAPABILITIES

The entry and landing phases of Shuttle flights create constraints on space construction projects, primarily in the areas of stowage/support of items returned to earth, timelines, and consumables planning.

The return stowage considerations for space construction projects tend to be unusual since return stowage configurations are likely to be widely different from ascent configurations. For example, different pallets or cradles might be required, and centers of mass of return payloads must still be held within the orbiter constraints. Load requirements are also different from those for ascent.

The planning of space construction processes also may be unusually sensitive to potential contingencies. A potentially large number of piece parts to be removed, in a sequence which is desirable to optimize construction, may pose problems of maintaining an acceptable location for the cargo center of gravity in case of contingency abort. In general, c.g. locations of cargo within the orbiter should be maintained within safe limits at all times (see Section 3.4). Provisions for on-orbit stowage may be required to achieve this goal, and provisions for jettison or rapid stowage must be included to permit re-configuration for safe descent.

Non-productive time required to attain a favorable orbit for entry after finishing construction activity may be a large portion of the total construction flight time, up to a full day. This time may significantly affect requirements for consumables, for crew, and orbiter power.

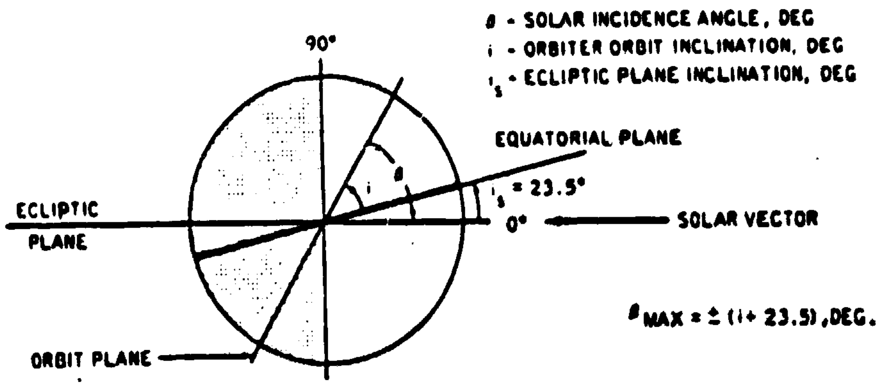
Constraints on Shuttle entry and landing initiation are listed in Table 2.4-1.

Table 2.4-1. Checklist: Constraints on Shuttle Entry and Landing Initiation

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) A thermal conditioning period (barbecue) is required prior to entry. The maximum time is 12 hours. (R-1) • See Figure 2.4-1 of this document.	Rockwell, 1979 (a): SSD 79-0123 Affects planning of schedules, weight of consumables.
( ) A hold period up to 24 hours may be required to bring the orbiter within cross-range of the landing site. (R-1) • Hold period is a function of duration of construction period on orbit and orbit parameters.	Rockwell, 1979 (a): SSD 79-0123 Affects weight and volume of supplies for on-orbit habitability, orbiter operation.

Table 2.4-1. Checklist: Constraints on Shuttle Entry and Landing Initiation (Cont.)

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES IMPLICATIONS
( ) Construction equipment/materials to be returned must be stowed to provide acceptable location of center of gravity. Contingency return conditions shall be considered. (R-1) •See Section 3.4 for specific constraints.	NASA, 1980 (a): JSC 07700, Vol. XIV Affects design of carriers, supports, volume, and size of payload.
( ) Payload bay doors must be closed [allow adequate time for manual backup to close doors (TBD)]. (R-1)	Timeline margins affected.
( ) The direct entry capability of the orbiter is a function of cargo weight and orbit inclination. (R-1) • Figure 2.4-2 presents only the direct entry capability (launch capability is not shown, and may be less than entry capability in some cases). • Figure 2.4-2 is based on no orbit thermal conditioning (barbecue) of the thermal protection system (TPS), except as noted in the right-hand curve.	NASA, 1980 (a): JSC 07700, Vol. XIV Of concern primarily for changeout of payloads. Analysis to determine need for barbecue is required.
( ) Return Payload. The orbiter can deorbit and land with a 14,515 kg (32,000 lb) maximum cargo weight. The overall flight capability will permit cargo weighing more than the nominal 14,515 kg (32,000 lb) design down limit, but no more than 29,484 kg (65,000 lb) to be returned under abort conditions. The circumstances under which they may occur include: • Return to launch site abort • Abort once around • Aborts from orbit operation due to a payload mulfunction. No mission with a landing cargo weight of more than 14,515 kg (32,000 lb) should be planned.	NASA, 1980 (a): JSC 07700, Vol. XIV Affects maximum weight of payload.



θ RANGE DEGREES	ORBITER ORIENTATION	HOLD CAPABILITY HOURS	PREENTRY THERMAL CONDITIONING REQUIREMENTS HOURS
0 TO 60	ANY	≥160	≤12
60 TO 90	A. OTHER THAN 3-AXIS INERTIAL HOLDS	CYCLES OF 6-HOUR HOLDS FOLLOWED BY 3 HOURS OF THERMAL CONDITIONING FOR WORST THERMAL ATTITUDES	≤7
	B. 3-AXIS INERTIAL HOLDS	≥160	≤12

Figure 2.4-1. Orbiter Attitude-Hold Capabilities

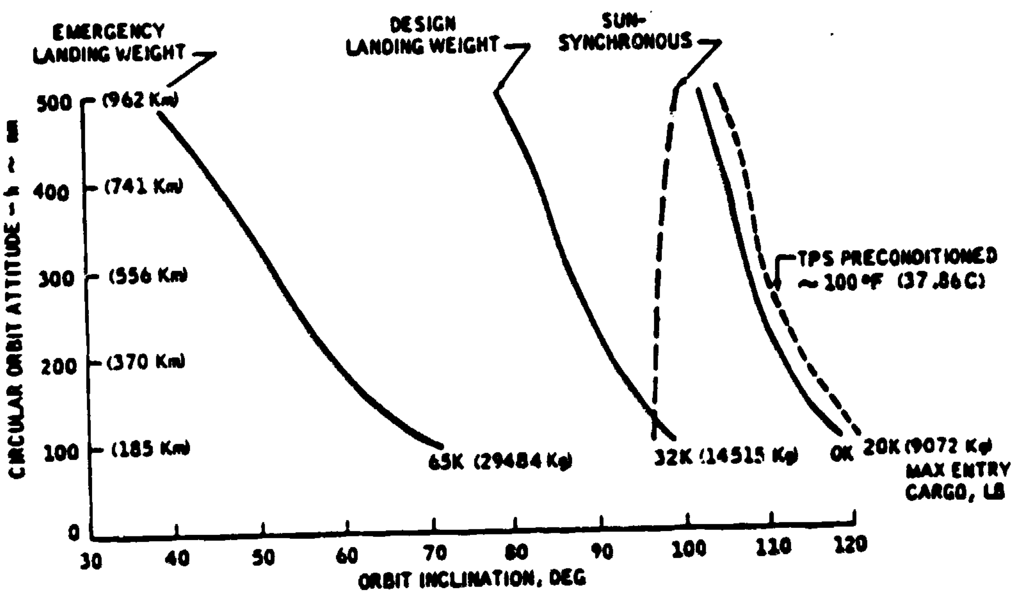


Figure 2.4-2. Preliminary Direct Entry Capability

2.4.1 Typical Descent Timelines

2.4.1.1 Description of Descent Profile

The descent profile is illustrated in Figure 2.4-3. The significant concerns for the designers of large space structures are the duration, loads and environments experienced by the crew and cargo. Duration influences energy requirements and crew supplies and, thus, weight and volume of cargo. Also, duration influences (and may be affected by) crew sleep scheduling on account of constraints on sleep periods discussed in Section 5.2.

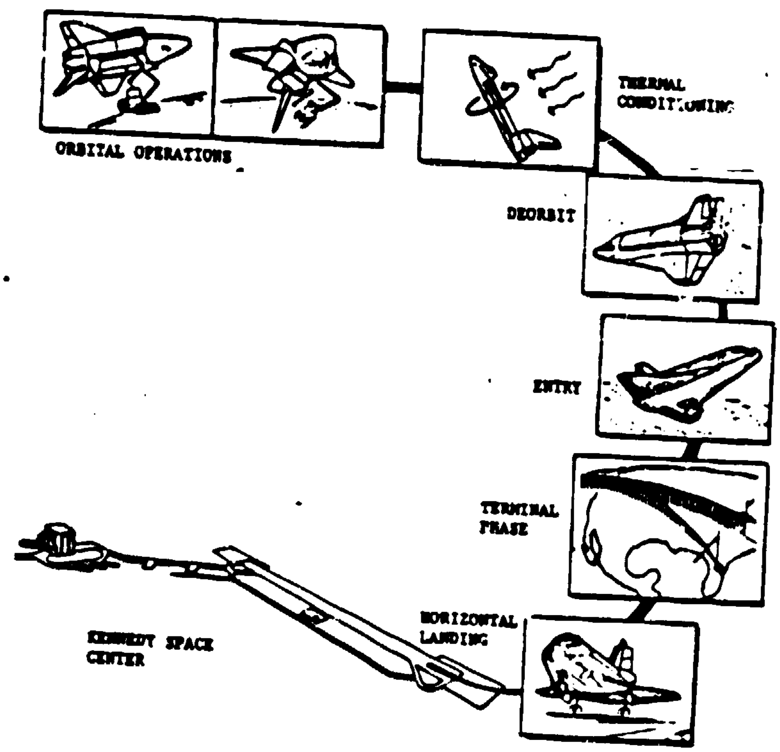


Figure 2.4-3. Descent Profile

2.4.1.2 Entry and Landing Timeline

A typical timeline description is presented in order to point out the general magnitudes of the different phases of entry. This discussion highlights the major effect of the need for thermal conditioning of the orbiter prior to entry.

The maximum time for this phase is 12 hours (Figure 2.4-1). However, considerably shorter time periods have been employed in other analyses.

An entry timeline showing a five-hour thermal conditioning or barbecue phase is shown in Table 2.4-2. Note that the event times assume that completion of rollout is the reference time zero, and that the events are listed in reverse chronological order from top to bottom of the table. All times shown are from completion of the roll at touchdown. Just as for the ascent profile, the events are based on a nominal representative entry profile. A detailed description of the events occurring during this period follows.

Table 2.4-2. Entry Timeline

Complete roll	00:00:00
Rollout	00:02:00
At 16000 ft 4.9 km	00:04:39
At 47000 ft 14 km	00:07:55
At 100000 ft 30 km	00:12:36
Entry 400000 ft 122 km	00:38:07
Entry burn	01:11:35
Maneuver to burn attitude	o1:21:35
IMU Alignment	01:36:35
Thermal Conditioning Mode	06:36:35
IMU Alignment	06:20:35
RCS Maneuver ΔV = 1 m/s	08:06:35
Separation (undocking)	08:26:35

	PHASE START G.E.T.			PHASE DURATION		
	HR	MIN.	SEC	HR	MIN	SEC
1. SEPARATION	08	26	35	01	50	00

After completion of operations, unberthing occurs at 08:26:35 and is accomplished with use of the manipulator arm. After release of the manipulator arm from the construction fixture, a minimal RCS pulse is used to start the orbiter moving away from the platform. At 08:06:35, the orbiter RCS jets are used to generate a posigrade ΔV of 1 m/s. During the ensuing revolution the orbiter first moves ahead, then above, and then drifts behind the platform at distance of about 15 km. During this drifting-away period, visual observation of the platform is maintained. The IMU is aligned using the star trackers at a convenient time during this period, and the state vector is updated using the one-way doppler technique and any available STDN station.

	PHASE START G.E.T.			PHASE DURATION		
	HR	MIN	SEC	HR	MIN	SEC
2. THERMAL CONDITIONING	06	36	35	05	15	00

All loose equipment is secured and stored, and fuel cell water is dumped. If necessary, the orbiter may be operated in a thermal barbecue mode for up to 12 hours at the end of this phase to minimize thermal gradients. A five-hour barbecue mode is shown here. The orbiter is next configured for deorbit including closing of cargo doors, forward RCS doors, and star tracker's protective doors. The APU's are activated for two minutes and the aerosurface controls checked.

	PHASE START G.E.T.			PHASE DURATION		
	HR	MIN	SEC	HR	MIN	SEC
3. DEORBIT	01	21	35	00	43	28

At 01:21:35, the orbiter maneuvers to the deorbit burn attitude, retrograde, heads-down. A single OMS engine is used with ignition at 01:11:35. Burn time is 411 seconds and resultant ΔV is 110 m/s. The orbiter is then maneuvered to a heads-up, nose-first, pitch-up attitude specified for entry interface.

	PHASE START G.E.T.			PHASE DURATION		
	HR	MIN	SEC	HR	MIN	SEC
4. ENTRY	00	38	07	00	30	12

The orbiter passes through 122 km altitude at 00:38:07. The APU's are activated and aero surfaces powered. Throughout the remainder of the entry the GN&C steers to reach the landing site and controls to minimize accelerations and thermal heating. The ammonia boiler is activated as the orbiter passes through 30 km altitude at 00:12:36. Entry is completed as the orbiter passes over the minimum energy point (MEP) at approximately 14 km altitude at 00:07:55.

	PHASE START G.E.T.			PHASE DURATION		
	HR	MIN	SEC	HR	MIN	SEC
5. DESCENT	00	07	55	00	03	16

The orbiter maneuvers as it descends under control based on terminal area energy management (TAEM). It acquires the projection of the final approach trajectory at approximately 4.9 km altitude at 00:04:39.

	PHASE START G.E.T.			PHASE DURATION		
	HR	MIN	SEC	HR	MIN	SEC
6. FINAL APPROACH	00	04	39	00	02	39

Final approach, like the descent, is entirely nominal.

	PHASE START G.E.T.			PHASE DURATION		
	HR	MIN	SEC	HR	MIN	SEC
7. ROLLOUT	00	02	00	00	02	00

Landing rollout is complete at 00:00:00.

	PHASE START G.E.T.			PHASE DURATION		
	HR	MIN	SEC	HR	MIN	SEC
8. POST-LANDING	00	00	00	00	13	00

GSE hookup requires 13 minutes to the point where the fuel cells may be shut down.

2.5 ABORTS

The Shuttle system provides intact abort throughout all mission phases. There are four basic modes: return to launch site (RTLS), abort once around (AOA), abort to orbit (ATO), and abort from orbit. The trajectory profiles are TBD. (R-1).

Abort load limit requirements are included in constraints listed for ascent (Section 2.0) and descent (Section 2.4). These also affect the attachment load limits listed in Section 3.0. Payloads which are not planned for return must be compatible with the environments encountered during STS abort and emergency landing [NASA 1979: NHB 1700.7].

2.6 PRESSURE AND VENTING

With the vents open, the cargo bay pressure closely follows the flight atmospheric pressure. The payload vent sequencing is as follows:

Prelaunch	Closed (vent no. 6 in purge position)
Lift-off (T = 0)	Closed
T + 10 seconds	All open
Orbit insertion	All open
On orbit	All open
Preentry preparation	All closed
Entry (high heat zone)	All closed
Atmospheric (75 000 ±5000 feet (23 ± 1.5 kilometers)) to landing	All open
Postlanding purge	Closed (vent no. 6 in purge position)

During the orbital phase, the cargo bay operates unpressurized. Pressures for other flight phases are shown in Figure 2.6-1.

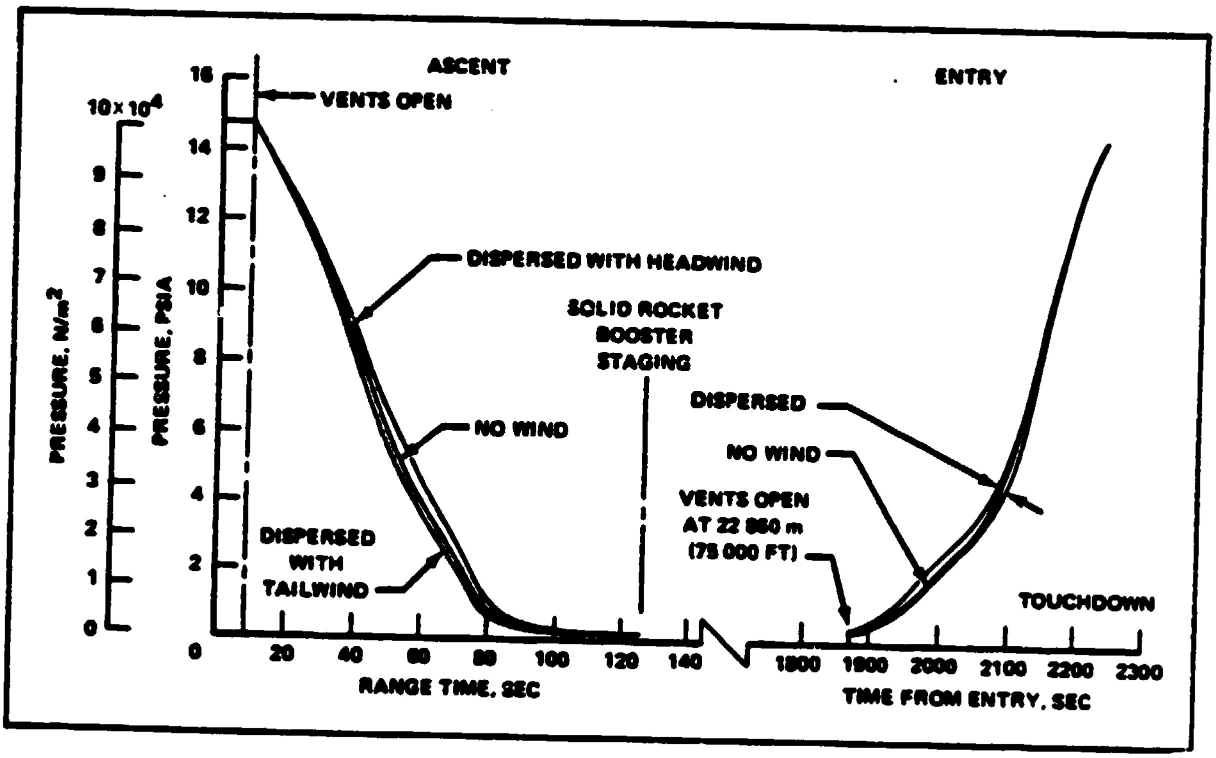


Figure 2.6-1. Cargo Bay Internal Pressure

3.0 PACKAGING CONSTRUCTION EQUIPMENT AND MATERIALS FOR SHUTTLE DELIVERY

This section is concerned primarily with designing for the volumetric constraints and physical interfaces between the Orbiter, the construction equipment and the construction materials as they concern delivery to orbit and return to earth. These requirements may also relate to topics covered in Section 4.4, such as design of the on-orbit attach points between the Orbiter and the construction equipment or the access and feasible pathways for extracting construction equipment and materials from the Orbiter cargo bay. The emphasis in this section is on geometry, mechanical connections, electrical connections, fluid connections and other possible interfaces. Loads and mass properties are also considered.

Sections 3.0, 4.0 and 5.0 together constitute descriptions of the majority of the Shuttle Orbiter features and services which are included in the general subject of Payload Accommodations.

3.1 ENVELOPE AVAILABLE TO PAYLOAD

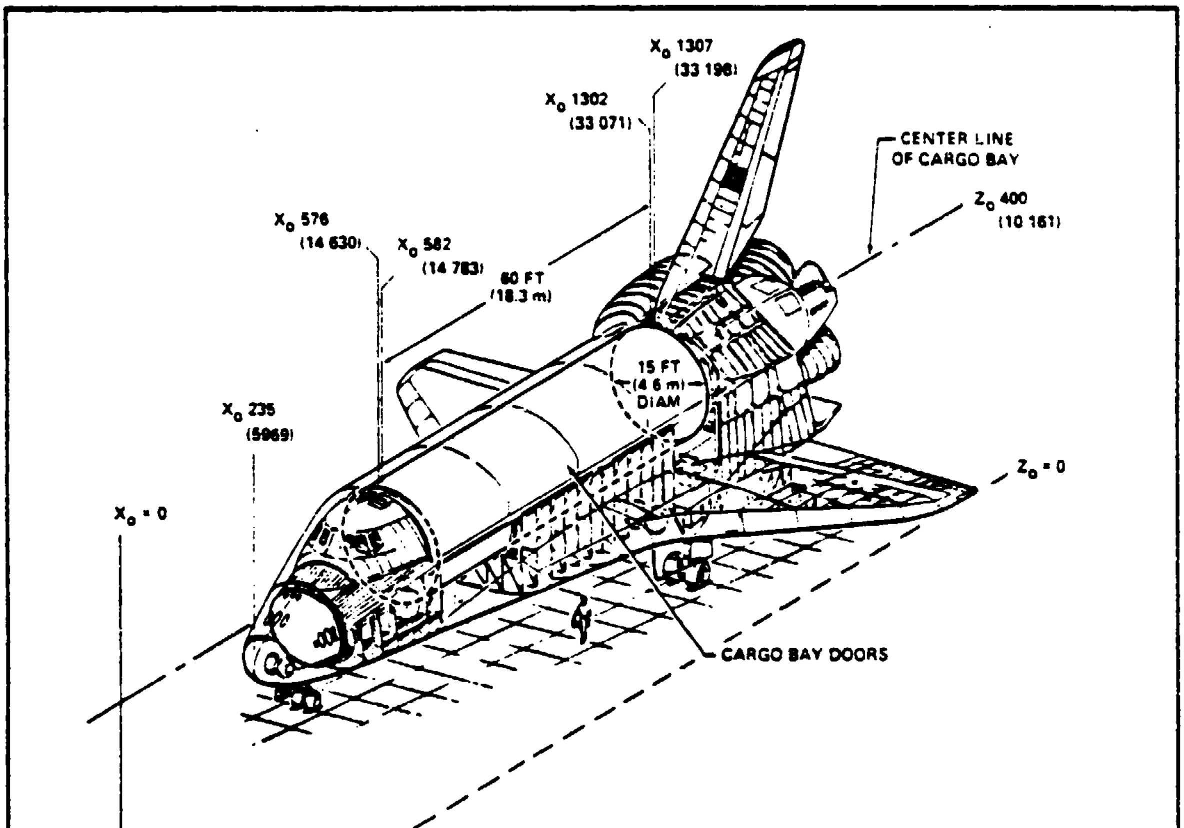
Typically, one of the most critical constraints in space construction project design is the volume available for carrying construction equipment and materials in the Orbiter cargo bay. Since the density of on-orbit construction equipment and materials is usually low, the payload volume permissible per launch (rather than the payload weight) often determines how many Shuttle flights are required. The cargo bay size also limits the maximum size of objects to be carried in each launch. Figures 3.1-1, 3.1-2 and 3.1-3 define the coordinates and major constraints on the spatial envelope available in the cargo bay for space construction equipment, materials, OMS kits, etc. Table 3.1-1 presents a checklist of key constraints relative to this subject, in order to further specify and qualify the volumetric limits.

In addition to the cargo bay, there are smaller volumes available in the crew cabin for appropriate payload-chargeable electronic display and control equipment, crew equipment and supplies and accommodations for additional crew which may be desired for construction or operations related specifically to large space structures. These are briefly summarized in Figures 3.1-6 and 3.1-7. A typical modular stowage box is described in Figure 3.1-8. Constraints are further defined in Table 3.1-2.

Finally, there are designated volumes under the liner of the cargo bay which are available for special payload-chargeable tank kits and plumbing, such as for cryogenic oxygen and hydrogen and gaseous nitrogen. These spaces are designated in Figures 3.1-9 and 3.1-10. Constraints applicable to these volumes are listed in Table 3.1-3.

The above noted volumetric capacities may be effectively reduced by the necessity to carry various standard equipment items (such as OMS kits on docking modules) or to provide access, particularly for contingency EVA. The following sections (Sections 3.1.1 through 3.1.3) identify these potential reductions and provide further information on the expected impacts.

"Caution: Not a controlled document! See appropriate reference documents for current data."



NOTE: Dimensions in parentheses are in millimeters, others are in inches

Figure 3.1-1. Orbiter Coordinate System and Cargo Bay Envelope (The dynamic clearance allowed between the vehicle and the payload at each end is also illustrated)

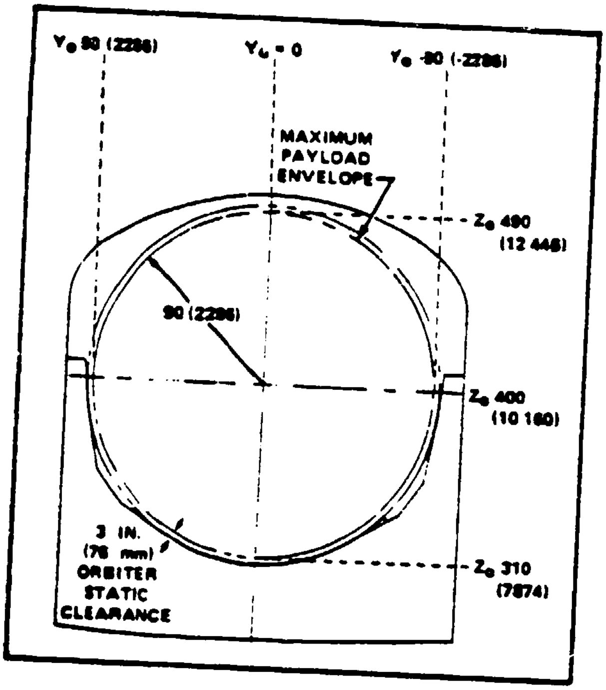
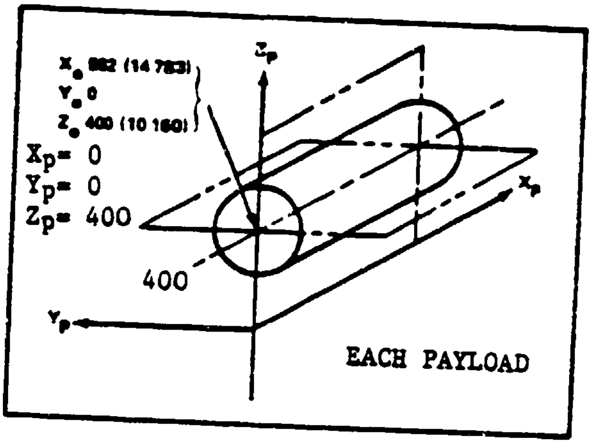


Figure 3.1-2. View of Payload Envelope Looking Aft



Reference: NASA 1980(a), Vol. XIV, p. 2-10

NOTE: X_o, Y_o, Z_o are orbiter coordinates
X_p, Y_p, Z_p are payload coordinates

Figure 3.1-3. Payload Coordinates, Showing Relationship to Orbiter Station on Each Axis

Table 3.1-1. Checklist: Orbiter Constraints on Envelope Available in Payload Bay

( ) GUIDELINES/•QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) The nominal payload clearance envelope in the Shuttle Orbiter payload bay measures 4572 by 18,233 mm (15 ft by 60 ft). This volume is the maximum allowable payload dynamic envelope, including payload deflections. (R-1)	NASA, 1980 (b): Determine largest construction equipment module dimensions (in stowed condition)
• In addition, a 76-mm (3-in.) clearance is provided between the payload envelope and the orbiter structure to account for orbiter deflections.	
• Payload support trunnions protrude outside the nominal envelope to support points on longeron fittings or on keel fittings (See Section 3.2).	
• Electrical and fluid interconnections penetrate the nominal envelope to orbiter structure attach points (Sections 3.8 & 3.9).	
• If a starboard side RMS arm is not installed, the space allocated for the arm stowage above the longeron may be available for payload stowage. (See PIDA, Section 4.6)	
( ) There is a small potential intrusion into the nominal 3" clearance envelope between payload and orbiter due to the Ku-band antenna stowage.	Rockwell 1980 (h) In-House Briefing Generally not a problem when EVA access is provided to MMU.
• Detail design study in progress at time of this document preparation. Figure 3.1-4 illustrates the proposed new stowage location concept (cross-section).	
• Detailed geometric and dynamic analysis is recommended where questions of critical clearance may exist.	

Table 3.1-1. Checklist: Orbiter Constraints on Envelope Available in Payload Bay (Cont.)

( ) GUIDELINES/•QUALIFICATIONS	REFERENCES IMPLICATIONS
( ) Cargo bay forward bulkhead purge ducts create an intrusion into the clearance envelope for payloads. • See Figure 3.1-5	NASA 1980 (a): JSC-07700, Vol. XIV (ICD 2-19001) Generally not a problem when berthing/docking module or EVA access is provided.
( ) During ascent and/or descent, the cargo bay liner may balloon into the cargo bay clearance envelope, exerting a pressure of less than .1 psi. • Maximum protrusion is 4.6 into the cargo envelope (radius of 85.4 inches from cargo bay centerline). • When required, special hardware will be provided by the orbiter for restraint of the liner for sensitive cargo.	NASA 1980 (a): JSC-07700, Vol. XIV (ICD 2-19001) See reference for details of areas affected.
( ) The cargo bay is covered with doors that open to expose the entire length and full width of the bay. (R-1) • Items mounted on the payload bay longeron must not obstruct this access.	NASA, 1980 (b): Design affected: hinge mechanisms on the mounting may be required to permit outboard rotation.

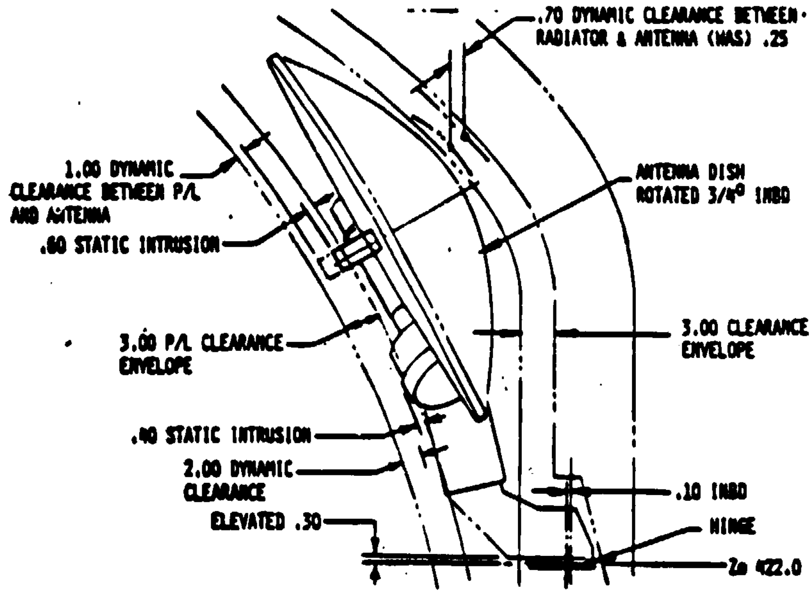


Figure 3.1-4. Payload Clearance Envelope Intrusions of Ku-Band Antenna

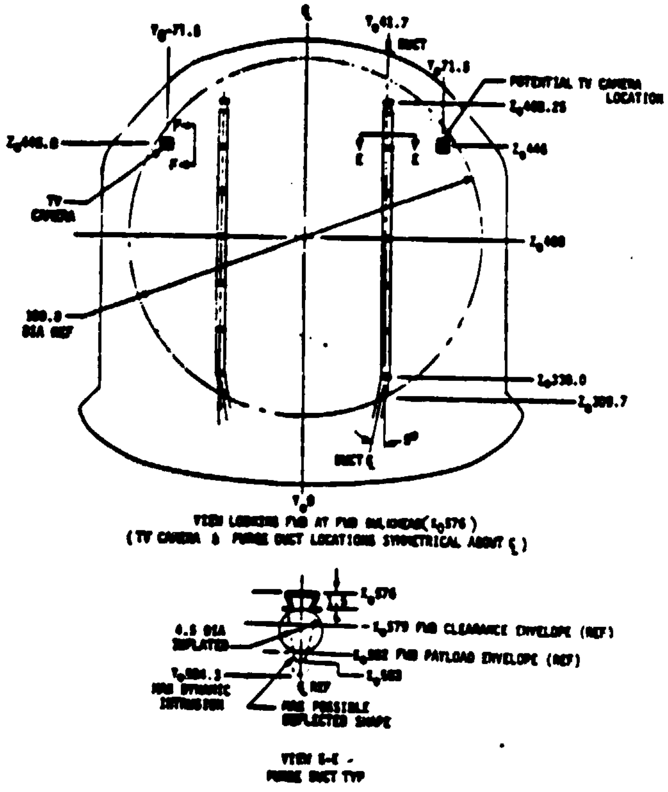
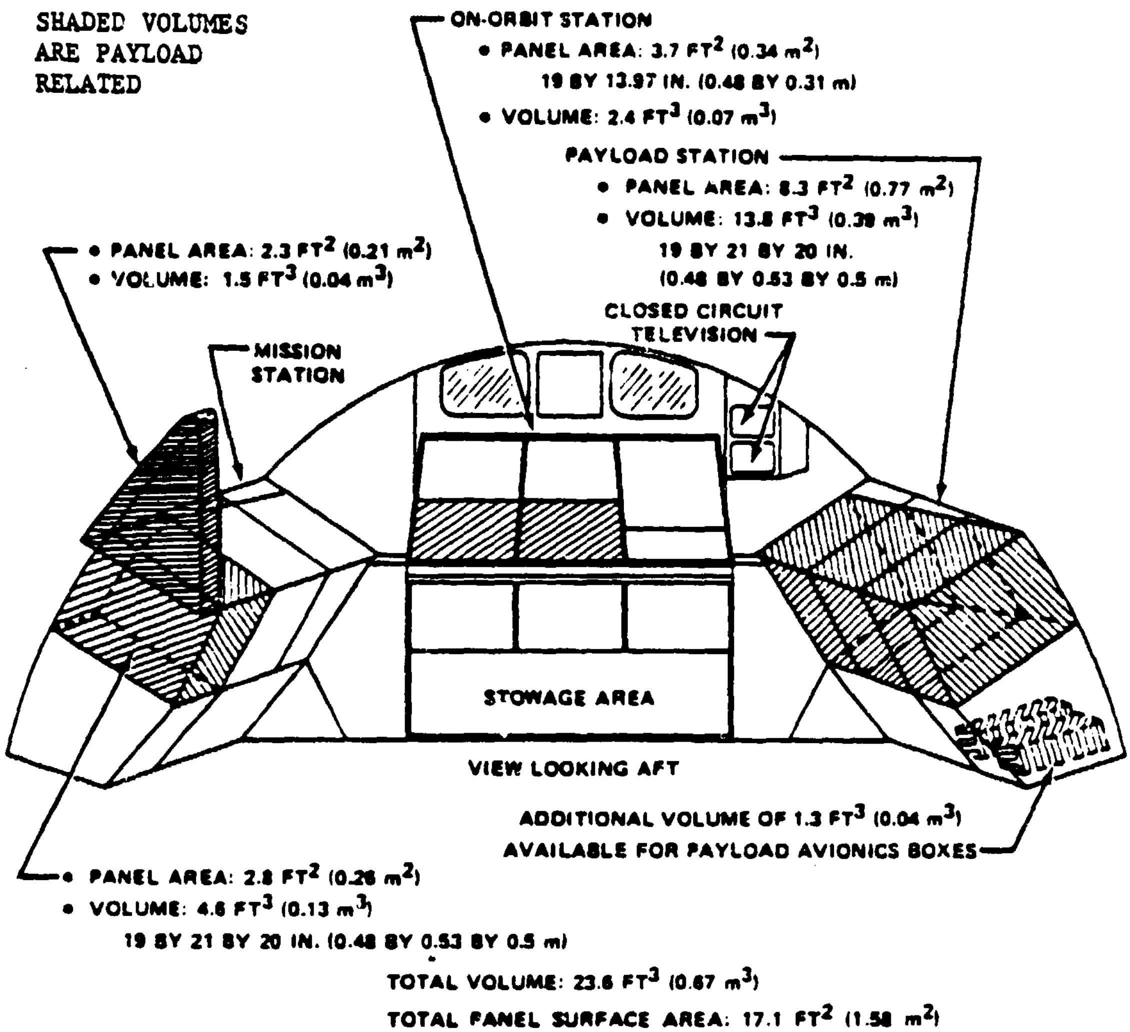


Figure 3.1-5. Payload Clearance Envelope Intrusions of Purge Ducts on Forward Bulkhead



NOTE: The payload space on the aft flight deck is intended primarily for control panels and stowage.

Also see: NASA 1980(a), ICD 2-19001, p. 3-10 and 3-15 for potential updated information.

Figure 3.1-6. Area and Volume Available for Payload Equipment or Controls in the Orbiter Aft Flight Deck

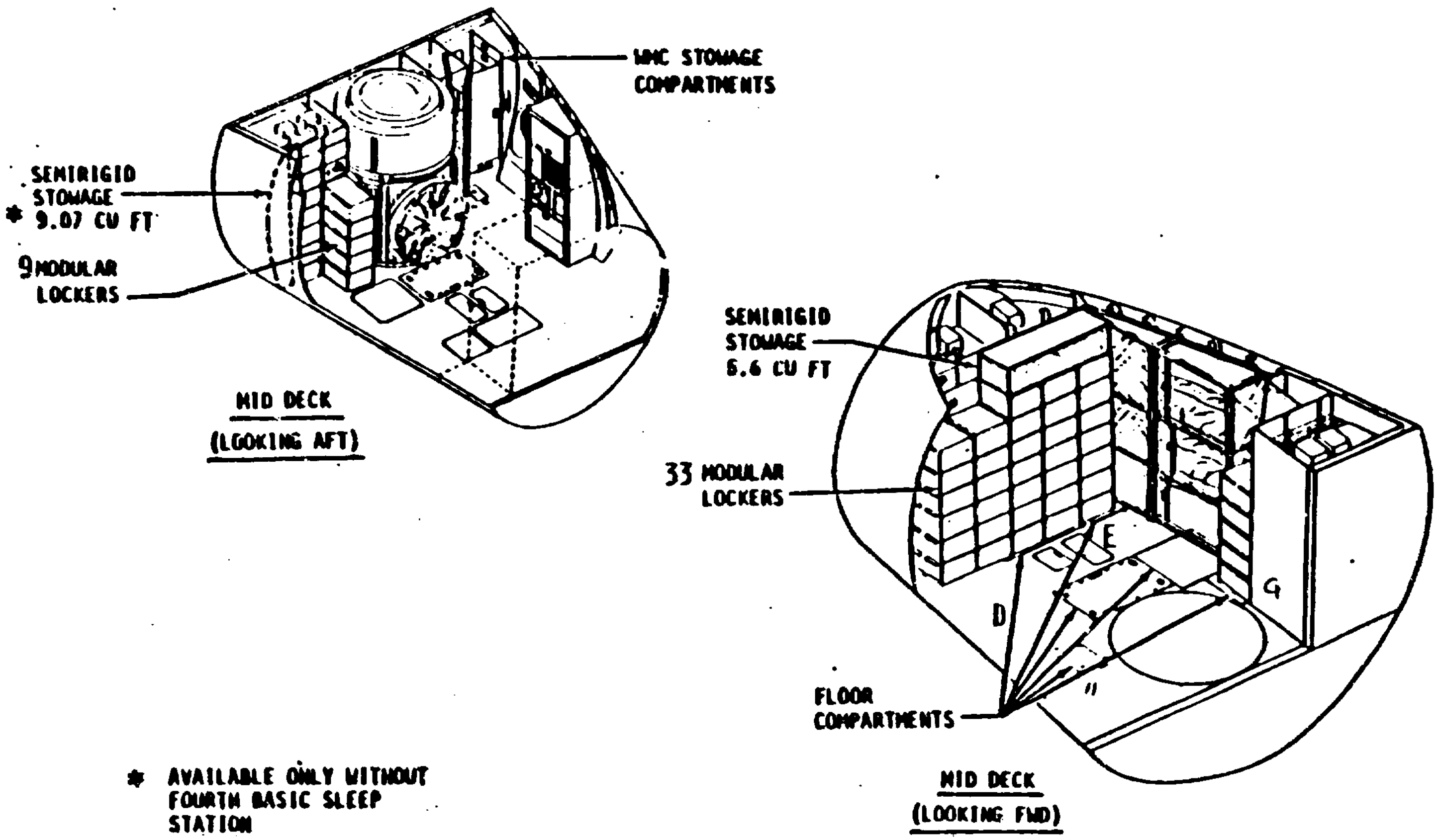


Figure 3.1-7. Stowage Volume Available for Crew Supplies, Including Additional Construction/Operations Crew Seating in the Orbiter Mid-Deck

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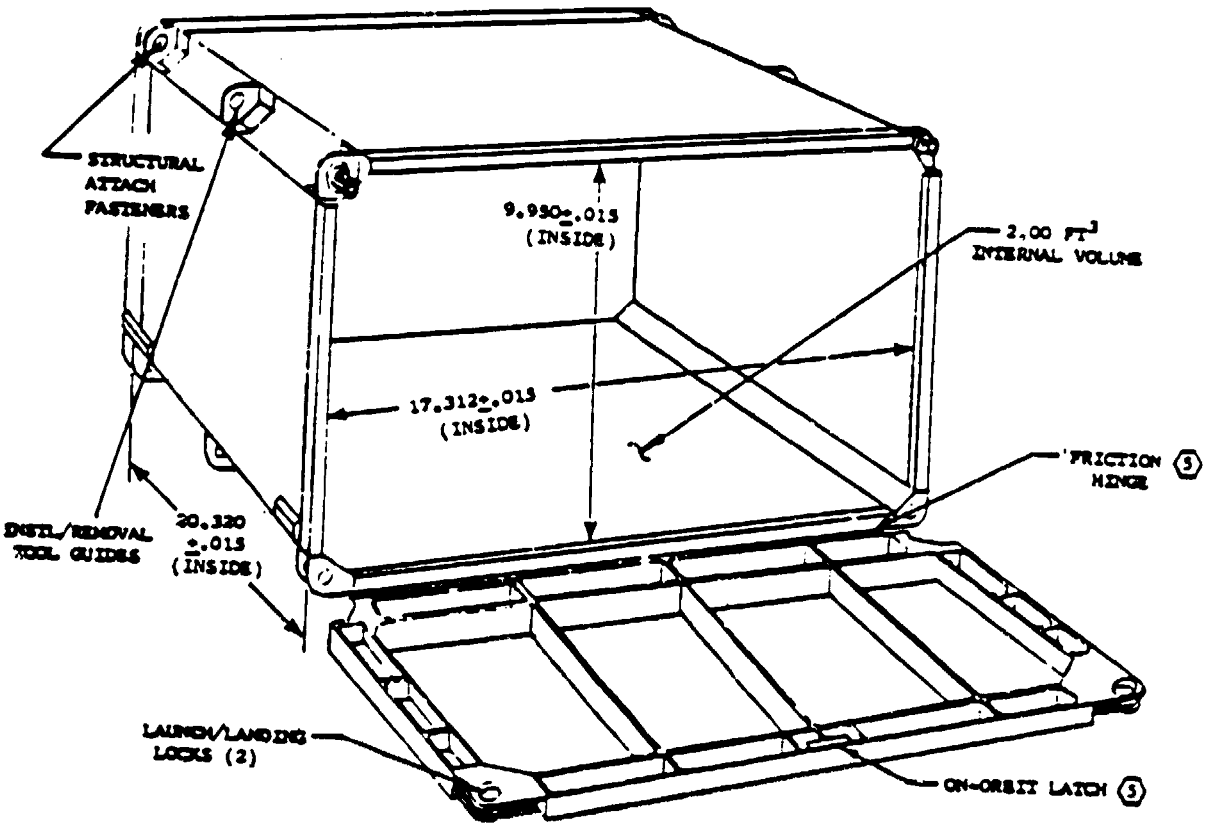


NOTES

1. STOWAGE CONTAINER HAS A DESIGN DENSITY OF 30 LBS/FT^3
2. LOCKER CONTENTS MUST BE PACKAGED TO PREVENT LOAD SHIFTING
3. INTERNAL LOCKER RESTRAINTS WILL NOT BE DESIGNED FOR CRASH CONDITIONS
4. DOOR IS FLUSH WITH BOTTOM OF LOCKER WHEN OPENED 90 DEGREES AND CAN OPEN 180 DEGREES (STRAIGHT DOWN)

(5) DOOR HAS FRICTION HINGE FOR ZERO-G OPERATION AND A MAGNETIC LATCH FOR TEMPORARY CLOSURE OF DOOR

Figure 3.1-8. Modular Stowage Unit for Crew Cabin Mid Deck

Table 3.1-2. Checklist: Orbiter Constraints on Envelope Available in Crew Cabin Mid-Deck

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) A minimum volume of 89 cubic feet shall be dedicated to cargo use in each mission for potential loose equipment. (R-1) • The volumes available are negotiated on a mission by mission basis. • Actual volume available may be less.	NASA 1980 (a): JSC-07700, Vol. XIV, Attachment 1
( ) All of the potential loose-equipment stowage locations and volumes for cargo use are defined and controlled in ICD-3-0027-03. (R-1)	NASA 1980 (a): (See above) Rockwell 1980: ICD-3-0027-03
( ) A limited number of identical modular stowage units are included in the above total volume dedicated to cargo use in each mission. (R-1) • Volume occupied by box structure is subtracted from available volume.	NASA 1980 (a): (See above) See Figure 3.1-8 for modular stowage unit dimensions.

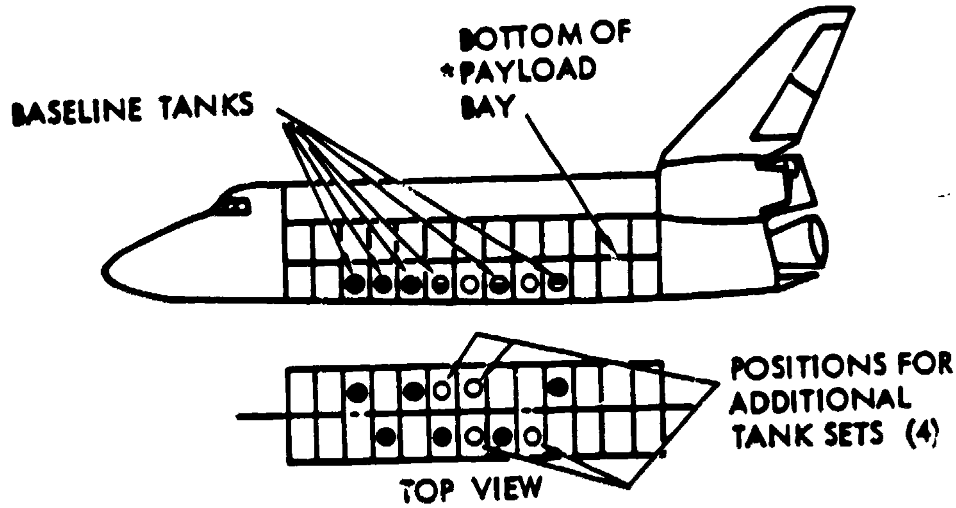


Figure 3.1-9. Positions for Additional Cryogenic Tank Sets Under Cargo Bay Liner

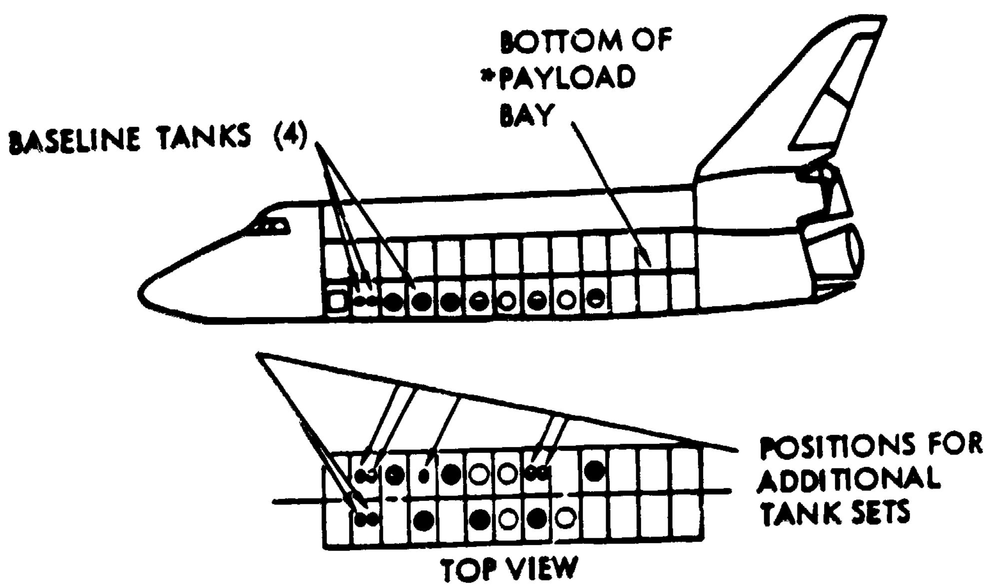


Figure 3.1-10. Stowage Volume for Additional Nitrogen Tanks Under Cargo Bay Liner

*NOTE: Different Shuttle documents may refer to the same space as either "cargo bay" or "payload bay."

Table 3.1-3. Checklist: Constraints on Nitrogen Stowage Under Payload Bay Liner

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) Baseline stowage provisions for nitrogen stowage provides four (4) tanks under the cargo bay liner. (R-1)	Rockwell 1980 (b): SSV80-1
( ) The volume occupied by one cryogenic hydrogen or oxygen tank under the payload bay liner can be used to install four N_2 tanks. (R-4)	Trade studies are required to optimize conflicting volume usage options.
( ) Additional nitrogen stowage could be provided by use of cryogenic stowage tanks. (R-4) . Cryogenic N_2 tank designs are not currently a standard Shuttle inventory capability.	Requires new engineering, test, qualification, and manufacturing of tank sets, as well as ground installation time.
( ) Nitrogen tank installations under payload bay liner are limited by available volume, taking into account other tank installation requirements and ground access requirements. (R-4)	
( ) Additional stowage is available for 12 to 14 nitrogen tanks under the cargo bay liner, assuming a passageway is provided for ground access. (R-4)	
( ) Additional plumbing and brackets must be designed and manufactured to provide for added N_2 stowage beyond the baseline four tanks under the payload bay liner. (R-4) . If required, a dedicated orbiter should be assigned for such special usage.	Requires new engineering, test, qualification and manufacturing as well as ground installation time.

3.1.1 Cargo Volume Reductions Due to OMS Kits

Figure 3.1-11 outlines the significant volume occupied by one or more orbital maneuvering system (OMS) ΔV kits. Such kits may be required by a particular mission to achieve a desired orbit with sufficient weight to meet the stated mission objectives. The integral orbital maneuver subsystem (OMS)

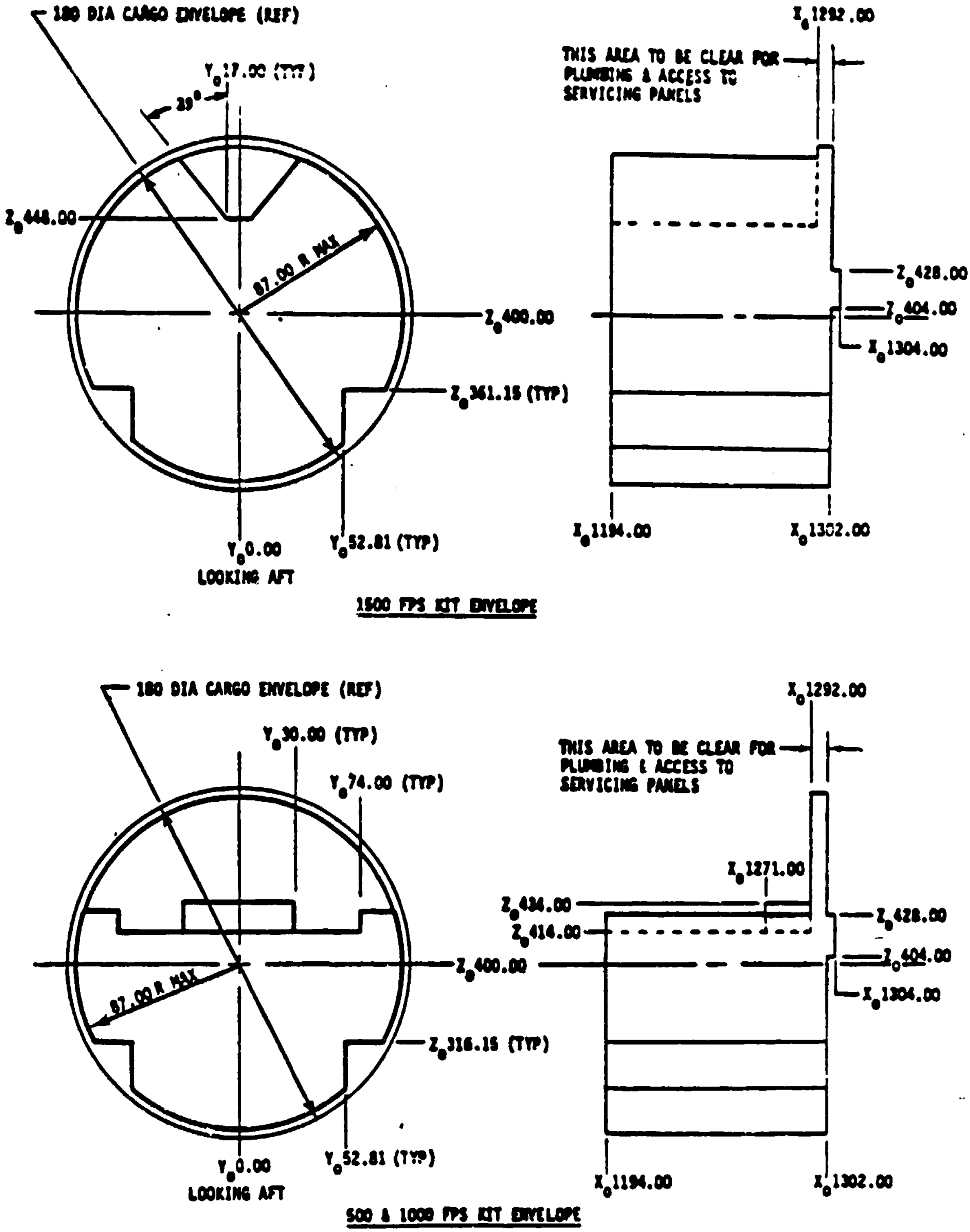


Figure 3.1-11. Payload Bay Volume Required for OMS Kits

propellant tanks aboard the orbiter provide approximately 1000 feet per second of velocity on-orbit. Auxiliary propellant kits must be employed whenever mission velocity requirements for the OMS exceed this value. Up to three sets of auxiliary tankage, each providing an additional delta V capability of 500 feet per second can be added to achieve a total delta V capability of 2500 feet per second. The auxiliary tankage, located in the cargo bay, utilizes the same type propellant tanks, helium bottles, and pressurization system components as the pods. The weight and the nominal added delta V are shown in the table below as a function of the number of kits employed. The data presented are the latest values of the baseline design. (See Section 2.3 for further discussion of these constraints).

Table 3.1-4. Weight and ΔV Capability of OMS Kits on Orbiter

	TOTAL ADDED WEIGHT *		NOMINAL ADDED ΔV (fps)
NUMBER OF KITS	DRY (lb)	FILLED** (lb)	
1	2978	15,380	500
2	3955	28,723	1000
3	5275	42,408	1500

*Does not include fittings (additional 397 lb)
**Includes all loaded propellants and gases

The installation and removal of OMS V kits shall be possible without impact to, or by, cargo elements installed forward of X_o 1191.

Data Source: NASA 1980(a), ICD 2-19001

3.1.2 Cargo Bay Volumes Required for Access and Miscellaneous Optional Equipment

In addition to possible impacts of OMS ΔV kits previously described, there are other potential volumetric requirements related to optional equipment which limit full use of the nominal orbiter payload bay volume for payload. These potential constraints are listed in Table 3.1-4.

Table 3.1-5. Checklist: Shuttle Constraints on Usage of Cargo Bay Stowage Volume due to Access and Miscellaneous Optional Equipment

( )	CONSTRAINTS/QUALIFICATIONS	REFERENCES / IMPLICATIONS
( )	The usable payload envelope is limited by access needed and items of supporting subsystems in the cargo bay which are chargeable to the payload volume. (R-1) Potentially applicable requirements are listed below. The following notes apply to all dynamic/thermal excursion data: NOTES: (1) Dynamic/thermal intrusions include orbiter deflections. (2) Development of a general set of orbiter deflection data, encompassing the complete cargo bay, is not currently planned. Such data will only be developed for specific flight manifested payloads, and will be maintained as reference information for future payload planning.	. NASA, 1980 (a): JSC-07700, Vol. XIV, (ICD 2-19001) . NASA, 1980 (b): STS User Handbook
( )	Access must be provided for contingency EVA on orbit. (R-1) . Provisions to remove or jettison payload items may be incorporated to provide such access. (R-5) . See Sections 4.8.1.1 through 4.8.1.3 for volumes associated with airlocks, tunnel adapters and docking modules plus associated access.	NASA 1980 (a): JSC-07700, Vol. XIV In general, airlocks and tunnels in cargo bay should be avoided, in order to provide for maximum payload stowage volume.
( )	When one or more Manned Maneuvering Units are installed, access must be provided in the cargo bay for donning and checkout of MMU as well as for stowage. (R-1) . See Section 4.8.2.2. . Access to MMU is not required until orbital operations begin.	May affect decision on usage of MMU, stowage, sequence of operations. Analysis of access sequence is required.

Table 3.1-5. Checklist: Shuttle Constraints on Usage of Cargo Bay Stowage Volume Due to Access and Miscellaneous Optional Equipment (Cont'd)

( ) CONSTRAINTS/QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) Stowage should consider access requirements to tool stowage (Modular Equipment Stowage Assembly). • Access to tool stowage may be required for closing cargo bay doors (contingency requirement). • See Section 4.8.2.4.	Tool stowage is (TBD) for operational vehicles.
( ) CCTV cameras, when mounted in the orbiter cargo bay, require a small intrusion into the nominal payload envelope volume. (R-1) • See Figure 3.1-12 and Figure 3.1-13. • Additional volume must be allowed if viewing through the TV cameras is required prior to removal of adjacent stowed equipment. • See Section 4.9.2 for TV viewing angles.	NASA 1980 (a): JSC-07700, Vol. XIV (ICD 2-19001) Requires analysis of clearance requirements for specific payloads.
( ) Handholds which are mounted on the aft bulkhead of the cargo bay require consideration of a dynamic intrusion into the payload clearance envelope. • See Figure 3.1-14. • Handholds are used to attach EVA body restraint for contingency access to cargo bay door latches. • OMS kit prevents installation of some handholds.	NASA 1980 (a): JSC-07700, Vol. XIV (ICD 2-19001)
( ) If a keel bridge fitting is required over the orbiter wing box (between Xo 1193.0 and Xo 1249) there is a resulting static protrusion of .4 inch into the payload envelope and a 5.3 inch dynamic/thermal protrusion into the payload envelope. • See Figure 3.1-15.	NASA 1980 (a): JSC-07700, Vol. XIV (ICD 2-19001) Analysis of clearance required for actual payload package. Avoid usage if possible in design of space-critical stowage.

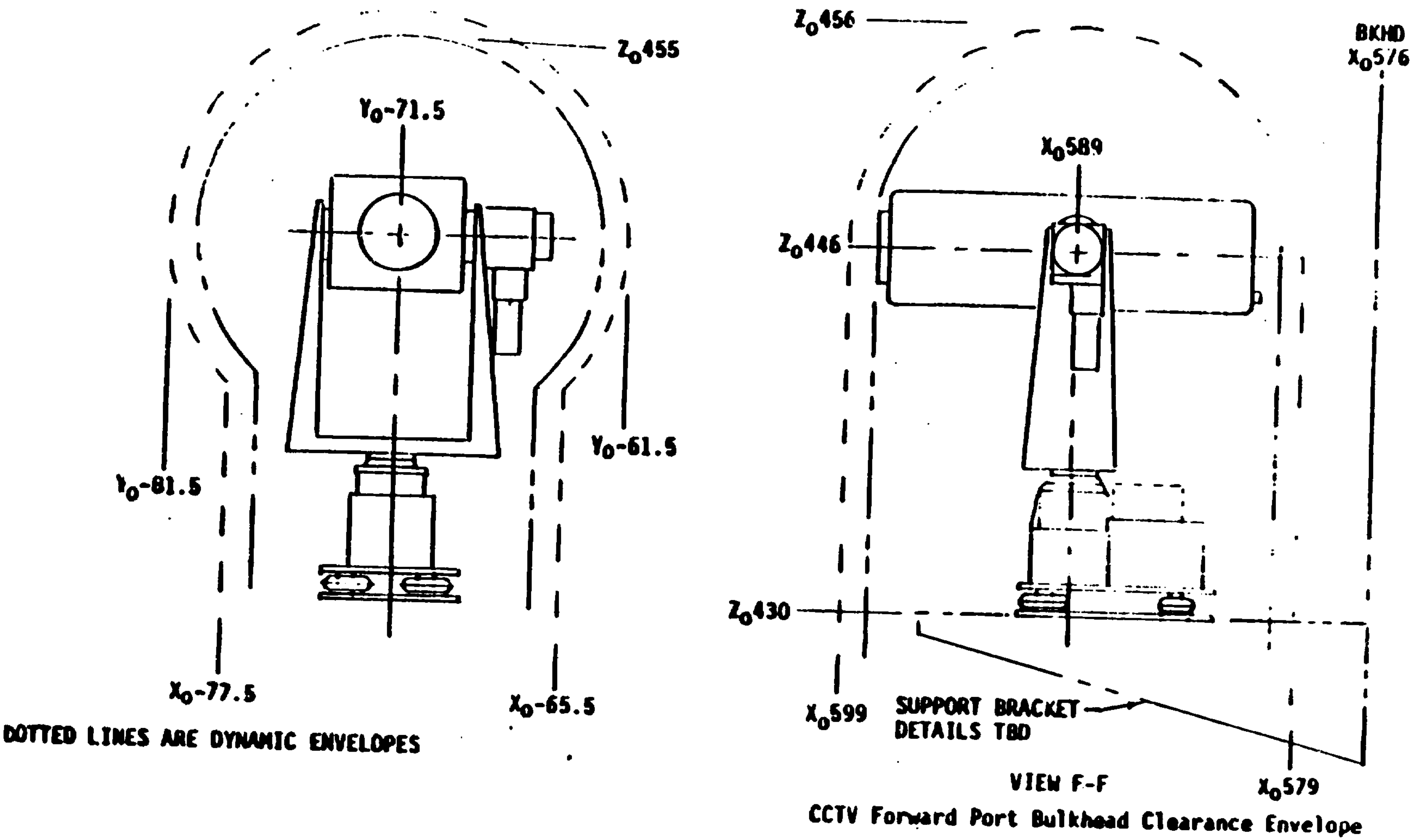
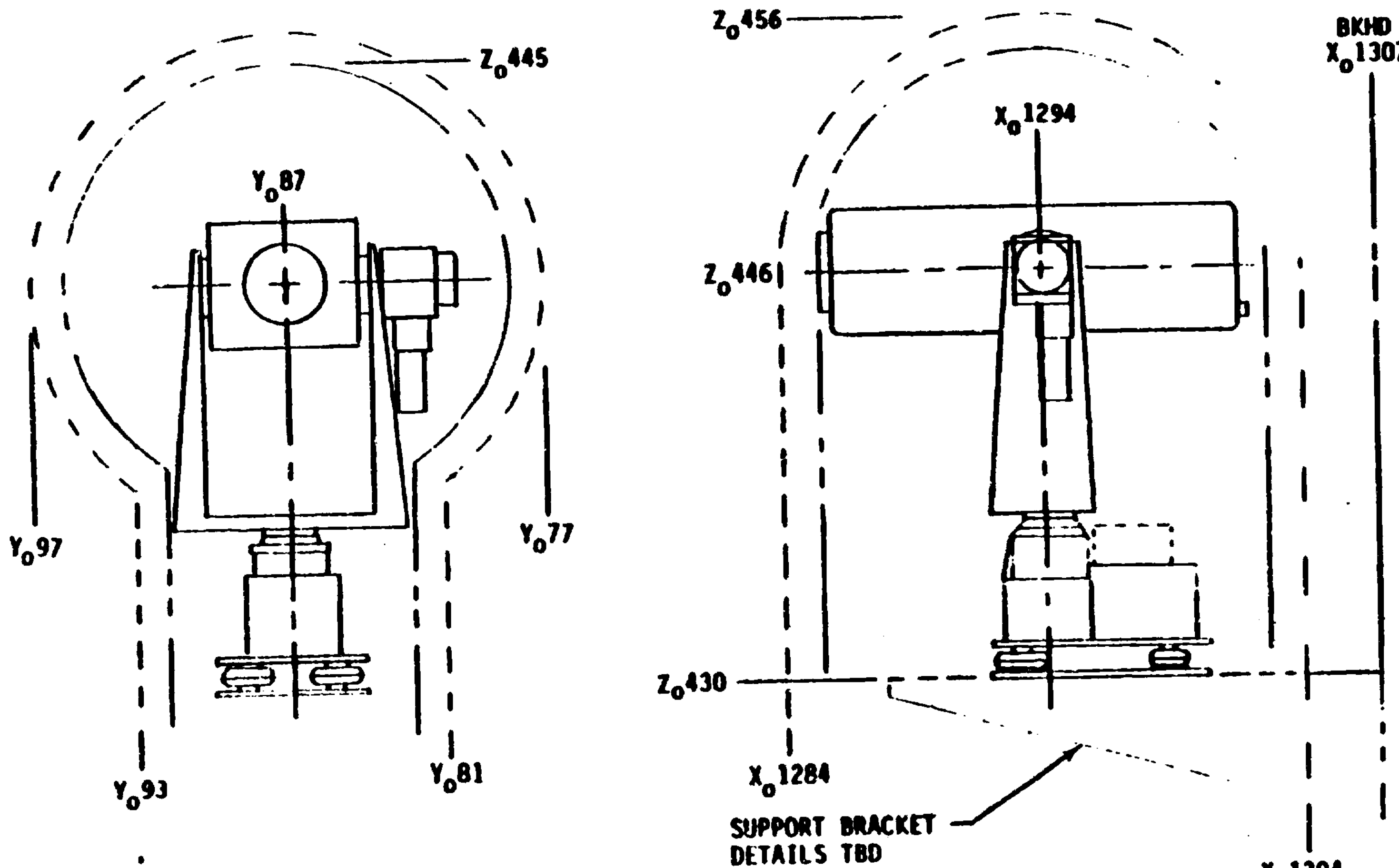


Figure 3.1-12. Payload Clearance Envelope Intrusions of Forward TV Cameras in Payload Bay



DOTTED LINES ARE DYNAMIC ENVELOPES

CCTV Aft Starboard Bulkhead Clearance Envelope

Figure 3.1-13. Payload Clearance Envelope Intrusions of Aft TV Camera in Payload Bay

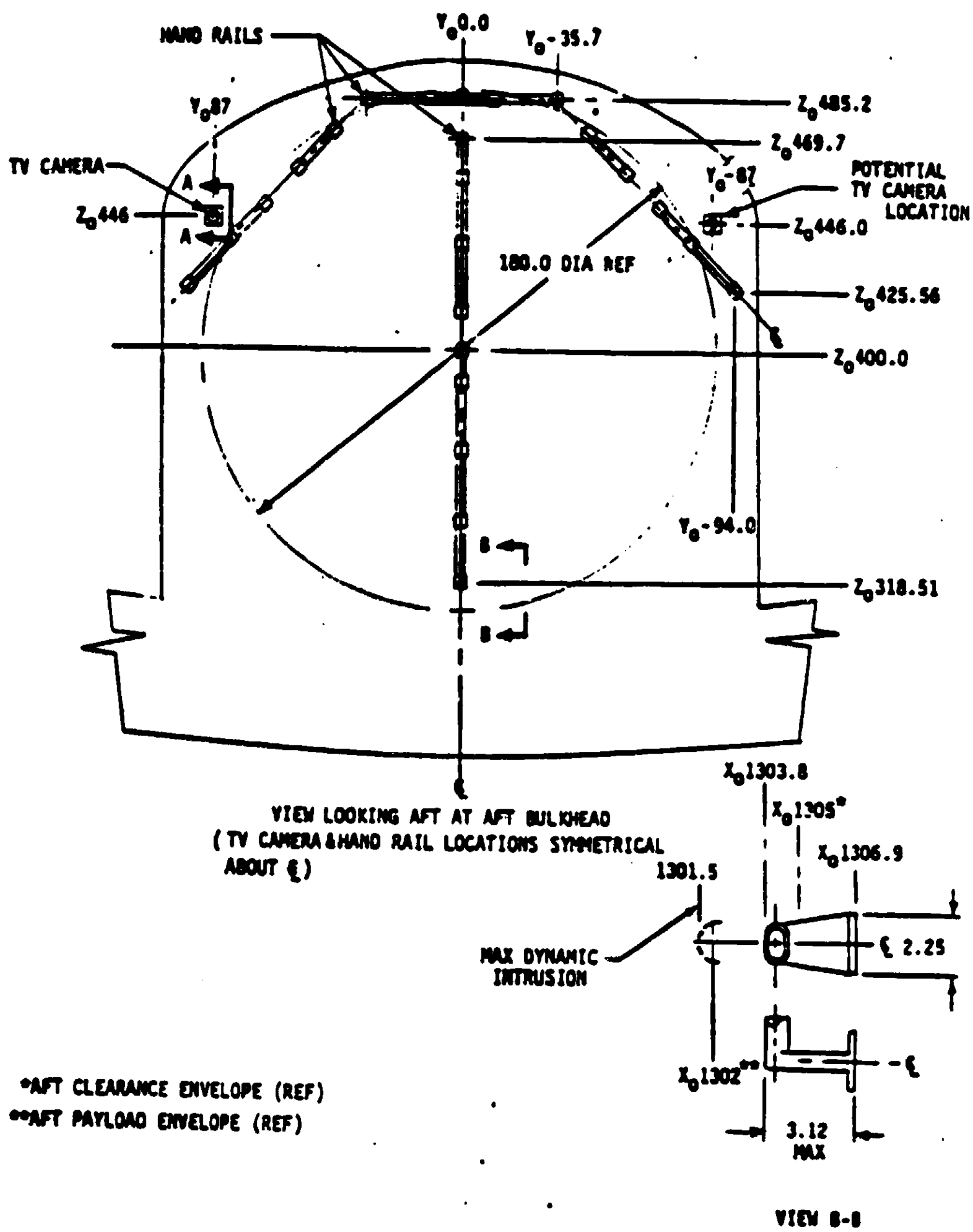
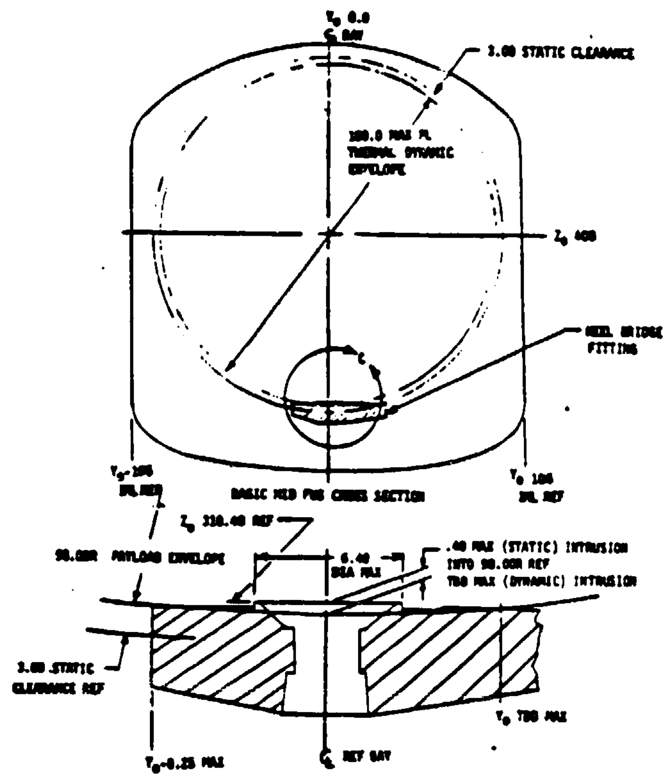


Figure 3.1-14. Payload Clearance Envelope Intrusions of Handholds on Aft Bulkhead of Cargo Bay



Detail C

Keel Bridge Fitting Over Wing Box Typ between X_o 1198.0 and X_o 1249.0

Figure 3 1-15. Payload Clearance Envelope Intrusions of Keel Fitting over Wing Box

3.2 SHUTTLE ORBITER/PAYLOAD MECHANICAL ATTACHMENT LOCATIONS AND FITTINGS

3.2.1 Attachment Locations

The orbiter provides structural support attachment points for carrier/payloads along the length of the cargo bay approximately as indicated on Figure 3.2-1. The attachment points are located at 3.933-inch increments along the main longerons and along the bottom centerline of the cargo bay on bridge fittings spanning adjacent fuselage frames. Of the potential 172 attach points on the longerons, 45 are not available due to the proximity of orbiter hardware; 127 may be used for carrier/payload attachment, and of these, 111 may be used for deployable payloads; 104 attach points are available along the keel, any of which may be used for deployable payloads. Longeron and keel attach points normally utilize the attach fittings described in Figure 3.2-2.

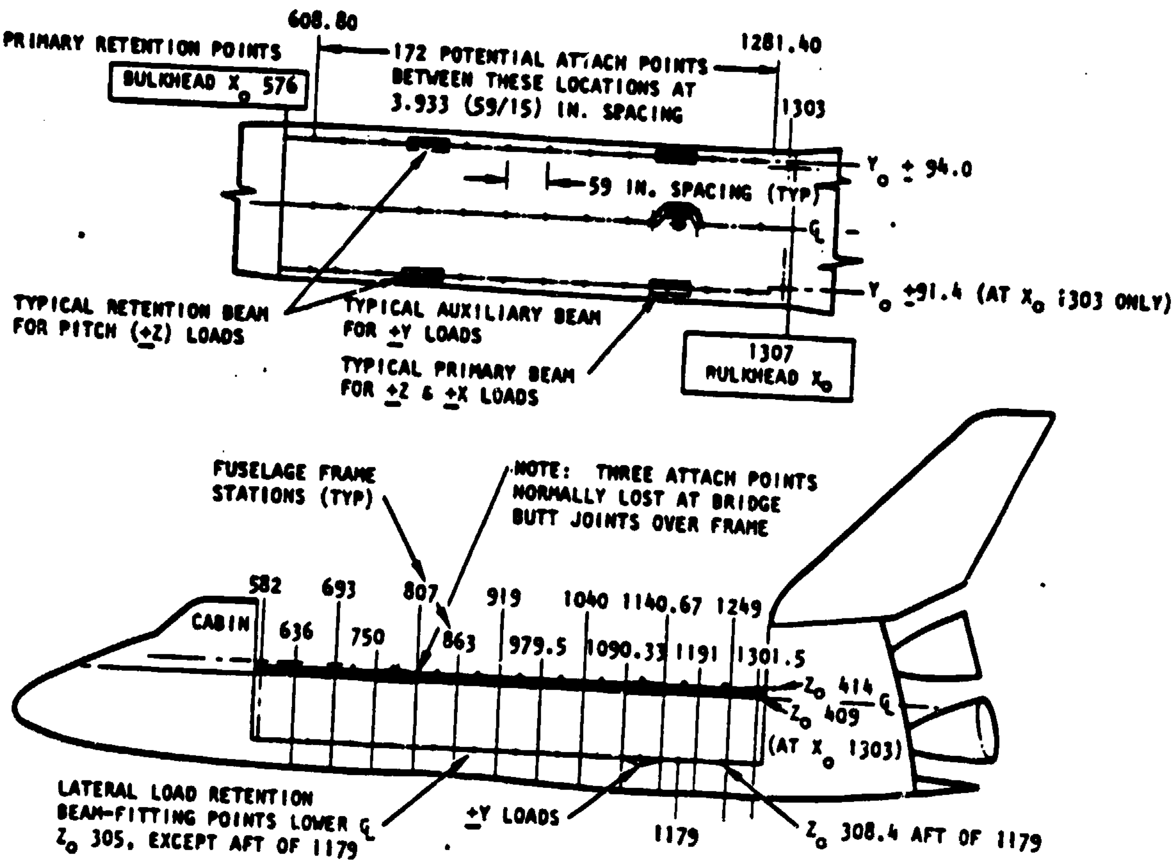


Figure 3.2-1. Standard Orbiter/Payload Attachment Locations (Approximate)*

Should the need arise, unique attach points could be provided on payload-supplied special bridge fittings. Special constraints on such attach point locations are described in Table 3.2-1.

*Specific locations are shown in NASA 1980(a), Table 3.3.1.1-1, Page 3-19.

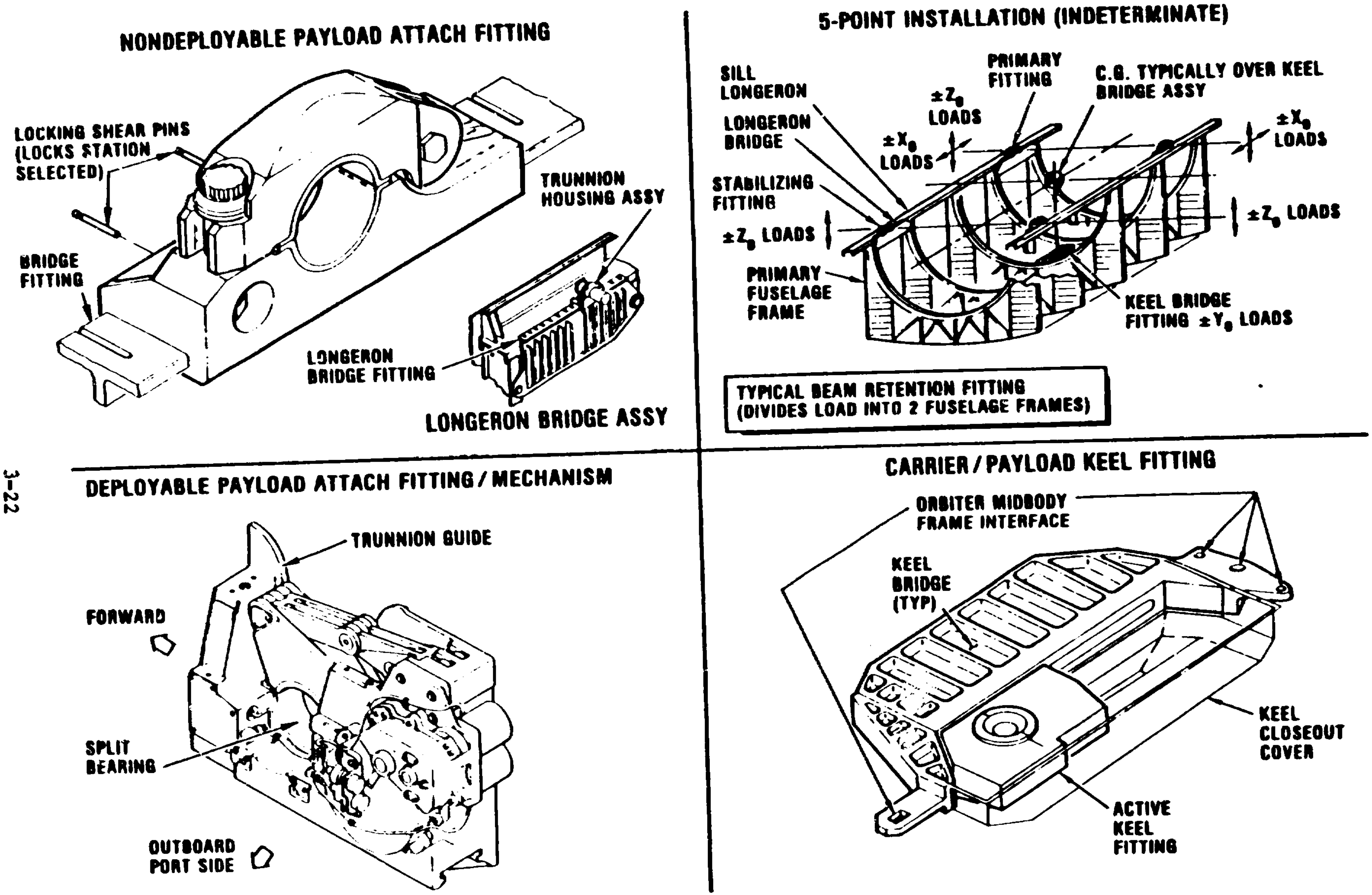


Figure 3.2-2. Standard Attach Fittings for Carriers/Payloads

Table 3.2-1. Checklist: Constraints on Orbiter Payload Attachment Locations

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) Attachment fittings shall be spaced to avoid radiator hose crossover locations, centered at: Xo = 839.36 and Xo = 1198.00 (Typical left and right side). (R-1)	Rockwell 1979 (a) Drawing No. V070-362003
( ) Attachment fittings shall be spaced to avoid cargo bay door actuator locations, centered at: Xo = 602.30 Xo = 966.35 Xo = 737.30 Xo = 1144.20 Xo = 903.80 Xo = 1264.20 (Typical left and right side) (R-1)	
( ) Attachment fittings shall be spaced to avoid electrical crossover locations centered at: Xo = 725.60 Xo = 1019.75 Xo = 794.95 Xo = 1084.75 Xo = 863.31 Xo = 1214.25 (Typical left and right side). (R-1) . A stiff structure design might preclude interference by reason of trunnion deflections.	
( ) Attachment fittings shall be located to avoid interference with the RMS support pedestals (if RMS is installed on right or left side). (R-1) . See Section 4.5 for RMS dimensions. NOTE: RMS and its supports rotate outboard for loading/unloading payloads, inboard for flight.	NASA 1980 (a): JSC-07700, Vol. XIV Four support pedestal locations per RMS arm to be considered.
( ) Attachment fittings shall be located to avoid interference with the Ku-band antennas (R-1) . Principal attach point to longeron at Station 589, right (starboard) side, Left side optional.	Rockwell 1979 (a): MC 409-0025

Table 3.2-1. Checklist: Constraints on Orbiter Payload Attachment Locations (Cont'd.)

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES / IMPLICATIONS
. Ku-Band antenna hinges outboard to permit clearance for loading/unloading payloads on the ground (recent revision).	
( ) Keel fittings limited over wing box.	
. See Figure 3.1-5.	
( ) Active (deployable) attach fittings have limited locations.	NASA 1980 (a): JSC-07700, Vol. XIV, ICD 2-19001
. See Reference ICD	
( ) Minimum distances between trunnions on adjacent cargo elements may be limited by payload ground handling mechanisms (PGHM)	Rockwell 1980 (f): Payload Mixing Study
. See Section 3.7.	

3.2.2 Shuttle Attach Fittings for Carriers/Payloads

Figure 3.2-2 illustrates the typical orbiter-supplied mechanical attach fittings. Carriers/payloads are supported along both sides of the cargo bay at points 14 inches above the bay centerline and along the bottom at the orbiter keel centerline. All attach fittings are outside the 93-inch radius of the orbiter dynamic envelope, except for keel fittings over the wing box (aft of X_o 1179).

The longeron (side) attach fittings are situated above the cargo bay sill longerons on detachable bridges that distribute the fitting loads into orbiter structure. The keel bridges also span adjacent fuselage frames and are detachable.

The carrier/payload attachments are of the trunnion/bearing/journal type. The longeron attach fitting has a split self-aligning bearing and is available either as a nonreleasing fitting whose hinged upper half is bolted closed before launch, or as a remotely actuated fitting which releases/secures the carrier/payload trunnion for on-orbit deployment and retrieval. The deployable fitting has a flared alignment guide to assist in retrieval, which is retracted when not in use. Both types of attach fitting engage the longeron bridge by a tee and slot which allow sliding in the X-direction. Since these trunnions must support both X- and Z-loads, shear pins are installed in one pair of fittings to prevent X-motion and thus transmit X-loads into the bridge and, thereby, into the orbiter. The other longeron attach fitting(s) remains free to slide and transmits only Z-loads.

The keel attach fitting also uses a tee-slot arrangement which makes it free to slide in the X-direction. The carrier/payload will provide a keel trunnion, nominally three inches in diameter, which will fit within a split bearing in the keel attach fitting. The trunnion will be free to move in the Z-direction, but will be restrained in the Y-direction. The keel attach fitting, which is currently being designed, will be remotely actuated open and closed to facilitate payload deployment and retrieval.

3.3 SHUTTLE CARRIER/PAYLOAD RETENTION SYSTEMS

In Figure 3.3-1, the three principal payload retention system concepts are illustrated. Longeron/sill attach fittings react either $\pm X_o$ and $\pm Z_o$ axes loads (primary) or Z_o axis loads only (stabilizing, on either side of the fuselage). The keel fitting (auxiliary, generally located near the payload c.g.) reacts Y_o axis loads, except as shown on the left in three-point determinate. The orbiter baseline, four-point determinate system with three longerons and one keel fittings, is shown in the middle. Addition of a second stabilizing longeron fitting makes the system indeterminate with four fittings on the longeron and one at the keel as shown on the right side in the five-point system. Two longeron fittings in plane with a keel fitting are presented in the three-point system on the keft. Keel fitting has $\pm X_o$ axis load capability as well as $\pm Y_o$ axis as required by this three-point approach.

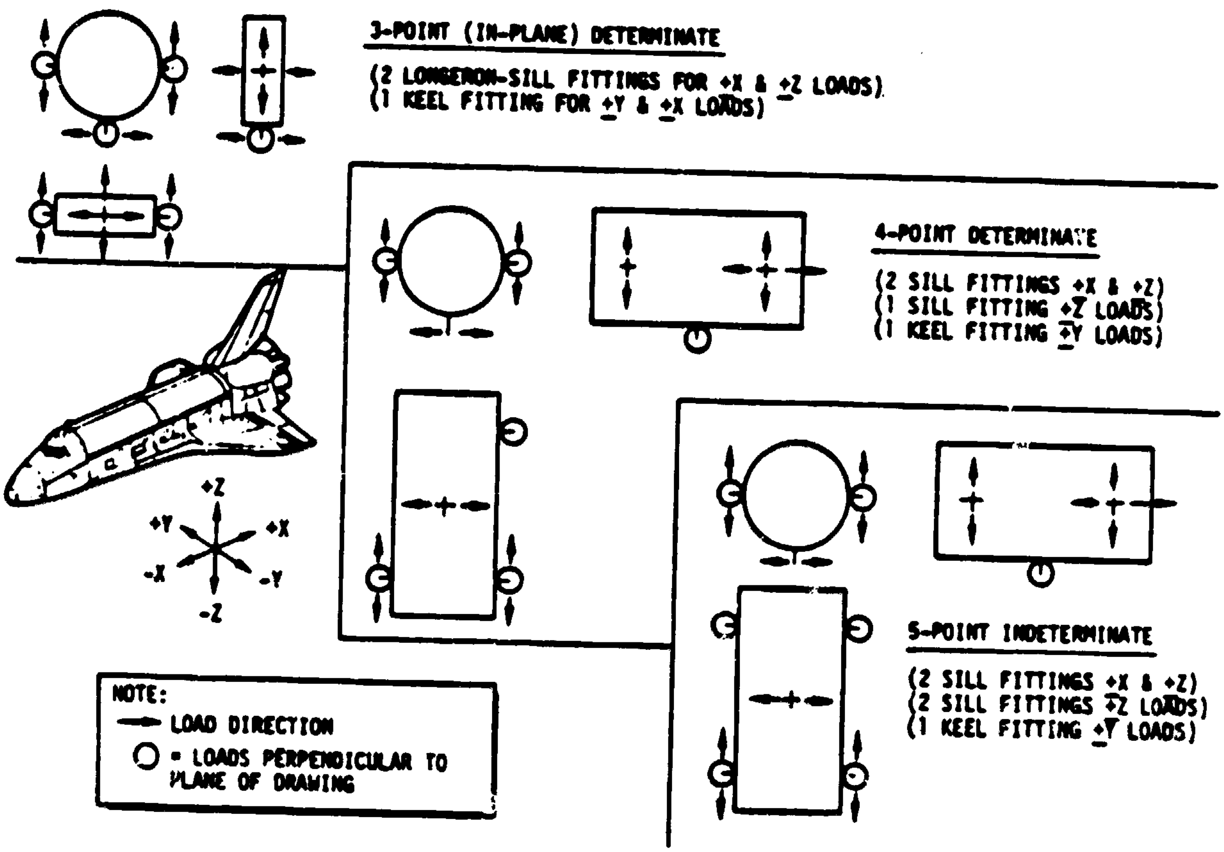


Figure 3.3-1. Carrier/Payload Retention Systems

3.4 CARGO CENTER-OF-GRAVITY ENVELOPE REQUIREMENTS

Carriers/payloads must be integrated into cargos which meet the c.g. constraints of Figure 3.4-1 due to orbiter aerodynamics. All payload chargeable items (e.g., orbiter maneuvering system kits, electrical power subsystem kits, spare parts), regardless of location (e.g., cargo bay, beneath the bay, in the cabin), must be included in the computation to obtain the location of the cargo center of gravity, which must be within the specified envelope for normal or abort reentries and landings. The envelope for weights above 32,000 pounds must be maintained for potential emergency landings. Typical small payloads will not be individually required to control their center of gravity within these limits. Except for large unique payloads, the cargo integrator will be able to manage cargo distribution to achieve an acceptable center of gravity for the entire cargo.

Longitudinal Constraints (X-Axis)

The current vehicle is designed to reenter and land with the total vehicle c.g. located between 65 and 67.5% of the body length. With a design-landing cargo weight of 32,000 lb, the cargo c.g. must fall between 32 and 45 feet from the forward end of the cargo bay.

Vertical Constraints (Z-Axis)

The vertical c.g. envelopes are measured in inches from the centerline of the cargo bay for cargo up to 65,000 lb. The outer envelope constraint is for the entire cargo, whereas the inner envelope (shaded) applies to cargo elements mounted in the cargo bay.

Lateral Constraints (Y-Axis)

The lateral cargo c.g. envelope is symmetrical about the orbiter centerline. Allowable excursions within this envelope are measured in inches rather than feet. The lateral c.g. excursion limit for a 10,000-lb cargo is ±8 in. As the weight increases, the allowable excursion decreases; it is ±3.5 inches for a 32,000-lb cargo and only ±2.5 inches for a 65,000-lb cargo.

3.5 SHUTTLE CARRIER/PAYLOAD DESIGN LIMIT-LOAD FACTORS

The preliminary limit-load factors/angular accelerations shown in Table 3.5-1 apply to rigid payloads attached directly to the orbiter at any location in the cargo bay. These load factors shall be used for preliminary design of carrier/payload primary structure and for determination of preliminary orbiter/carrier interface loads. Load factors at specific points within the payload will depend upon payload design characteristics and mounting methods. Payloads that are cantilevered or that have substantial internal flexibility may experience higher load factors than those shown in the table.

Typical load factors for liftoff and landing are presented. However, during these events, external forces are highly transient and significant elastic response occurs. Carrier/payload responses will depend upon carrier/payload geometry, stiffness, and mass characteristics. Therefore, until suf-

C-2

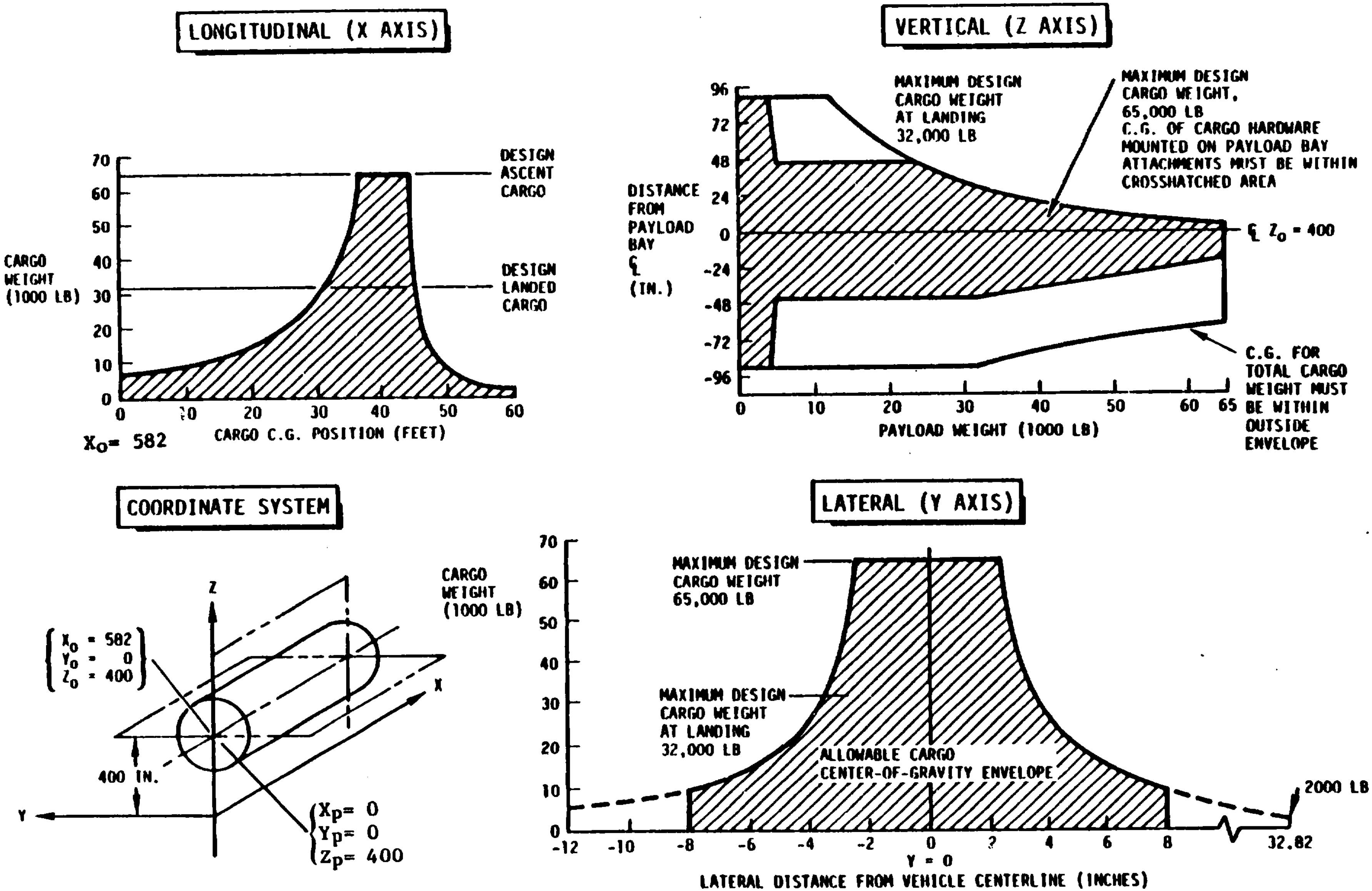


Figure 3.4-1. Cargo Center-Of-Gravity Envelope Requirements

Table 3.5-1. Carrier/Payload Design Limit-Load Factors/Angular Accelerations

QUASI-STEADY FLIGHT EVENTS***	LIMIT-LOAD FACTOR*, g			ANGULAR ACCELERATION** RAD/SEC^2			
FLIGHT EVENT	N_X (+ AFT)	N_Y (+ RIGHT)	N_Z (+UP)	$\ddot{\theta}_X$ (+ RIGHT WING DN)	$\ddot{\theta}_Y$ (+ NOSE UP)	$\ddot{\theta}_Z$ (+ NOSE LEFT)	CARRIER / PAYLOAD WEIGHT
ASCENT							
• LIFTOFF	-0.2 -3.2	±0.1	±2.5	±0.1	±0.15	±0.15	
• BOOST MAX, N_X INTEG VEHICLE	-2.9	±0.6	-0.1	±0.2	±0.25	±0.25	UP TO 65K LB
• SRB POST-STAGING	-1.1	±0.12	-0.6				
• BOOST MAX, N_X ORBITER	-3.17	0.0	-0.6	±0.25	±0.25		
DESCENT							
• TAEM: PITCH MANEUVER	1.01 -0.15	0	2.5	0	0	0	
	0.25	0	2.5	0	-0.11	0	
	0.97	0	-1.0	0	0	0	UP TO 32K LB
	0						
• TAEM: ROLL MANEUVER	0.65	±0.12	1.98	±1.28	0.02	±0.13	
• TAEM: YAW MANEUVER	0.90 0.03	±1.25 ±1.24	1.0 1.0	0 0	0 0	0 ±0.12	
• LANDING	1.8 -2.0	±1.0 ±1.0	4.2 -0.3	±0.25 ±0.25	1.25 0.75	±0.3 ±0.3	

* LIMIT-LOAD FACTOR IS DEFINED AS THE TOTAL EXTERNALLY APPLIED LOAD PER UNIT WEIGHT AT THE c.g. OF THE CARRIER/PAYLOAD AND CARRIES THE SIGN OF THE EXTERNALLY APPLIED LOAD.

** CENTER OF ROTATION IS AT CARRIER/PAYLOAD c.g.

***REFERENCE ICD 2-19001 FOR ADDITIONAL CONDITIONS.

ficient carrier/payload case history is collected, final design values for orbiter/carrier/payload interface forces and carrier/payload design loads must be determined by coupled orbiter/carrier/payload dynamic analyses for these transient flight events.

Emergency Landing Ultimate Load Factors

To provide for crew safety during emergency landing, the large equipment items, pressure vessels, payload attachments, and supporting structure must withstand the loads associated with the following ultimate load factors, acting separately:

Longitudinal (+ Aft)	Lateral (+ Right)	Vertical (+ Up)
+ 4.5 - 1.5	+ 1.5 - 1.5	+ 4.5 - 2.0

3.6 MAXIMUM ALLOWABLE CARRIER/PAYLOAD-IMPOSED LIMIT LOADS

The carrier/payload load factors and maximum allowable carrier/payload-imposed limit loads are related for identical flight and landing conditions. Hence, if the payload weight and its c.g. position are known, the payload attachment point reactions can be estimated and compared with the allowable limit loads at the designated attach points. This, in turn, will guide the positioning of a payload within the cargo bay, the design of the appropriate payload cradle, and/or the payload design.

Representative maximum axial, vertical, and lateral limit loads which can be imposed on the longeron and keel bridges of the orbiter during the various flight events are presented in Table 3.6-1. To determine allowable loads for discrete payload attach points, refer to Appendix D of JSC 07700, Vol. XIV, Payload Accommodations.

Allowable Axial (±X) Limit Loads

During ascent, critical longeron compression loading occurs (1) at SRB post-staging, where payload inertial loads combine with thrust misalignment loads; and (2) at orbiter maximum g. During descent, critical longeron local tension loading occurs at TAEM and at landing, where payload inertial loads combine with thermal and maneuvering loads.

The noted axial load capability applies at every point along the longeron bridge. However, during ascent, if X-loads are applied to a longeron at two or more attach points, the +X-loads are cumulative, and their sum cannot exceed the allowable load for the aftmost employed payload attach point.

Allowable Vertical (±Z) Limit Loads

The tabulated $\pm Z_v$ loads apply only near, or at, the center of each bridge span; are valid for all flight conditions; and are noncumulative—whereas, the $-Z_s$ values, which are valid only during landing, apply at any position along the bridge and are cumulative about the main landing gear station at X_o 1180 (Z_s loads are cumulative forward or aft to this station from the ends of the cargo bay).

Table 3.6-1. Maximum Allowable Carrier/Payload-Imposed Limit Loads

BRIDGE SPAN FRAME TO FRAME	LIMIT LONGERON BRIDGE LOAD PER VEHICLE SIDE, 1000 LB							LIMIT KEEL BRIDGE LOAD, 1000 LB*
	ASCENT, +X		TAEM, -X		VERTICAL LOADS, ALL FLIGHT CONDITIONS		VERTICAL SHEAR LOADS, LANDING	
	SRB POST-STAGING	MAXIMUM g ORBITER	SYM PITCH MANEUVER	UNSYM YAW MANEUVER & LANDING	$+Z_V$	$-Z_V$	$-Z_S$**	ALL FLT & LANDING CONDITIONS, ±Y
582-636	5.9	17		±7	53	-65	-19.3	9
636-693	5.9	17		±7	58	-72	-57.2	18
693-750	5.9	17		±7	61	-79	-59.5	33
750-807	11.4	33		±15	70	-78	-67.9	46
807-863	11.8	34		±15	64	-76	-73.9	56
863-919	18.7	54		±19	60	-76	-73.9	57
919-979.5	19.0	55	TBD	±27	76	-76	-83.2	70
979.5-1040	19.7	57		±25	72	-77	-90.5	80
1040-1090.33	24.2	70		±33	71	-77	-96.9	46
1090.33-1140.67	36.0	104		±50	98	-109	-105.4	72
1140.67-1191	41.6	120		±76	121	-121	-111.9	68
1191-1249	41.6	120		±76	121	-121	-103.9	56
1249-1301.5	38.1	110		±50	121	-121	-94.0	-

*LIMITED BY CAPABILITY OF KEEL BRIDGE SUPPORT STRUCTURE IN ±X DIRECTION

**GIVEN Z_S VALUE IS AT FORWARD FRAME OF BRIDGE SPAN (i.e., $-Z_S$ = -19.3 AT FRAME 582)

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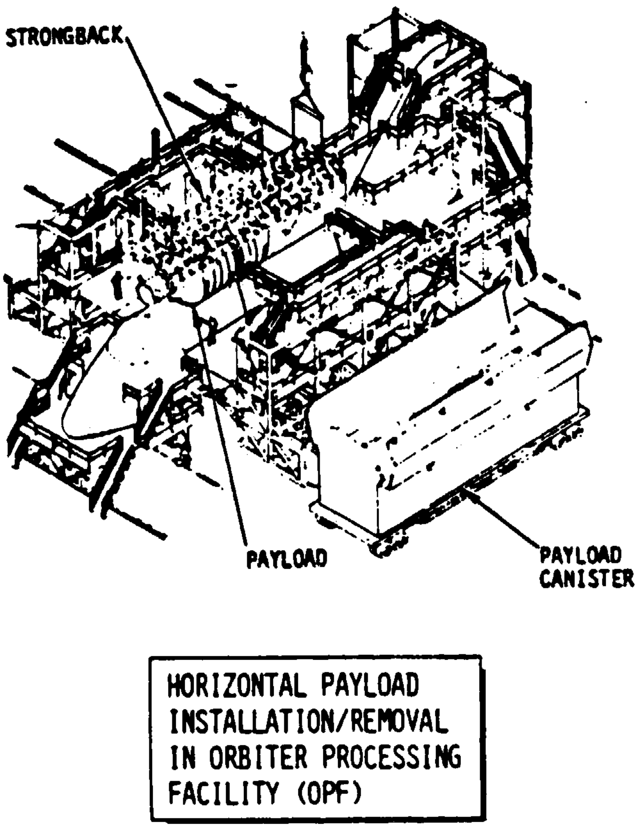
Allowable Lateral (±Y) Limit Loads

The values in Table 3.6-1 are valid for all flight and landing conditions and are noncumulative. The ±Y capability is limited by the axial load capability of the bridge support structure, where an axial load results from the friction force between the payload attach fitting and the bridge.

3.7 CARRIER/PAYLOAD INSTALLATION (KSC)

The concepts for carrier/payload installation and removal with the orbiter in a horizontal position at the Orbiter Processing Facility (OPF) or in the vertical position at the Payload Changeout Room (PCR) at the launch pad for KSC are illustrated in Figure 3.7-1.



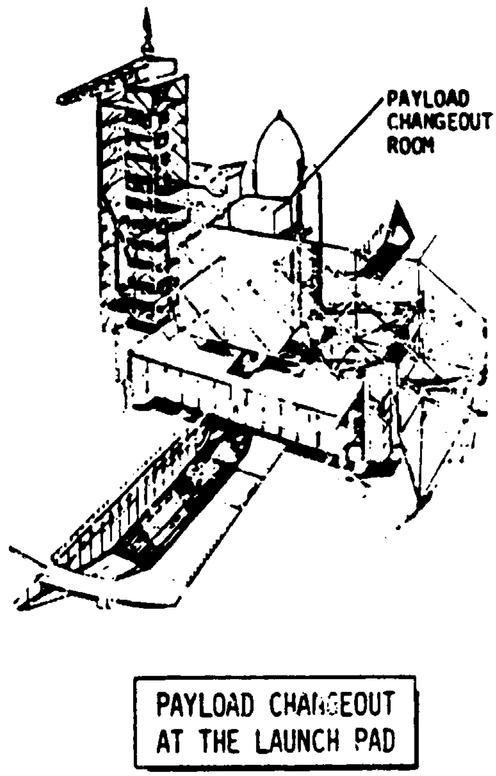


Figure 3.7-1. Carrier/Payload Installation (KSC)

Upon installation of mission kits and orbiter checkout, the cargo bay will be cleaned and carriers/payloads installed. Interfaces between the orbiter and carriers/payloads will be verified by integrated tests. It may be necessary to provide for simulating equipment functions not operable in the earth environment; as a result, carrier/payloads should facilitate simple and end-to-end functional tests. Unique carrier/payload ground support equipment should be easily integratable into integrated orbiter-cargo checkout operations. Payloads should be designed to minimize the need for ground servicing, calibration, and operation, and operation once installed in the orbiter. Payloads are not normally accessible after closure of the cargo bay doors, except through the orbiter cabin.

A carrier/payload can also be installed with the orbiter in the vertical position at the Payload Changeout Room, which encloses the cargo bay. Installation or removal of carrier/payloads at the PCR is with the use of the payload ground handling mechanism (PGHM). The minimum distances between PGHM trunnion arms may be greater than actual distances between trunnions on adjacent cargo elements as installed in the cargo bay. Constraints on these clearances are shown in Figure 3.7-2 (Rockwell 1980 (f)). Although these constraints do not apply to horizontal payload installation, they affect capability to remove or changeout payloads once the orbiter is stacked vertically on the launch pad. As a general guideline, PGHM restrictions should be considered.

In both the OPF and PCR, cargo bay temperature and humidity are controlled using filtered air at 70 $\pm$ 5°F, and 45 $\pm$ 5% RH. The final purge on the launch pad prior to launch is performed using dry nitrogen.

On the launch pad, following installation, payload signals can be monitored via an umbilical interface.

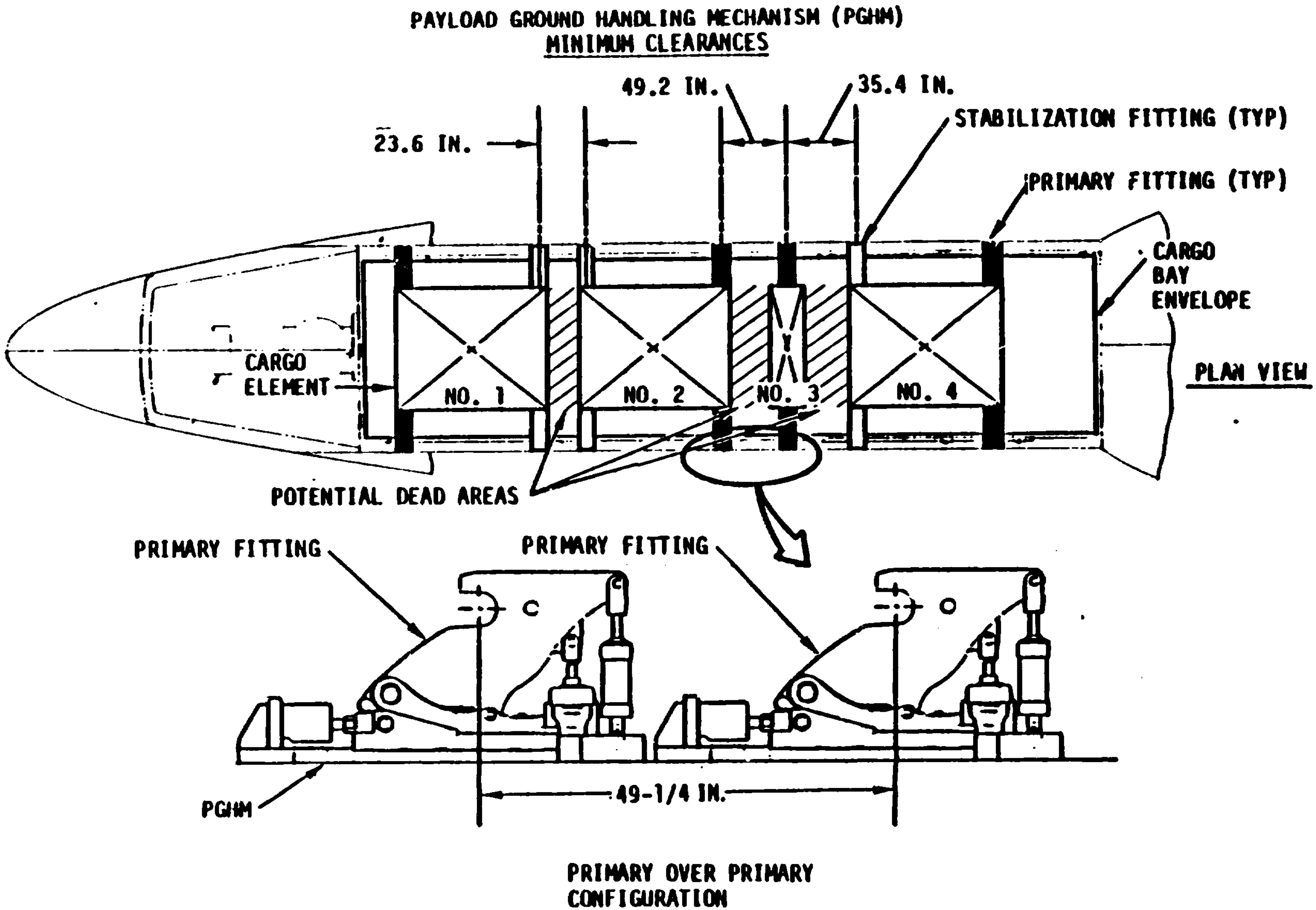


Figure 3.7-2. PGHM Installation & Removal Clearances

3.8 ELECTRICAL INTERFACES

Packaging and installation of equipment and materials for delivery by the Shuttle orbiter will typically require electrical connections to the orbiter for power, data, and control functions. Of particular concern in space construction are potentials for (1) disconnecting and connecting power, data, and signal lines in space, and (2) routing, deploying, and retracting flexible electrical umbilical lines. As in the case of any orbiter payload, all electrical lines, connectors, and support attachments must withstand the 3-g accelerations during ascent as well as the vibration and the descent/landing loads. This section deals primarily with identifying the physical locations and electrical load carrying capabilities of the electrical interfaces. However, it is recognized that the design of the payload side of these interfaces must be based primarily on the orbital functional requirements and their interactions with the overall electrical power system characteristics of the orbiter, which are described more fully in Section 4.10.

3.8.1 Orbiter/Cargo Power Interface Characteristics

Table 3.8-1 summarizes characteristics of orbiter/cargo power interface capabilities, which can also be viewed as forming constraints on the number and type of orbiter/cargo power interface connections. Further amplification is offered by the summary statements in Table 3.8-2 and supporting illustrations (Figures 3.8-1 through 3.8-5) referred to therein. Additional information and special qualifications are available in Space Shuttle System Payload Accommodations [NASA, 1980 (a)].

In general, the designer should strive to use the Standard Mixed Cargo Harness (SMCH) whenever possible, since this will save extensive development time, checkout/integration time and costs. The SMCH is described in Roels (1979), and Rockwell Interface Revision Notice SD-176A (Jones, 1980).

3.8.2 Orbiter/Cargo Command and Data Handling Interfaces

Shuttle constraints regarding command and data (also called signal and control) cables from the orbiter to the payload are similar in many respects to those for electrical power (see Section 3.8.1). However, orbiter command and data cables are separated from power cables, to the degree feasible, to reduce electromagnetic interference. Also, different routings are required to assure capability to route such lines to the control and display equipment in the crew cabin. Table 3.8-3 provides a checklist of pertinent constraints and references to explanatory illustrations. Some repetition of constraints in Table 3.8-2 will be noted, since these serve as a reminder of those special integration considerations involving power cables and fluid lines. Similar considerations apply to use of the SMCH as for the power interfaces.

3.8.3 Ground Umbilical Connections

Power and signal interface capabilities are also provided from ground facilities to the orbiter payload prior to launch. Constraints on such power and signal lines are listed in Table 3.8-4.

Table 3.8-1. Orbiter/Cargo Power Interface Characteristics

INTERFACE	ORBITER STATION	MISSION PHASE	VOLTAGE RANGE (VOLTS)**	POWER (KW) MAXIMUM CONTINUOUS	POWER (KW) PEAK	TIME LIMITATION ON PEAK POWER	ATCS CONFIGURATION	COMMENTS
MID-BAY POWER CONNECTOR	645	GROUND OPERATION (GSE POWER)	27.2-32	1.0	1.5	15 MIN/3 HR	5200 BTU/HR	NORMAL CHECKOUT
			27.2-32	6	TBD	15 MIN/3 HR	W/O RAD KIT	ORBITER POWERED DOWN
			27.2-32	7	12	15 MIN/3 HR	WITH RAD KIT	ORBITER POWERED DOWN
AUXILIARY POWER A & B	645 AFT FLIGHT DECK		26.2-32	0.4	25 AMP	2 SEC*	ANY	0.4 KW MAX AUX A & B COMBINED
			25.7-32	0.2				0.2 KW MAX AUX A & B COMBINED IN AFT FLIGHT DECK
AFT BUS B & C	1307		20-32	1.5	2	15 MIN/3 HR	ANY	
CABIN PAYLOAD BUS	TBD		25.7-32	0.35	0.42	15 MIN/3 HR	ANY	NORMAL CHECKOUT
			25.7-32	0.75	1	15 MIN/3 HR	ANY	ORBITER POWERED DOWN
			115 ±5 VAC	690 VA (3 ϕ)	1000 VA	2 MIN/3 HR	ANY	ORBITER POWERED DOWN
MID-BAY POWER CONNECTOR	645	PRELAUNCH/ ASCENT/ DESCENT/ POST-LANDING	27-32	1	1.5	2 MIN/PHASE	ANY	
AUXILIARY POWER A&B	645 AFT FLIGHT DECK		26.1-32	0.4	25 AMP	2 SEC*	ANY	0.4 KW MAX AUX A & B COMBINED
			25.7-32	0.2				0.2 KW MAX AUX A&B COMBINED IN AFT FLIGHT DECK
AFT BUS B & C	1307		25.7-32	1	1.5	2 MIN/PHASE	ANY	
CABIN PAYLOAD BUS	TBD		24.2-32	0.35	0.42	2 MIN/PHASE	ANY	AC POWER NOT AVAILABLE, PRELAUNCH & ASCENT
			115 ±5 VAC	0.35	0.42	2 MIN/PHASE	ANY	
MID-BAY POWER CONNECTOR NORMAL	645	ON-ORBIT PAYLOAD OPERATIONS	27.2-32	7	12	15 MIN/3 HR	WITH RAD KIT	DEDICATED FC MODE—ORBITER POWERED DOWN
			27.2-32	6	TBD	15 MIN/3 HR	W/O RAD KIT	DEDICATED FC OR TIME SHARE ORBITER BUS WITH 3 FCs OPERATING—ORBITER POWERED DOWN
BACKUP	645		26.8-32	5	8	15 MIN/3 HR	ANY	TIME SHARE POWER—1 FC FAILED
AUXILIARY POWER A & B	645 AFT FLIGHT DECK		26.1-32	0.4	25 AMP	2 SEC*	ANY	0.4 KW MAX AUX A & B COMBINED
			25.7-32	0.2				0.2 KW MAX AUX A & B COMBINED IN AFT FLIGHT DECK
AFT BUS B & C	1307		24-32	1.5	2	15 MIN/3 HR	ANY	
CABIN PAYLOAD BUS	AFT FLIGHT DECK		24.2-32	0.75	1.0	15 MIN/3 HR	ANY	
			115 ±5 VAC	690 VA (3 ϕ)	1000 VA	2 MIN/3 HR	ANY	

*REMOTE POWER CONTROLLERS (RPC) WILL AUTOMATICALLY OPEN CIRCUIT WHEN 25 AMP IS REACHED. **SUBTRACT ΔV BETWEEN ORBITER & PAYLOAD

Table 3.8-2. Checklist: Constaints Involving Orbiter/Cargo Electrical Power Interfaces for Payload Packaging/Installation and Ascent/Descent Modes

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) Total cargo bay power is limited to 1 kW during orbiter power-up mode, i.e., ascent and checkout. (R-1) • This power is the sum of all power from mid-bay, auxiliary, and aft buses.	Rockwell, 1980 (b): SSV 80-1 Limits power for environmental control, heating, etc., of cargo in bay. Limits total energy demand.
( ) Total power available to all payloads, including that in the cabin, is 1350 watts continuous during ascent and descent configuration. (R-1) • Up to 350 watts of this may be consumed by payload-unique equipment in the crew cabin.	Rockwell, 1980 (b): SSV 80-1 Defines limits for displays and controls in ascent/descent. Limits total energy demand.
( ) Total peak power for payload during ascent or descent is 1920 watts. (R-1) • Up to 420 watts of this may be consumed by payload-unique equipment in the crew cabin.	Rockwell, 1980 (b): SSV 80-1 Limits total energy demand.

Table 3.8-2. Checklist: Constraints Involving Orbiter/Cargo Electrical Power Interfaces for Payload Packaging/Installation and Ascent/Descent Modes (Cont.)

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) Orbiter subsystems are in the ascent or descent operational configurations during a portion of the prelaunch checkout, throughout the launch, ascent to orbit, entry, and landing phases of a mission; and up to 30 minutes of post-landing time. (4-1)	Rockwell, 1980 (b): SSV 80-1 Data useful for defining total mission energy requirement, design of equipment.
( ) Use of power from all sources is limited by heat rejection capability. (R-1) • See Section 4.13 for discussion of heat rejection capability.	Rockwell, 1980 (b): SSV 80-1 Implies consideration of water for boiling to reject heat.
( ) Power buses must be isolated on the payload side of the electrical interface, except for the auxiliary buses (EMI/EMC considerations) (R-1) • Electrical conduction analysis and disposition is required whenever electrical power is run to payload structure while the payload item is mechanically connected to orbiter structure. • See noted reference for electrical bonding requirements.	Rockwell, 1980 (b): SSV 80-1 Affects detail design of structure, EMI/EMC analysis efforts and procedures. Required for all electrical equipment in payload.
( ) Payload shall use standard mixed cargo harness (SMCH) whenever possible. (R-1) • See Figure 3.8-1. • See noted reference for details of SMCH design requirements. • Payload may supply unique cable to power supply panels at additional expense (not recommended).	•Roels, D.P., 1979: SMCH-DRD-0001 •Rockwell, 1980 (d) Affects design of cable connections, deployment capability, weight, cost.

Table 3.8-2. Checklist: Constraints Involving Orbiter/Cargo Electrical Interfaces for Payload Packaging/Installation and Ascent/Descent Modes (Cont.)

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) Electrical power can be supplied to payloads in the cargo bay at the following. (R-1)	Rockwell, 1980 (b): SSV80-1
• Mid-bay and auxiliary power connectors are located below the starboard door longeron at Station X_o645. There are two main power connectors and two auxiliary power connectors at this location. See Figure 3.8-2.	NASA, 1980 (a): JSC-07700, Vol. XIV ICD 2-19001
• Secondary interfaces are provided at Station X_o1203 on the starboard side on the wire tray. See Figure 3.8-3.	Affects design of interface harness from payload to power connectors.
( ) Payload-unique equipment in the crew cabin receives power from two additional interfaces. (R-1)	NASA 1980 (a);
• Location: See noted reference.	Affects design of display and control equipment in crew cabin (aft flight deck)
( ) Where two payloads share the same orbiter power bus, protected feeder lines with independent switching, where required, can be provided to each. (R-1)	Rockwell, 1980 (b): SSV 80-1 Affects switching function design
( ) Equipment requiring maximum power demand (up to 7 kW continuous with 12 kW peak for 5 min/3 hr operation) must be connected to the mid-bay power connectors. (R-1)	Rockwell, 1980 (b): SSV 80-1 Affects design of harness to construction fixtures, heaters, beam-builders

Table 3.8-2. Checklist: Constraints Involving Orbiter/Cargo Electrical Power Interfaces for Payload Packaging/Installation, and Ascent/Descent Modes (Cont.)

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) The cargo element user shall provide power, signal, and fluid interfaces, as required, within 540-in.^{3} volumes forward of each longeron trunnion as illustrated in Figure 3.8-4. The cargo element shall provide support for fluid lines and power and signal cables within 18 in. of the interfaces located above with adequate allowance for mating and demating of connectors. The cargo element shall also provide additional cable and fluid line support, as required, such that there shall be no unsupported lengths of cables or fluid lines greater than 18 inches. (R-1) • Requires integration with signal and fluid interfaces.	NASA, 1980 (a): JSC 07700, Vol. XIV ICD 2-19001 Affects design of the electrical cable support. Also influences other provisions for connections to orbiter.
( ) The orbiter shall provide scar in the mid-body for the attachment of cargo element hardware at locations shown in Figure 3.8-5. The load-carrying capability shall be such that a 10-lb weight can be carried at each scar location simultaneously. Use of this scar by each cargo element shall be limited to that specified in its unique ICD. (R-1)	NASA, 1980 (a). Influences attachment design for the electrical cables to payload from orbiter.
( ) Electrical cable trays are provided in the mid-body (starboard and port sides) for support of the SMCH or cables connecting two or more payload elements. (R-1) • See Figure 3.8-1. • If standard orbiter cable trays are not used, the payload must provide its own.	NASA, 1980 (a): Affects design of electrical support equipment for payload bay packaging.

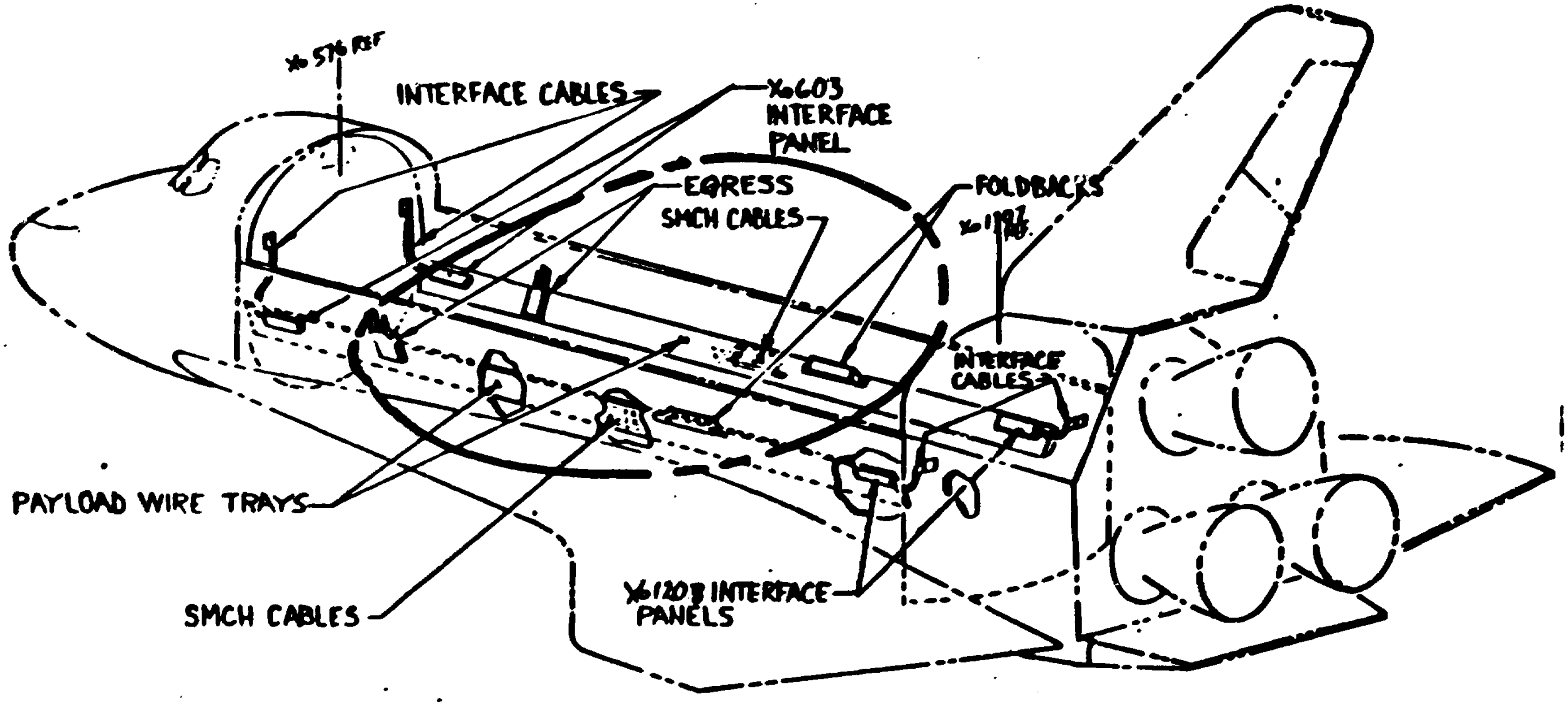
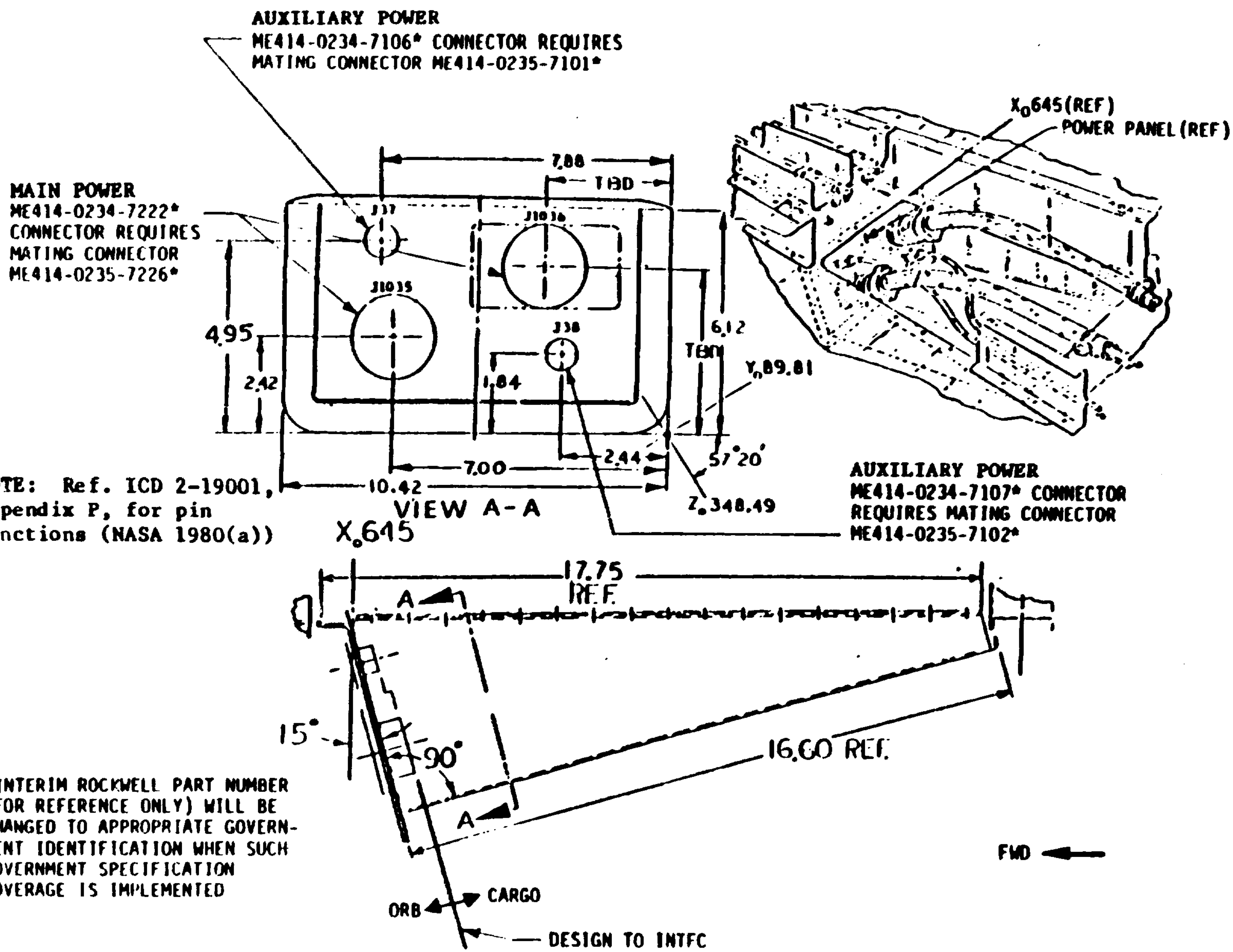


Figure 3.8-1. Standard Mixed Cargo Harness Kit (SMCH)



NOTE: Ref. ICD 2-19001, Appendix P, for pin functions (NASA 1980(a))

*INTERIM ROCKWELL PART NUMBER (FOR REFERENCE ONLY) WILL BE CHANGED TO APPROPRIATE GOVERNMENT IDENTIFICATION WHEN SUCH GOVERNMENT SPECIFICATION COVERAGE IS IMPLEMENTED

Figure 3.8-2. Payload Electrical Power Service Panel (X_o 645)

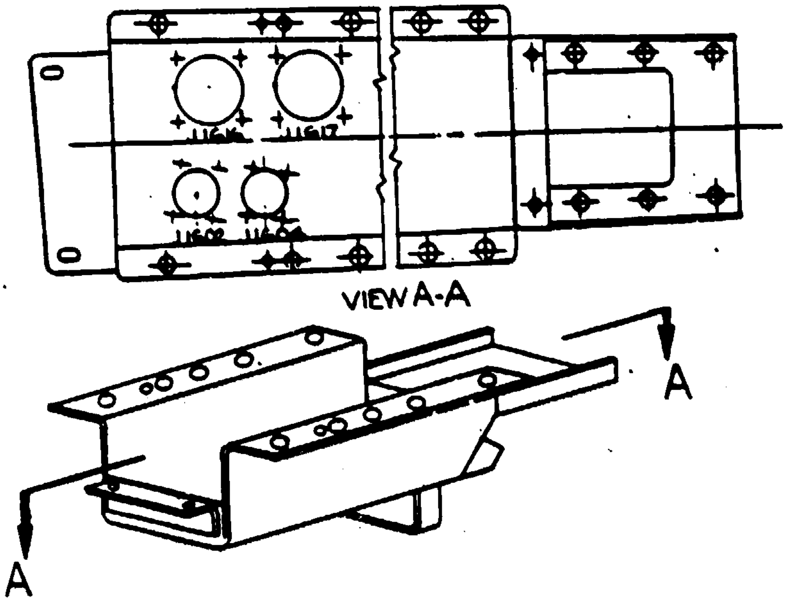
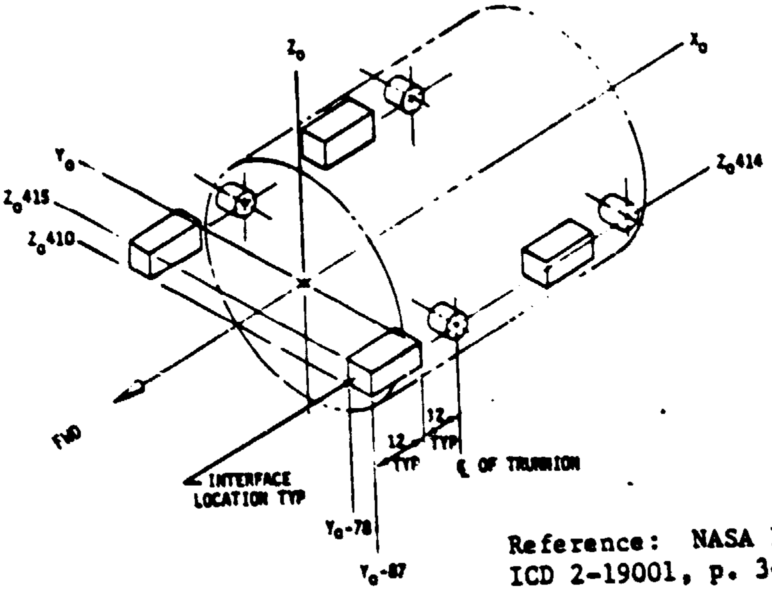
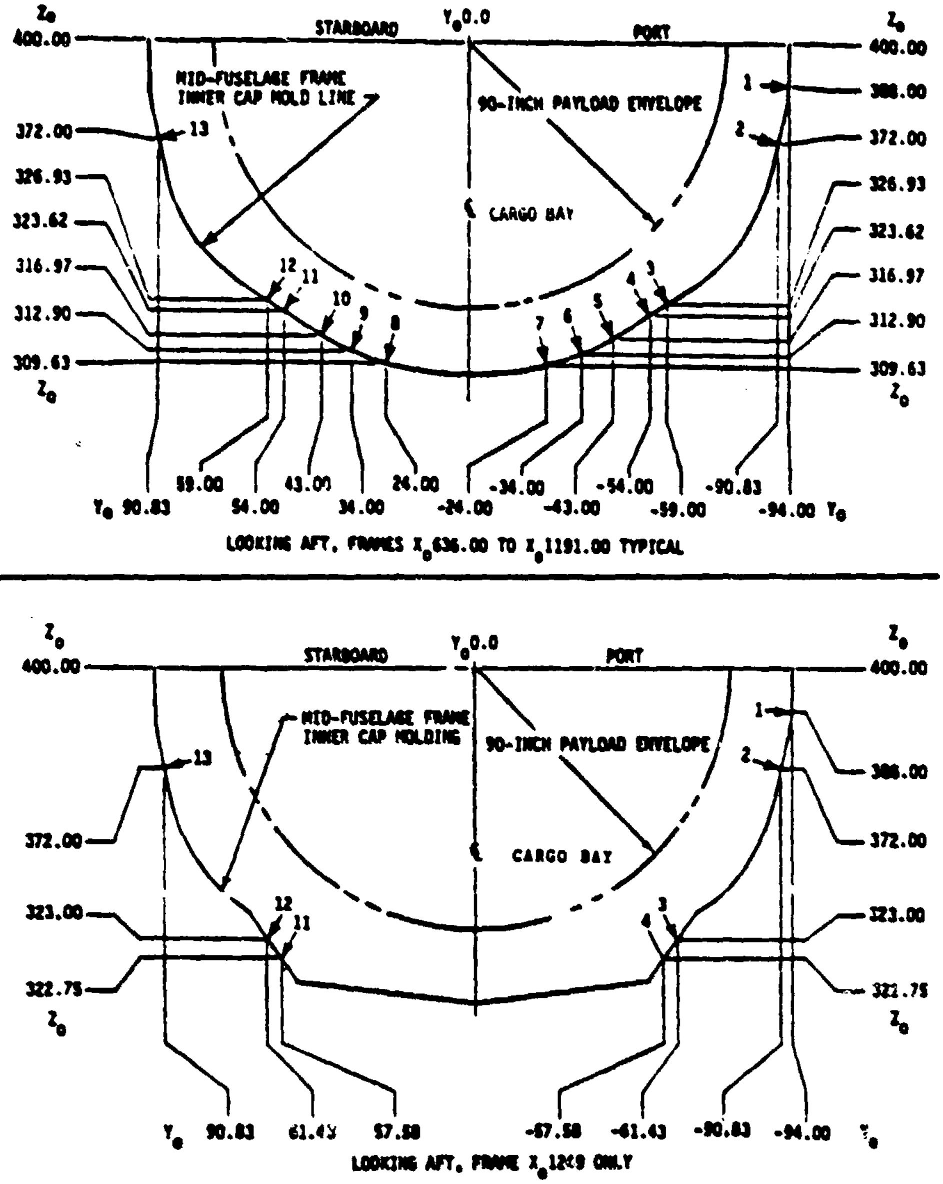


Figure 3.8-3. Electrical Power Interface Panel at Station X_o 1203, Starboard Side



Reference: NASA 1980(a), ICD 2-19001, p. 3-45A

Figure 3.8-4. Power, Signal, and Fluid Interface Locations at Cargo Element



NOTE: For specific details of hole patterns at each scar attach point, see Space Shuttle System Payload Accommodations, JSC-07700, Vol. XIV, Rev. F, ICD 2-19001, p. 3-45B.

Figure 3.8-5. Mid-Body Scar Attachment Provisions

Table 3.8-3. Checklist: Constraints Involving Cargo/Orbiter Signal and Control Cables in the Orbiter Payload Bay

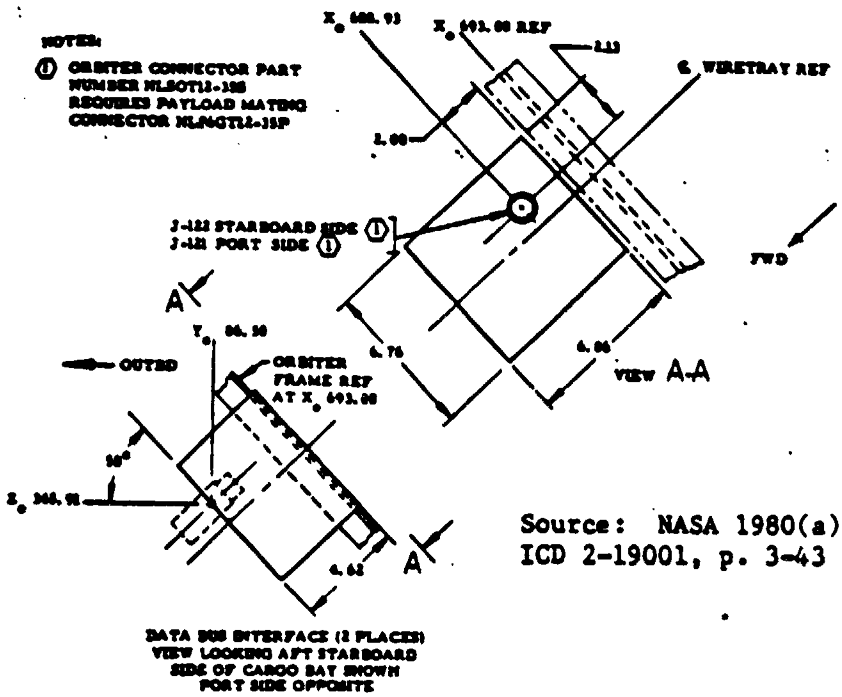
( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) Data Bus Interface at X_o 693. The mid-body data bus access panel port and starboard interface are defined in Figure 3.8-6. These panels are not available for cargo use unless the requirement is defined in the payload-unique ICD. (R-1)	NASA, 1980 (a): JSC 07700, Vol. XIV, ICD 2-19001
• See Figure 3.8-6 for specific locations of this interface.	Affects design of data cabling and routing
( ) A portion of the orbiter standard mixed cargo harness (SMCH) is provided for connecting payload data and command to the orbiter at Station X_o693. Details are provided in the reference. (R-1)	Roels, D. P., 1979: SMCH-DRD-0001 PIRN 176A to ICD2-19001, NASA 1980(a)
• Payload designers shall design for utilization of the SMCH.	Affects routing of data and command lines
( ) SMCH signal and control cables to payloads shall interface with panels located on the port side of the cargo bay at forward Station X_o603 and aft Station X_o1203 (Figure 3.8-7).	NASA, 1980 (a): JSC 07700, Vol. XIV, ICD 2-19001
( ) SMCH cables shall be available as GFE from Station X_o603 directly to payloads in the cargo bay. See the noted reference for additonal details involving locations, pin layouts, keyways and matching connectors.	NASA, 1980 (a): JSC 07700, Vol. XIV, ICD 2-19001 Affects design of attach cables
( ) Signal and control cables may, under some conditions, interface directly with service panels at the forward bulkhead (Sta. 576) and the aft bulkhead (Sta. 1307). In all cases, these must be agreed to and defined in payload-unique ICD's. (R-1)	NASA, 1980 (a): JSC 07700, Vol. XIV, ICD 2-19001 Rockwell, 1980 (b): SSV 80-1
• See Figure 3.8-8.	

Table 3.8-3 Checklist: Constraints Involving Cargo/Orbiter Signal and Command Cables in the Orbiter Payload Bay (Cont.)

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) All signal and control cables to payloads shall be routed through the standard orbiter payload bay cable trays up to the egress mount. • Inter-payload cables also may be routed through these cable trays. • Special cable supports may be supplied by payload. • An egress must be installed to provide an optimum routing for the payload wire harness/coax cables to interface with a payload.	NASA, 1980 (a): JSC 07700, Vol. XIV, ICD 2-19001 Same as for power, but EMI analysis may dictate which side of bay.
( ) Power, signal and fluid interfaces at cargo elements. The cargo element user shall provide power, signal and fluid interfaces as required, within 540 in.^{3} volumes forward of each longeron trunnion as shown in Figure 3.8-4. The cargo element shall provide support for fluid lines and power and signal cables within 18 in. of the interfaces located above with adequate allowance for mating and demating of connectors. The cargo element shall also provide additional cable and fluid line support, as required, such that there shall be no unsupported lengths of cables or fluid lines greater than 18 in. (R-1) • Requires integration with power and fluid interfaces.	NASA, 1980 (a): JSC 07700, Vol. XIV, ICD 2-19001 Affects design of the electrical cable support. Also influences other provisions for connections to orbiter.
( ) Mid-Body Scar. The orbiter shall provide scar in the mid-body for the attachment of cargo element hardware at locations shown in Figure 3.8-5. The load-carrying capability shall be such that a 10-lb weight can be carried at each scar location simultaneously. Use of this scar by each cargo element shall be limited to that specified in its unique ICD. (R-1)	NASA, 1980 (a): JSC 07700, Vol. XIV, ICD 2-19001 Influences attachment design for the electrical cables to payload from orbiter.

Table 3.8-3. Checklist: Constraints Involving Cargo/Orbiter Signal and Control Cables in the Orbiter Payload Bay (Cont.)

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) All cargo bay electrical bonding interfaces shall be prepared in accordance with the noted reference.	NASA, 1980 (a): JSC 07700, VOL. XIV, ICD 2-19001



Source: NASA 1980(a), ICD 2-19001, p. 3-43

Figure 3.8-6. Data Bus Interface (X_o693)

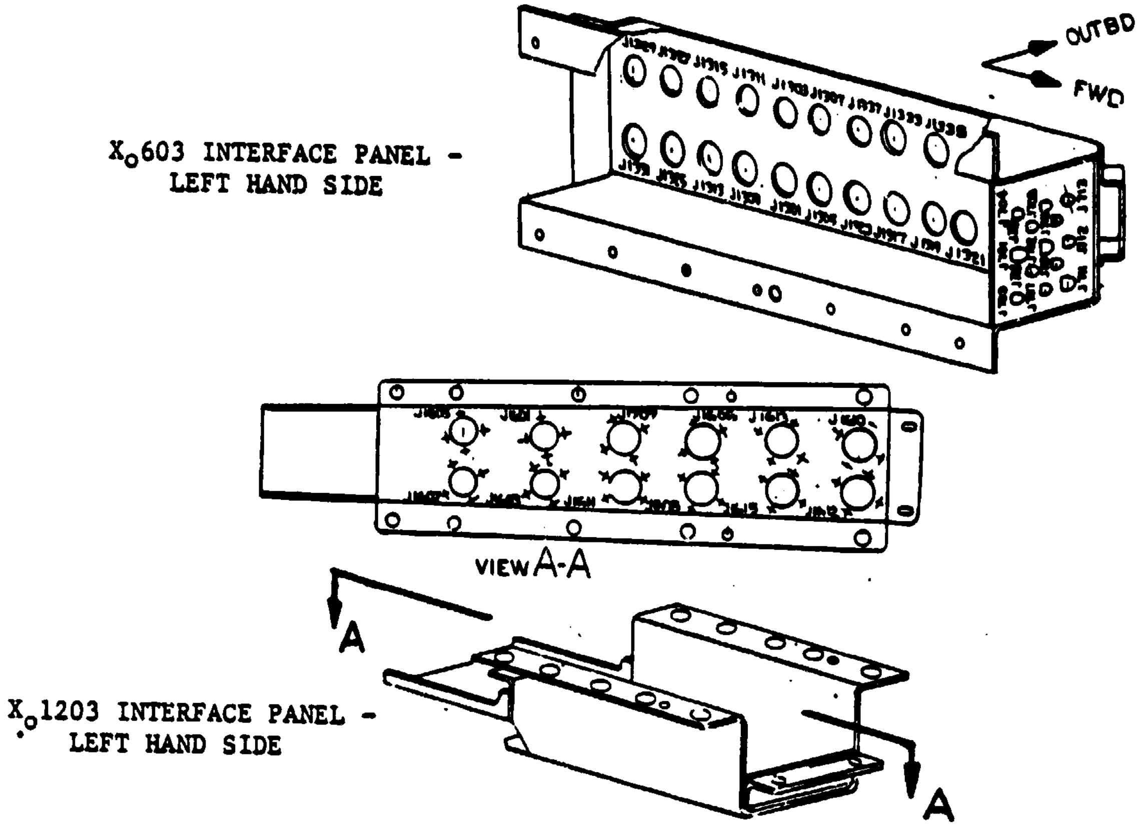


Figure 3.8-7. Forward (X_o603) and Aft (X_o1203) Interface Panels

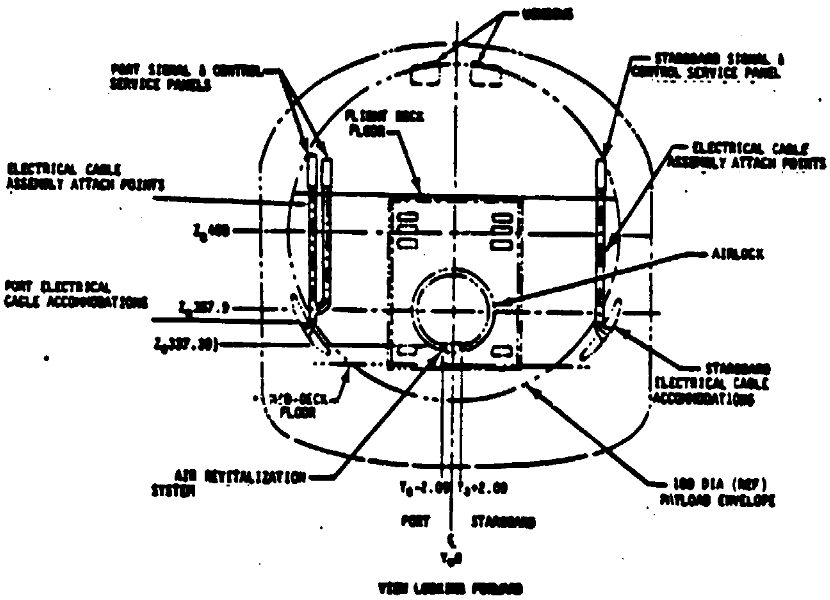


Figure 3.8-8. Locations of Signal and Control Service Panels at Station X_o576 Bulkhead

Table 3.8-4. Checklist: Constraints Involving Electrical Ground Umbilical Connections to Payloads

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) The orbiter shall provide for installation of electrical (and fluid) services to the cargo from GSE in the mid-body prelaunch umbilical panel as shown in Figure 3.8-9. (R-1)	NASA, 1980 (a): JSC 07700, Vol. XIV, ICD 2-19001, Par. 3.3.6, Figure 3.3.6-1
( ) The port and starboard (T-4) umbilical panels (Payload/OMS Delta V Umbilical Panels and Dump Provisions) are shown in Figure 3.8-10. (R-1)	NASA, 1980 (a): JSC 07700, Vol. XIV, ICD 2-19001, Par. 3.3.9
( ) The T-0 Umbilical Panels are shown in Figure 3.8-11. The port side is for fuel. The starboard side is for oxidizer. (R-1)	Rockwell, 1980 (b): SSV 80-1, Page 223

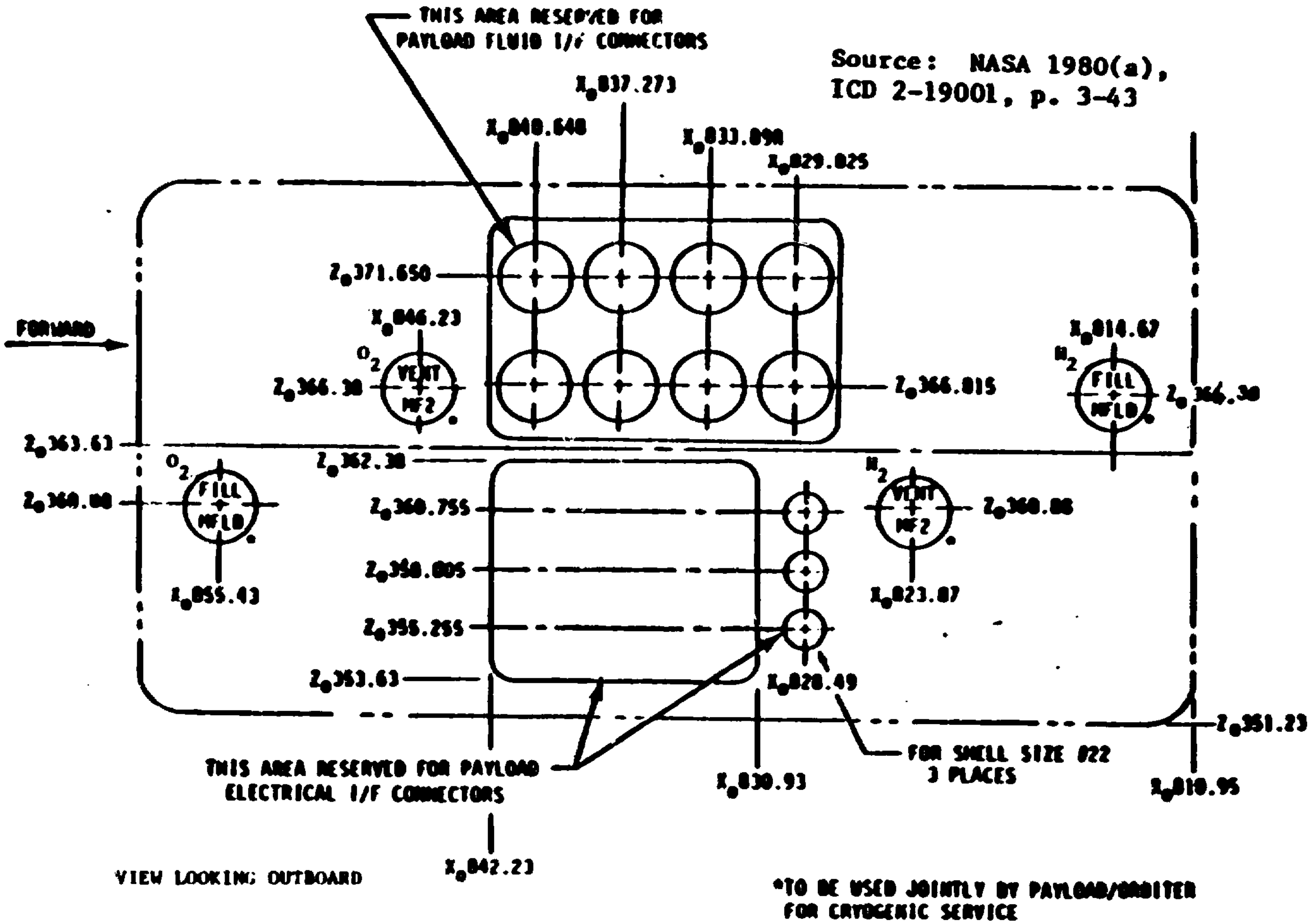
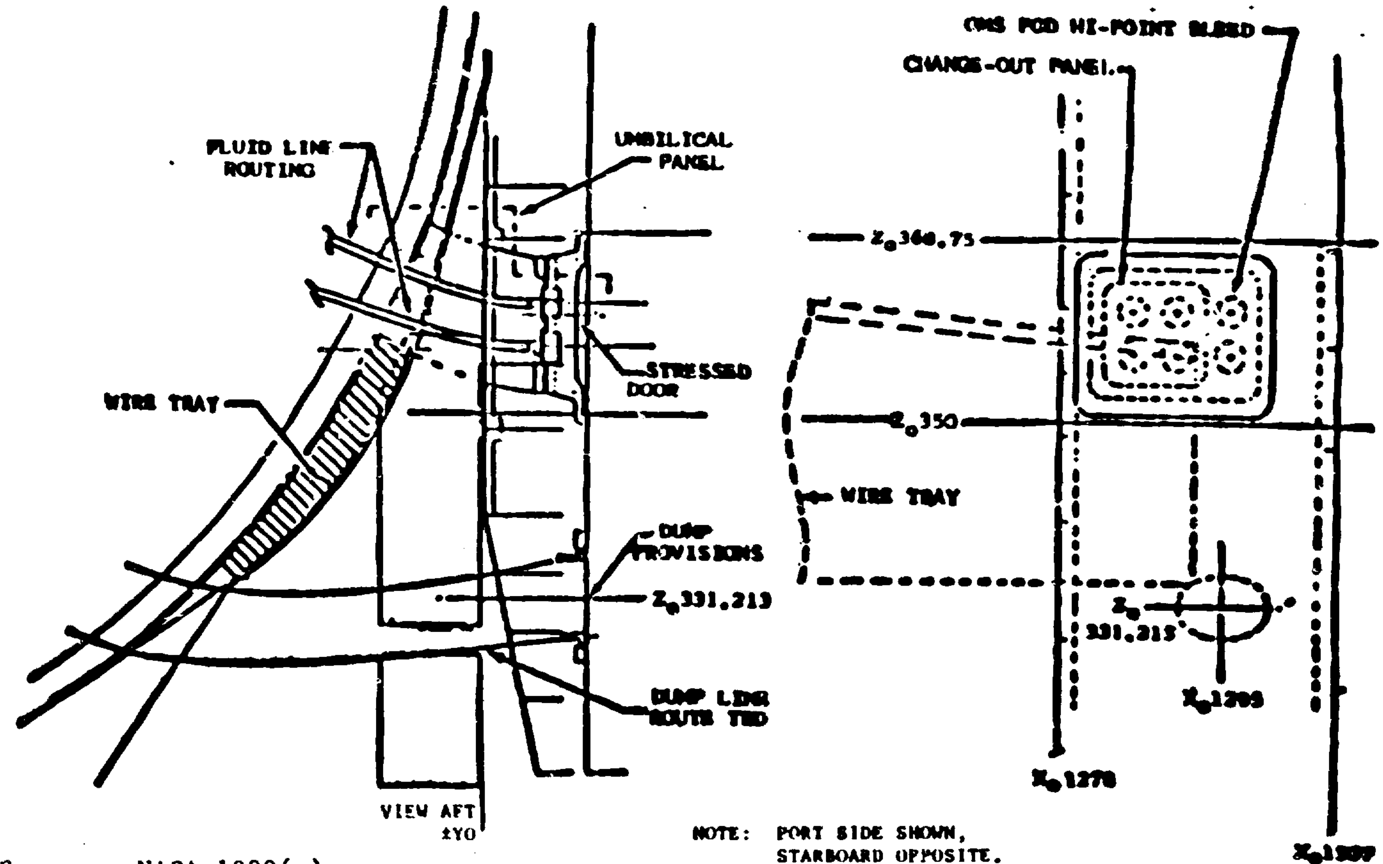


Figure 3.8-9. Mid-Body Prelaunch Umbilical Panel
(Port Side at Y_o 100.20)



Source: NASA 1980(a), ICD 2-19001, p. 3-47

Figure 3.8-10. T-4 Payload/OMS Delta V Umbilical Panels and Dump Provisions (Routing Concepts)

Design for large space structure components evaluated to date of this publication generally have not required extensive concern for services from GSE to payload items. Therefore, further details of these Shuttle system capabilities do not appear in this document.

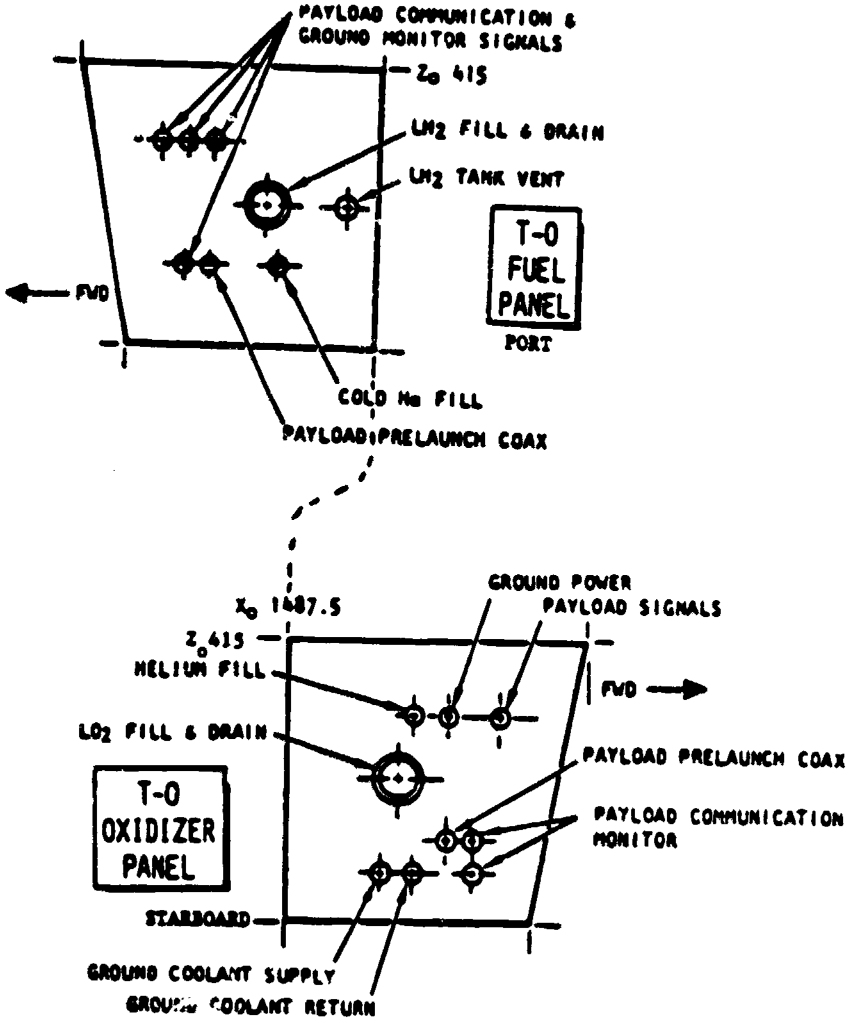


Figure 3.8-11. T-0 Umbilical Panels

3.9 FLUID INTERFACES

Some payloads associated with large space structures may require cooling services provided by the orbiter. In such cases, fluid lines need to be connected between the payload and the orbiter heat exchanger. Decisions on the size and locations of such lines will depend on the cooling capability described later in Section 4.13.

In addition, the orbiter can provide O_2/N_2 to the payloads, if desired.

Provisions for the above fluid connections are combined at a heat exchanger coolant and O_2/N_2 supply interface panel in the payload bay, at Station X_o 636 (see Figure 3.9-1 and NASA 1980(a), ICD 2-19001, p. 3-41). The constraints applicable to fluid lines are further amplified and summarized in Table 3.9-1.

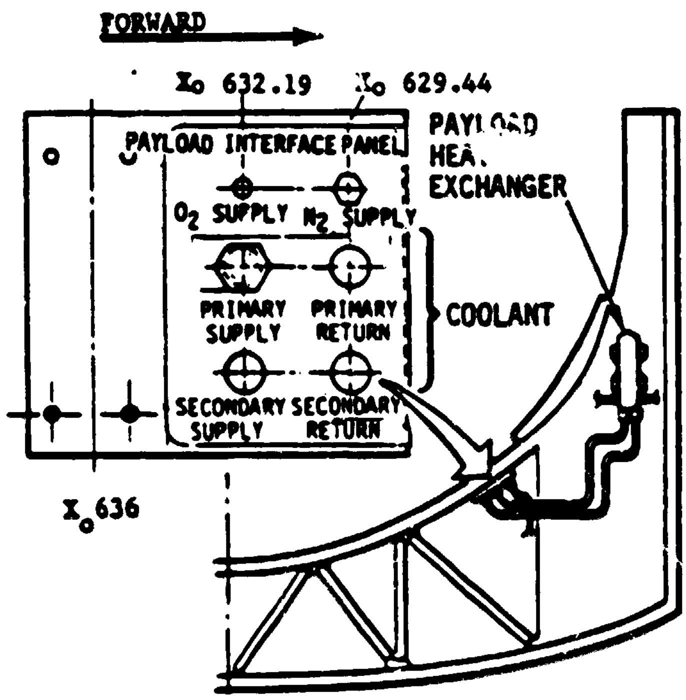


Figure 3.9-1. Heat Exchanger Coolant and O_2/N_2 Supply Interface Panel at X_o 636

Table 3.9-1. Checklist: Capabilities and Constraints Involving Fluid Interfaces between Payloads and Orbiter

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) A service panel is provided in the cargo bay for connecting coolant and O_2/N_2 lines from payloads to the orbiter. (R-1)	NASA, 1980 (a): JSC-07700, Vol. XIV, ICD 2-19001
• The panel is nominally located at Station X_0 636 on the port side.	
• See Figure 3.9-1.	
( ) Cargo element fluid lines shall be attached by payload supplied hardware to scar (mounting provisions) provided in the cargo bay. (R-1)	NASA, 1980 (a): JSC-07700, Vol. XIV, ICD 2-19001
• See Figure 3.8-5 and Table 3.8-3 for load-carrying capabilities. • Scar provisions may be utilized for either electrical or fluid lines	Affects attachments design and routing of fluid lines for payload connections to orbiter.
( ) Power, signal and fluid interfaces at cargo elements. The cargo element user shall provide power, signal and fluid interfaces, as required, within 540 in^3 volumes forward of each longeron trunnion as illustrated in Figure 3.8-2. The cargo element shall provide support for fluid lines and power and signal cables within 18 in. of the interfaces located above with adequate allowance for mating and demating of connectors. The cargo element shall also provide additional cable and fluid line support, as required, such that there shall be no unsupported lengths of cables or fluid lines greater than 18 inches. (R-1)	NASA, 1980 (a): JSC-07700, Vol. XIV, ICD 2-19001 Affects design of the fluid line supports. Also influences other provisions for connections to orbiter.
• Requires integration with electrical power and signal cables to payload from orbiter.	

3.10 CONTAMINATION CONTROL

The subject of contamination control in relation to design of large space structures is considered a concern more closely related to on-orbit operations than to preparations for delivery. The majority of information regarding this subject is provided in Section 4.14, which includes purging and venting throughout the orbiter flight from pre-launch to post-landing. For the most part, the individual payload designer will be responsible for packaging his equipment to protect it from atmospheric contamination and corrosion. However, no special shuttle constraints or guidelines pertaining to such packaging have been identified at this time.

4.0 DESIGNING FOR SHUTTLE ORBIT OPERATION CAPABILITIES

The shuttle constraints and guidelines in this section are those most clearly, directly, and uniquely related to design of large space structures and installed systems. That is, the emphasis in Section 4.0 is on the potential on-orbit operations involving use of Orbiter systems, components, and crew for constructing, maintaining and otherwise working with large space structures. The presentation begins with considerations of berthing, docking and separation. This is followed by the closely related concerns for attitude control. The remainder of Section 4.0 primarily concerns payload handling and system interface characteristics which may be of concern to designers of large space structures.

4.1 BERTHING, DOCKING AND SEPARATION

Large area space systems will typically require more than one Shuttle flight to complete the construction process. Thus, the capability for revisit docking/berthing to the partially completed space structure is a basic construction requirement. It follows that a means to safely separate the Orbiter from the partially or completely constructed space project also must be available. This section deals primarily with Shuttle provisions, constraints and guidelines involving the Orbiter control and interfacing hardware for berthing, docking and separation. The associated issues of attitude control of the Orbiter, large space structure and combination of both are covered in Section 4.2.

4.1.1 Definitions

Equipment and procedures constraints and guidelines are significantly different according to the means of joining the Shuttle Orbiter to the large space structure. The following definitions are essential to subsequent discussions:

Docking is defined as the joining in space of two spacecraft or spacecraft modules by maneuvering one into contact with the other, at the docking interface, using reaction control thrusters.

Berthing is defined as the joining in space of two spacecraft or spacecraft modules by maneuvering one into contact with the other, at the berthing interface, using a manipulator.

In the context of this document, berthing and its reverse, unberthing, of modules to the large space structure are considered under the heading of "payload deployment and handling", discussed mostly in Sections 4.4.4 and 4.5 through 4.7. The following discussion deals with joining and separation of the Orbiter to the main large space structure.

"Caution: Not a controlled document! See appropriate reference documents for current data."

The basic capabilities of the Orbiter to provide the revisit requirement are outlined in the following paragraphs. Individual space contractor projects may differ in their revisit needs and terminal closure geometries. Thus, key capabilities and considerations are discussed for both berthing and docking type revisits.

4.1.2 <u>On-Orbit RCS Control Modes</u>

The Orbiter flight control system is mechanised to perform a number of different rotational and translational maneuvers options. These are summarized in Table 4.1-1. Any primary RCS translation mode may be selected in combination with any primary RCS rotational control mode. Only rotational control modes are available with the vernier RCS system and combined primary and vernier RCS modes are not possible. In addition to the modes identified in the table, an alternative mode for applying small ΔV's in the +Z body direction is being implemented. This mode is called $\pm X$ PRCS or "low-Z" braking in various sources and is principally used to minimize plume impingement effects of proximity braking maneuvers. It utilizes the $+Z_B$ component of thrust associated with the forward and aft X-thrusters of the primary RCS system (See Table 4.1-2). Both fore and aft-thrusters are fired simultaneously to cancel their X_B components of thrust and to produce a net $\pm Z_B$ component of thrust. This mode is relatively fuel inefficient (approximately 9 times the propellant used by direct Z-thruster firing) but results in significant reductions in plume impingement effects for many target vehicle configurations.

Most revisit terminal closure operations will be performed with the relative closure motion along the Orbiter Z-body axis to allow visual alignment of the target vehicle with the COAS (crewmans optical alignment sight) which is installed in the starboard overhead window in the aft flight deck (See Sections 4.9.3 and 5.1). During this closure maneuver the Orbiter attitude will be controlled to match the target vehicle attitude, either using inertial hold (IH) or local vertical hold (LVH), with the Orbiter docking/berthing port face aligned to the target vehicle port. Initial phases of the terminal closure will likely use acceleration command modes to establish the basic closing velocity and alignment. Subsequent trim maneuvers will likely employ pulse command modes at or near the minimum impulse levels of the system.

Although specific piloting techniques remain to be developed through simulation activity and on-orbit flight experience, the above concepts are representative of the current best insights into the terminal closure operation. It may be further noted that certain target vehicle characteristics tend to influence the preferred closure path. For example, V-bar ($\bar{V}$) approaches (along the orbit velocity vector) are preferred for activity stabilized target vehicles because of the ease with which initial station keeping operations can be performed prior to the terminal closure maneuver. On the other hand, R-bar ($\bar{R}$) approaches (along the earth radius vector) are preferred for lightly damped, gravity gradient stabilized target vehicles. This approach utilizes orbit mechanics effects to assist in final braking maneuvers thus reducing plume induced motions on the target vehicle. Thus, project related factors must be considered in resolving overall system requirements imposed by the revisit requirement.

The ranges of attitude rates and dead bands and translation ΔV's currently specified for the Orbiter are summarized in Table 4.1-2.

Table 4.1-1. On-Orbit RCS Control Mode

	Mode	Manual or Auto		Selectable per Axis		Vernier or Primary RCS		Comments
		M	A	Yes	No	V	P	
ROTATION	Accelerations command	X		X		X	X	Continuous acceleration per selected axis while RHC out-of-detent; otherwise, free drift
	Pulse command	X		X		X	X	Single pulse per selected axis per deflection of the RHC: pulse size selectable (down to 8 msec)
	Discrete rate command/attitude hold	X		X		X	X	While RHC out-of-detent, rotates at selectable maneuver rate; otherwise maintains attitude hold
	Automatic (discrete rate command attitude hold)[a]		X		X	X	X	Same as above; however accepts G&N attitude and rate commands (interruptable by RHC deflection)
TRANSLATION	Acceleration command	X		X			X	Continuous acceleration per selected axis while THC out-of-detent
	Pulse command	X		X			X	Single pulse per selected axis per deflection of the THC; pulse size selectable (down to 8 msec)

[a]G&N commands while in automatic rotation mode provide functions of (1) tracking (e.g., local vertical), (2) barbecue (thermal control), (3) attitude reorientation maneuvering, and (4) possible steering commands during automatic translation

Reference: NASA 1980(b), Vol. I

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Table 4.1-2. Dapload Parameters*

Parameter	Unit	Each Dapload contains sep. values for PRIM/VERN RCS	Each Dapload contains sep. values for ROLL/PITCH/YAW	Values automatically updated, when vehicle inertia diagonal updated	Range
Threshold for open-loop RCS off-axis coupling compensation firing	deg/sec	yes	no	no	PRIM: .00 to .99 VERN: .000 to .999
Attitude deadband, RCS	deg	no	yes	no	0.010 to 40.000
Maneuver rate, automatic and manual discrete rate RCS rotation	deg/sec	yes	no	no	PRIM: 0.050 to 2.000 VERN: 0.002 to 1.000
Pitch acceleration option (nominal, low using fwd RCS, low using aft RCS)	none	no	no	no	1 to 3
Yaw acceleration option (nominal, low using fwd RCS, low using aft RCS)	none	no	no	no	1 to 3
Payload configuration (payload not extended, any of five payload-extended configurations)	none	no	no	no	0 to 5
Rate deadband, RCS rotation	deg/sec	yes	no	no	PRIM: 0.2 to 5.00 VERN: 0.01 to .500
Manual RCS rotation pulse size	deg/sec	yes	no	no	PRIM: 0.04 to 1.00 VERN: 0.001 to 0.500
Manual RCS translation pulse size	ft/sec	no	no	no	0.01 to 5.00
Nominal average control accel, RCS rotation, payload not extended, primary RCS	deg/sec^2	no	yes	yes	TBD
Nominal avg. control accel, RCS rotation, payload not extended, vernier RCS	deg/sec^2	no	yes	yes	TBD

Reference: Rockwell 1980(a)

*Parameters to be loaded into digital autopilot system.

4.1.3 Orbiter Docking Capabilities

Current Orbiter requirements call for the capability of crew-controlled docking and undocking to other compatible orbiting elements during either daylight or darkness conditions. Further, the Orbiter will be the active vehicle in terms of both closure trajectory management and docking mechanisms. Impact attenuating and active latching elements will be on the Orbiter side of the docking interface. Figure 4.1-1 illustrates the current design concept for these devices. They are to be incorporated in a standard docking module which is installed aft of the forward bulkhead of Orbiter payload bay as illustrated in Figure 4.1-2. (Note: These illustrations combine information from MCR 5546 (Rockwell 1980(e)) and JSC-07700 (NASA 1980(A)) at the time of preparation of this report. Alternative docking adapter concepts have been studied for use in conjunction with large power modules and large space structures. An example is a McDonnell Douglas (1980) study of an advanced science and application space platform. This study includes example analyses of dynamic interaction forces at time of docking. Studies were also underway at the time to divide the docking module hardware into three modular parts, a tunnel adapter, a docking module and a support structure, as shown in Figure 4.1-3. The proposed geometry would be altered slightly from that of Figure 4.1-2, as indicated in Figures 4.1-4 and 4.1-5. The upper, docking module structure could also be replaced with an open framework concept, similar to that proposed for a 25-kW power system reference configuration (Figure 4.1-6), when unmanned operations are considered (no shirtsleeve transfer of crew to habitable module). Further implications of docking module concepts on crew EVA operations are discussed in Section 4.8.1.

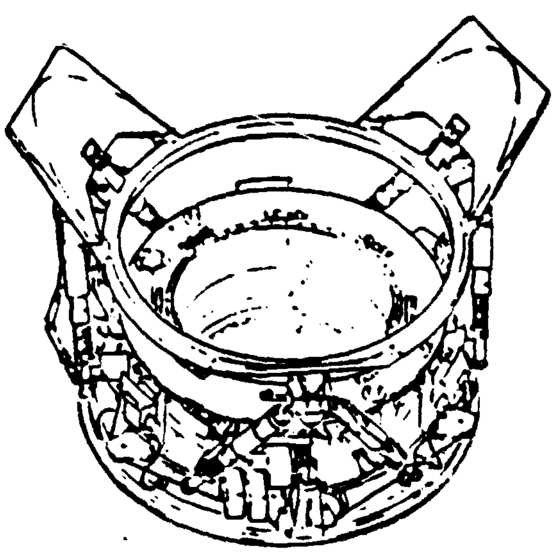


Figure 4.1-1. Docking/Berthing Port Mechanism Concept

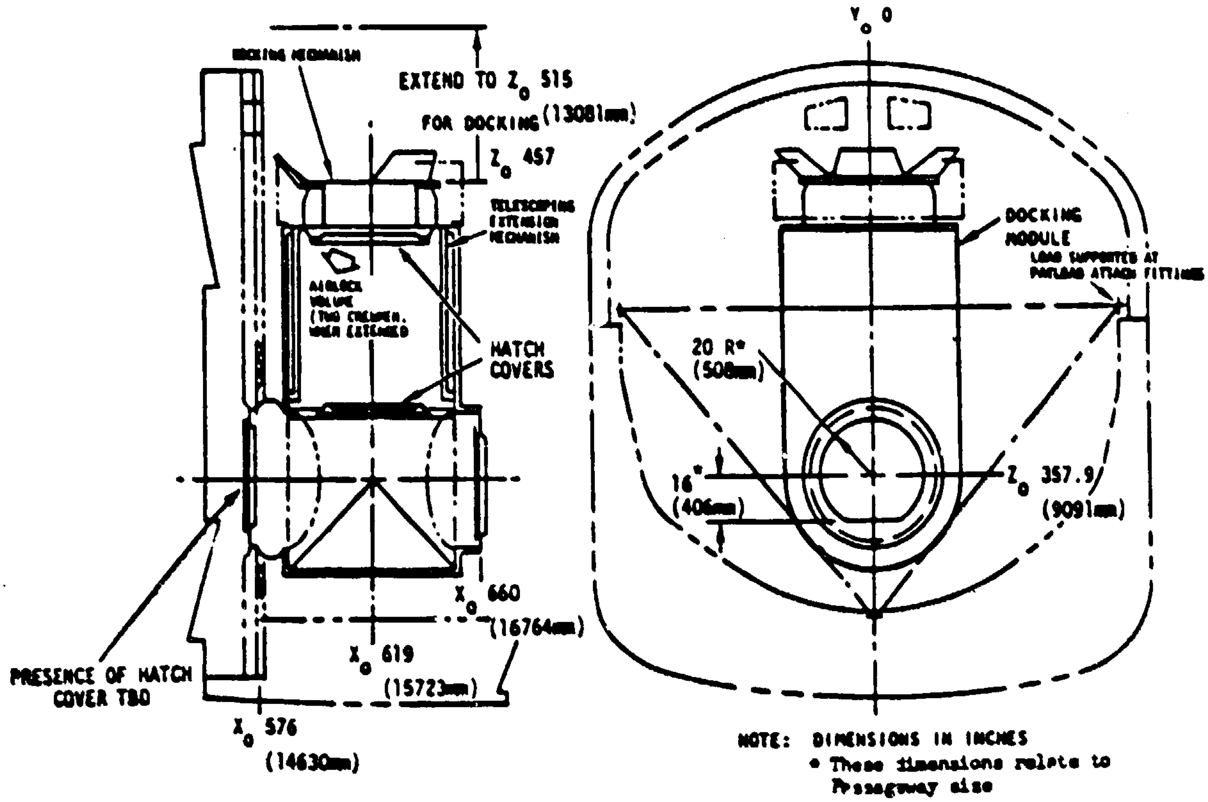


Figure 4.1-2. Baseline Docking Module Concept for Orbiter

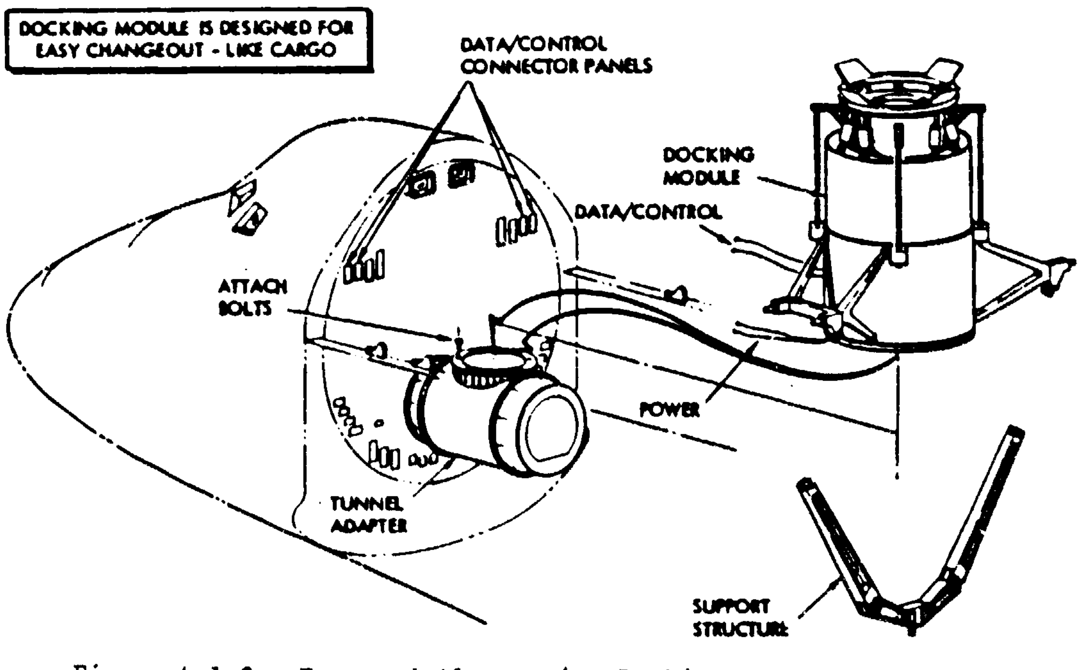


Figure 4.1-3. Proposed Alternative Docking Module Concept

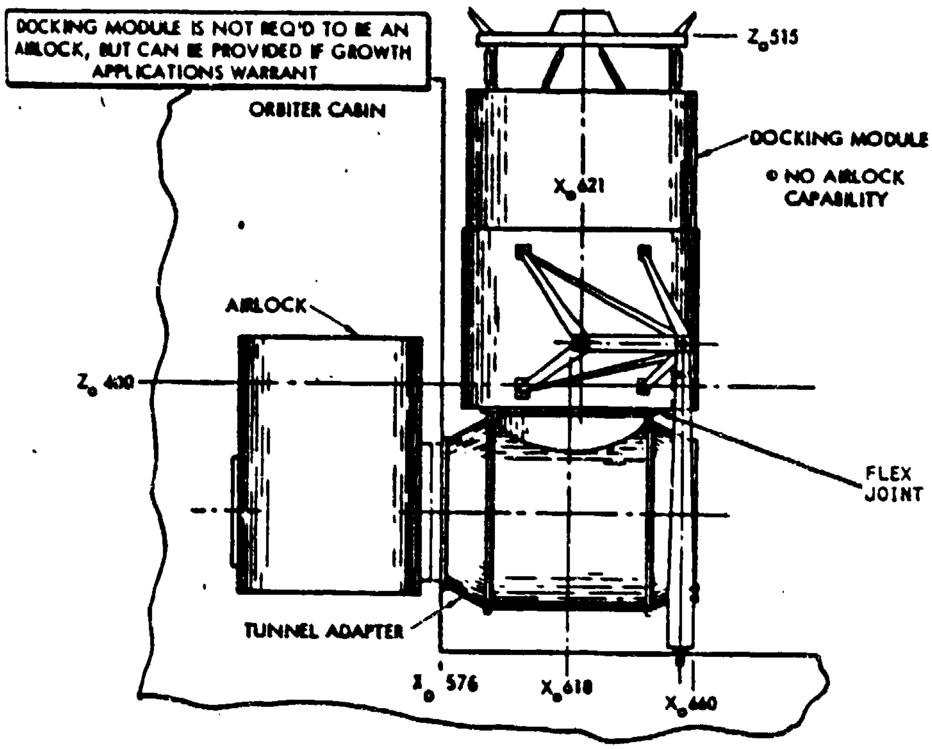


Figure 4.1-4. Nominal Orbiter Arrangement Concept for Proposed Alternative (Sideview)

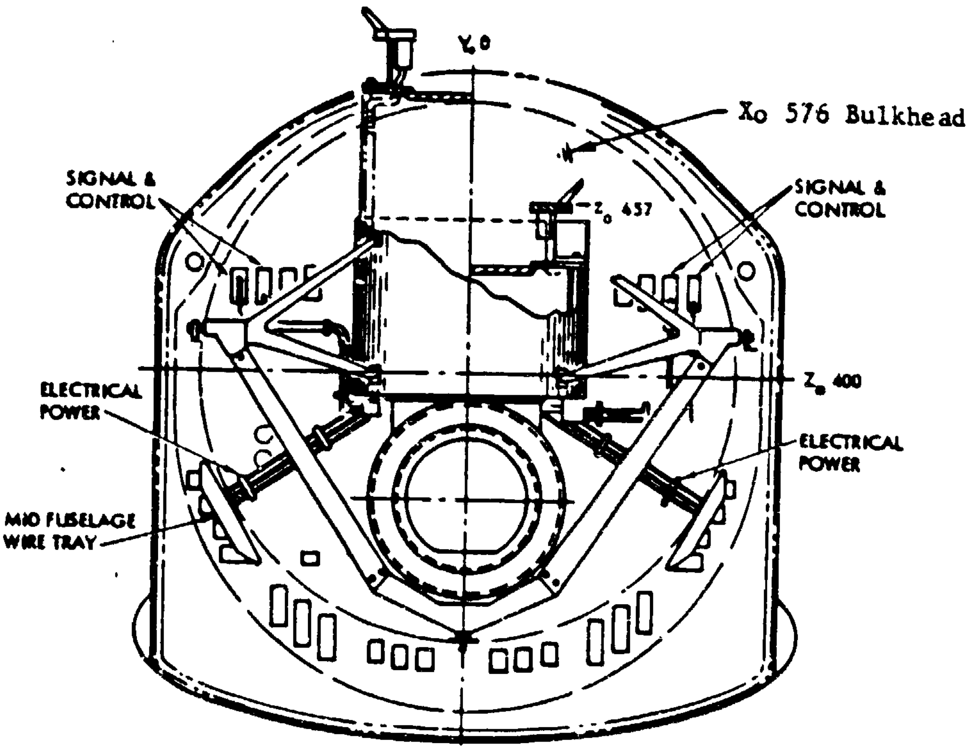


Figure 4.1-5. Utilities Interface Routing - View Looking Forward at Proposed Alternative Docking Module

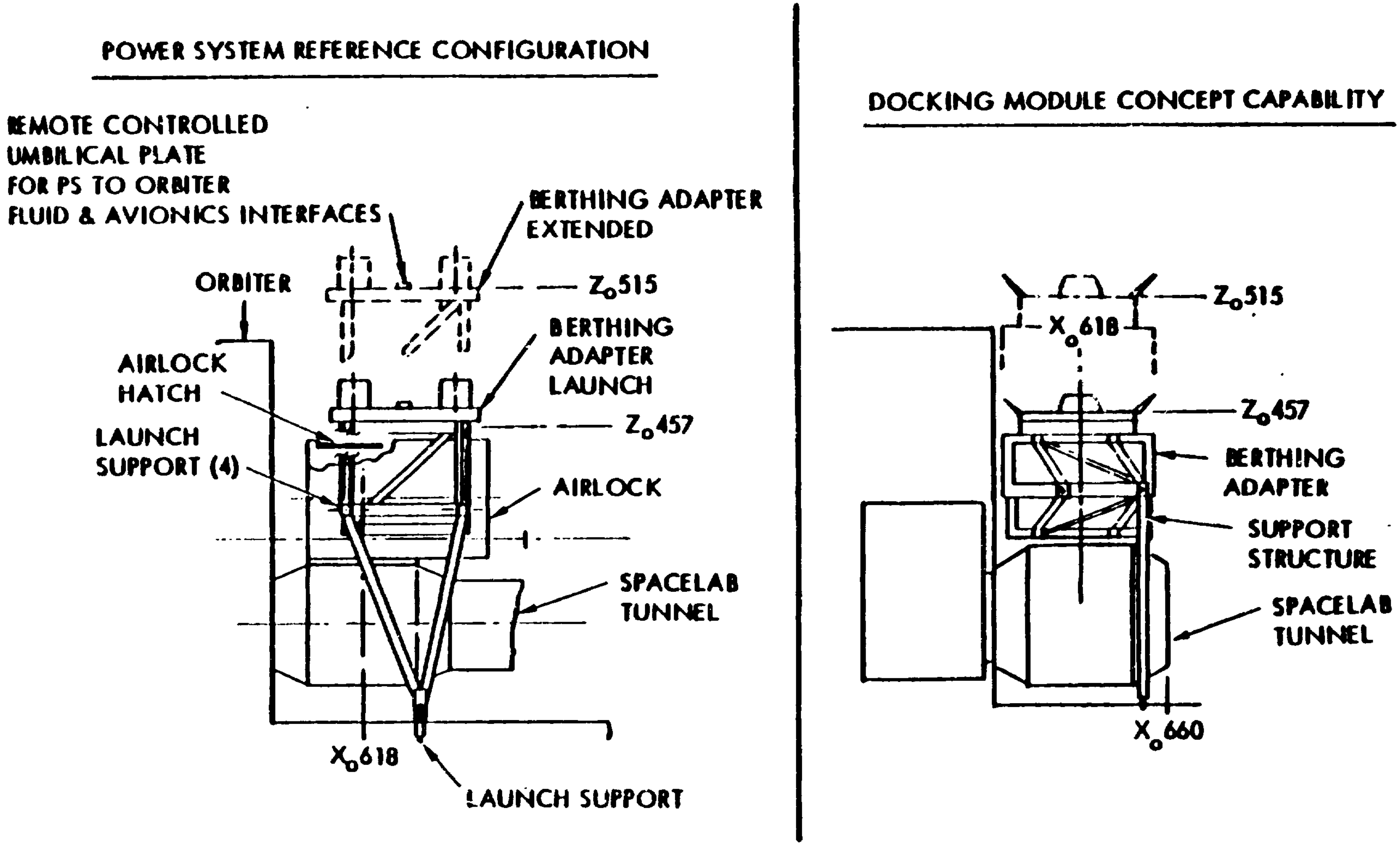


Figure 4.1-6. Alternative Docking System for Unmanned Operations

However, it is expected that the alternative hardware will not affect significantly the following comments regarding attitude control during docking and berthing. Within this general concept, the Orbiter vehicle shall cause "capture" of the payload and after initial docking contact, continued force may be applied to the vehicle until "hard" capture is completed and verified. Table 4.1-3 specifies the range of docking contact condition currently specified for the Orbiter (Para. 3.2.3.6.4.1, Orbiter End Item Spec MJ070-0001-1B). Concept studies covering a variety of space systems have investigated docking requirements and design implications using slightly different values for the contact conditions. An example is shown in Table 4.1-4. The envelope of contact conditions presented in these tables is representative of the likely final Orbiter capabilities and is suitable for preliminary analyses of docking type revisit operations.

Table 4.1-3. Orbiter/Payload Contact Conditions

Parameter	
Relative contact velocity (-Z)	0.5 fps 0.05 fps - lower limit
Relative lateral velocity ($\dot{X}$, $\dot{Y}$)	(0. $\pm$ 0.1) fps
Relative angular velocity (3 axes)	$\pm$1.0 deg/sec about any axis
Relative lateral misalignment (Orbiter X, Y)	0 $\pm$ 0.5 ft
Relative angular misalignment (Orbiter pitch, roll)	0 $\pm$ 5 deg about each axis
Relative rotation misalignment (Orbiter yaw)	0 $\pm$ 7 deg

Reference: Orbiter Vehicle End Item Specification for the Space Shuttle System, Part I, Performance and Design Requirements, Spec. No. MJ070-0001-1B

Table 4.1-4. Example of Docking Contact Conditions

Axial Closing Velocity	0.16 - 0.5 fps
Lateral Velocity	$\leq$ 0.2 fps
Angular Velocity	$\leq$ 0.6 deg/sec
Lateral Misalignment	$\leq$ 0.75 ft
Angular Misalignment	$\leq$ 5.0 deg (roll) $\leq$ 6.0 deg (pitch/yaw)

4.1.4 Orbiter/RMS Berthing Capabilities

An alternative to the direct docking approach for revisit operations is the use of the Remote Manipulator System (RMS) to perform the final physical attachment between the Orbiter and the target vehicle. With this concept the terminal closure is performed in much the same manner as for direct docking, that is, with $\bar{V}$ or $\bar{R}$ approaches using closing vehicles along the Orbiter Z-body axis. However, in this case a braking maneuver to null the main closing velocity is performed when the target vehicle is within the reach envelope of the RMS (See Section 4.5.2). Following the braking maneuver, the RMS is used to grasp and remove residual motions between the Orbiter and the target vehicle and then to reposition, orient and berth the target vehicle to the Orbiter.

The currently defined capabilities of the RMS to track, grasp and arrest the residual motions of a target vehicle are summarized in Table 4.1-5. These capabilities are based on RMS operational scenarios involving the deployment, handling and retrieval of payloads (See Section 4.5) and not on the problem of berthing the Orbiter to large free-flying systems. Thus, additional analyses of the berthing operations and procedures must be performed. Many project unique factors must be considered as illustrated in the following simplified example problem.

Table 4.1-5. RMS Capabilities for Berthing

Passive Stabilized Target	$\leq$15 inches $\leq$0.05 inches/sec
Active Stabilized Target	
Deadboard	$\leq$1 deg (all axes)
Rate	$\leq$0.1 deg/sec (all axes)
Graple Motion	$\leq$3.0 inches
Relative Velocity Orbiter/Target	$\leq$0.1 fps

Reference: RMS Users Conference, 30 May - 1 June 1979.

Example Motion Arrest with the RMS

The fundamental revisit "problem" is depicted in Figure 4.1-7. It is a two-body problem with each body having its own mass properties and six-degree-of freedom (DOF) motion. The Shuttle must be flown in a fail-safe approach trajectory that eliminates the possibility of an inadvertent collision with the target system. This requires precise ΔV control of the terminal closure path while simultaneously maintaining line-of-sight visibility to the target. This precision control is hampered by attitude control coupling with translation control due to RCS thruster geometry and minimum impulse size.

The target vehicle will typically be librating with amplitudes and rates that could be significantly affected by plume impingement from proximity RCS firings. The RMS must reach out, track, and engage the target structure and then arrest its relative motion. The capability of the Orbiter/RMS combination to perform this sequence is greatly affected by the mass properties, hook-up geometry, and dynamic motion of the large area system to be engaged.

As a preliminary step in looking at this problem, a simplified model of the engagement dynamics was formulated by Rockwell International (1979a). This model and the resulting safe engagement requirements are shown in Figure 4.1-8. In this model, the Orbiter was assumed to represent an infinite mass, thereby reducing the situation to a one-body problem. The target mass properties and engagement geometry are shown in the figure. It was further assumed that the translation component of relative motion (X) must be arrested within 6 meters (20 feet). This allows ample time (100 to 200 seconds) for the RMS to track and grasp the target and then stop it before it travels out of reach. Similarly, the rotational motion was assumed to be arrested within a rotational angle of 30 degrees, thereby allowing adequate clearance between the Orbiter and the target structure. Further, these "stopping" actions were constrained by the 67 newton (15-pound) tip force and 800 newton-meter (600 ft-lb) joint hinge moment limits associated with the RMS.

The resulting safe closure conditions are shown at the lower right of the figure. Relative X's range from 0.03 to 0.06 mps (0.1 to 0.2 fps) in combination with relative 0's of 0.05 and 0.01 deg/sec. These are based on a single DOF Model along with many other assumptions. Allowance for additional degrees of freedom will likely reduce these limits. Thus, careful attention must be focused on the librating motion of the partially completed space platform.

4.1.5 Construction Project Stabilization Between Visits

Both the direct docking and the RMS berthing techniques require reasonably stabilized target vehicles to meet the revisit requirements. Librating motions induced by various disturbance environments must be reduced to acceptable levels for safe berthing or docking operations with the Shuttle orbiter. Many space construction projects will be of sufficient size/magnitude that installation of their functioning attitude control system will not be possible on the first construction mission. Thus, some auxiliary system is required which may either be part of the construction system or a temporary unit installed on the project system.

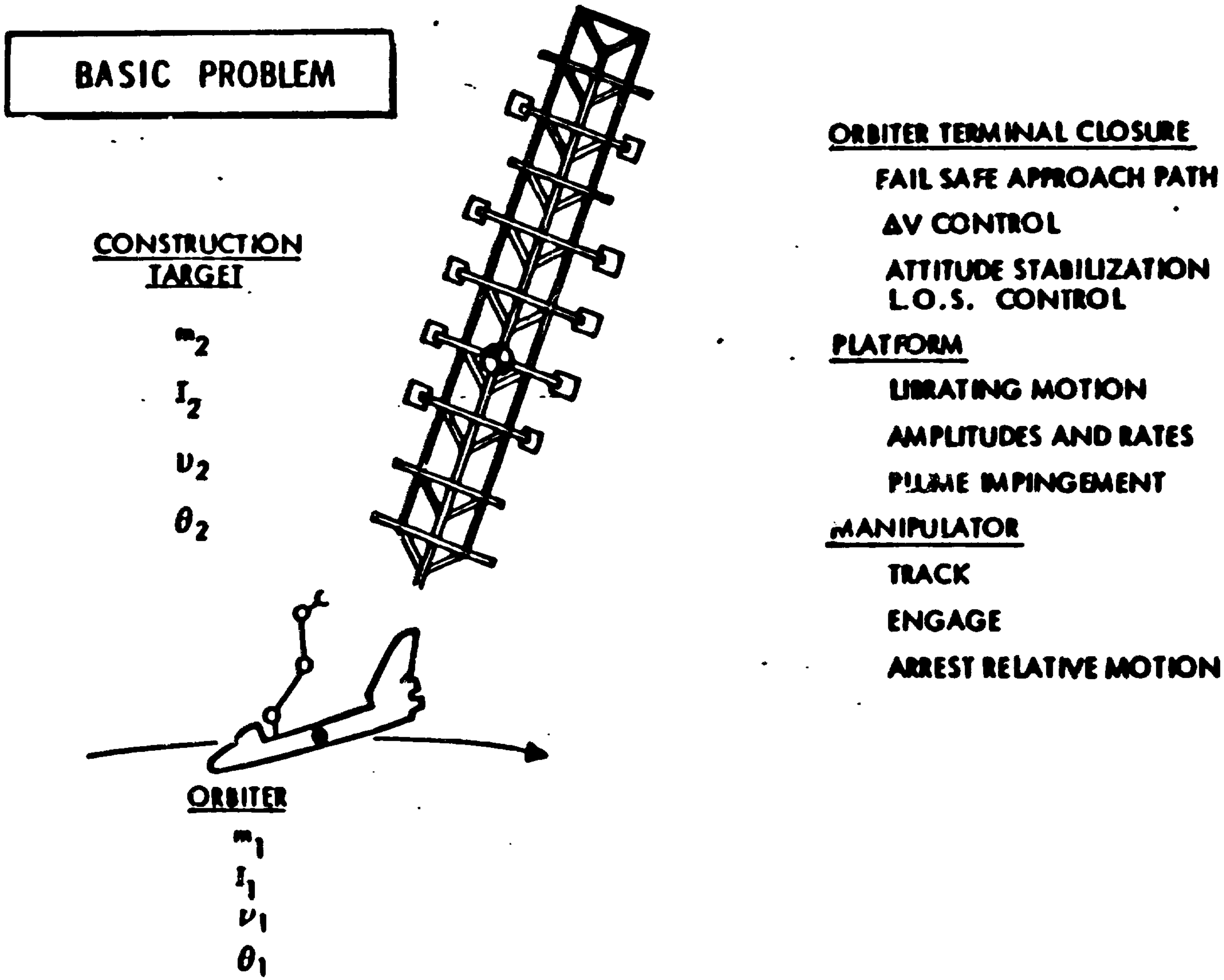


Figure 4.1-7. The Revisit/Berthing Problem

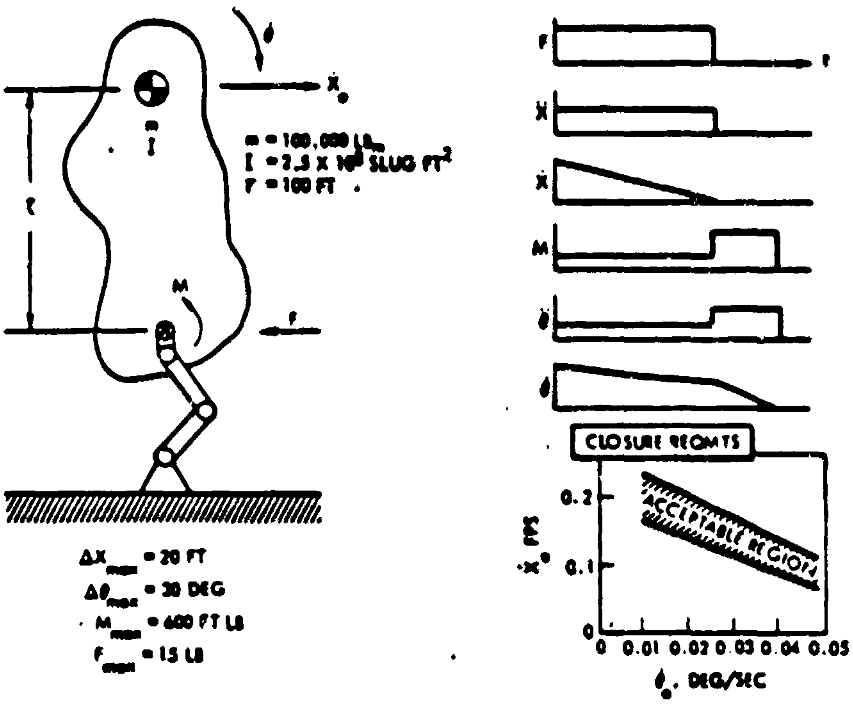
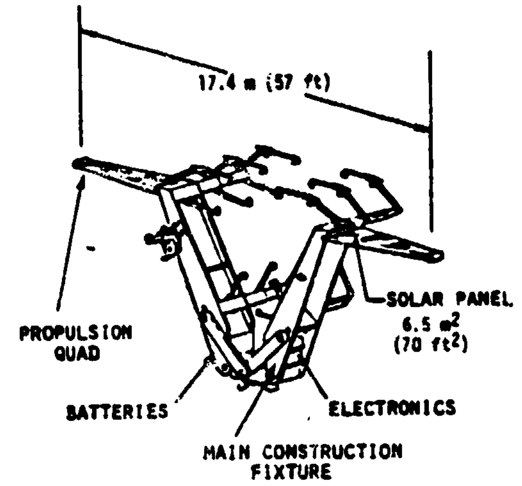


Figure 4.1-8. Safe Terminal Closure Criteria

The choice of an auxiliary system concept for meeting the intended stabilization requirements between construction visits is dependent upon many project-related factors. These include project size, shape and mass properties, the relative location of the revisit interface with respect to its c.g. and mass properties and the structural integrity of the partially completed construction project at the end of each construction mission. Possible sensitivities to plume impingement effects from proximity firings of orbiter RCS thruster must also be considered. Depending upon these factors, the revisit stabilization concepts could range from simple, low-cost libration damping concepts to full capabilities attitude control systems in which designated orientation could be commanded. An operational advantage of the designated attitude capability is that all terminal closures by the orbiter for the revisit operations could utilize a preferred standardized V-bar or R-bar approach path. The low cost libration damper concept would likely require a different thermal approach path for each libration damped trim attitude associated with the configuration mass properties at the end of each construction mission.

An example of a simple libration damper concept is illustrated in Figure 4.1-9. This is a cold gas system sized to support the construction of a 400 foot linear tri-beam platform with the build up mass properties and thruster geometry as presented in Figure 4.1-10. Three construction flights are nominally required to complete the construction of this example platform. The cold gas capacity in this example system is sized to damp 0.1 degrees per second libration rates (threshold of tumbling motion) about all three axes prior to each revisit. The total propellant of 128 lbs (two 64 lb tanks) allows one



COLD GAS SYSTEM
- 0.5 LB_F THRUSTERS
- 128 LB N_2 PROPELLANT
- 25-IN.-DIA. TANKS
- 5 THRUSTERS PER QUAD
- PULSE WIDTH, 0.2-50 SEC

POWER SYSTEM
- 100 W BATTERY/SOLAR ARRAY
- 100 W BATTERY (ONE DAY)

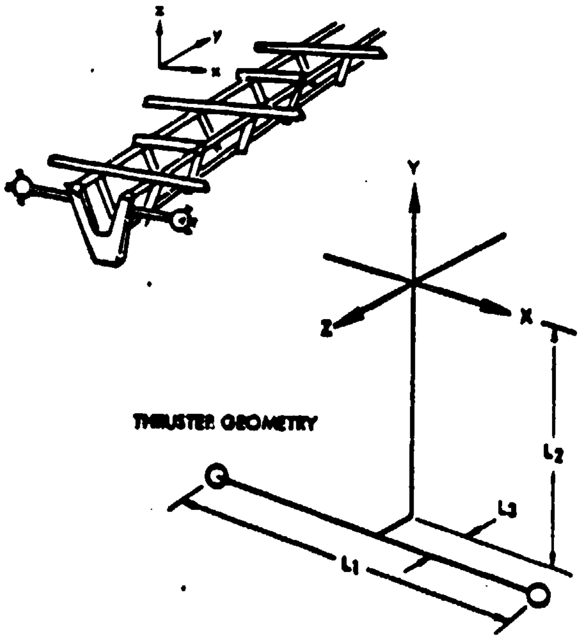
TT&C SYSTEM
- FOUR FLIGHT MODES
  (1) READY ALARM (3) HOLD DAMPING
  (2) ACTIVE DAMPING (4) RENDEZVOUS

RENDEZVOUS AIDS
- TRANSPONDER (TT&C)
- LIGHTS

RMS GRASP FIXTURE

- 3 CONSTRUCTION FLIGHTS PLUS CONTINGENCIES AND MARGIN
- GRAVITY-GRADIENT CAPTURE MOMENTUM (0.1 DEG/SEC PER AXIS)
- RESIDUAL RATES LESS THAN 0.01 DEG/SEC

Figure 4.1-9. Example Cold Gas Libration Damper Concept



MASS PROPERTIES

PARAMETER	CASE 1	CASE 2	CASE 3
WT (LB)	4000	21000	63000
Ixx (Slug Ft^2)	700	14×10^6	84×10^6
Iyy (Slug Ft^2)	3700	7.5×10^4	1.2×10^6
Izz (Slug Ft^2)	2900	14×10^6	85×10^6
L_1 (Ft)	57	57	57
L_2 (Ft)	0	184	325
L_3 (Ft)	10.4	-4.7	-0.2

CASE 1: FIXTURE ALONE

CASE 2: END FLIGHT #1

CASE 3: END FLIGHT #3

Figure 4.1-10. Libration Damper Configuration Characteristics contingency revisit (a total of four construction flights) plus a 100 percent margin. The thrusters and pulse width range are sized to fully damp the libration motions within one day and to produce residual motions less than 0.01 degrees per second. A solar array/battery electrical power system sized to provide 100 watts continuous power and 200 watts peak power for one day is included. This provides power for the damping electronics, command and control and rendezvous sides.

4.1.6 Orbiter Plume Effects

In addition to the effects on the orbiter motion, each RCS thruster creates a plume with potential effects on nearby large structures. These effects can be of significance to plans for docking and separation, on-orbit control of orbiter/structure combinations and contamination or damage of sensors, solar arrays or other components of large space systems.

The full range of potential effects are highly dependent on factors of mass distribution, geometry, and properties of materials too complex and extensive to consider in depth here. However, a brief summary of plume characteristics is presented here to suggest the magnitude of possible effects. The general locations of RCS thrusters are shown in Figure 4.1-11.

The identification and location of RCS thrusters and the forces which they apply along the various axes are presented in Section 4.2 (Figure 4.2-1 and Table 4.2-2), which deals with attitude control of combined orbiter and structure assemblies. Designers of large space structures may be concerned about orbiter plume contamination, plume pressures, plume flow rates and plume gas temperatures. Although the general issue of contamination is discussed in Section 4.14 of this report, details of RCS plume contributions to contamination are described here for convenience and coherence of presentation. The following graphical data appear in Section 11 of ICD 2-19001 [NASA, 1980 (a)].

4.1.6.1 Primary RCS Engine Plumes

The engine plume characteristics described in the following figures corresponds to the effective engine thrust vector, rather than the engine/nozzle centerline due to the effects of nozzle scarfing to conform to the orbiter outer moldline. Typical, plume, constant-density contours generated by a single engine operating in a vacuum are shown in Figure 4.1-12. Figure 4.1-13 shows typical, contaminants (liquid/solid phase), constant mass-flux-rate contours throughout the plume shown in Figure 4.1-12. With the nominal RCS nozzle configuration (ε defined in Table 4.2-2) and the computed non-gas (liquid/solid) phase flow rate, the contaminants plume has the maximum angular displacement (ϕ_M) defined in Table 4.2-2, and it lies entirely within the gas-phase envelope. The distribution of contaminants in this plume is such that the largest mass particles are contained within a cone defined by the effective nozzle expansion angle ϕ_N. Composition of the contaminants is defined in Paragraph 10.6.3 of ICD 2-19001 [NASA, 1980 (a)]. Figure 4.1-14 shows the nominal gas static temperature, as a function of the density constant K_ρ, throughout the plume shown in Figure 4.1-12. Figure 4.1-15 shows the nominal plume gas Mach number, as a function of the density constant K_ρ, throughout the plume shown in Figure 4.1-12. Figure 4.1-16 shows the nominal relationship between gas static pressure and impingement force on a normal flat plate (computed by means of the Diffuse

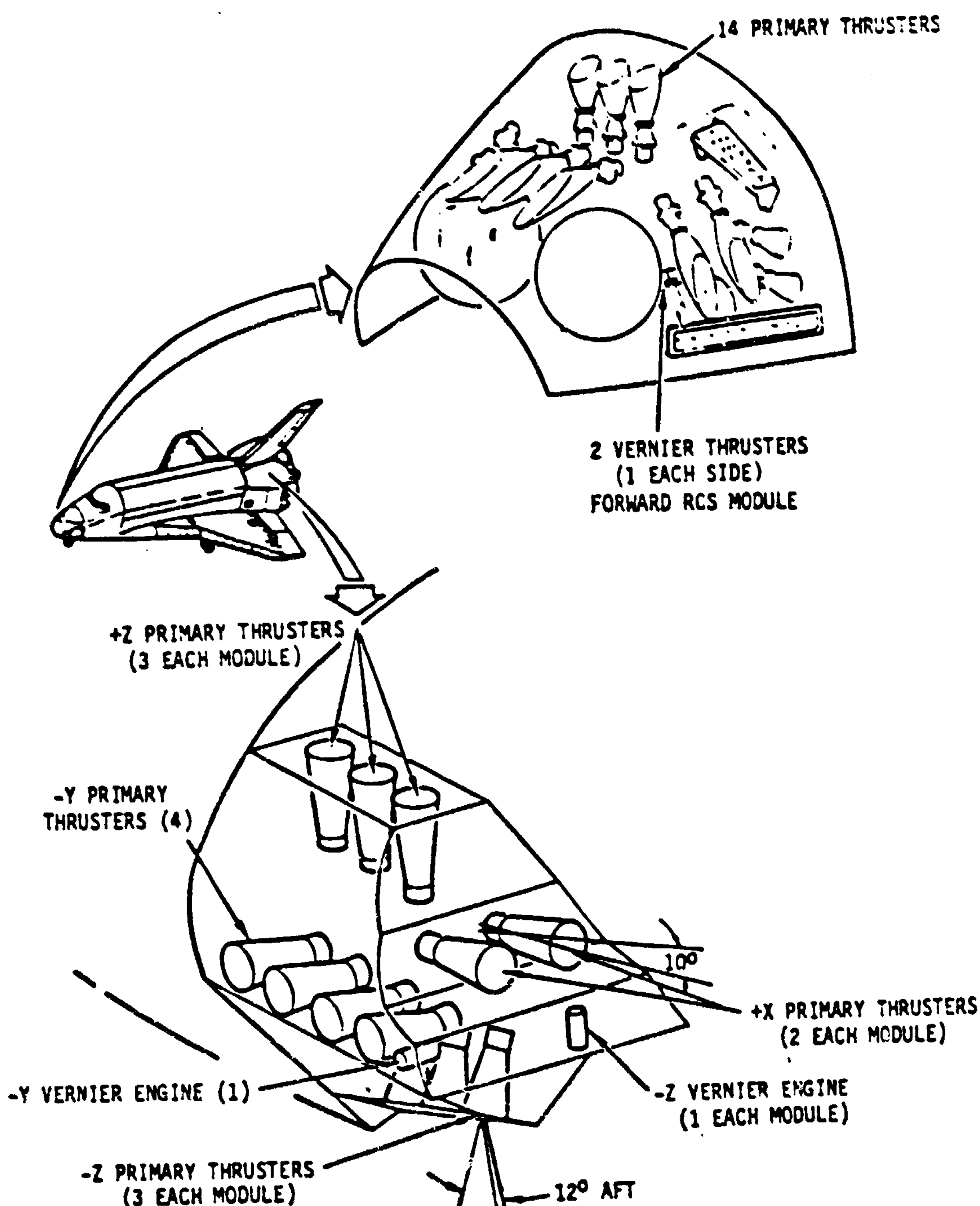
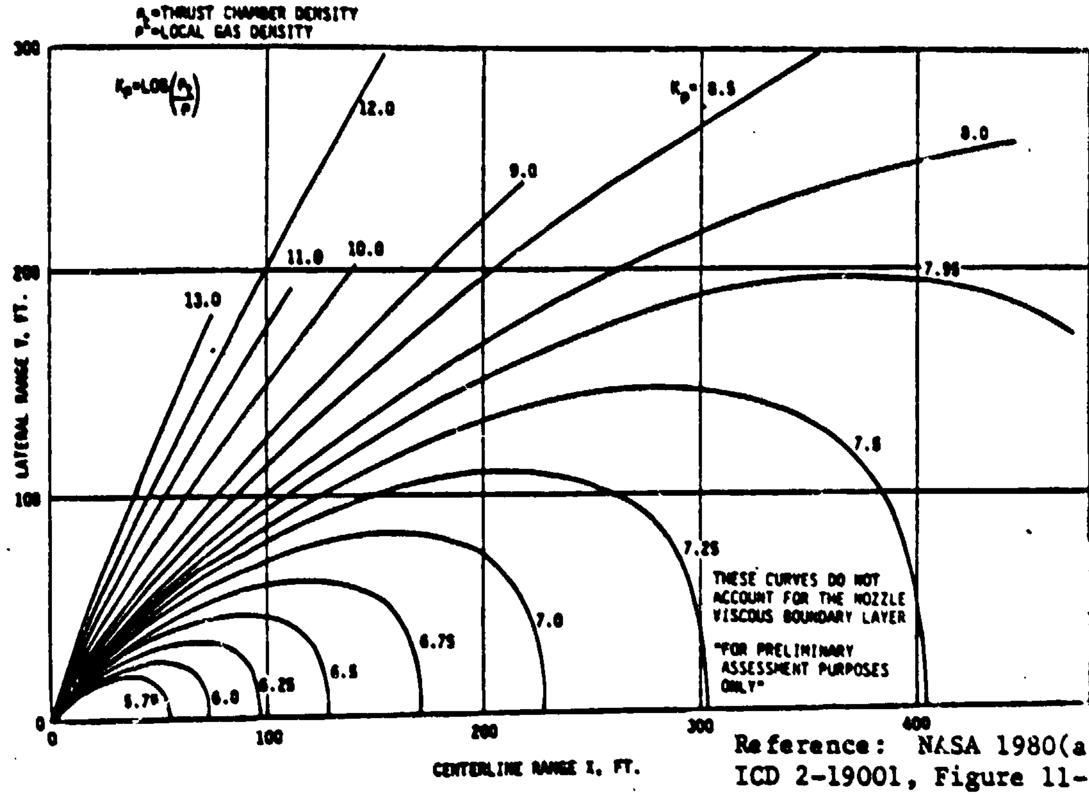
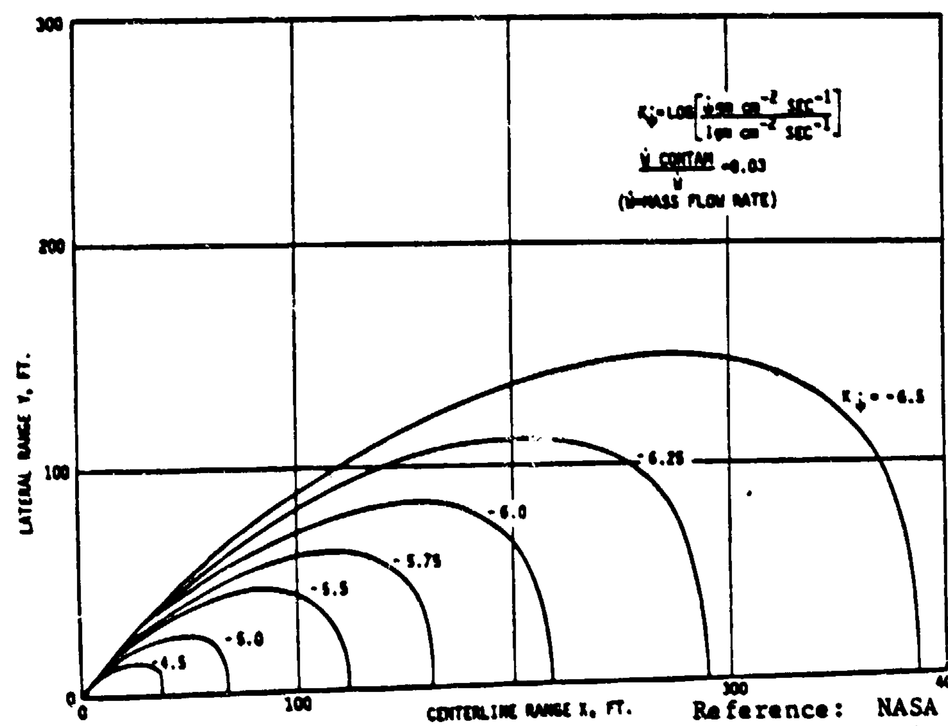


Figure 4.1-11. RCS/VRCS Thruster Locations
and Orientations



Reference: NASA 1980(a), ICD 2-19001, Figure 11-16

Figure 4.1-12. RCS plume Constant Contours - Kρ



Reference: NASA 1980(a), ICD 2-19001, Figure 11-17

Figure 4.1-13. RCS Contaminants Constant Mass-Flux-Rate Conturs

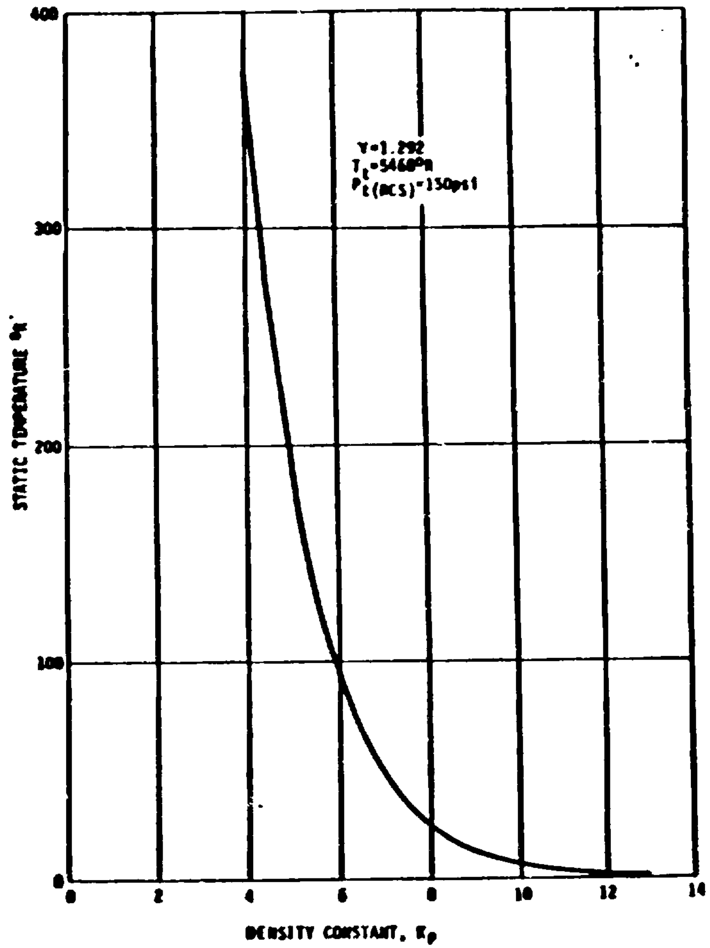


Figure 4.1-14. RCS Plume Static Temperature (Deg R) as a Function of Density Constant, K_ρ

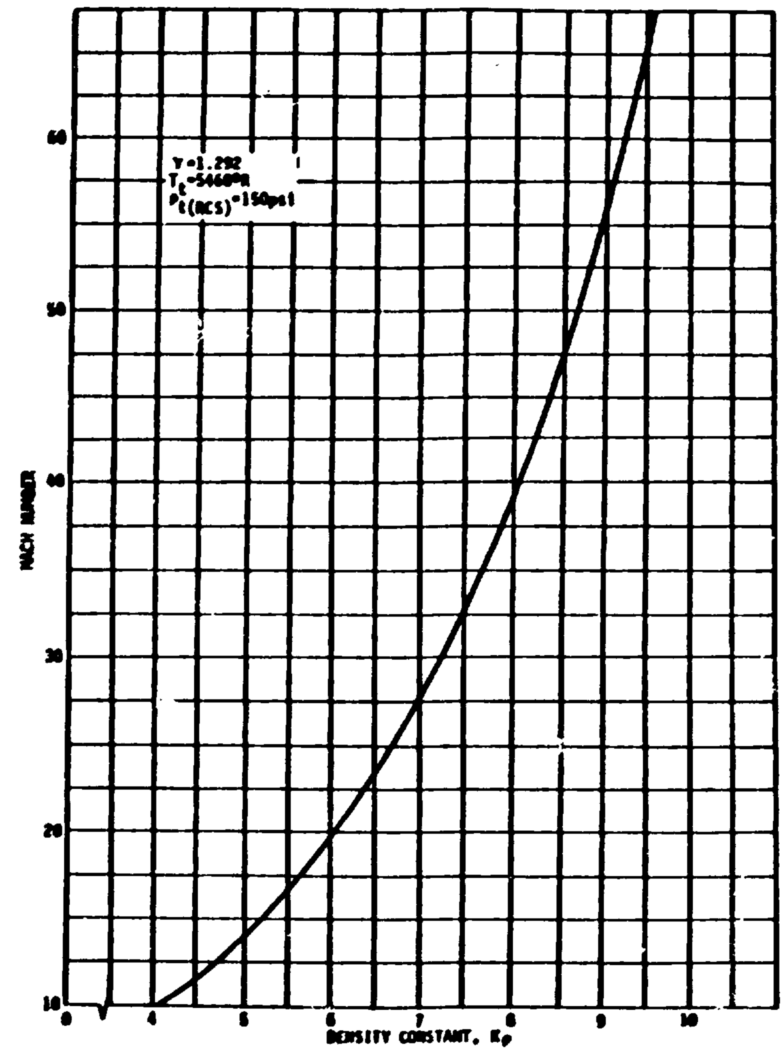


Figure 4.1-15. RCS Plume Mach Number as a Function of Density Constant, K_ρ

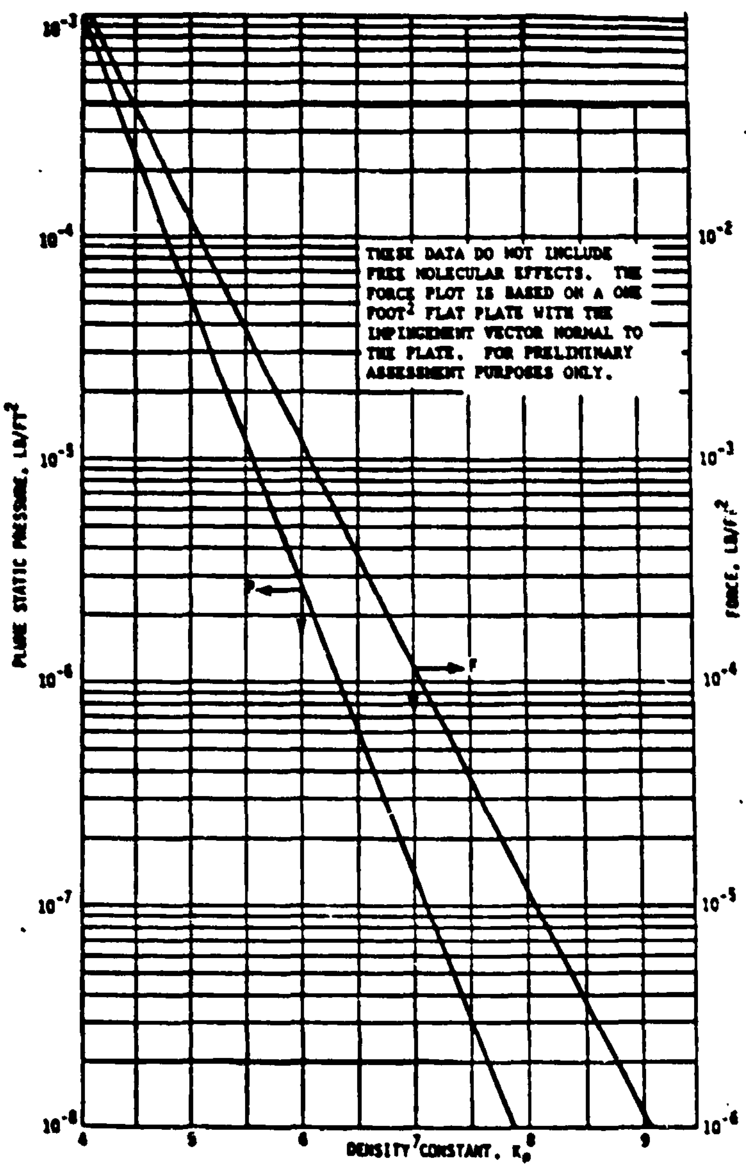


Figure 4.1-16. RCS Plume Static Pressure and Impingement Force as a Function of Density Constant, K_ρ

Newtonian Gas Dynamic Interaction Model), both as functions of the density constant K_ρ, throughout the plume shown in Figure 4.1-12. The referenced figures were generated by source flow analysis, represent single engine only (multiple engine effects not considered), valid for continuum flow only, free molecular flow not considered, and are presented for preliminary assessment use only.

4.1.6.2 Vernier RCS (VRCS) Engine Plumes

The engine plume characteristics described in the following figures corresponds to the effective engine thrust vector, rather than the engine/nozzle centerline due to the effects of nozzle scarfing to conform to the orbiter outer moldline. Typical, plume, constant-density contours generated by a single engine operating in a vacuum are shown in Figure 4.1-17. Figure 4.1-18 shows typical, contaminants (liquid/solid phase), constant mass-flux-rate contours throughout the plume shown in Figure 4.1-17. With the nominal VRCS nozzle configuration (ϵ, defined in Table 4.2-2) and the computed non-gas (liquid/solid) phase flow rate, the contaminants plume has the maximum angular displacement (ϕ_M) defined in Table 4.2-2, and it lies entirely within the gas-phase envelope. The distribution of contaminants in this plume is such that the largest mass particles are contained within a cone defined by the effective nozzle expansion angle ϕ_N. Composition of the contaminants is defined in Paragraph 10.6.3 of ICD 2-19001. Figure 4.8-19 shows the nominal gas static temperature, as a function of the density constant $K\rho$, throughout the plume shown in Figure 4.1-17. Figure 4.1-20 shows the nominal plume gas Mach number, as a function of thedensity constant $K\rho$, throughout the plume shown in Figure 4.1-17. Figure 4.1-21 shows the nominal relationship between gas static pressure and impingement force on a normal gas plate (computed by means of the Diffuse Newtonian Gas Dynamic Interaction Model), both as a function of the density constant K , throughout the plume shown in Figure 4.1-17. The referenced figures were generated by source flow analysis, represent single engine only (multiple engine effects not considered), valid for continuum flow only, free molecular flow not considered, and are presented for preliminary assessment purposes only.

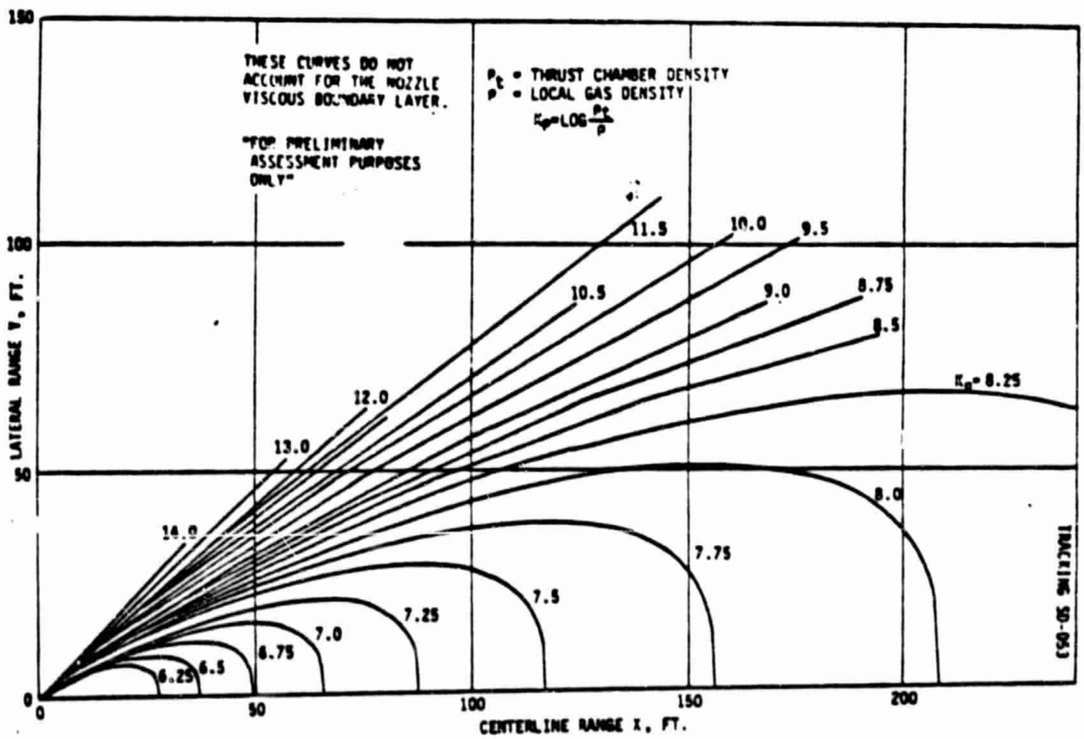


Figure 4.1-17. VRCS Constant Density Contours, $K\rho$

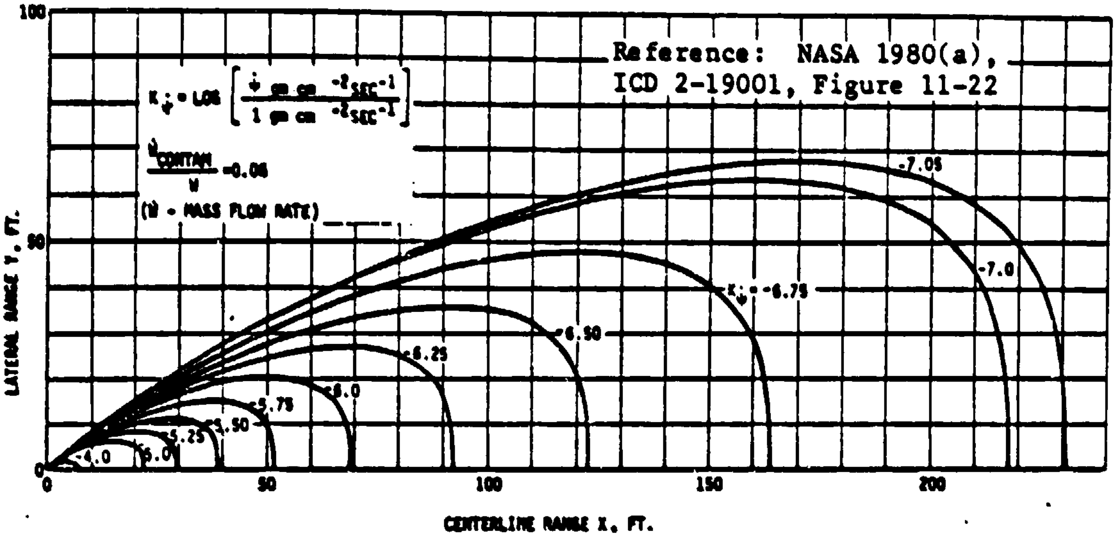


Figure 4.1-18. VRCS Contaminants Constant Mass-Flux-Rate Contours

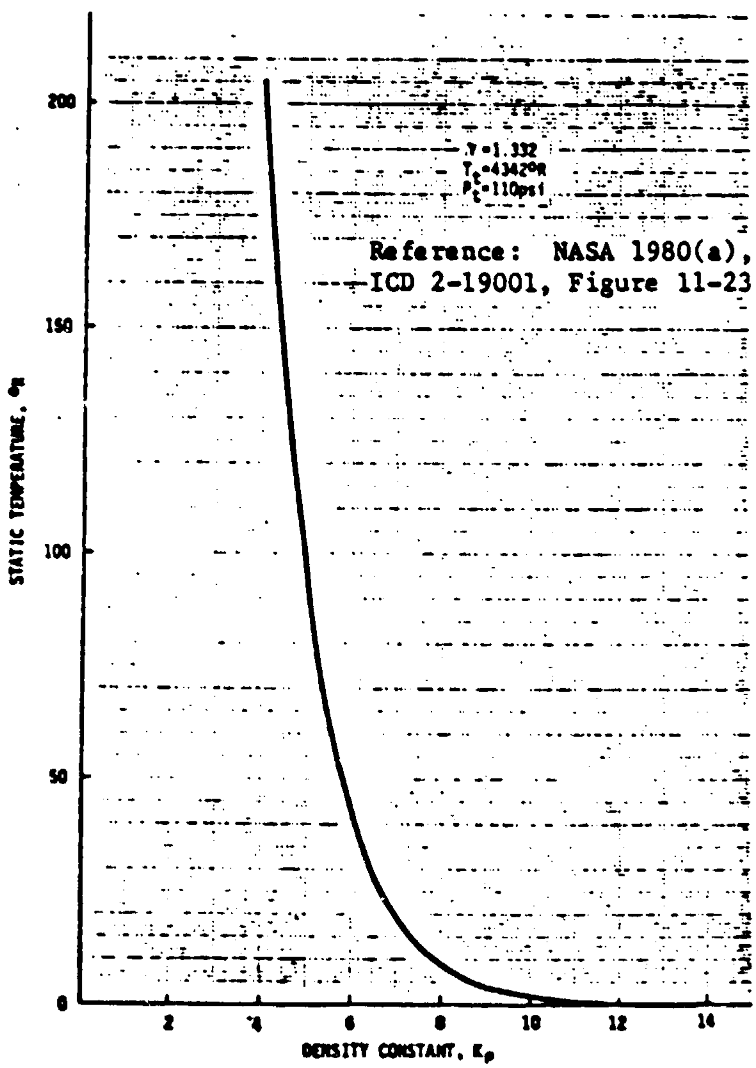


Figure 4.1-19. VRCS Plume Static Temperature Vs. Density Constant, Kρ

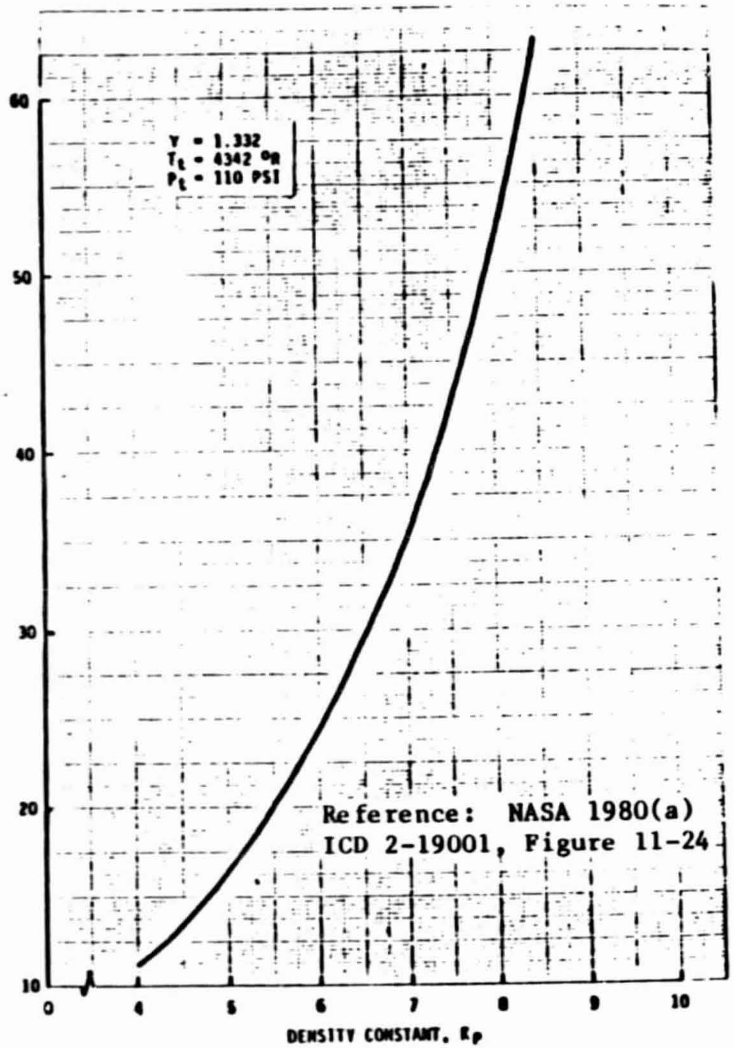


Figure 4.1-20. VRCS Plume Mach Number Vs. Density Constant, Kρ

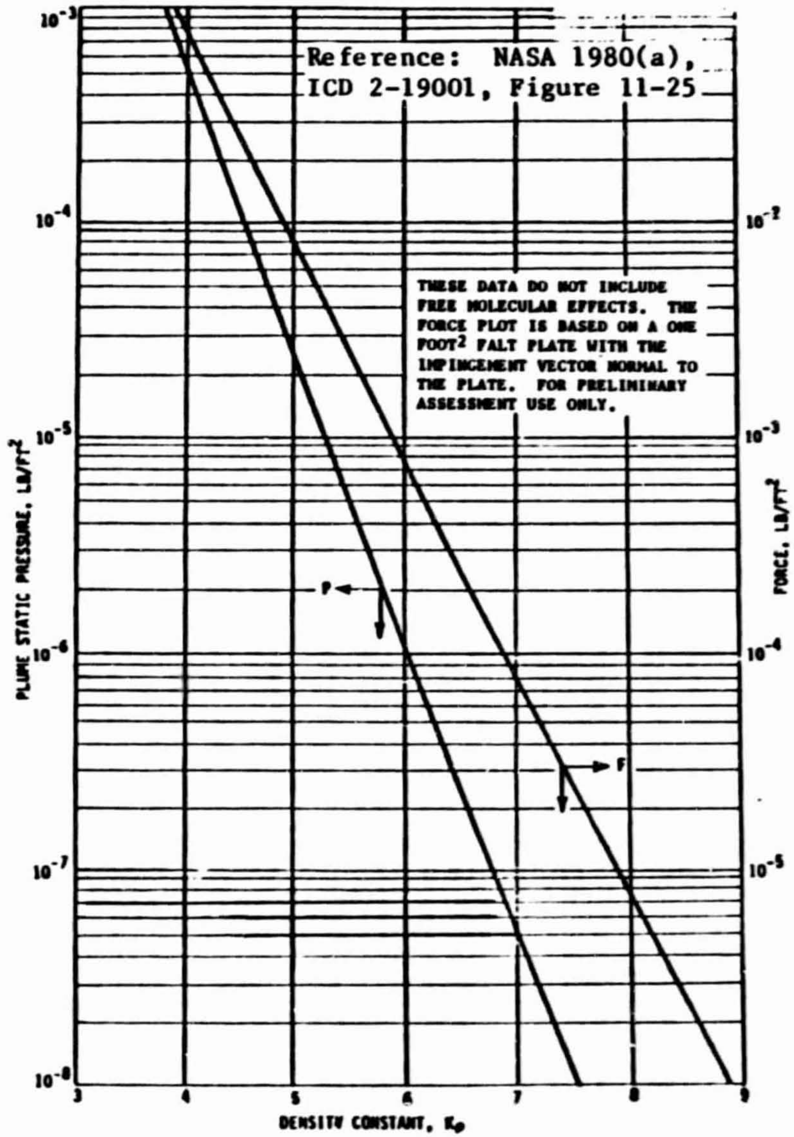


Figure 4.1-21. VRCS Plume Static Pressure and Impingement Force Vs. Density Constant, Kρ

4.2 ATTITUDE CONTROL DURING CONSTRUCTION AND ORBITER-TENDED OPERATIONS

Construction in space of large spacecraft will result in large changes in mass distribution and configuration. Likewise, revisit operations will involve coupling the mass of the orbiter to that of the orbiting spacecraft, and possibly result in installation of relatively large masses on the construction project. Thus, the attitude control system design must meet a variety of conditions. The following discussion reviews potential control requirements and control system alternatives leading to the selection of an acceptable control method for space construction operation and subsequent operations. It is recognized that each type of large space structure can have unique pointing and stabilization requirements, which must be achieved within orbiter constraints. However, a case is made for the consideration of free-drift or gravity-gradient stabilization, allowing for relatively large (but slow) variations in attitude during space construction for projects which are not constrained by pointing requirements during construction.

4.2.1 Control Requirements and Approaches

Spacecraft control requirements during construction, maintenance, and refurbishment of large projects arise from the typical pointing and stability constraints of operational spacecraft and possible special constraints resulting from the construction process. Some typical attitude control issues and requirements pertinent to design of large space structures are listed in Table 4.2-1, together with potential Shuttle constraints and guideline topics which will be addressed in this section.

Table 4.2-1. Attitude Control Issues and Requirements for Design of Large Space Structures

Issue or Requirements	Potential Constraints
Large disturbances are undesirable during delicate construction operations. • Orbiter RCS inputs can create problems • Libration of spacecraft can induce additional loads • Null-torque orientations are highly desirable	• Specific vernier RCS thrust loads • Specific primary RCS thrust loads (larger) • Stability of the combined orbiter and large space structure • Time constraints on construction/operation
Orbiter docking can induce loads, changes in combined moments of inertia, stability condition, and rotational/translational rates.	• Orbiter VRCS and primary RCS as means to null undesirable induced motions
Orbiter berthing can induce loads, changes in combined moments of inertia, stability condition, and rotational/translational rates.	• Orbiter VRCS and primary RCS as means to null undesirable induced motions

Table 4.2-1. Attitude Control Issues and Requirements for Design of Large Space Structures (Cont.)

Issue or Requirements	Potential Constraints
Orbiting space structure attitude rates must be nulled to safe residual levels at time of docking or berthing of orbiter to spacecraft.	• Space project requires means to null natural and induced librating motions.
Combined orbiter/space structure attitude rates must be within safe limits at the time of separation of the orbiter from the structure.	• Combined orbiter/project requires means to null or constraint attitude rates to safe levels. • VRCS logic and control capability are limited.
Attitude control may be required during construction for solar array pointing, sensor or antenna pointing, illumination control or thermal control. • RCS options • CMG options • Libration damper options • Other systems	• Specific capabilities of vernier RCS or primary RCS on orbiter.
Large, lightweight structures optimized for the space environment can withstand only limited load inputs for attitude control.	• Specific capabilities of vernier RCS or primary RCS on orbiter
Movement of large masses relative to orbiter during construction can induce rotational rates on combined orbiter/space structure system.	• Orbiter RCS capabilities (if used) • Vision interruption due to lighting (orbiter orientation) • Loads between orbiter and structure • Crew and system safety—work schedule interruption due to motion

Within these issues and constraints, there are several fundamental options which may be considered in establishing suitable attitude control concepts for space construction.

Option 1. Use of Existing Orbiter RCS System

If it has been determined that a given construction project requires specified pointing and/or project orientations during the construction process, the use of existing orbiter RCS systems—either primary or vernier—and their related flight control software becomes a prime option. However, the use of the orbiter primary RCS system can introduce significant loads and disturbance forces into the construction process. Use of the vernier RCS system would greatly reduce the disturbance force levels, but because of its non-coupled thruster geometry, control authority in certain directions will be lost for combined orbiter/project c.g. outside of the cargo bay region. Either of these options may be considered for construction projects requiring specified pointing and/or construction orientations. Figure 4.2-1 and Table 4.2-2 provide the orbiter RCS system characteristics needed for control capability and induced loads assessment for application to any construction project. Figure 4.2-1 identifies thruster location and firing direction codes for all 38 primary and 6 vernier thrusters. Table 4.2-2 defines the thrust components in orbiter body axes and the location of thrust application for each of the 44 RCS thrusters. These data will allow full assessment of the impact of orbiter RCS thruster firings during the construction of any project system.

As a refinement to the option of using the orbiter RCS systems for attitude control during construction, if point/orientation requirements are not continuous and/or the RCS induced disturbances cause discrete event problems, consideration could be given to designing the construction system and process around the constrained use of RCS firings. With this approach, the basic construction process could be briefly interrupted for needed RCS thruster operations. If further precautions are necessary, the construction system and project could be designed to be temporarily rigidized or "safed" to tolerate these RCS induced disturbances.

Option 2. Dedicated Attitude Control System

If the disturbance environment created by the orbiter RCS systems is too severe, an alternative option for projects requiring controlled orientations during construction could be the use of a dedicated attitude control system. This could be integral with the construction system or a separate unit temporarily installed at some convenient location on the construction project. It could be designed to produce a "softer" disturbance environment than the orbiter PRCS and could be configured around the use of pure thrust couples to provide adequate control torques over a wide range of project configurations and mass properties.

Option 3. Modified Orbiter RCS System

Another option for satisfying the need for controlled orientations during construction is the possibility of introducing modifications to the orbiter RCS/flight control subsystems. These could include adding vernier thruster, changes to software (and possibly computer hardware), reducing thruster minimum impulse size, etc. The problems and costs involved in changing fully qualified, space-rated systems such as the orbiter make this a low priority option. However, under very special construction project circumstances and/or in combination with other Shuttle applications program needs, this option would be a viable candidate solution to attitude stabilization and control needs.

REFERENCE: *Shuttle Operational Data Book, JSC-08934 (Vol. I), Rev. B, Feb. 1980.*

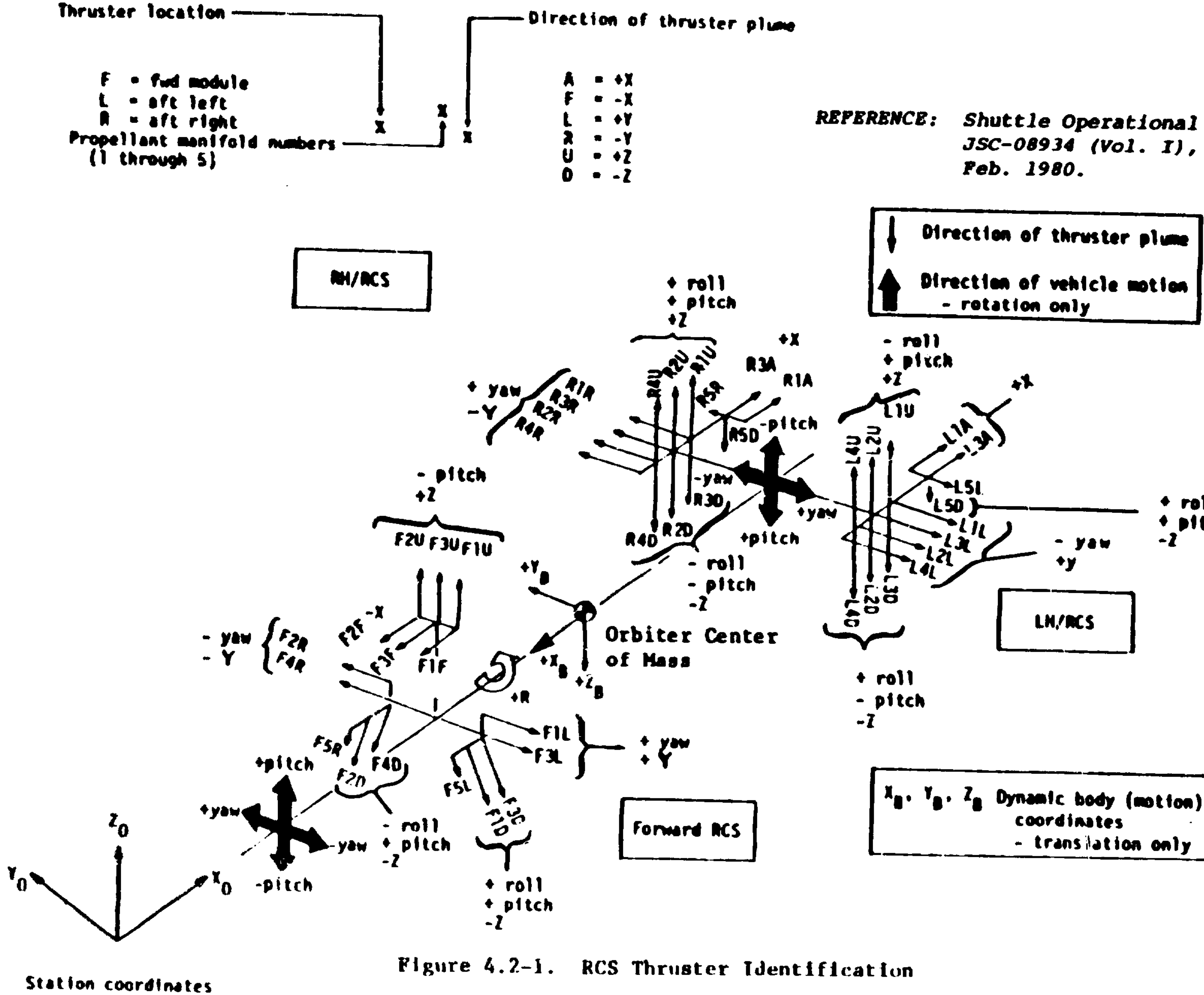


Figure 4.2-1. RCS Thruster Identification

Table 4.2-2. Nominal Thrust Components and Application Locations

Thruster Number	Thrust Components, lb[a]			Resultant Thrust, lb	Thrust Application[b]		
	FX_B	FY_B	FZ_B		X_0	Y_0	Z_0
F2F	• -879.4	-26.2	119.9	887.9	306.72	14.65	392.96
F3F	• -879.5	0.0	122.7	888.0	306.72	0.0	394.45
F1F	• -879.4	26.2	119.9	887.9	306.72	-14.65	392.96
F1L	-26.3	• 873.6	18.2	874.2	362.67	-69.50	373.73
F3L	-21.0	• 870.3	0.5	870.6	364.71	-71.65	359.25
F2R	-26.3	• -873.6	18.2	874.2	362.67	69.50	373.73
F4R	-21.0	• -870.3	0.5	870.6	364.71	71.65	359.25
F2U	-32.3	-11.7	874.4	875.1	350.93	14.39	413.46
F3U	-31.9	0.0	873.5	874.1	350.92	0.0	414.53
F1U	-32.3	11.7	874.4	875.1	350.93	-14.39	413.46
F2D	-28.0	• -616.4	-639.5	888.6	333.84	61.42	356.95
F1D	-28.0	• 616.4	-639.5	888.6	333.84	-61.42	356.95
F4D	-24.8	• -612.6	-639.4	885.9	348.44	66.23	358.44
F3D	-24.8	• 612.6	-639.4	885.9	348.44	-66.23	358.44
F5R	-0.8	• -17.0	-17.6	24.5	324.35	59.70	350.12
F5L	-0.8	• 17.0	-17.6	24.5	324.35	-59.70	350.12
R3A	• 856.8	0.0	151.1	870.0	1555.29	137.00	473.06
R1A	• 856.8	0.0	151.1	870.0	1555.29	124.00	473.06
L3A	• 856.8	0.0	151.1	870.0	1555.29	-137.00	473.06
L1A	• 856.8	0.0	151.1	870.0	1555.29	-124.00	473.06
L4L	0.0	• 870.5	-22.4	870.8	1516.00	-149.87	459.00
L2L	0.0	• 870.5	-22.4	870.8	1529.00	-149.87	459.00
L3L	0.0	• 870.5	-22.4	870.8	1542.00	-149.87	459.00
L1L	0.0	• 870.5	-22.4	870.8	1555.00	-149.87	459.00
R4R	0.0	• -870.5	-22.4	870.8	1516.00	149.87	459.00
R2R	0.0	• -870.5	-22.4	870.8	1529.00	149.87	459.00
R3R	0.0	• -870.5	-22.4	870.8	1542.00	149.87	459.00
R1R	0.0	• -870.5	-22.4	870.8	1555.00	149.87	459.00
L4U	0.0	0.0	870.0	870.0	1516.00	-132.00	480.50
L2U	0.0	0.0	870.0	870.0	1529.00	-132.00	480.50
L1U	0.0	0.0	870.0	870.0	1542.00	-132.00	480.50
R4U	0.0	0.0	870.0	870.0	1516.00	132.00	480.50
R2U	0.0	0.0	870.0	870.0	1529.00	132.00	480.50
R1U	0.0	0.0	870.0	870.0	1542.00	132.00	480.50
L4D	170.4	291.8	-801.7	870.0	1516.00	-111.95	437.40
L2D	170.4	291.8	-801.7	870.0	1529.00	-111.00	440.00
L3D	170.4	291.8	-801.7	870.0	1542.00	-110.06	442.60
R4D	170.4	-291.8	-801.7	870.0	1516.00	111.95	437.40
R2D	170.4	-291.8	-801.7	870.0	1529.00	111.00	440.00
R3D	170.4	-291.8	-801.7	870.0	1542.00	110.06	442.60
L5D *	0.0	0.0	-24.0	24.0	1565.00	-118.00	455.44
R5D *	0.0	0.0	-24.0	24.0	1565.00	118.00	455.44
L5L *	0.0	• 24.0	-0.6	24.0	1565.00	-149.87	459.00
R5R *	0.0	• -24.0	-0.6	24.0	1565.00	149.87	459.00

*Vernier RCS thrusters

• Principal Contributor(s)

[a] Motion coordinates (Figure 4.2-1)

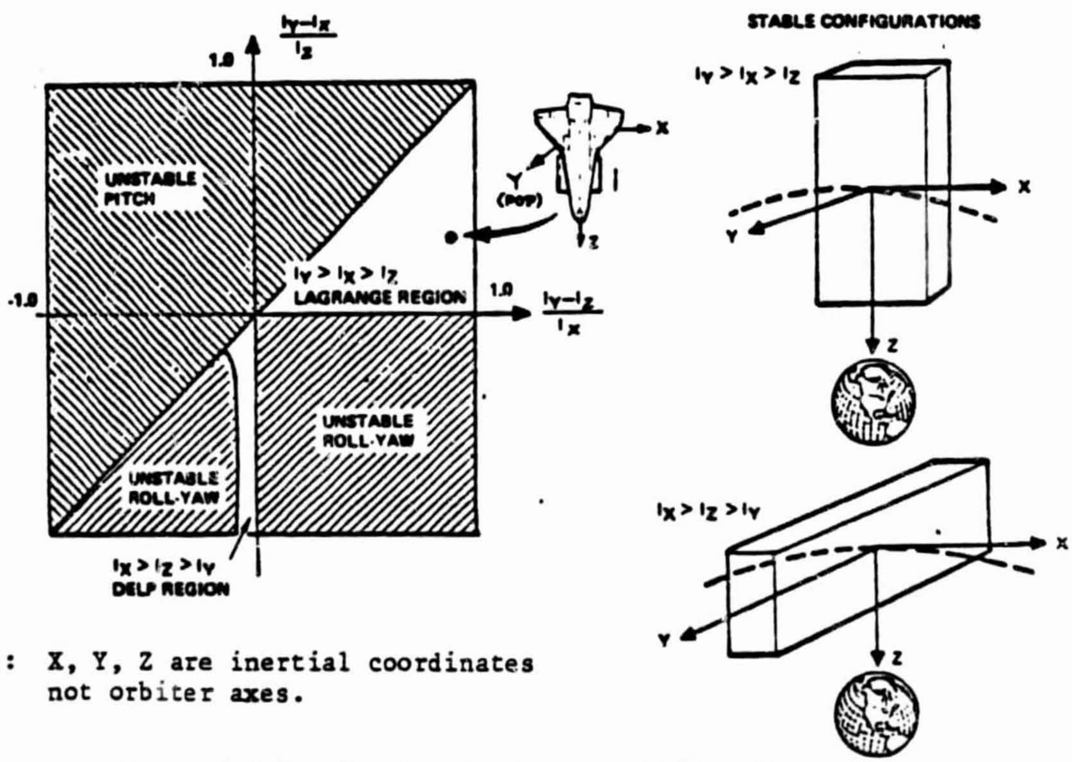
[b] Station coordinates (Figure 4.2-1)

Option 4. Free-Drift Construction Attitude

In addition to the above options for attitude control during construction, for those project systems which are less demanding in their stabilization and pointing needs it may be possible to employ free-drift gravity-gradient oriented construction operations. With this technique, the orbiter RCS firings are inhibited and the combined orbiter/construction project is allowed to "free drift" in the natural gravity-gradient aero torque environment. Preliminary studies have shown the resulting librational motions are very slow—typically less than 0.1 degree/second—and that construction induced disturbances can be controlled to inhibit full tumbling motion. The librating motions are judged to be sufficiently slow as to not cause visual distractions to the construction operations and that these motions tend to average out the thermal environments important to orbiter heat rejection capability and construction operations. An example analysis of the free-drift techniques applied to the construction of a 200-m space platform is presented in the following discussion.

4.2.2 Gravity-Gradient Stabilization During Construction (Example)

For satellites in circular orbits and those that rely on gravity-gradient stabilization, the principal axes of inertia must be aligned with the radial, tangential, and normal axes of the orbit in order for the satellite to maintain a fixed attitude in the orbiting frame. The question of stability in this attitude was addressed by D. B. DeBra and R. H. Delp (1961). They developed the stability diagram shown in Figure 4.2-2. This diagram shows two regions for which a gravity-gradient satellite is stable—Liapunov stable in the Lagrange region, and infinitisimally stable in the Delp region.



Note: X, Y, Z are inertial coordinates not orbiter axes.

Figure 4.2-2. Gravity-Gradient Stability Diagram

The analysis leading to this diagram assumes that (1) the spacecraft is a rigid body with constant mass properties and geometry, (2) the only forces on the satellite result from an inverse-square gravity field, (3) the body is small enough that the attitude motions do not significantly affect orbital motion, (4) the orbit is circular, and (5) the attitude deviations from the equilibrium position are small. These conditions are satisfied during the space construction process only prior to and after the period of internal motion resulting from erection and fabrication of the system. During that period, Conditions (1), (2), and (5) are violated. However, the stability diagram remains useful in establishing the orientation at the start of construction and the allowable extent of construction in terms of changes in moments of inertia. Figure 4.2-3 illustrates the relative motion of the orbiter with respect to the construction project during the majority of construction time. The structure is a tri-beam, with a beam building machine used to fabricate each beam in space. The beams are connected by cross members of similar fabrication.

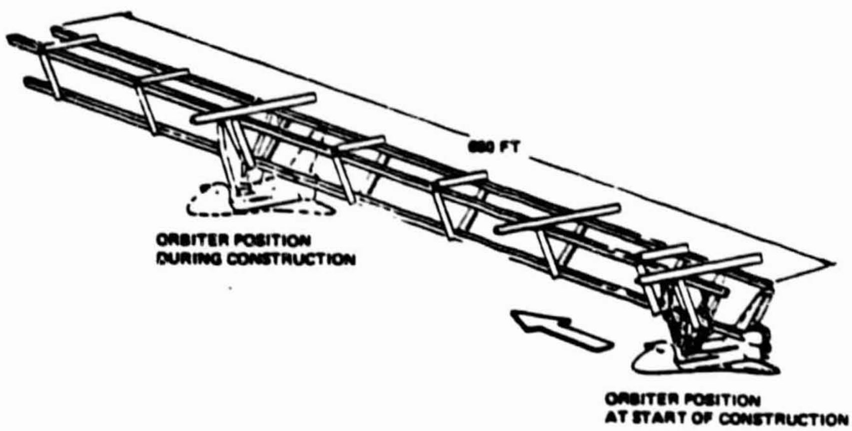


Figure 4.2-3. Construction Scenario

The structure is built from a fixture attached to the orbiter's docking port. The construction scenario is based on several orbiter flights and includes the following major events during these flights.

- Erection and deployment of the construction fixture containing a beam fabrication machine.
- Fabrication of three longitudinal beams extending outward and parallel to the orbiter pitch axis.
- Attachment of cross members to the longitudinal beams while the structure is translated back through the construction fixture.
- Translation of the structure in the original direction of beam fabrication while such items as wire bundles and junction boxes are being attached.

- Reversal of the direction of structure motion a second time and addition of payload and subsystem components such as antennas and control system elements.

The salient events of space construction are now examined in terms of the gravity-gradient stability diagram. The construction scenario assumes that a 36,300-kg (80,000-lb), 200-m (660-ft) long space fabricated tri-beam structure is built from the Space Shuttle orbiter, and five 4500-kg (10,000-lb) elements are added to the structure. These elements represent, for example, large communication antennas or large subsystem modules. These events, selected because they demonstrate the significant gravity-gradient problems, are listed below.

- Orbiter unattached to the structure. This occurs before a construction starts or before docking or after undocking.

- Erection and positioning of the construction fixture.

- Initial fabrication of a tri-beam assembly.

- Move the structure through the construction fixture. This could be for the purpose of putting the fixture at the opposite end of the construction or adding relatively massless items along the length of the structure.

- Move the structure through the fixture and add five 4500-kg (10,000-lb) elements, equally spaced, to the structure.

The stable gravity-gradient orientation of an unattached orbiter is nose down (or up) and wings parallel to the orbit plane. This orientation and the location of the inertial ratios $(I_y-I_x)/I_z$ and $(I_y-I_z)/I_x$ are shown on Figure 4.2-2. The erection and positioning of the construction fixture starts with the Shuttle in this orientation. The process is assumed to proceed in four steps as shown in Figure 4.2-4, which also shows the stability diagram for this operation. It can be seen that the orbiter-fixture combination becomes unstable because I_x, the orbiter's pitch moment of inertia, becomes larger than I_y, the Shuttle's yaw moment of inertia. Attitude and rate histories are shown in Figures 4.2-5 and 4.2-6. Attitude, described in terms of Euler angles of pitch, roll, yaw sequence, and body rates diverge in roll and yaw as predicted by Figure 4.2-4.

The third case is the initial fabrication of a tri-beam assembly. Assuming that the angular rates are small enough, construction can proceed from the orientation at the end of the previously discussed event. If not, then active control would be required to damp the librations and maintain stability in this unstable region. An alternative would be to reorient the system into the stable region and then damp the residual rates to an acceptable level. In any case, the subsequent construction, the third event, would cause the inertia ratios to move into the third quadrant of the diagram. Although this is an unstable region, slow changes in the relative magnitudes of the principal moments of inertia and their directions relative to body fixed coordinates will not excite large libration amplitudes. A second method of limiting these amplitudes is to provide active damping. This could come from operation of the

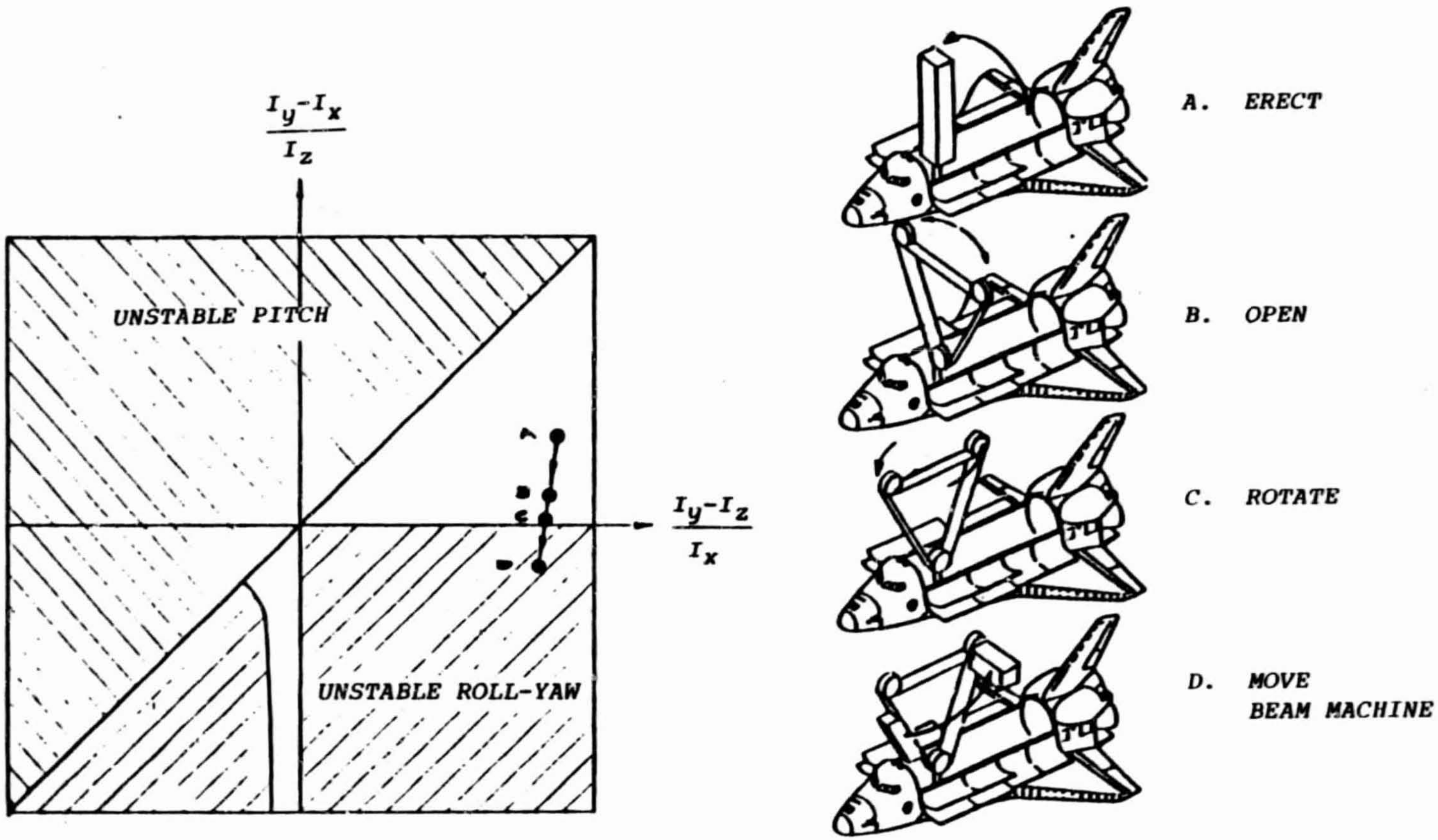


Figure 4.2-4. Gravity-Gradient Stability Diagram for Fixture Erection Case

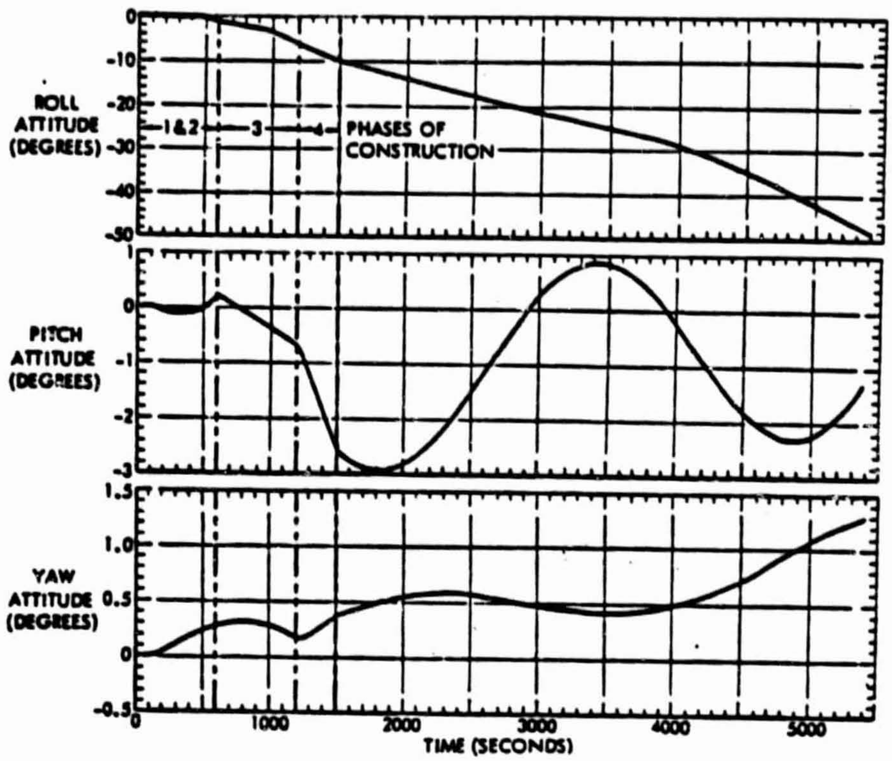


Figure 4.2-5. Attitude Histories for Fixture Erection Case

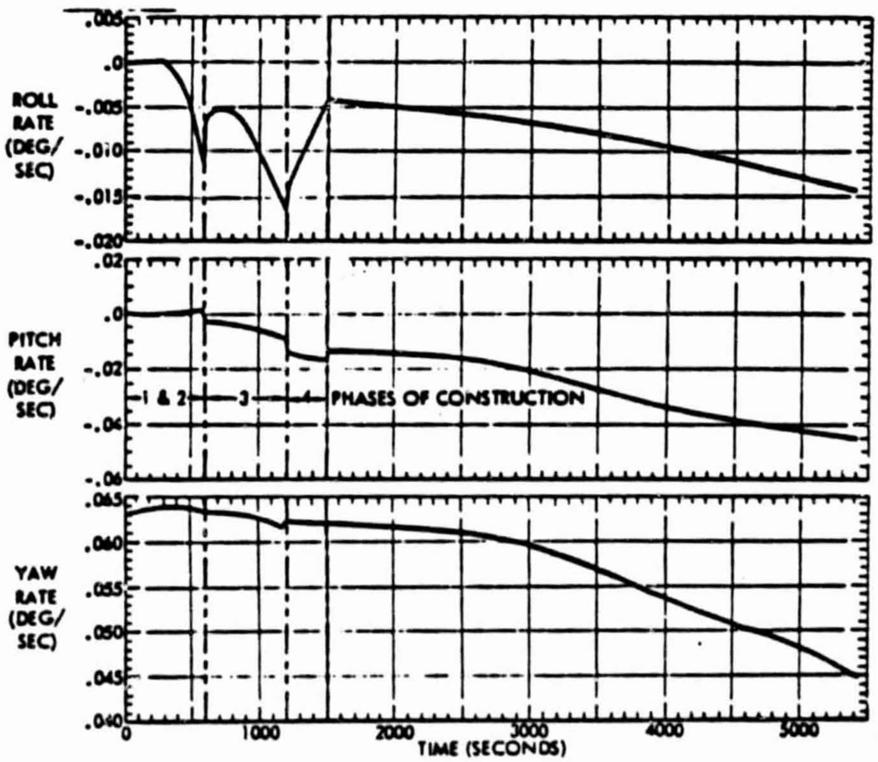


Figure 4.2-6. Body Rate Histories for Fixture Erection Case

orbiter reaction control systems in a damper mode, or from the use of a simple and inexpensive add-on reaction control system.

The fourth case is the movement of the structure through the construction fixture while adding relatively massless items (wire bundles and junction boxes). Two cases were considered in order to determine the effect of speed of construction on attitude dynamics. For each of the cases, it is assumed that construction starts at the gravity-gradient stable orientation as shown in Figure 4.2-7. The loci of the inertia ratios are also shown on this figure.

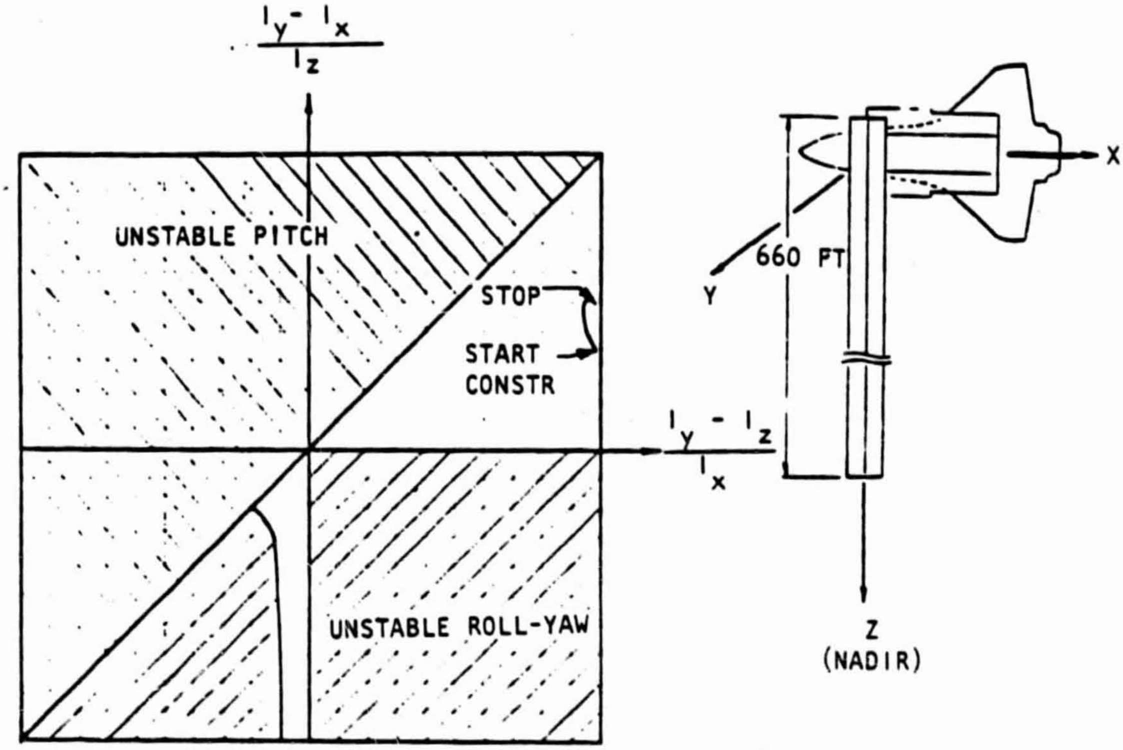


Figure 4.2-7. Gravity-Gradient Stability Diagram for Simple Translation of Structure

The first case (Figures 4.2-8 through 4.2-11) is for a construction speed, or translation of the structure through the construction fixture, of one meter per minute. The second case (Figures 4.2-12 through 4.2-14) is for a construction speed of about 1/3 meter per minute (1.0 ft/min.). The initial conditions for each are such that the principal axes of inertia are aligned with the radial, tangential, and normal axes of a circular orbit at an altitude of 300 nmi. The Euler angles describing the orientation of the body axes are $\theta = -3.556°$, $\phi = -0.6738°$, and $\psi = 16.18°$. The initial body rates are $p = 0.0175°/sec$, $q = 0.0602°/sec$, and $r = 0.007°/sec$. This places the long axis of the structure parallel to the local vertical and the orbiter roll axis parallel to the velocity vector. Construction starts at 96 minutes in each case and ends at 296 minutes for the fast case and at 756 minutes for the slow case.

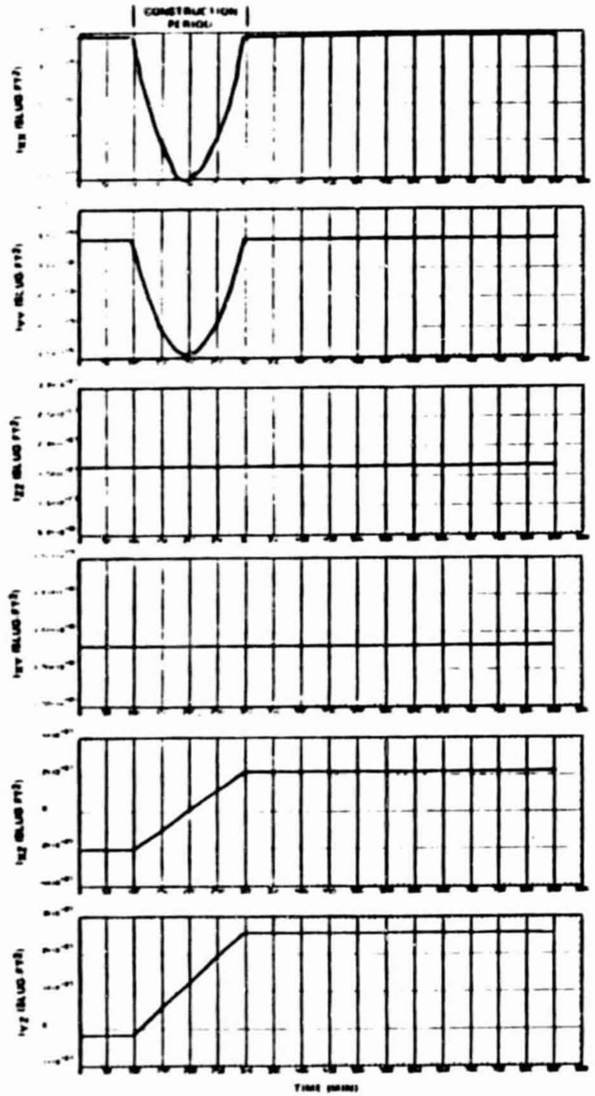


Figure 4.2-8. Moment of Inertia for Fast Construction Case

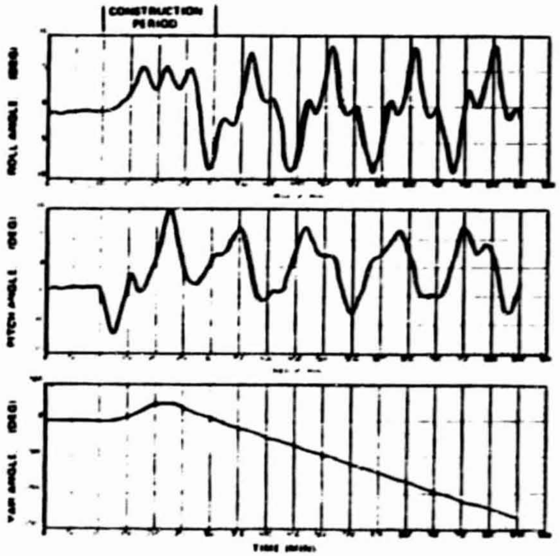


Figure 4.2-9. Euler Angle Histories for Fast Construction Case

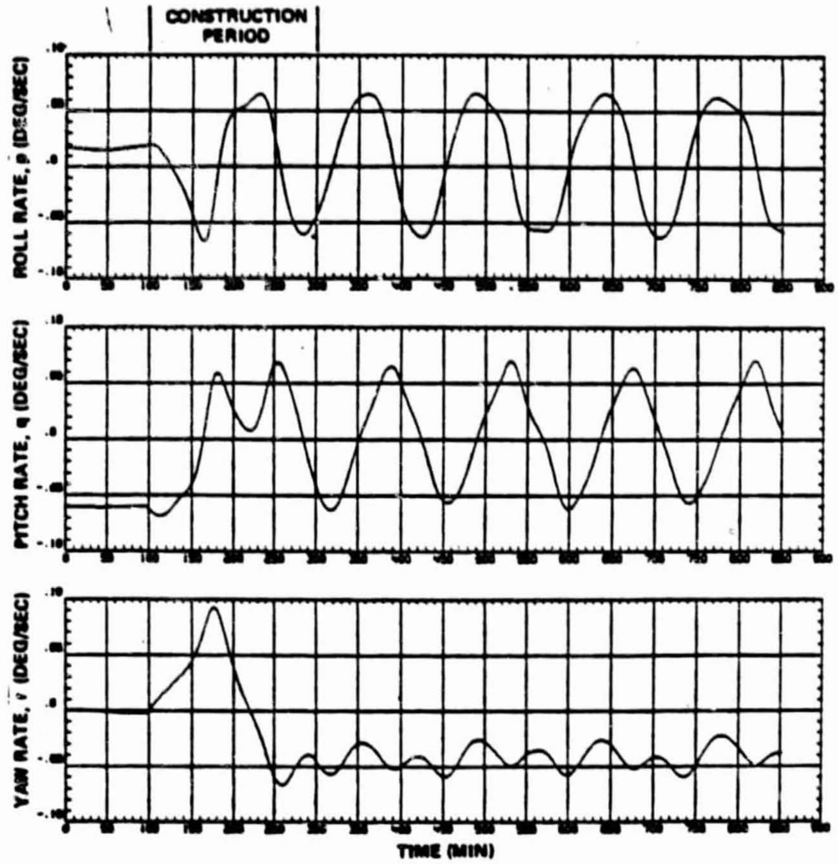


Figure 4.2-10. Body Rate Histories for Fast Construction Case

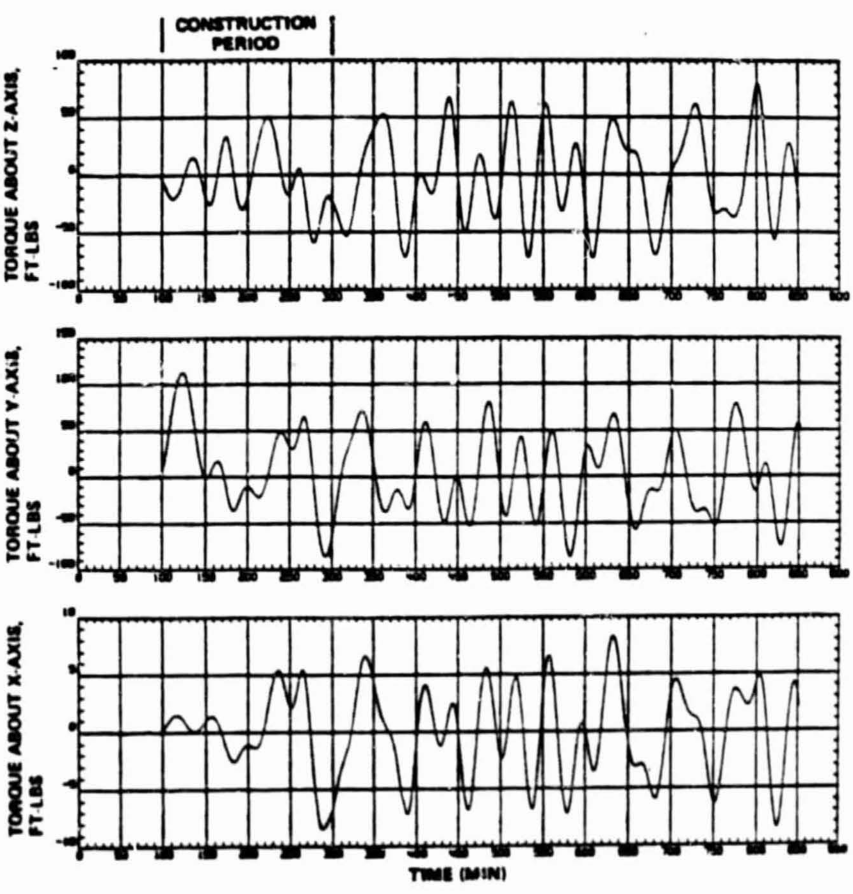


Figure 4.2-11. Gravity Gradient Torque Histories for Slow Construction Case

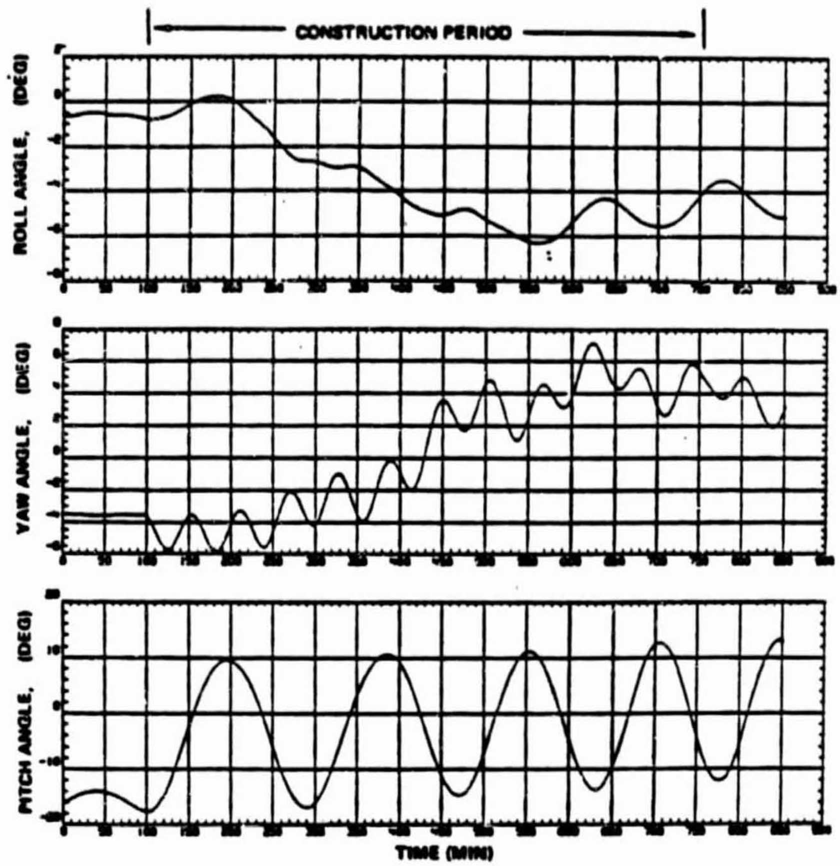


Figure 4.2-12. Euler Angle Histories for Slow Construction Case

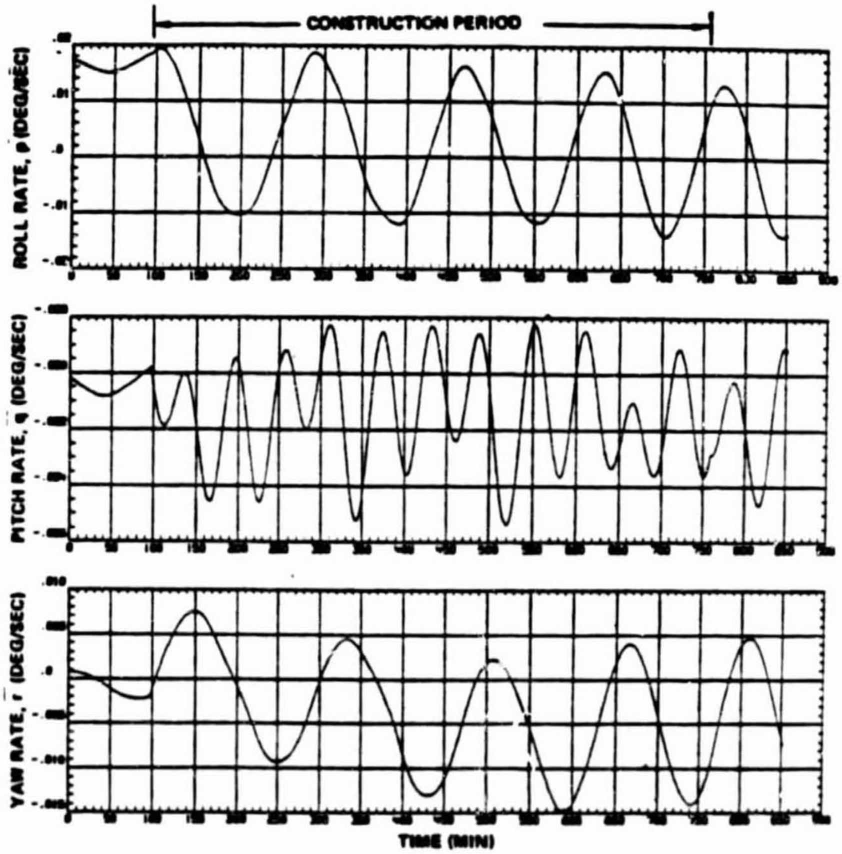


Figure 4.2-13. Body Rate Histories for Slow Construction Case

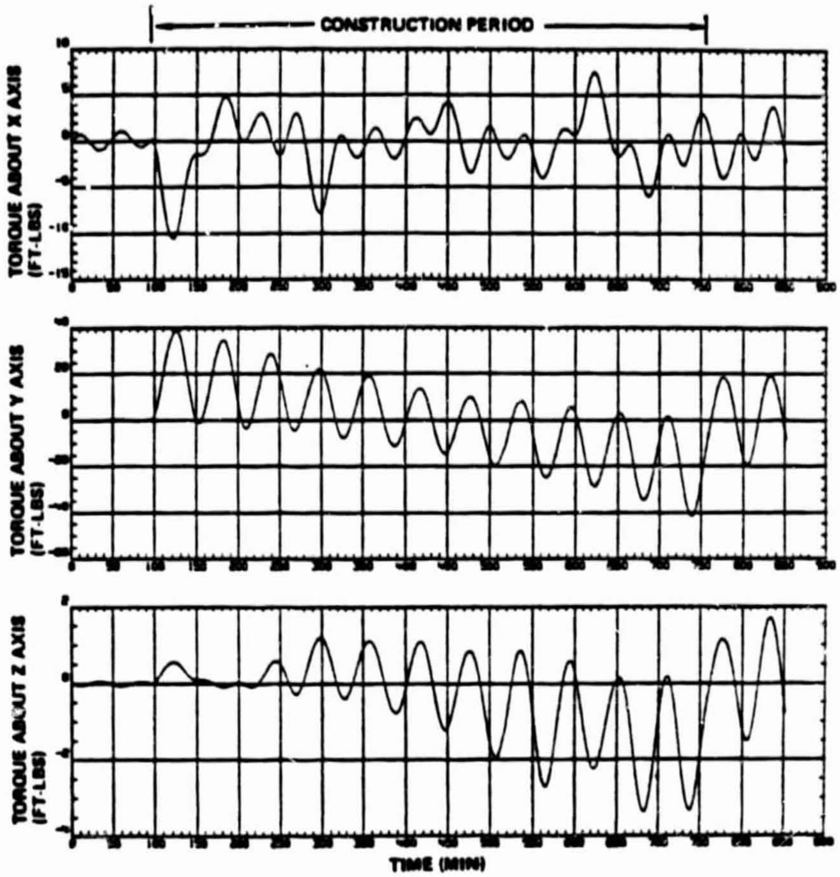


Figure 4.2-14. Gravity-Gradient Torque Histories for Slow Construction Case

Figure 4.2-8 shows the elements of the inertia matrix as a function of time for fast construction. The change in the elements is a result of the translation of the structure through the construction fixture. The moment of inertia history for the slow case is the same, but occurs over the longer period of construction. A comparison of the Euler angle histories for the fast and slow cases is shown in Figures 4.2-9 and 4.2-12, respectively. The Euler angle histories for the fast construction case show that the spacecraft is rotating in yaw with unconstrained motion about the local vertical. The use of the slower construction speed eliminates the tumbling and reduces pitch and roll librations. Hence, it is concluded that some degree of construction speed modulation can prevent tumbling and reduce the amplitude of libration and that use of gravity-gradient orientations during space construction will permit relatively long periods of uncontrolled, disturbance-free construction.

Figures 4.2-10 and 4.2-13 show the body rate histories for the fast and slow cases, respectively. Rates for the fast case are generally much larger than for the slow case—five times larger in roll rate, and ten times larger in yaw rate. The gravity-gradient torques about the body axes are shown in Figures 4.2-11 and 4.2-14. The torques in the fast case are larger than in the slow case, which is expected because the attitude excursion and, hence, out-of-trim conditions are larger.

Another case analyzed, but not described in detail herein, is similar to the previous fast construction case with the exception that the translation motion is stopped periodically to add five 4500-kg (10,000-lb) elements to the structure. The results showed that there is a large change in yaw attitude which is the axis with the weakest gravity-gradient stiffness. However, all body rates are small, not exceeding 0.07 degree/sec.

The results of the simulations of these several cases show that, for construction scenarios where large changes in mass distribution and dynamic changes in configuration exist, long periods of flight are possible without use of active control systems. A comparison of two identical cases with and without movement of masses from the cargo bay to the structure shows that there is an effect on the dynamics due to small inertia changes occurring periodically.

Further information on these analyses is available in the study report [Rockwell, 1979 (a)] and a technical paper by Oglevie, Quartararo, Sampson and Abramson (1979).

4.3 SYSTEM SAFETY GUIDELINES

This section includes discussion of special safety issues applicable to design of large space structures built out of the orbiter. In particular, those involving on-orbit operations, or those which are not specifically discussed in the NASA Document NHB 1700.7 are of concern herein.

4.3.1 Jettison Requirements

Use of the Shuttle Orbiter for construction and operations involving large space structures generates a special series of safety concerns involving contingency separation of the orbiter and payload.

The basic safety requirement is stated in the NASA Safety Policy Document NHB 1700.7 (NASA 1979), as follows:

> "EXTENDABLE PAYLOADS. Any payload which may be operable in a fashion which could prevent closure of the cargo bay doors shall be provided with primary and backup methods of clearing the cargo bay door envelope. The primary method may be either retraction or jettison and shall be controlled by remote initiation. When the primary method is retraction, the backup technique must be either remote jettison or removal by EVA. . ."

EVA requirements are stated in NASA 10615 (NASA 1976). They are outlined briefly in Section 4.3.2.

In particular, the space construction mode could create an unusually varied and extensive set of configurational conditions which are not compatible with immediate return to earth. Therefore, more than normal attention should be directed toward planning for simple and reliable modes of recovery should an abort from orbit become necessary. Stability and momentum conditions may preclude simple jettisonning in some cases, due to close clearances or actual interferences with direct orbiter separation. Also, once separation is accomplished, there may be a need to re-stow some undeployed cargo in order to achieve a satisfactory vehicle center of gravity and to be able to shut the cargo bay doors.

4.3.2 EVA Access Requirements

From a safety standpoint there are certain minimum functional requirements for contingency EVA operations. These involve two basic types: (1) access for closing cargo bay doors and configuring for return to earth, and (2) rescue operations.

The overall clearance volume requirements for EVA access are pictured in Section 4.8 of this report. Further specific constraints are listed in Table 4.3-1.

Within the context of construction of large space structures there is also another aspect of concern regarding EVA access. This is the potential for entanglement of hangup of EVA suit or backpack, etc. within the space-

Table 4.3-1. Checklist: EVA Access Constraints for Contingency Situations

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) Access volume shall be provided within the cargo bay for EVA crew to close the orbiter cargo bay doors, using tools provided for this purpose, in the contingency event that normal controls in the cabin are not capable of closing and latching the doors for safe descent. (R-1)	NASA 1980 (a): JSC-07700, Vol. XIV Also see NASA 1976: JSC-10615
. Payload components which could block such access shall be removed by normal means or jettisonned.	
. See Section 4.8.1 for EVA egress pathway requirements.	
. Access volume is required to reach and remove tools in the Modular Equipment Stowage Assembly (MESA).	
. See Section 4.8.2.2 for EVA tools description and locations.	
( ) Handholds/handrails shall be provided along all contingency EVA pathways to assist crew mobility. (R-1)	NASA 1976: JSC-10615
. Existing handholds on orbiter foreward and aft bulkhead and cargo bay doors, and structure may be utilized if accessible under the given contingency conditions.	
. See Section 4.8.1.5 for handhold design requirements.	
( ) EVA safety tether provisions shall be incorporated with handrail design to assure crew security during contingency EVA. (R-1)	NASA 1976: JSC-10615
. See Section 4.8.1.5 for safety tether/slide wire requirements and standard provisions.	

craft structure. To minimize this possibility, care must be directed toward eliminating snags, protrusions, lines and/or wedging situations within and along the path of EVA motion or workspaces. In addition, structure within reach or contacted by the suit must have rounded corners as defined in JSC-10615 (NASA 1976) to prevent puncture or tearing and consequent total loss of pressure within the EVA suits.

4.3.3 Rescue and Contingency Provisions Requirements

Designers of large space structures payloads generally will not be deeply concerned with orbiter crew rescue modes if they meet the above-discussed requirement for separation of the orbiter from the space structure and for EVA access. However, they should provide for contingency rescue of EVA crew members who inadvertantly become lodged, entangled or otherwise involved in large space structure without capability to return to the orbiter airlock alone. Such provisions must include standby rescue personnel, either as EVA "buddy" operators or as crew within the cabin who are prepared to don suits and egress through the airlock as soon as possible. Such crew pre-breathing and suit donning timeline considerations must be included in planning crew assignments, manpower loading, equipment and provisions for airlock repressurizations, as discussed in Section 4.8.4.

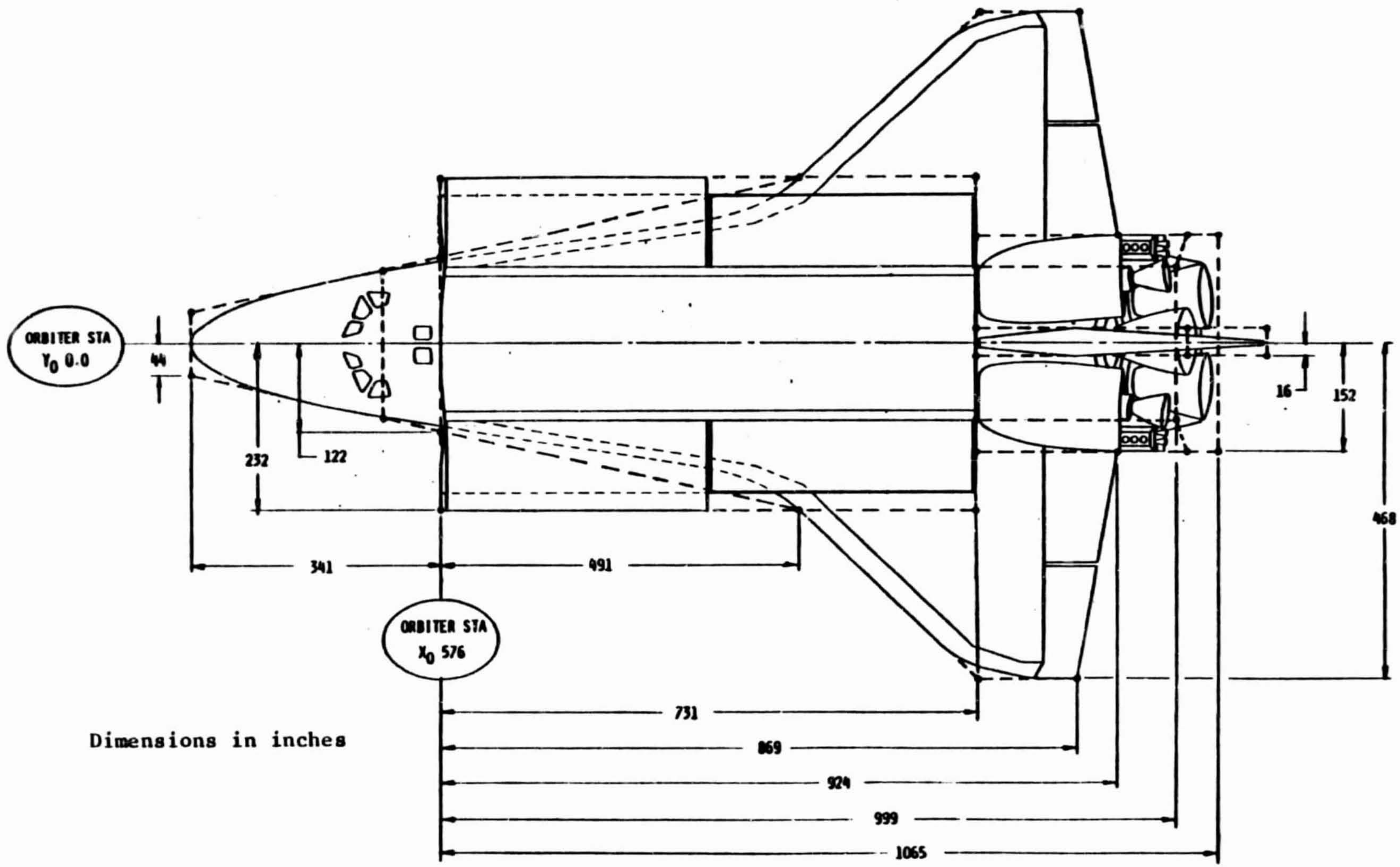
4.4 PAYLOAD HANDLING CAPABILITIES/CONSTRAINTS AND GUIDELINES

The following sections (Sections 4.4.1 through 4.4.4) deal with the general topics of deploying structure, setting up construction equipment, handling, assembling and checking large space structure materials and modules, and re-stowing components for return to earth. The various options for Shuttle equipment and personnel to help perform such tasks are described briefly in Section 4.4.4, and are described in greater detail in Sections 4.5 through 4.8. Support services constraints are discussed in Sections 4.9 through 4.14. Some overall crew scheduling operations considerations involving payload handling also are included in Section 5.0.

4.4.1 Construction/Deployment Orientation and Clearance Constraints

This section outlines the overall, gross geometric constraints for large space structures attached to the Shuttle Orbiter while on orbit. These constraints have a major impact on the possible shapes which can be constructed or deployed from the orbiter and which directions can be used for translating structure with respect to the orbiter. Such limitations are primarily based on structural outlines of the orbiter. Additional concerns involve antenna blockage, radiator blockage, TV and window viewing, lighting, aerodynamic drag, center of mass and moment of inertia. Detail considerations of attach points and close tolerance clearances are considered in Section 4.4.2.

Figure 4.4-1 provides overall, three-dimensional constraints for the orbiter in the normal on-orbit operational condition. The cargo bay doors and radiators are fully opened and deployed. Figure 4.4-2 (from Reference NASA (1)) provides another means of defining the structural configuration in terms of viewing angles from a single point in the center of the cargo bay. Tables 4.4-1 and 4.4-2 provide a checklist of additional constraints and guidelines



Dimensions in inches

PLAN VIEW

Figure 4.4-1A. Orbiter Structure Geometric Constraints on Design of Large Space Structures

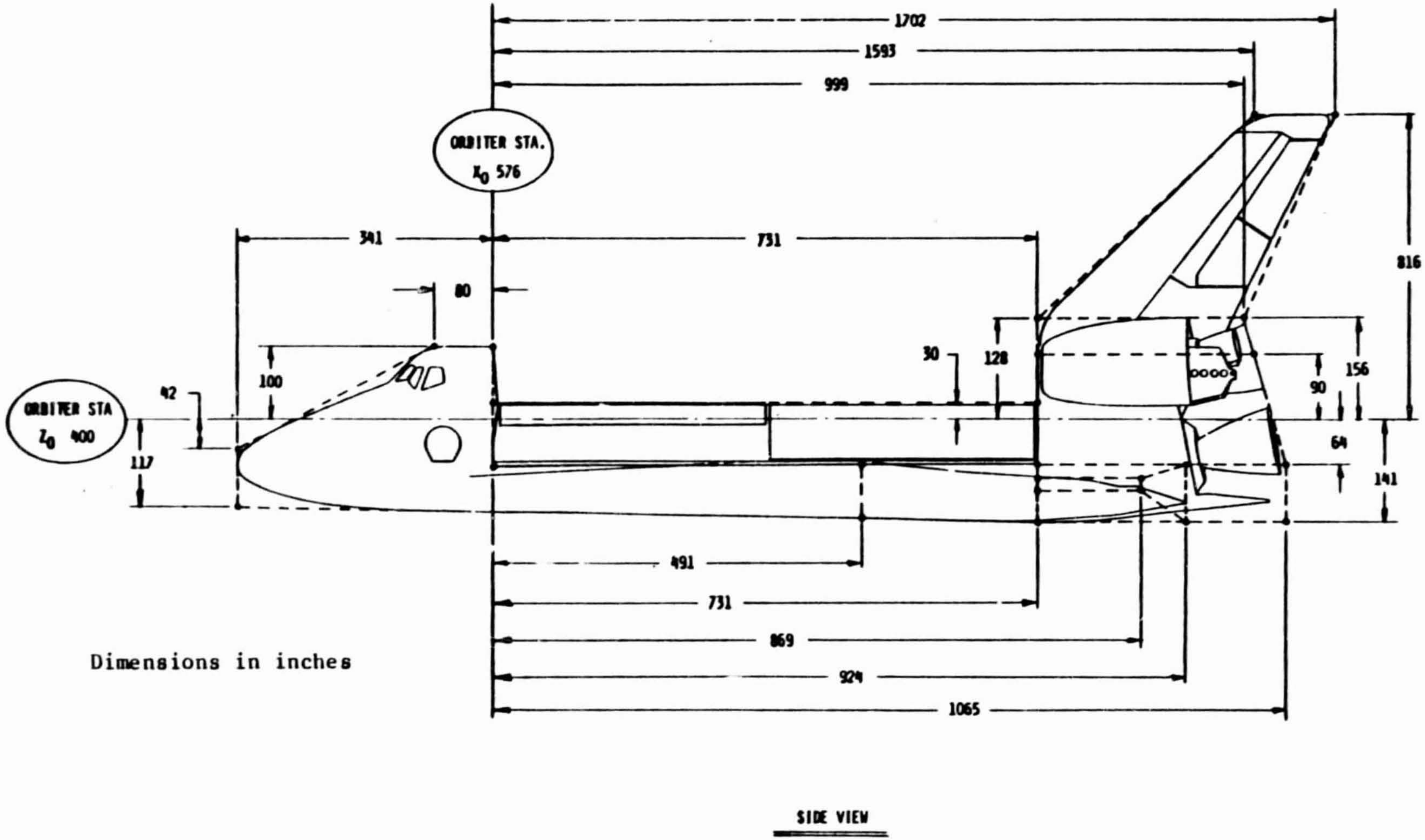


Figure 4.4-1B. Orbiter Structure Geometric Constraints on Design of Large Space Structures

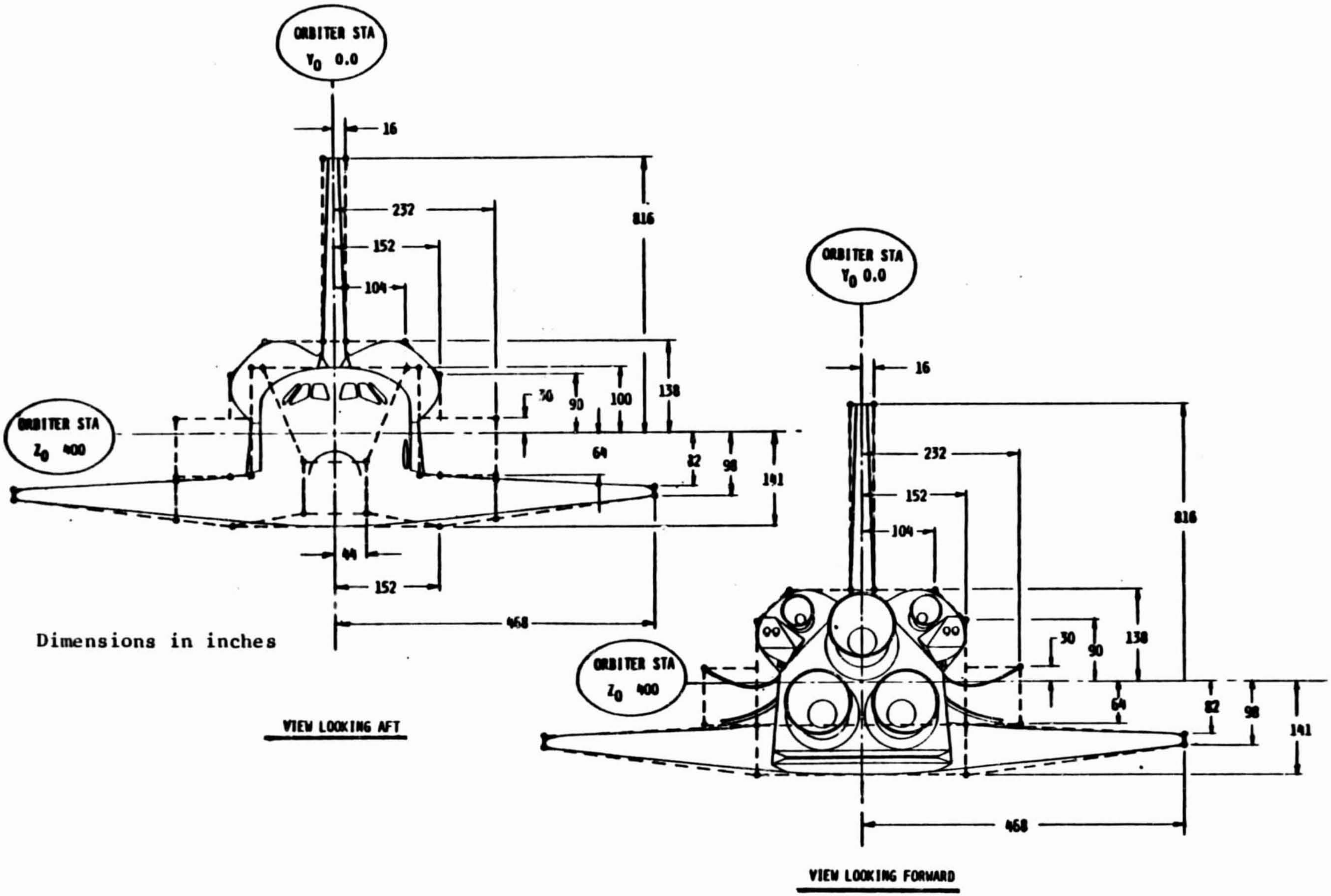
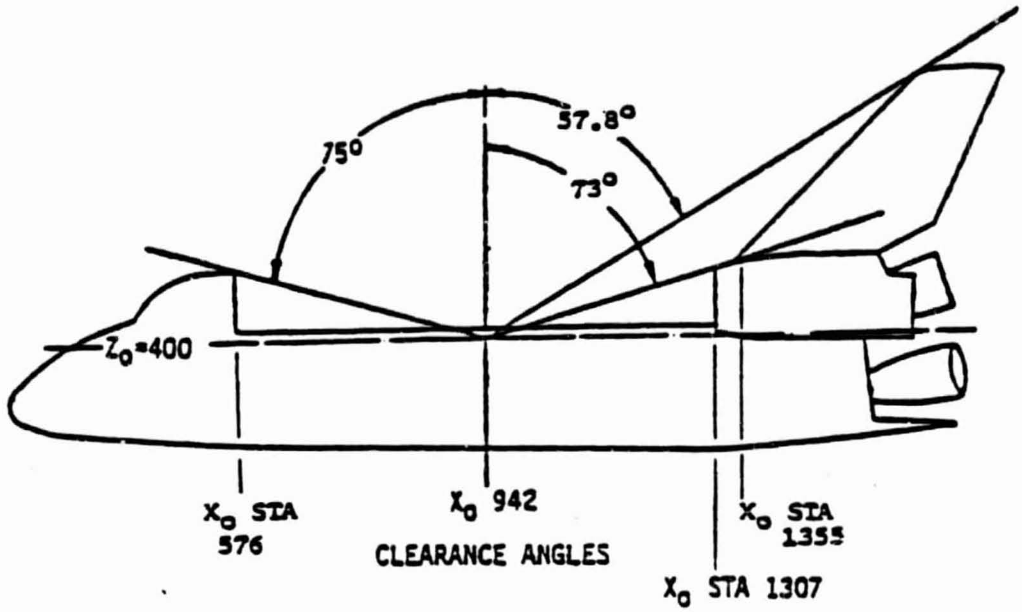


Figure 4.4-1C. Orbiter Structure Geometric Constraints on Design of Large Space Structures



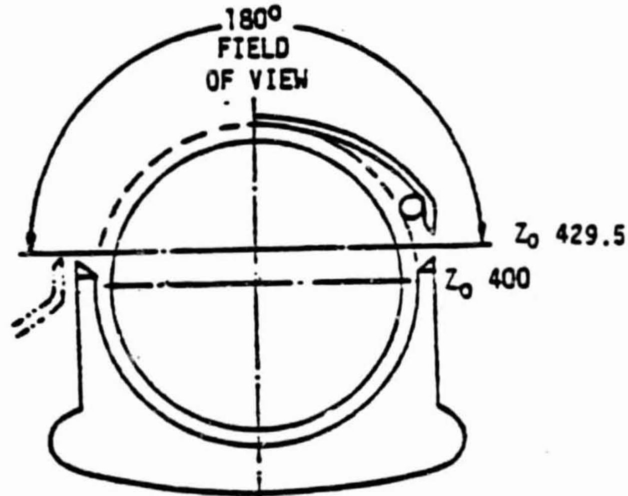


Figure 4.4-2. Cargo Bay Field of View

Table 4.4-1. Checklist: Overall Orbiter Geometric Constra its on Design of Large Space Structures

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) Large space structure configurations and orientations shall not block communications/antenna radiation or reception envelopes when the orbiter is used as a communications base during construction, maintenance or other orbiter-tended operations. (R-4) . Further analysis required for specific cases. . See Figure 4.4-3 for antenna locations. . See Section 4.4.2 for Ku-Band antenna envelope and locations.	. Constraints configuration of structure . Potential need for auxiliary antennas early in construction period, located on large space structure and hard-lines connected to orbiter.
( ) Large space structure configurations should not block viewing angles of orbiter star trackers in nose of vehicle. (R-4) . See Figure 4.4-4.	. Constrains design for guidance and control systems.
( ) Large space structures shall not unduly block orbiter heat rejection from cargo bay doors. (R-4) . Blockage is a function of open area in structure and size. See Section 4.13 for parametric estimates of reduction in heat rejection.	. Constrains design of large space structure, orientation and clearance from orbiter.
( ) Large space structures, when deployed or under construction, shall not block egress for contingency EVA. (R-4) . Jettison or other means for removal of blockage may be permitted. NASA safety reviews may be required to resolve these problems. See Sections 4.3.2 and 4.3.3.	Affects location of docking/berthing ports and support structure, stowage and construction fixtures. (Requirement was derived for this document).

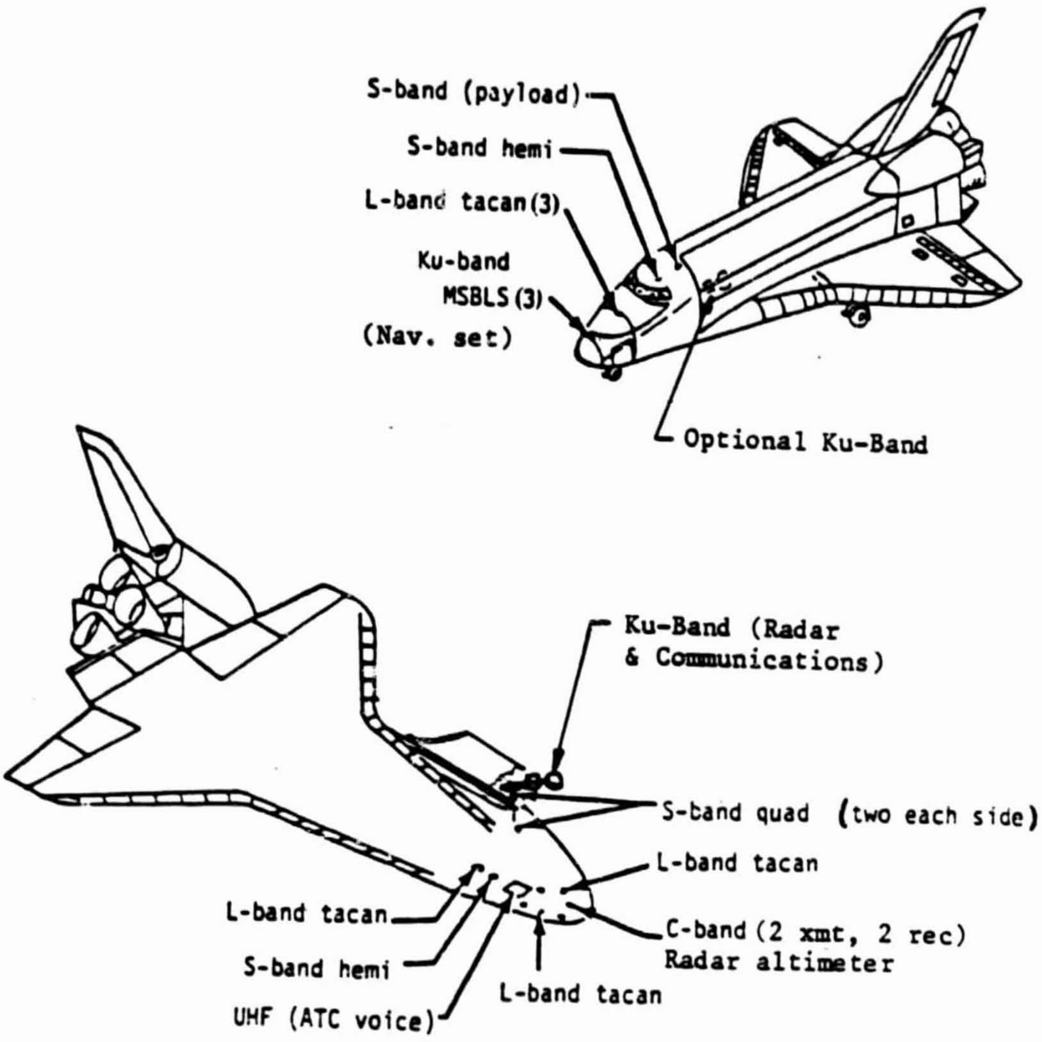
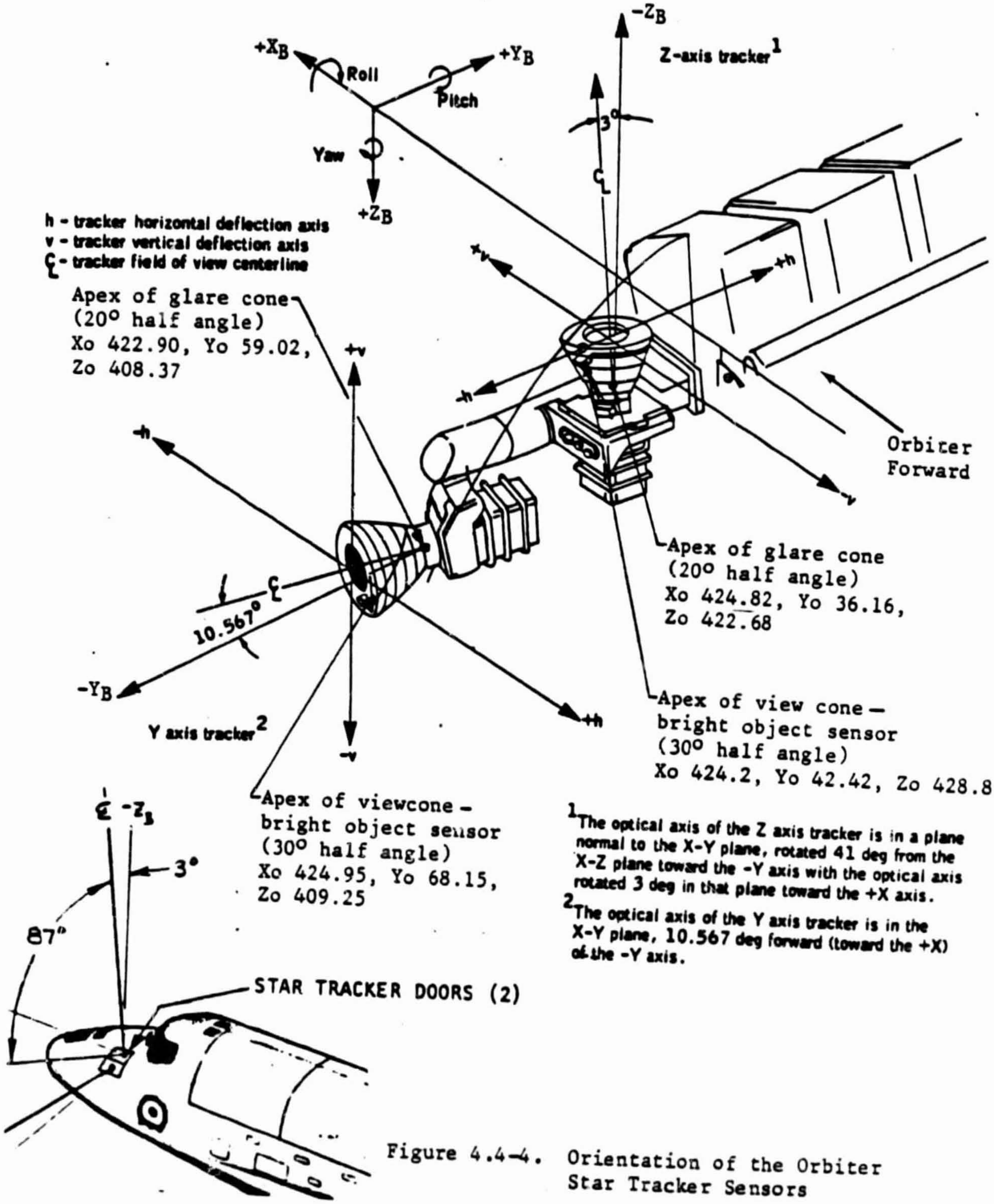


Figure 4.4-3. Antenna Locations



[1] The optical axis of the Z axis tracker is in a plane normal to the X-Y plane, rotated 41 deg from the X-Z plane toward the -Y axis with the optical axis rotated 3 deg in that plane toward the +X axis.

[2] The optical axis of the Y axis tracker is in the X-Y plane, 10.567 deg forward (toward the +X) of the -Y axis.

Figure 4.4-4. Orientation of the Orbiter Star Tracker Sensors

Table 4.4-2. Checklist: Guidelines for Large Space Structure Orientation and Configuration with Respect to Overall Orbiter Geometry and Functions

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) Large space structures should be configured to avoid plumes of orbiter RCS when the orbiter control system is used for attitude stabilization. (R-3) • Each case is subject to chemical, thermal and thrust dispersion analysis to determine degree of impingement permissible. • See Section 4.1.6 for plume data.	Constraints configuration of structure. (Guideline was derived for this document, based on previous contractor experience)
( ) When possible, space structures should be located to permit visual observation from orbiter cabin windows. (R-3) • See Section 4.9.6 for window viewing angles. • Closed circuit TV is probably the major method for providing visibility for large space structure construction and operations.	General Dynamics 1979: Affects design of construction fixtures, shape of large space structures.

(respectively) relating to the subjects previously noted, with reference to Figures, Tables or other documents which may further define the constraints or guidelines applicable.

4.4.2 Support Interfaces Constraints

This section provides information concerning specific, detail configuration constraints for on-orbit payload attachments and deployment considerations. These considerations are potential modifications and additions to the basic requirements outlined in Section 3.2, which emphasized payload supports and connections constraints for delivery to orbit and return. On-orbit conditions differ in respect to the configuration of opened payload bay doors and radiators, deployed positions for RMS and antennas, EVA operations, and operational requirements for various other systems, such as TV cameras and payload installation and deployment aids.

In addition, it is possible to consider temporary use of payload support latches on orbit without regard for load criteria imposed for lift-off or descent and landing.

Table 4.4-3 lists constraints in the subject category. The majority of these constraints relate to hardware installed on or near the payload bay longerons of the orbiter. Further description follows.

4.4.2.1 Ku-Band Antenna Deployment Clearance and Obscuration

In addition to the deployment considerations clearance may need to be provided for the full range of motion of the Ku-Band antenna dish. Such geometric constraints have not been specified to date. However, control drawings of the mechanisms are available from which such envelopes can be developed (Hughes 1979).

If the Ku-band antenna is to be operated in conjunction with the attached large space structure, consideration must be given to the antenna viewing angles which may be obscured. The line of sight obscuration zone for the high gain antenna beam, created by the orbiter structure and a maximum size package payload, is defined by Tables 4.4-4 and 4.4-5 which present the pertinant analysis constants and contour data respectively. The contour outlines are depicted graphically in Figure 4.4-6. By definition, the information given applies to the Communications A/Radar Deployed Assembly when in the fully deployed position. The numbered contour segments identified in Figure 4.4-6 refer to specific curve segments using a procedure defined in the technical document MC409-0025.

Note that the boundary defined by the numbered contour segments is 5° outside the boundary defined by the orbiter and maximum payload envelope, i.e., the boundary defined by the numbered contour segments includes a 5° guard band around the orbiter and payload envelope.

Table 4.4-3. Checklist: Detail Constraints on Support Attachments, Payload Deployment Clearances and Equipment Configurations for On-Orbit Operations

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) Clearance shall be provided for deployment and operation of the Ku-band antenna(s). (R-1) • Normal installation is starboard side, centerline mounted at Station X_o 589. • Port side installation is (TBD). • See Section 4.11.4. • See Figure 4.4-5 for deployment envelope. • See Section 4.4.2.1 for obscuration envelope details.	Rockwell 1978 (a): MC 409-0025 Location forward of radiators seldom poses configuration constraints, but analysis is required.
( ) Clearance shall be provided for deployment and operation of the RMS when this item is installed. (R-1) • RMS shoulder joint is deployed outboard to provide added clearance for large payload extraction and stowage while on orbit. • See Section 4.5 for RMS details.	NASA 1980 (a): JSC-07700, Vol. XIV (ICD 2-19001) Affects design of assembly fixtures attached to longerons.
( ) Clearance shall be provided for the slide-wire for EVA safety tethers, located above the longeron. (R-2) • See details of operational slidewire concepts in Section 4.8.1.5. (NOTE: Final design still in progress)	Rockwell ERB Briefing (Unpublished) Affects detail design for EVA provisions and assembly fixture extending out from longerons.
( ) Attach points near the longerons for on-orbit re-locations of construction fixtures, etc. should avoid interferences with items listed in Table 3.2-1. • Radiator hose crossovers. • Cargo bay door actuators. • Electrical crossovers.	Rockwell 1979 (c): Drawing V070-362003 Design should consider on-orbit configuration of these devices.

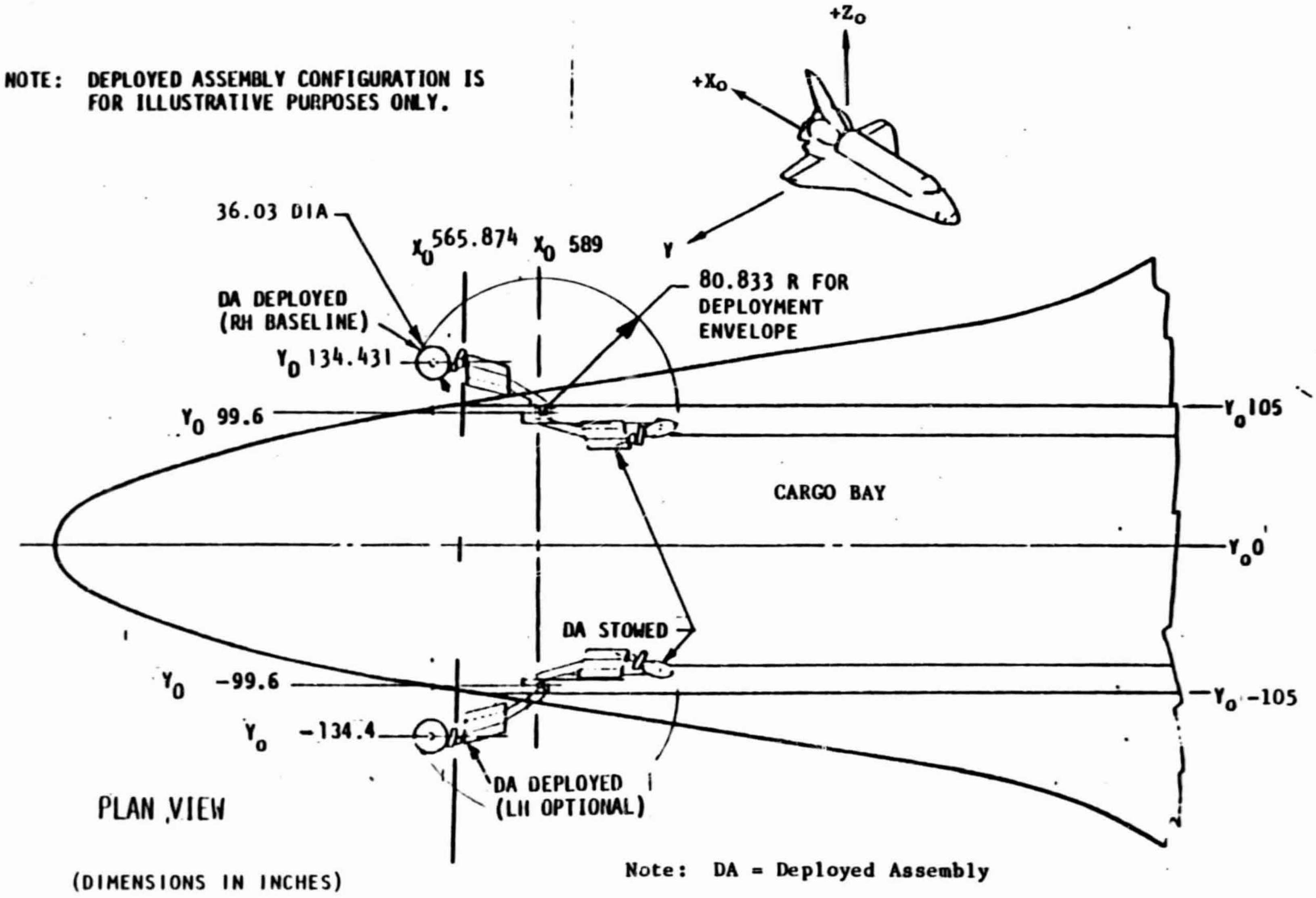


Figure 4.4-5. Ku-Band Communications Equipment Location

Table 4.4-4. Constants Used in High Gain Antenna Analysis Obscuration Zone

Pivot point location (intersection of α and β axes)	X_o: 568.841 in. Y_o: 135.365 in. Z_o: 443.875 in.
α-axis (structural coordinates)	(-0.3907311285, 0.9205048535, 0.0)
Beam focus point offset from pivot point	21.01 in.
Beam half-width	18.015 in.
Rotation limits	$0^\circ \leq \alpha \leq 150^\circ$ or $0^\circ \geq \alpha \geq -210^\circ$ $+75^\circ \geq \beta \geq -90^\circ$
Payload cylinder envelope	15 ft diameter 60 ft length Centroid: X_o = 930.0 in Y_o = 0.0 in Z_o = 400.0 in

Table 4.4-5. Obscuration Zone Contour Data

POLYNOMIAL COEFFICIENTS FOR EQUATION

$$\text{Roll} = \sum_{i=o}^{4} a_i \cdot (\text{Pitch})^i \quad (\text{Degrees})$$

Curve Number	Pitch Range (Degrees)	a_o	a_1	a_2	a_3	a_4
1	$-1<P<3$	-113.25	-22.25	0.	0.	0.
2	$-58<P<-1$	-91.105263	-0.105263	0.	0.	0.
3	$-85<P<-58$	-125.814815	-0.703704	0.	0.	0.
4	$-85<P<-80$	1039.0	13.0	0.	0.	0.
5	$-80<P<-64$	99.0	1.25	0.	0.	0.
6	$-66.5<P<-64$	-864.2	-13.8	0.	0.	0.
7	$-66.5<P<-23$	58.392298	-0.278239	-5.282522E-3	0.	0.
8	$-23<P<-19$	21.75	-1.75	0.	0.	0.
9	$-19<P<24$	55.0	0.	0.	0.	0.
10	$24<P<70$	49.793625	0.301218	-8.595869E-3	2.085508E-4	0.
11	$(100<P<140)$	$(P = 70)$				
12	$50<P<70$	-11005.08532	757.881672	-19.192845	0.215179	-9.024330E-4
13	$0<P<50$	175.0	-0.22	0.	0.	0.
14	$(175<P<80)$	$(P = 0)$				

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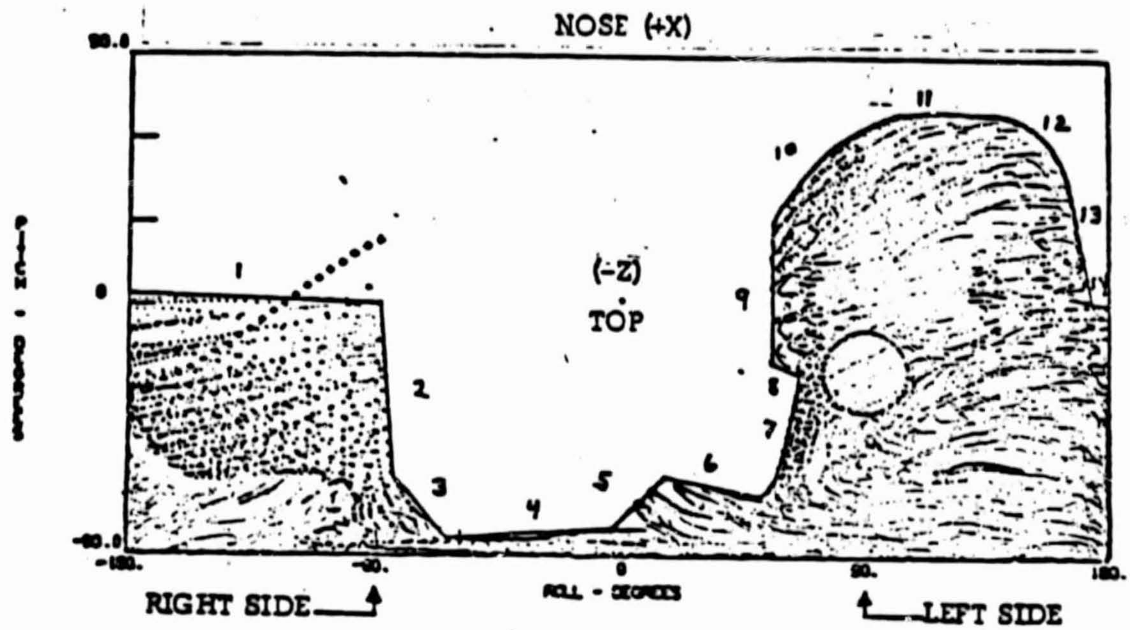


Figure 4.4-6. Obscuration Zone Contour for Antenna Main Beam LOS

4.4.2.2 S-Band Obscuration Considerations

The S-band antennas mounted on the top of the crew cabin are most likely to be obscured by metal portions of large space structures and construction fixtures which extend above the payload bay. These antennas nominally have a hemispherical view envelope. Note that alternate S-band antennas are available on the sides of the crew cabin (see Figure 4.4-3).

4.4.3 Procedures Constraints and Options

A small number of procedural constraints must be considered for payload deployment and stowage operations. These mainly relate to items hinged to the cargo bay longerons in such a way as to restrict deployment of large payloads. Such constraints are listed in Table 4.4-6. This table does not include opening of the cargo bay doors and deployment of attached radiators, since such activity is normally performed as soon as feasible upon reaching orbital altitude, regardless of payload.

4.4.4 Equipment Options

The Space Transportation System has under development, or consideration for development, several aids which potentially can be used for deployment of payloads, including the various construction, maintenance and checkout functions contemplated for large space structures. Although these devices were originally conceived for more limited payload handling usages, there are many operations involving large space structures which are similar in some respects to the original design requirements.

The major group of devices related to the payload handling operations is called the Payload Deployment and Retrieval System (PDRS). This includes "all orbiter and payload systems required for accomplishing the deployment and/or retrieval of payloads, on-orbit" (NASA 1980 (a)). It is further described as "those orbiter systems which are utilized uniquely for payload deployment, retrieval, or special handling operations. This includes the Remote Manipulator System (RMS) and the payload retention system. Other orbiter systems which are required for deployment and/or retrieval but are also used for other purposes include the guidance, navigation and control system, the closed circuit television system, and cargo bay lights." (NASA 1980 (a)).

In the broad context of construction and operations involving large space structures one may include such systems as the Manned Maneuvering Unit (MMU), the Crew Space Suits (Extravehicular Maneuvering Unit (EMU)) and associated mobility aids, and several devices considered for future development. Table 4.4-7 lists the actively used devices (or sets of devices), together with brief descriptions and suggested usages for large space structures. Further details about the equipment are developed in Sections 4.5 through 4.8. In addition, one may consider the payload handling system to include the active payload retention latches, and various cradles and pallets.

Table 4.4-6. Checklist: Constraints on Procedures for On-Orbit Payload Handling Operations

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) The Ku-Band antenna generally must be deployed prior to removing large payloads from the vicinity of the forward end of the payload bay.	Only affects payloads which extend forward of RMS shoulder.
( ) In general, the RMS arm must be rotated outboard on its supports before checkout and deployment for use. Checkout required prior to use. • See Section 4.5 for timelines.	Included in checkout of RMS prior to payload operations.
( ) The EVA slidewire may need to be deployed prior to removal of large payloads. • See Section 4.8.1.5 for dimensional and functional interfaces.	Rockwell Briefing (Unpublished)
( ) Obstructing equipment near Xo 576 bulkhead may need to be removed prior to EVA.	Function of stowage and operations requirements.
( ) The orbiter primary RCS thrusters may not be fired while the RMS arm is transporting payloads. • Specific analysis would need to be performed for the arm position and masses involved.	Rockwell Briefing (Unpublished) • The RMS arm may be overstressed under such conditions. • Generally is not a problem for construction.

Table 4.4-7. Equipment Options to Assist Payload Deployment, Construction, Maintenance, Checkout and Stowage on Orbit

EQUIPMENT DESCRIPTION	POTENTIAL USAGES
o <u>Remote Manipulator System</u> Control and monitoring devices, 50 ft. long manipulator arm(s) and basic payload-handling end effector. The RMS is a versatile, remotely controlled device, much like a crane, which can remove payloads from the vicinity of the cargo bay and return them, perform a variety of transport and assembly functions and act as a camera or light stand. It is more fully described in Section 4.5. The RMS is a funded system under development for Space Shuttle operations as a standard item.	o Transport modules o Transport beams o Hold tools o Transport tools, containers or beam builder machines. o Transport EVA crew on open cherry picker. o Perform assembly and removal operations. o Assist in inspection o Perform berthing operations (orbiter to spacecraft or module to module).
o <u>Payload Installation and Deployment Aid (PIDA)</u> Removal and installation of very large payload items relative to the payload bay may be very difficult for an RMS alone, due to small clearances and limited visibility. A proposed method for facilitating such operations uses remotely controlled PIDA arms, which would lift a large payload above the payload bay longerons where it can be more readily handled by the RMS. The PIDA concept prototype as developed by NASA, is described in Section 4.6. The PIDA is not currently funded for Shuttle usage, but is considered a likely candidate to solve future payload handling problems related to large space structures.	o Move large payload modules from payload bay stowage to location clear of longerons. o Return large payload modules to stowed position in payload bay. o Potential aid for attaching modules to structure. o Holding fixture for performing maintenance and repair.

Table 4.4-7. Equipment Options to Assist Payload Deployment, Construction, Maintenance, Checkout and Stowage on Orbit (Continued)

EQUIPMENT DESCRIPTION	POTENTIAL USAGES
o <u>Flight Support System (for MMS)</u> The MMS Flight Support System consists of a set of cradles, one of which contains a remotely controlled tilting and positioning mechanism which could be useful for limited handling of certain types of large space structure components. This system is more fully described in Section 4.7.1.	o Tilt large payload modules to vertical position, perpendicular to payload bay axis. o Rotate in X-Y plane.
o <u>Spacelab Pallets</u> Handling of components for large space structures may be facilitated by mounting them on structural pallets such as used in the spacelab program as described in Section 4.7.2.	o Provides structural base for multiple parts stowage, sensor sets, and other groups of mechanisms.
o <u>Manned Maneuvering Unit</u> The manned maneuvering unit is a small space vehicle which is donned by EVA (pressure suited) astronauts for transport operations in the vicinity of the orbiter. It is more fully described in Section 4.8.2.3, as a part of the EVA aids. The MMU is a fully funded item for the Space Shuttle program.	o Manned transport of small objects (by EVA crew). Transport large objects using two crew. o Stabilize EVA crew man for assembly work, inspection or removal. o Especially useful for operations beyond reach of the RMS. o Assist in maintenance operations.
o <u>Open Cherry Picker (Manned Remote Work Station)</u> The open cherry picker concept consists of an open support platform which can be attached to the end of a manipulator arm (such as the RMS). An EVA astronaut is supported at his feet on the platform, and is provided with manipulator arm controls, lights, tool supports and a stabilizer arm, which can grasp payload structures. Prototypes of the open cherry picker have been	o Assist installation of modules and removal of modules. o Assist transport of moderate size modules and beams. o Transport EVA crew within reach envelope of manipulator. o Stabilize crew for manual assembly, removal, checkout, and adjustment operations.

Table 4.4-7. Equipment Options to Assist Payload Deployment, Construction, Maintenance, Checkout and Stowage on Orbit (Continued)

EQUIPMENT DESCRIPTION	POTENTIAL USAGES
tested, but the project is not yet funded as an approved Space Shuttle component. The open cherry picker concept is more fully described in Section 4.8.2.4.	
o <u>Extravehicular Mobility Unit (EMU) & EVA Mobility Aids</u>	
The orbiter crew can don full pressure suits (Extravehicular Mobility Units) and manually handle equipment and materials for assembling and performing other operations involving large space structures. In so doing, they will utilize handholds, handrails, tethers, foot supports and tools especially designed for aiding mobility and providing body stabilization in space. Various aspects of EVA operations are described in Section 4.8. A 4-psi EMU is currently undergoing funding development for Space Shuttle operations. The orbiter provides a baseline set of handholds and tethers for EVA operations in the cargo bay.	o Transport moderate sized objects. o Guide large objects for joining. o Perform assembly/maintenance o Make electrical connections and fluid connections. o Inspect, monitor o Adjust o Latch/unlatch o Release/deploy o Retract, remove objects. o Control manipulators.
o <u>Special Devices</u>	
A wide range of unique items has been studied for performing various phases of construction and other operations involving large space structures. Such devices include: fixtures, jigs, dispensers (rolls, magazines), rail conveyors, special handling and positioning aids, teleoperator devices, cranes and self-maneuvering vehicles. These devices usually are conceived as unique to the particular project,	o As required for defined operations

Table 4.4-7. Equipment Options to Assist Payload Deployment, Construction, Maintenance, Checkout and Stowage on Orbit (Continued)

EQUIPMENT DESCRIPTION	POTENTIAL USAGES
although they may have other, more general uses. They may interface with orbiter structures and subsystems or only with the large space structure project. Such devices are not described in this document because they are not currently considered as a part of the Shuttle system.	

4.5 REMOTE MANIPULATOR SYSTEM CONSTRAINTS AND GUIDELINES

The Shuttle Remote Manipulator System (RMS) is the most significant, operationally useful portion of the payload deployment and retrieval system (PDRS).

It is the primary means for remotely handling carriers and payloads, and for performing additional operations of which a manipulator is capable. RMS (as defined herein) consists of the manipulator retention latches, the manipulator positioning mechanism, the manipulator jettison subsystem control and monitoring devices, the manipulator arm, and a basic payload-handling end effector as shown in Figure 4.5-5. The cargo bay portion of the RMS is 50 feet in length and is mounted on the port (standard) or starboard (optional) side of the vehicle. The RMS arm is stowed outside the cargo bay envelope. However, the stowage position restricts the nominal clearance space for removal of large payloads. Therefore, the arm is supported on hinged retention mechanisms permitting it to be rotated outboard as shown in Figure 4.5-3. The RMS arm accommodates lights, TV cameras and electronics for illumination and viewing of remote locations as described in Section 4.9.

Additional dimensional details of the RMS arm are shown in Figure 4.5-4. These dimensions must be considered in planning for clearances in transport and assembly operations using the RMS.

A second (optional) manipulator arm can be installed on the starboard longeron if compatible with STS operational constraints. The weight of the second manipulator is weight chargeable to the payload. This weight is 905 lbs., including the standard end effector and TV at the wrist (TV also mounted at the elbow is an additional 28 lbs). Capability is provided to operate two manipulators in serial-only (non-simultaneous) operations. Capability is provided to hold the payload with the manipulator arm in a chosen position while operating the second manipulator arm.

The capability is provided to jettison each manipulator arm assembly. Sufficient redundancy is provided to insure that the payload can be released prior to RMS arm jettison (NASA 1980 (a)).

4.5.1 General RMS Capabilities

Table 4.5-1 lists RMS general capabilities and constraints applicable to large space projects construction, maintenance, checkout and related operations. More detailed and specific constraints and capabilities relating to reach and performance are presented herein.

4.5.2 RMS Reach and Performance Characteristics

A highly critical constraint on use of the RMS for space construction and many other operations involving large space structures design is the RMS functional reach capability.

Figures 4.5-10 through 4.5-12 depict the maximum reach capability of the RMS. These figures indicate the end effector tip reach limitations, but do not imply that the full arm torque/force capability exists along the reach arcs described. The limits shown in the figures are actually "contours" with

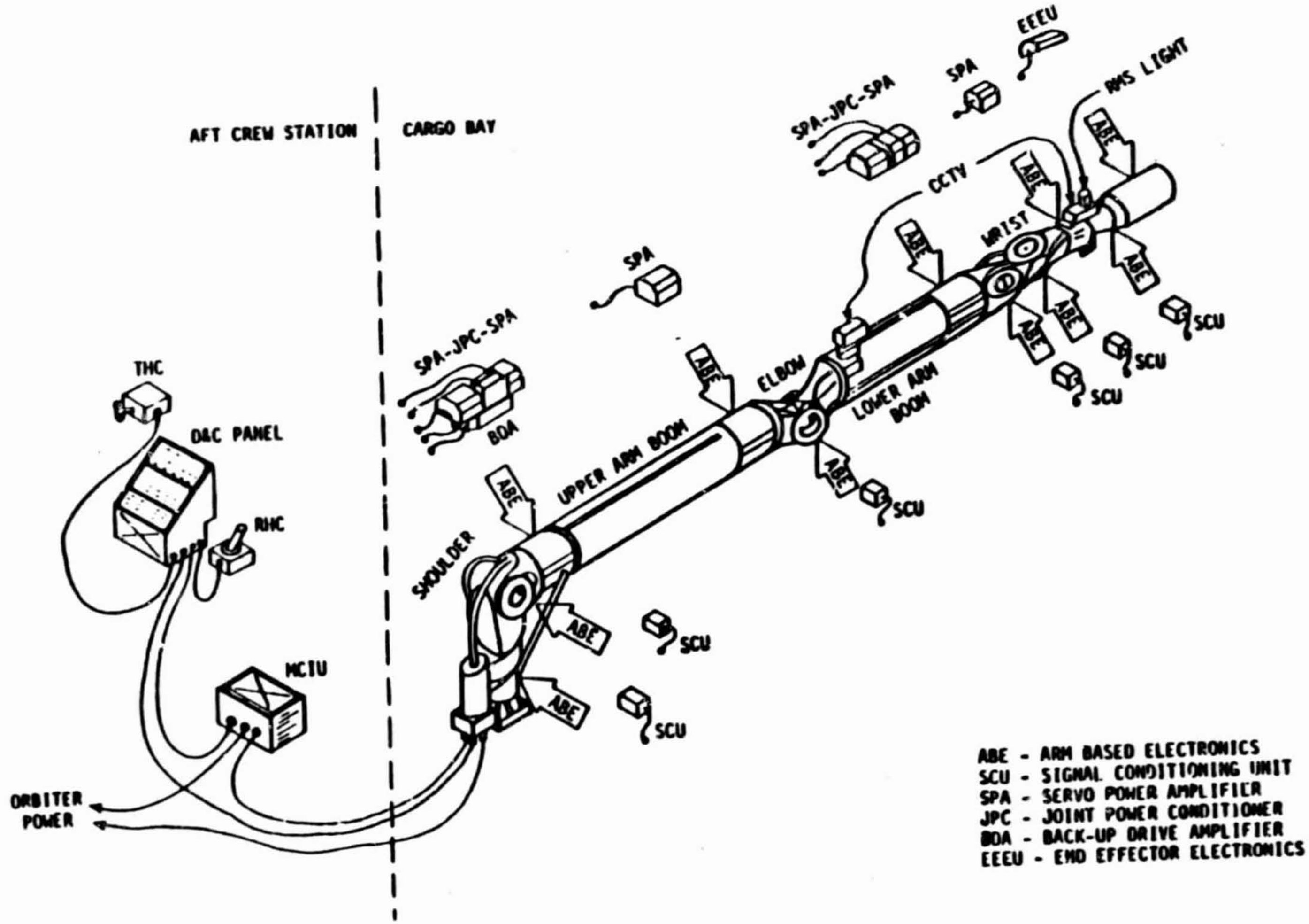


Figure 4.5-1. Orbiter Remote Manipulator System Electronics and Drives

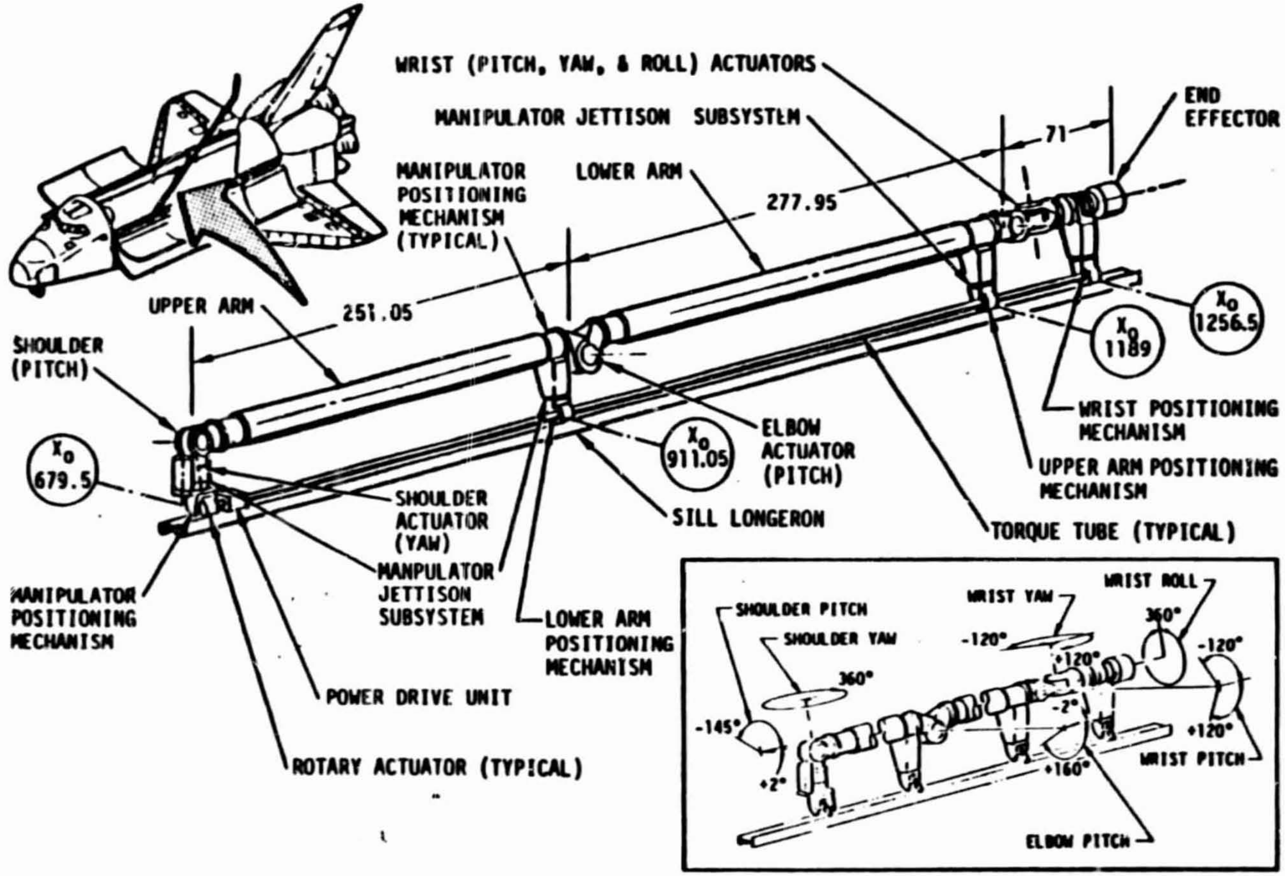


Figure 4.5-2. Remote Manipulator System Mechanisms

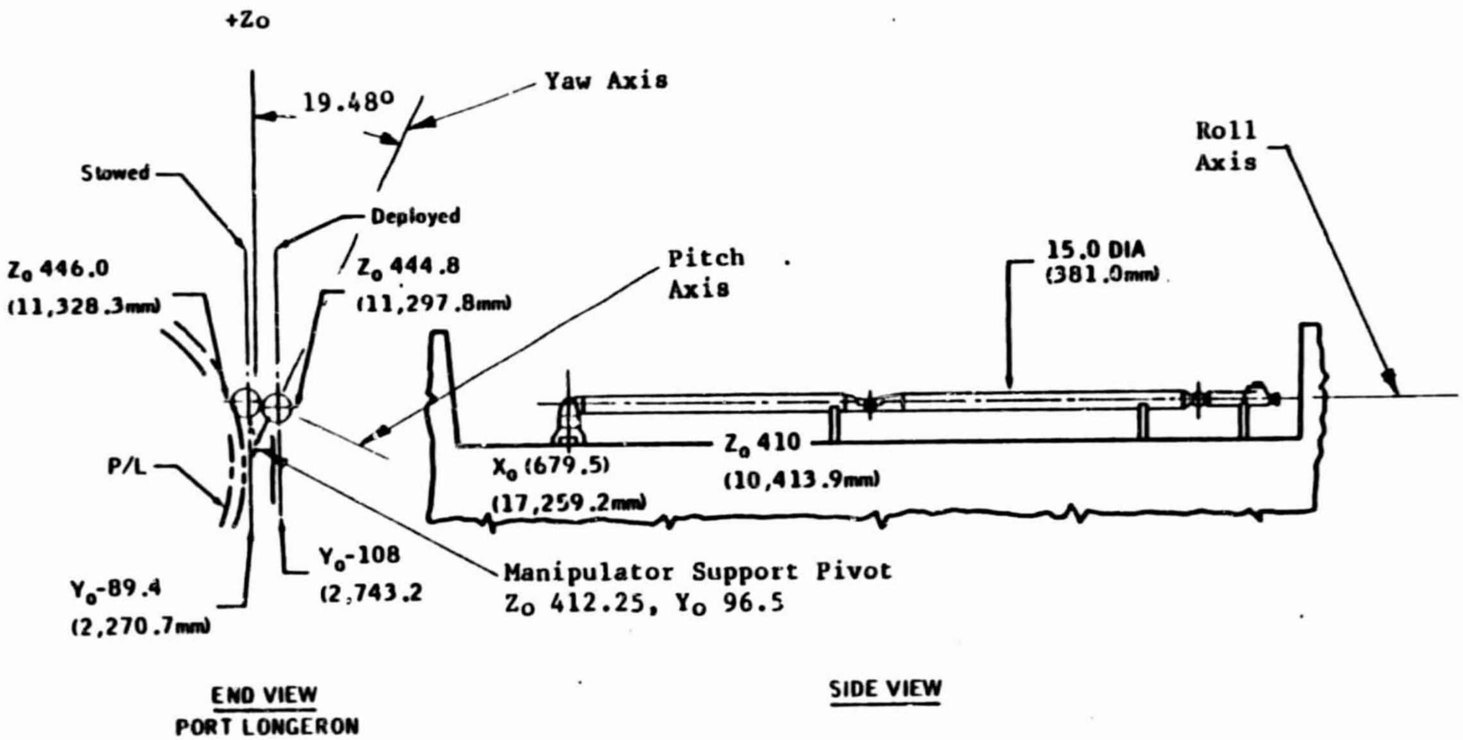


Figure 4.5-3. RMS Locations, Stowed and Deployed

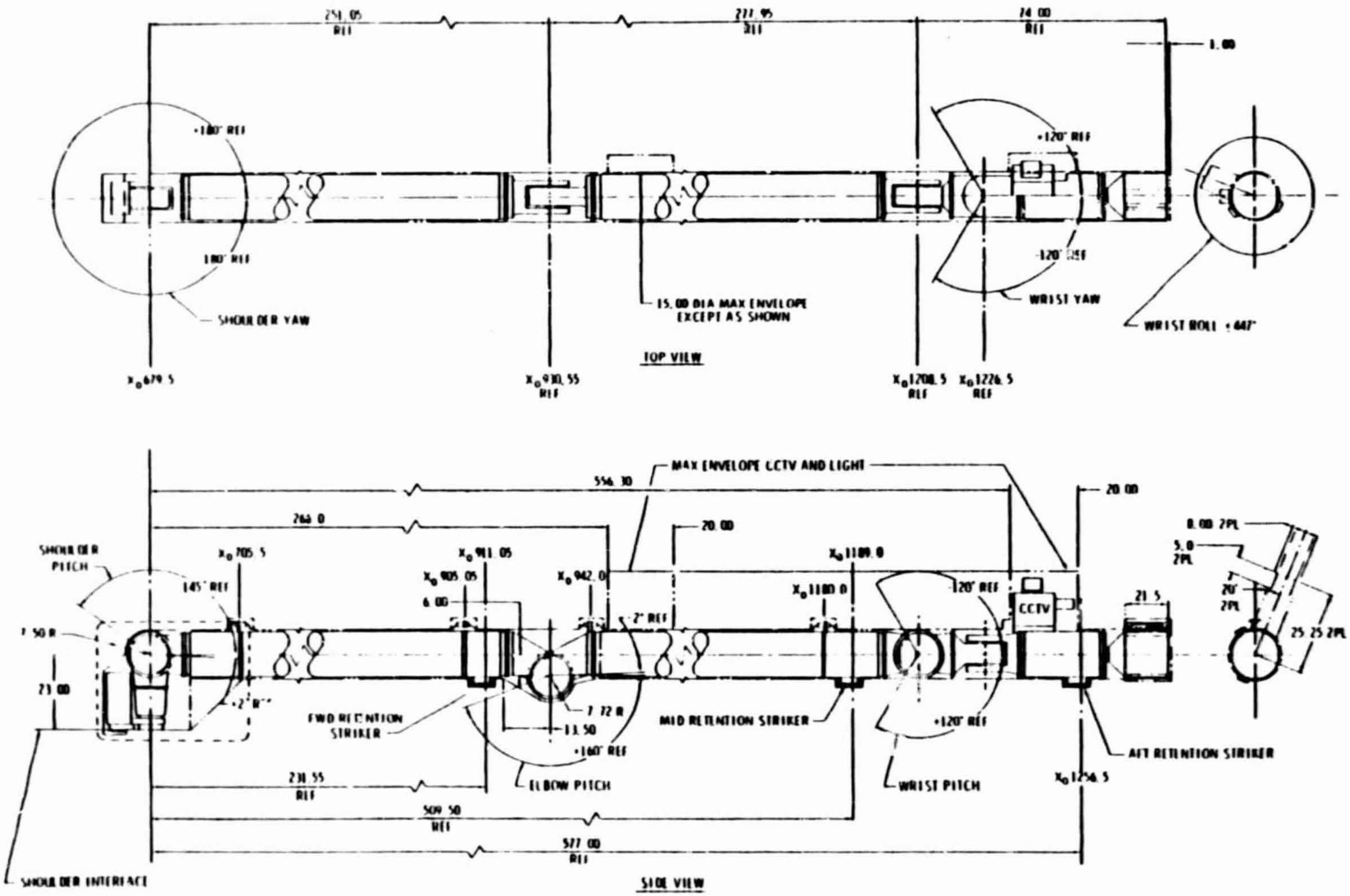


Figure 4.5-4. RMS Arm Dimensions and Joint Angle Limits

Table 4.5-1. Checklist: RMS General Capabilities and Constraints

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) In orbit, the RMS is capable of deploying a maximum envelope (approximately 15 feet diameter x 60 feet long), maximum weight 65,000 lbs (29,484 kg) payload. (R-1)	NASA 1980 (a)
( ) Under normal operational conditions, the RMS is capable of retrieving a 32,000 lb (14,515 kg) payload and placing it in a position for engagement with the cargo retention system in the cargo bay for return to earth. (R-1)	
• Under clearly defined contingency conditions, the RMS is capable of retrieving a maximum weight payload (65,000 lbs) in a non-time constrained operation. (The requirement for retrieval of a payload weighing more than 32,000 lbs could be to correct a malfunction in the payload and subsequently redeploy the payload. The orbiter entry and landing is normally constrained to payloads weighing less than 32,000 lbs.).	
( ) Deployment of a maximum envelope, maximum weight payload can be accomplished in approximately 25 minutes from release of payload tiedown to release of the payload at the RMS fully deployed position. (R-1)	
( ) The RMS is capable of supporting up to a maximum weight payload in the preplanned deployed position under the attitude stabilization loads imposed by the orbiter vernier RCS (operating and minimum impulse mode). (R-1)	
. Also see Section 4.5.2.	
( ) Within the operational reach limits of the RMS, the orbiter vehicle will have the capability to deploy and/or retrieve single or multiple payload elements on-orbit during a single flight. Within defined limitations, the RMS may also be used to place payloads on, or dock payloads with, a suitably configured and stabilized body. (R-1)	NASA 1980 (a) Capability is useful for construction of large space structures

Table 4.5-1. Checklist: RMS General Capabilities and Constraints (Continued)

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) Standard end effector provided with the RMS. (R-1) . See Figure 4.5-5, a perspective illustration of the end effector.	
( ) The standard snare end effector is designed to mate with a target and grapple fixture (provided by the payload). (R-1) . See Figures 4.5-6, 4.5-7 and 4.5-8 for dimensions and grapple procedure details.	
( ) The RMS also has the on-orbit capability of grappling a special purpose end effector (payload provided) and providing an electrical connection across the interface for control of the special end effector. This connection may also be used to provide power and/or signals to payloads, if the payload provides the compliance and mating connector within its grapple fixture. The electrical connector is fitted on the outside of the standard end effector at the end effector/payload interface as indicated in Figure 4.5-5. The power for the special purpose end effector or payload is taken from the 28V arm power bus. Wiring is provided from the orbiter flight deck on-orbit station distribution panel to the RMS shoulder interface, and from there to the face of the standard end effector. (R-1)	
( ) Controls and displays for command or signals to the special purpose end effector or to payloads must be provided by the payload. The wire gaging and quantities available for this interface are shown in Figure 4.5-9. (R-1)	NASA 1980 (a) Affects design of displays and controls for on-orbit operations.
( ) On-orbit stowage of any special purpose end effector must be provided by the payloads. The RMS standard end effector may be exchanged on the ground with a special end effector for use on-orbit. (R-1)	Affects design of special end effectors and systems for space construction.

Table 4.5-1. Checklist: RMS General Capabilities and Constraints (Continued)

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) Insofar as it is intrinsic in the RMS design for payload deployment and retrieval, the RMS may also be used to perform other tasks in support of payload servicing and as an aid in translating an EVA crewman to assist in extra vehicular activities. An EVA hand-hold is an integral part of the RMS end effector. (R-1) . Also see Sections 4.8.2.1 and 4.8.2.4, which discuss EVA assistance by the RMS.	

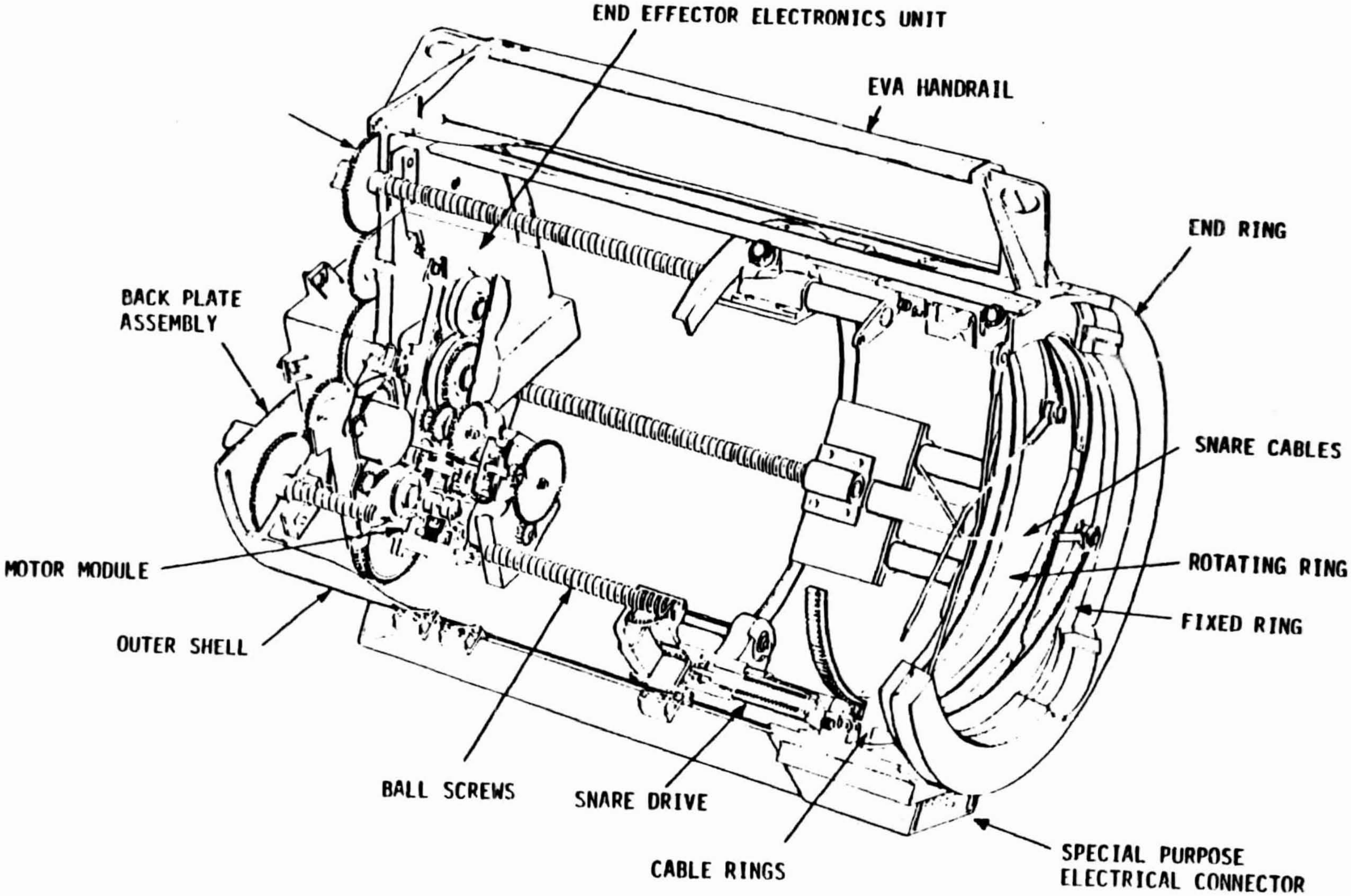


Figure 4.5-5. Standard End Effector

C-3

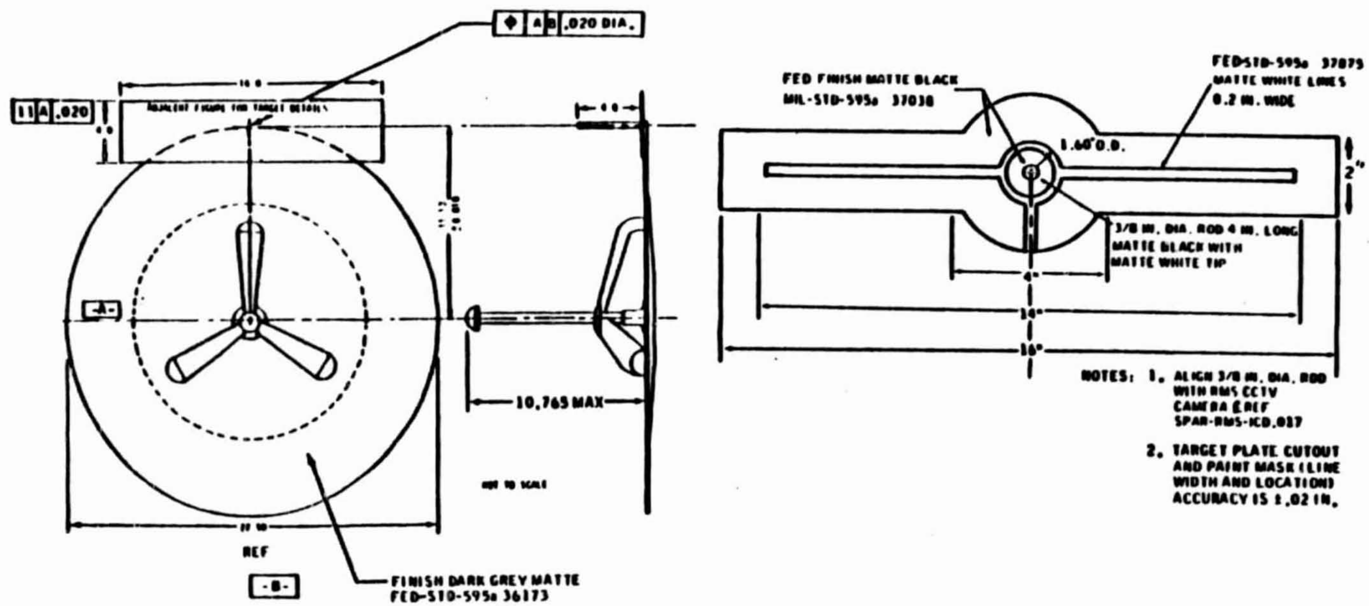


Figure 4.5-6. Grapple Fixture and Target

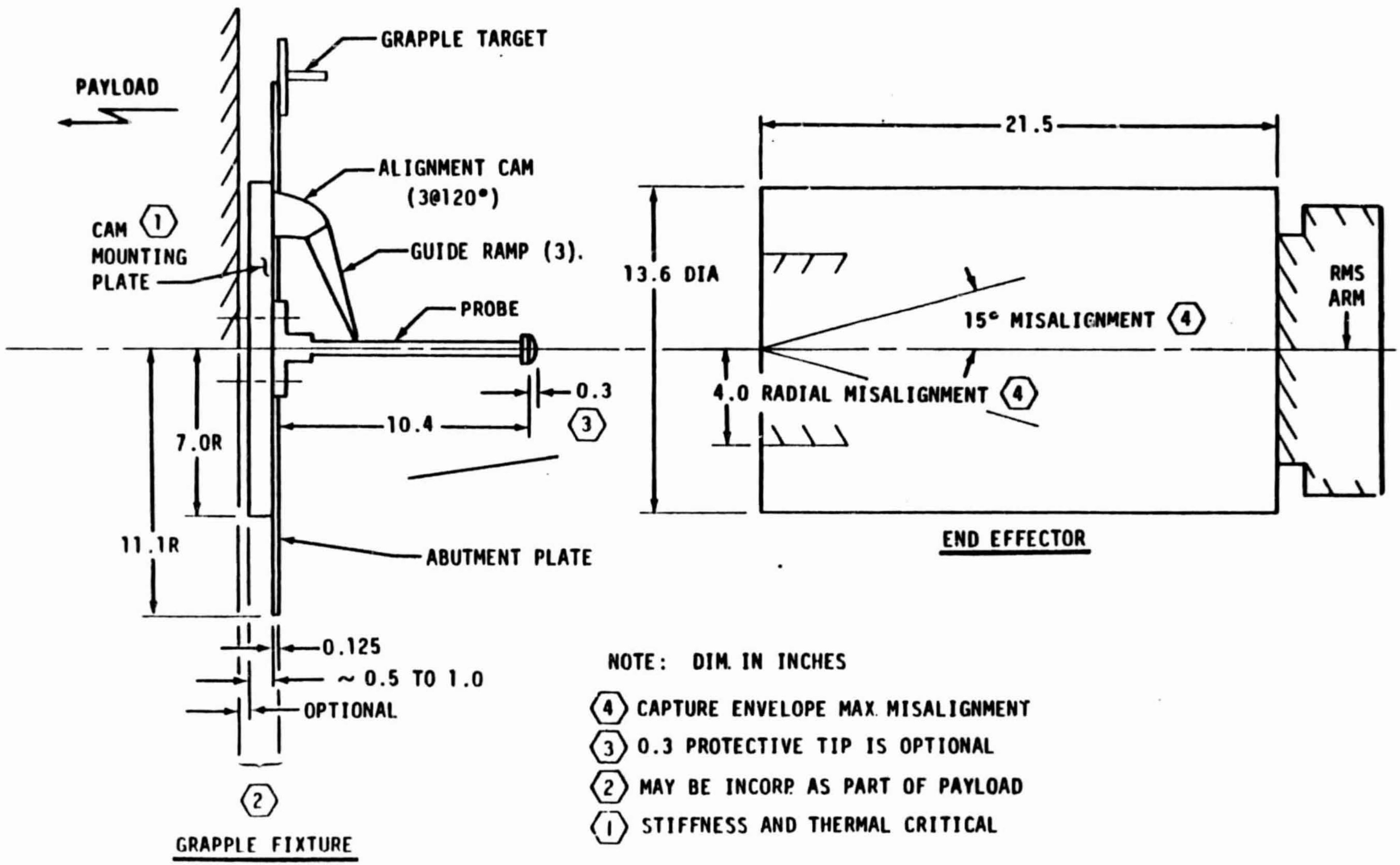


Figure 4.5-7. RMS Standard End Effector and Grapple Fixture Envelope Schematic

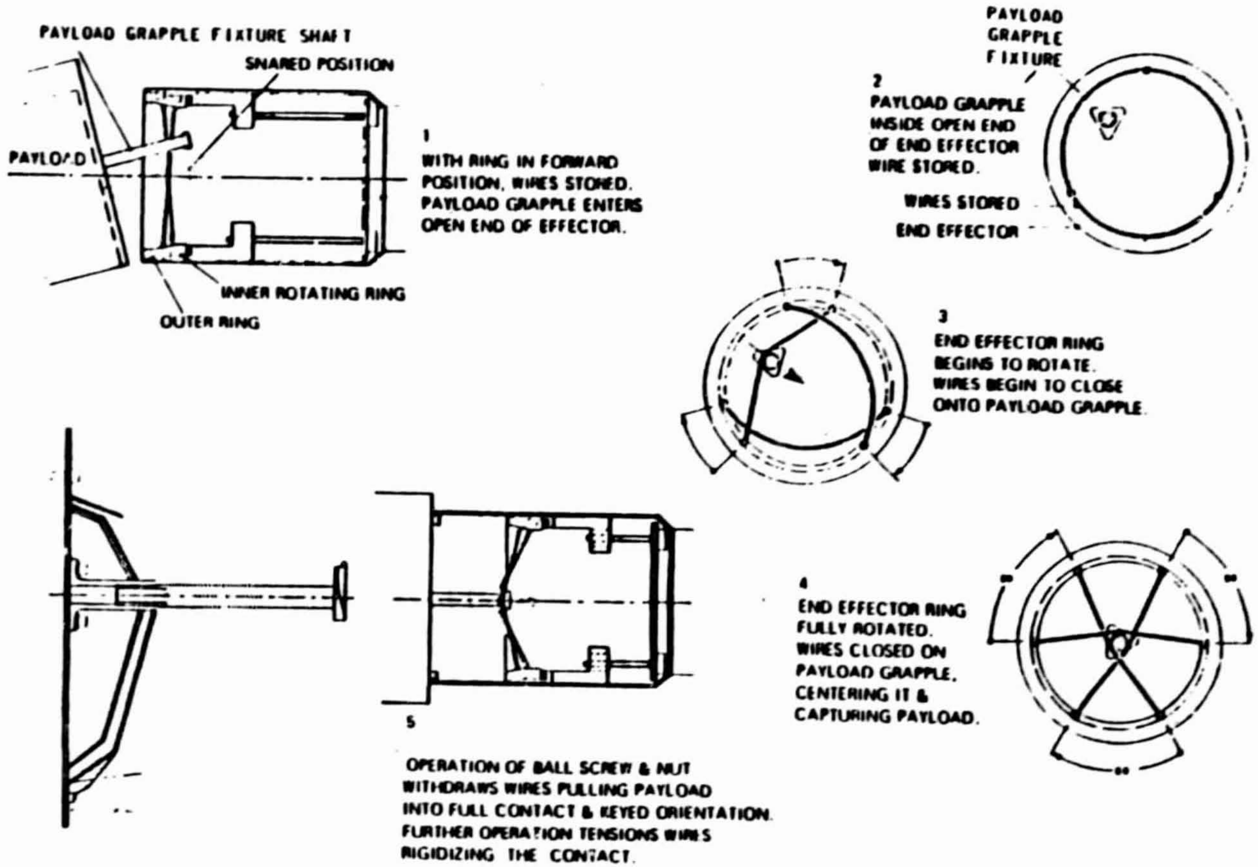


Figure 4.5-8. Standard End Effector - Capture and Rigidize Sequence

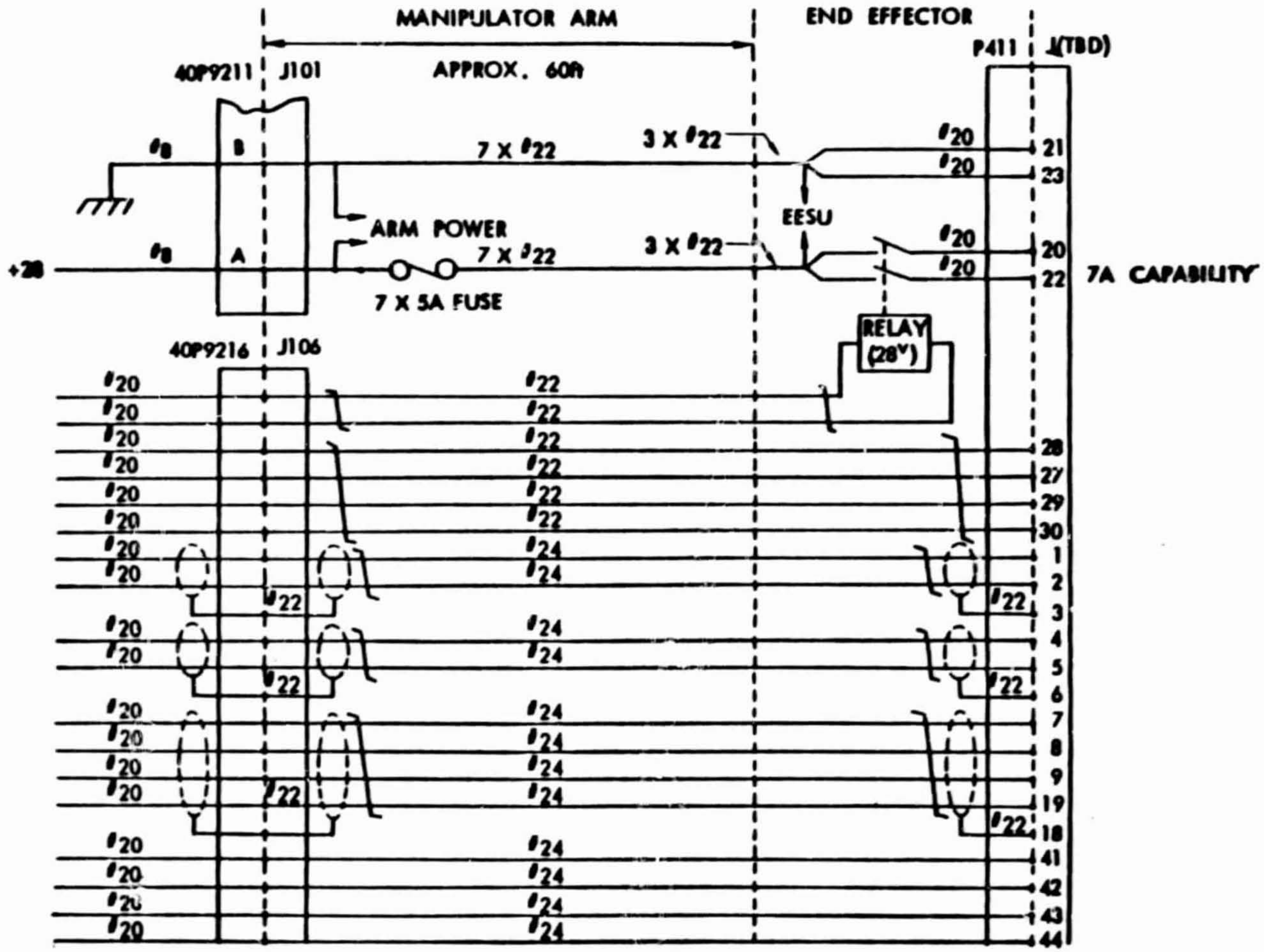


Figure 4.5-9. RMS Standard End Effector/Special Purpose End Effector Electrical Interface (NASA 1980(a), Para. 8.1.1.1(b))

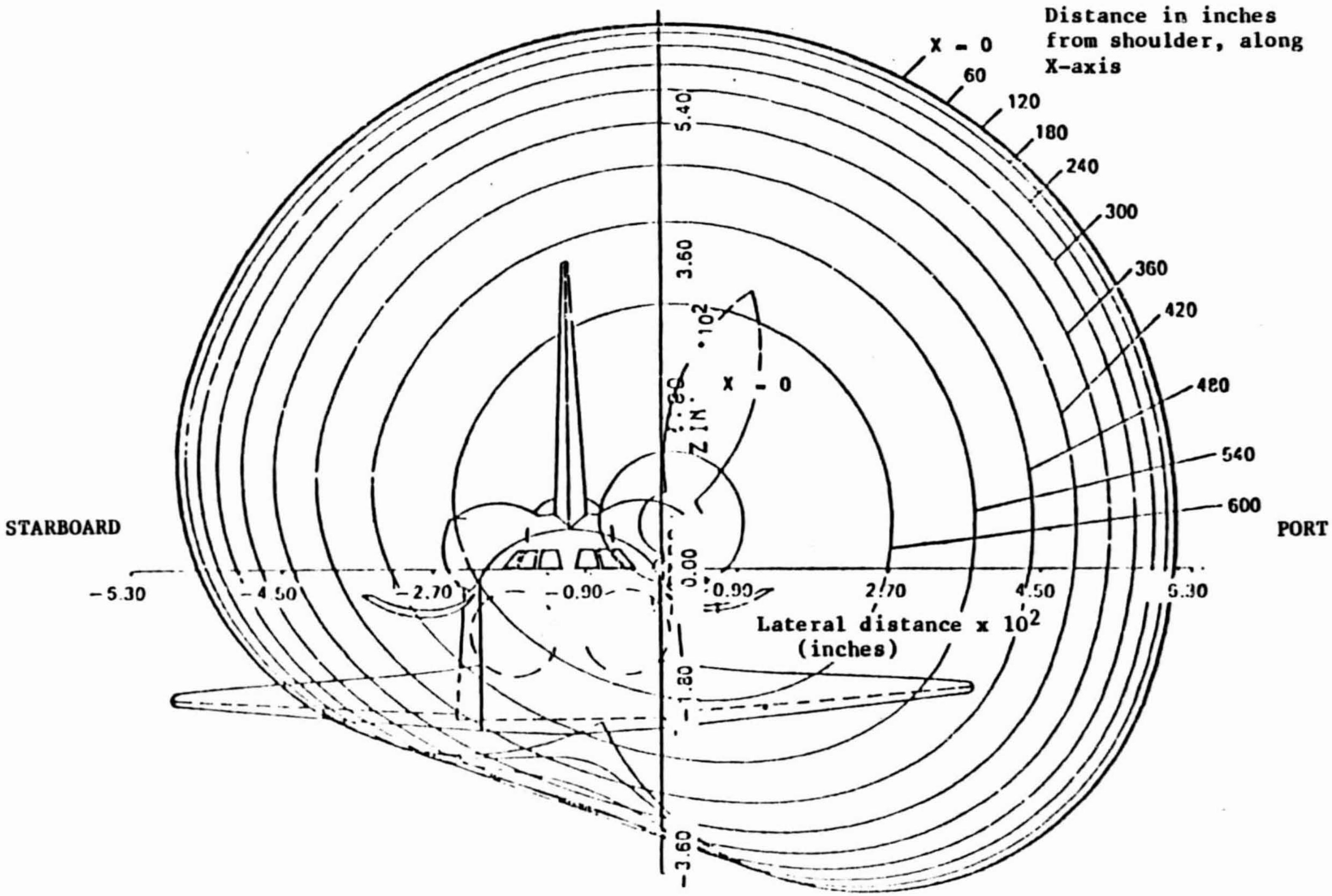


Figure 4.5-10. Front View of Orbiter and X-Contours

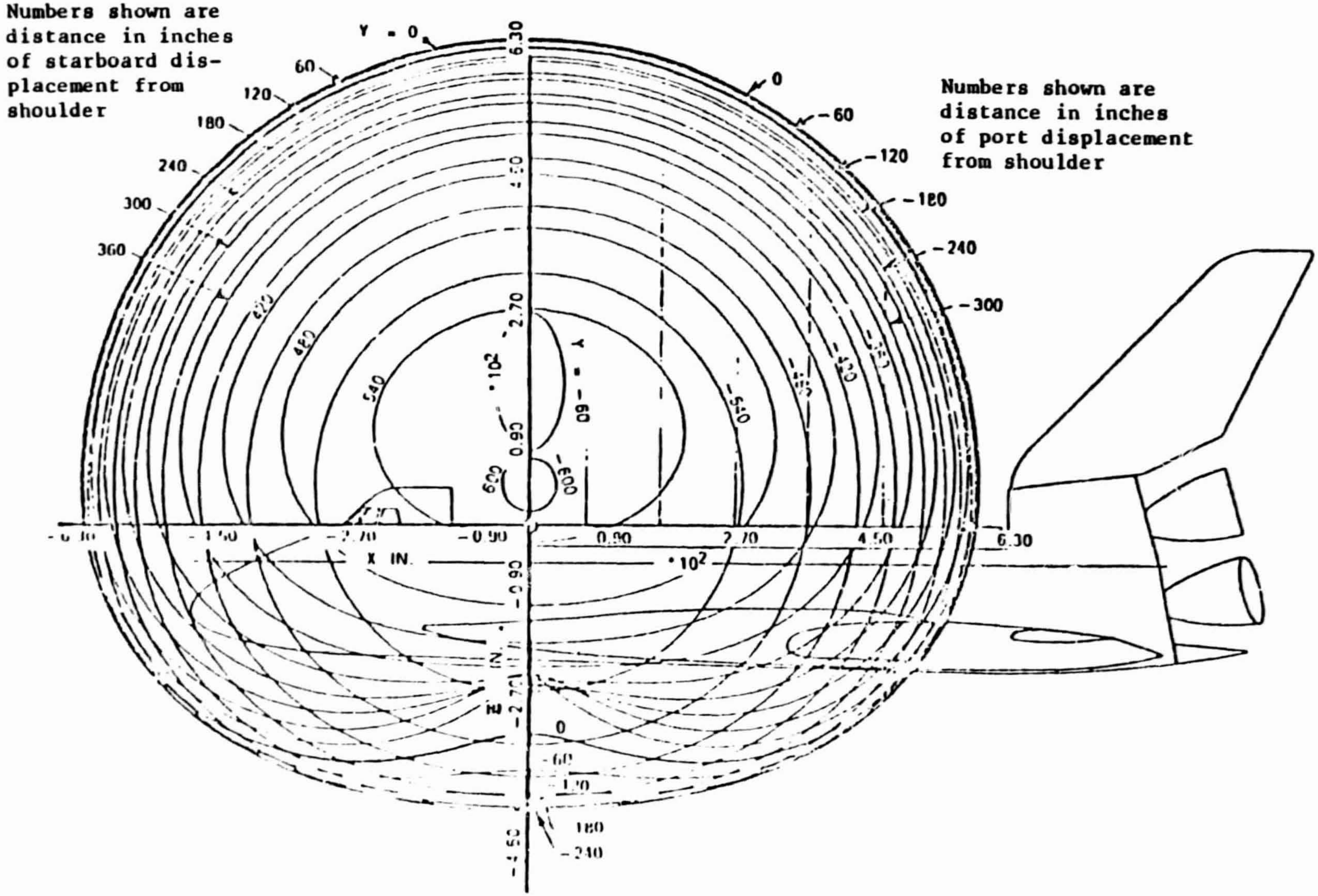


Figure 4.5-11. Side View of Orbiter and Y-Contours (Looking starboard)

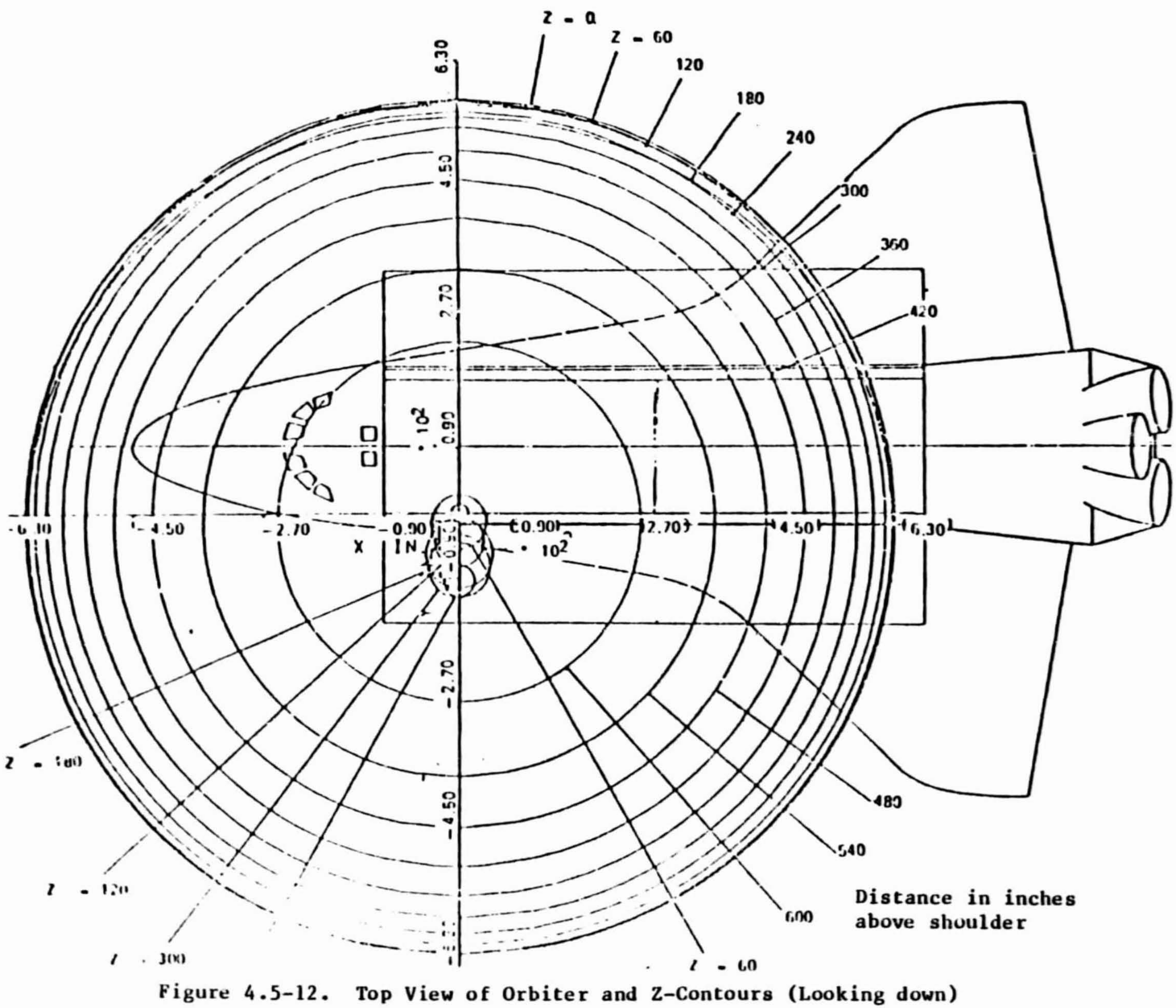


Figure 4.5-12. Top View of Orbiter and Z-Contours (Looking down)

respect to the axis that is orthogonal to the plane of the paper. The contours shown do not account for orbiter structure/RMS interference. Total reach accessibility within the contour envelopes may, therefore, not be available. The actual reach capability for a flight or payload task will be analyzed prior to flight. Figure 4.5-13 indicates the reach restrictions caused by the orbiter cabin and payload bay longerons, doors, etc. This figure also shows normal visible areas thru the upper window of the maximum reach envelope.

Among the significant constraints on RMS reach are the limitations imposed by specific orientations of the end effector. Figure 4.5-14 depicts the reachable area of the cargo bay volume when the end effector is vertically oriented. Table 4.5-2 lists key RMS performance characteristics.

A key concern of payload handling operations independent of large space structures is the capability of the RMS to withstand VRCS and PRCS thrusting loads. Such cases are unlikely to concern designers of large space structures, since most module handling will be done while the orbiter is attached to the large structure, and RCS firings will be inhibited. However, Table 4.5-2 and Table 4.5-3 include such constraints. Also, further summarization is offered in Table 4.5-4 for various arm loading conditions. In all loaded arm cases, the arm must not be in singularity or reach limit zones.

In general, the VRCS can be used to orient the orbiter/RMS/payload combination under the normal circumstances; the orbiter can stationkeep with the unloaded arm in a "poise to capture" position (which requires the PRCS). The PRCS cannot be used to maneuver the orbiter/RMS/payload if the payload exceeds 32,000 lbs. The orbiter/RMS/payload can be maneuvered with the low thrust modes of the PRCS, if the payload is very light (below 1,000 lbs). Unique orbiter/RMS/payload maneuvering requirements must be evaluated according to specific mission characteristics.

4.5.3 RMS Control System

The following descriptive material is abstracted from JSC-07700, Vol. XIV (NASA 1980 (a)). Control of the RMS is effected by the operator from the RMS D&C panel in the aft flight deck. The operator has access to four prime control modes, in which he has varying degrees of software support, and a back-up mode which completely by-passes the control and display software. The control modes that can be selected by the operator are as follows:

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

(2) Automatic Mode - The manipulator arm movement can be controlled automatically along a prespecified trajectory. This trajectory is defined by a series of predefined positions and orientations stored in the GPC. The operator can select up to four preprogrammed automatic trajectories from the D&C panel mode select rotary switch. Up to 200 points (total)

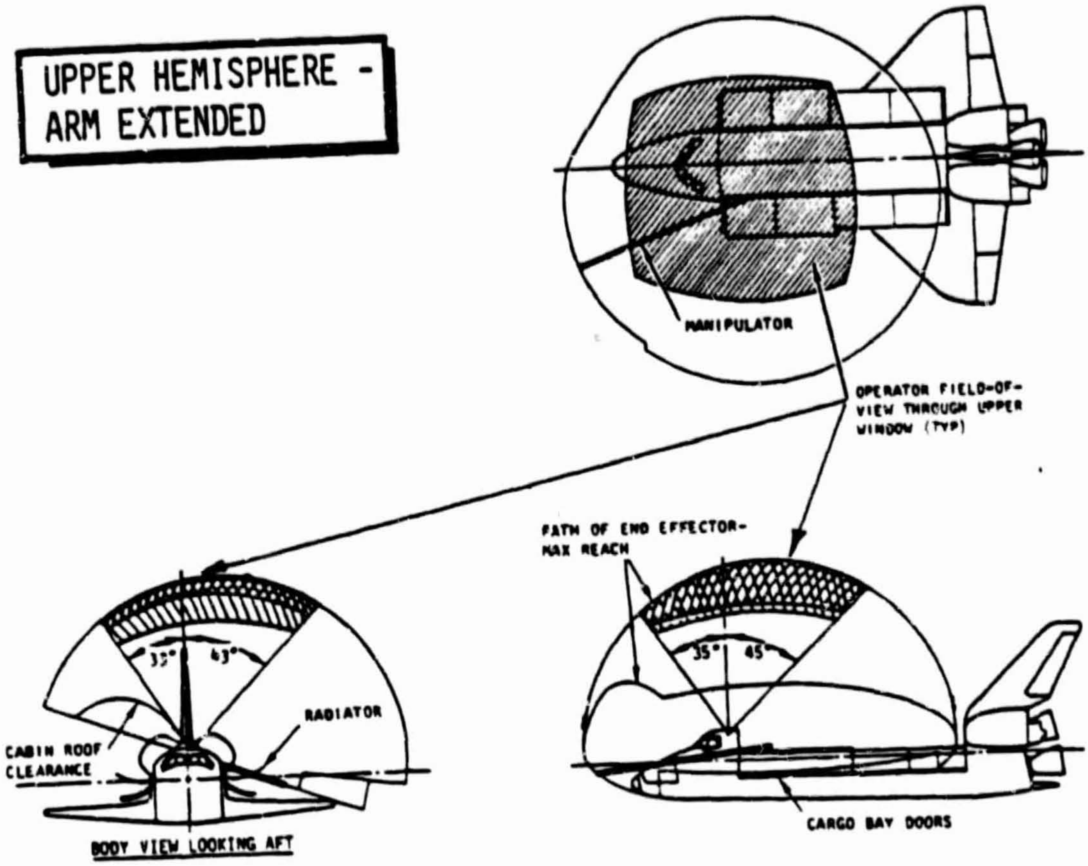


Figure 4.5-13. Maximum Reach Envelope for RMS Arm (Excluding Elbow Bend Capability)

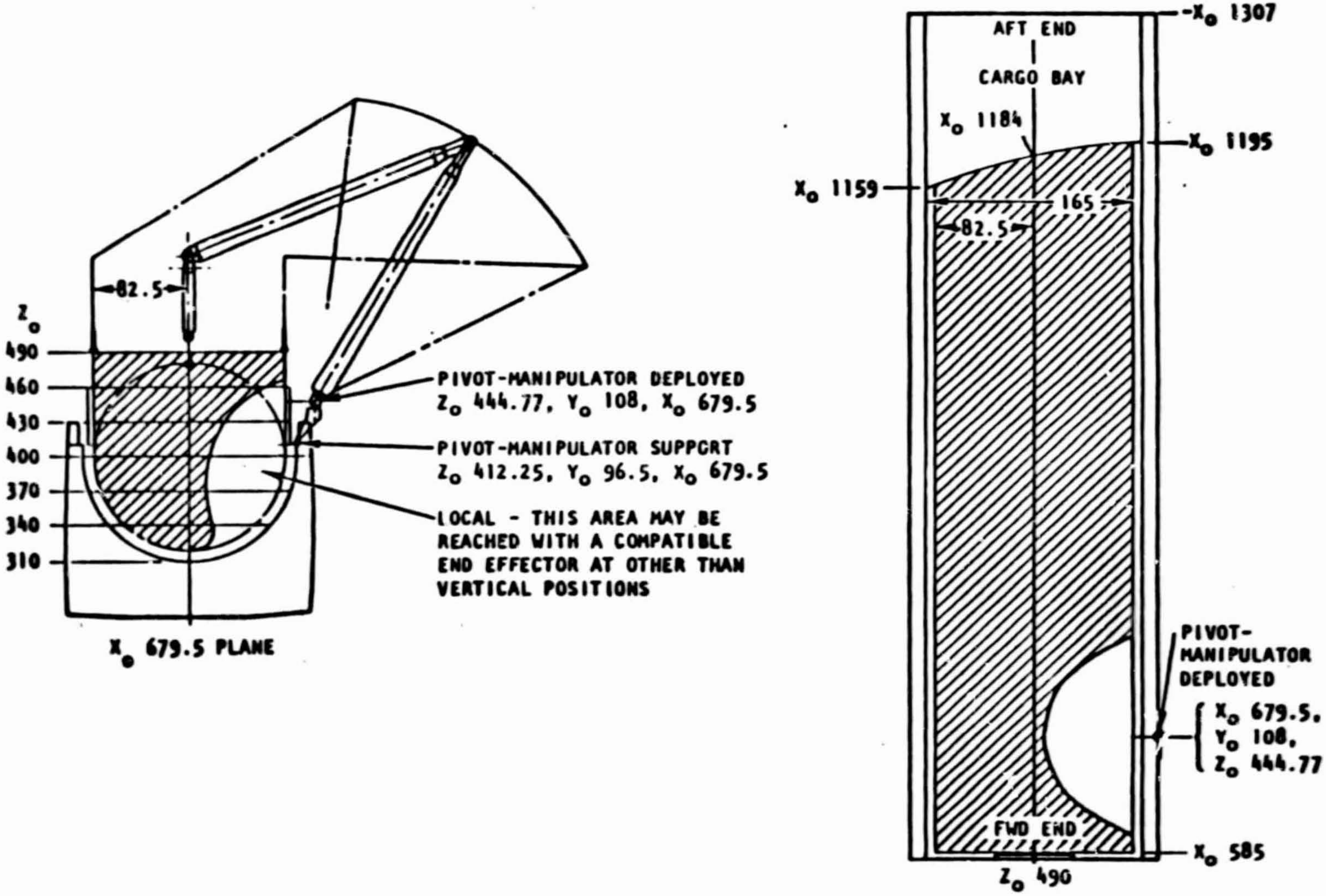


Figure 4.5-14. RMS Capability to Reach Surfaces of Payload Bay Envelope with End Effector Vertically Downward

Table 4.5-2. Checklist: Key RMS Performance Characteristics Constraining Design for Large Space Structures

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) The velocity of the RMS end effector is limited to 2 ft/sec for the unloaded arm, to 0.2 ft/sec when a homogeneous, maximum envelope (32,000 lbs) payload is attached to it and to other values based on the mass properties of the payload when other payloads are attached.	NASA 1980 (a)
( ) At a time not less than 10 minutes after deactivation of the orbiter RCS, the RMS will be capable of releasing maximum envelope payloads of 65,000 lbs with the following accuracy: • Attitude within 5 degrees of specified orientation relative to the orbiter structural coordinate system. Additional attitude error contributions by the orbiter may include the 2 degree uncertainty between the orbiter structural coordinate system and its NAV base, the vehicle drift resulting from the vernier attitude control capability, plus gravity gradient torque effects. The latter two will depend on mission timelines and payload thermal constraints (shadowing attitude requirements) and so must be determined on an individual flight design basis. • Angular rate of the payload relative to the orbiter less than or equal to 0.015 degrees per second. Additional angular rate errors contributed by the orbiter may include the vernier attitude control rate uncertainty of 0.01 degree per second plus gravity gradient torque effects. The gravity gradient effects could be significant, but are dependent on the required orbiter attitude pointing. The total system angular rate should be defined on a flight design basis. • Linear motion of less than 0.10 feet per second, relative to orbiter shoulder attach point.	NASA 1980 (a)

Table 4.5-2. Checklist: Key RMS Performance Characteristics Constraining Design for Large Space Structures (Continued)

( ) GUIDELINES/•QUALIFICATIONS	REFERENCES IMPLICATIONS
( ) In the automatic mode, the RMS is capable of accurately positioning the end effector (loaded or unloaded) within ± 2.0 inches (50.8mm) and $\pm 1^{\circ}$ relative to the shoulder attach point. . Longeron mechanical inaccuracies and thermal distortions at the shoulder attachment could add as much as an additional 7-inch error in radius of position plus 1 degree in end effector orientation. . The end-to-end position inaccuracies are dependent on actual RMS geometry and may require analysis for the planned application. . In the manual augmented mode the end effector positioning accuracy is primarily a function of operator visibility.	NASA 1980 (a)
( ) The manipulator arm will transmit, when fully extended and attached to a payload, loads not exceeding the following: . A combined 12 lb. shear force and 160 ft. lb. bending moment at the end effector. . A 230 ft. lb. torque about the end effector axis. An example of the forces and torques that be applied by the end effector for various arm configurations, are shown in Table 4.5-3.	NASA 1980 (a)
( ) The manipulator arm is capable of operating (when exposed to direct and/or reflected sunlight) for not less than: . 30 minutes when operating in the cargo bay . 120 minutes when operating outside the cargo bay.	NASA 1980 (a)

Table 4.5-2. Checklist: Key RMS Performance Characteristics Constraining Design for Large Space Structures (Continued)

( ) GUIDELINES/•QUALIFICATIONS	REFERENCES IMPLICATIONS
( ) The unloaded arm can operate with no restrictions on VRCS (Vernier RCS) except that the arm must not be in singularity or reach limit zones. The only restrictions on PRCS (Primary RCS) firings for the unloaded RMS are that the arm cannot be simultaneously maneuvered with thruster firings, it must be outside the singularity and reach limit warning zones, and no high-thrust level Z axis or pitch thruster firings of duration longer than TBD seconds are allowed.	NASA 1980 (a)
( ) The arm can handle a payload of 32,000 lbs or less and withstand PRCS firings if the loaded arm is not being maneuvered; only single minimum impulse PRCS are allowed, however, with intervals between firings sufficient to allow RMS motion to settle.	NASA 1980 (a)
( ) For payloads greater than 32,000 lbs, no PRCS firings are allowed. No restrictions are required for VRCS firings (for orbiter attitude changes or stabilization) other than no VRCS firings are allowed if the loaded arm is in a singularity or reach limit zone.	NASA 1980 (a)

Table 4.5-3. Force/Torque Capability At End Effector

	Torque Range (ft/lbs) Min.	Torque Range (ft/lbs) Max.	Force (lbs) Min.	Force (lbs) Max.	Condition
Shoulder Yaw	772	1158	15.44	23.2	Straight Arm
Shoulder Pitch	772	1158	15.44	23.2	Straight Arm
Elbow Pitch	528	792	18.41	27.3	Bent Arm Overall Length <42 ft.
Wrist Pitch	231	347	37.97	57.0	Bent Arm Overall Length <20 ft.
Wrist Yaw	231	347	54.35	81.6	Bent Arm Overall Length <14 ft.
Wrist Roll	231	347	-		

NOTE: All values are quotes for the arm under steady state rigid body static condition. E.G. In Cargo Bay - And Single Joint Drive

Reference: NASA 1980(a), Vol. XIV, Table 8.1

Table 4.5-4. Summary of RMS Capability to Withstand VRCS and PRCS Thrusting

Source: NASA 1980(a)

SYSTEM	RMS LOAD CONDITION	CAPABILITY ✓ = Ok
VRCS	Unloaded	✓
	32,000 Lb. $\geq$ Payload	✓
	32,000 Lb. $<$ Payload $\leq$ 65,000 lb.	✓
PRCS	Unloaded	✓, if arm is not being maneuvered and no pitch of Z-thrusting longer than TBD sec.
	32,000 Lb. $\geq$ Payload	✓, if single min-pulse firing and arm not being maneuvered.
	32,000 Lb. $<$ Payload $\leq$ 65,000 lb.	No

Reference: NASA 1980(a), Vol. XIV, p. 8-5 can be stored for auto trajectories, each point defined by orbiter reference position X, Y, Z, plus yaw, pitch, roll orientation.

A second type of automatic trajectory can be initiated by the RMS operator through the D&C select switch and the GPC keyboard. This is the operator commanded auto sequence mode and is initiated by input of the required position and orientation of the end effector or payload. A straight line trajectory is then performed from the current position and orientation to the desired position and orientation.

The above automatic sequence capabilities are available to be negotiated by payloads on an individual basis.

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

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

(5) Back-Up Drive Mode - Back-up control of individual joints by operator commands through unique hardwired channels. No position data is displayed.

The combined operations of the six joints of the manipulator arm, through one of the appropriate control modes above, enables the operator to move the end effector in six degrees of freedom (three degrees of motion in translation, three in rotation). The coordinate systems relating these travel directions are shown in Figure 4.5-15. In the manual modes, the operator commands movement of the end effector using the THC and RHC in the selected coordinate system. Operations in the automatic control mode will utilize the orbiter reference coordinate system.

4.5.4 RMS Software

The RMS software, under which most RMS operations are performed, resides in the orbiter General Purpose Computer (GPC). (NASA 1980 (a)). The RMS software performs the following functions:

(1) Translates operator commands into RMS arm operations and motions.

(2) Monitors RMS status.

(3) Performs display computational tasks for information to the RMS operator, including caution and warning.

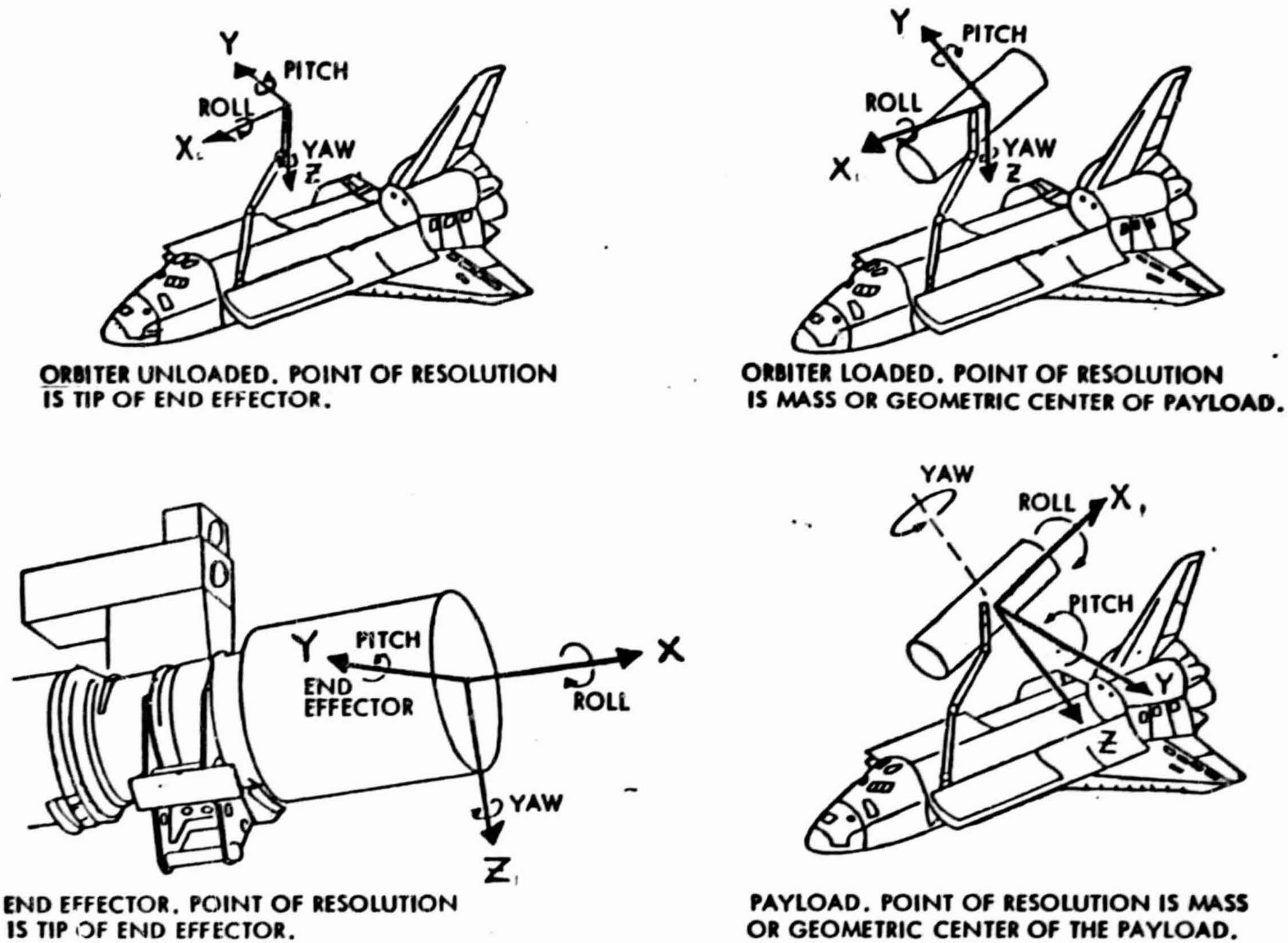


Figure 4.5-15. Control Coordinate Reference Systems

Control algorithms contained in the RMS software convert operator commands (normally input by the hand controllers at the D&C panel) into output rates resolved for each joint of the arm. The rate demands to the joint servos are output within limits defined according to arm and individual joint loading conditions present at the time of computation.

4.5.4.1 Initialization Data

Parameters with which the RMS software is initialized may vary from flight to flight. These parameters may be RMS hardware dependent (generally called I-loads) or flight and payload dependent (generally called Level C data). The hardware dependent parameters include:

(a) End effector length
(b) Hand controller biases
(c) Tachometer biases, etc.

The flight and payload dependent parameters include those listed in Table 4.5-5.

Table 4.5-5. Flight and Payload Dependent Parameters for RMS

	Coarse	Vernier
Maximum Payload Translation Rate	0.2 fps	0.01 fps
Maximum Payload Rotation Rate	0.0083 rad/sec	0.00415 rad/sec
Joint Angle Course Rate Limits	---	---
Joint Angle Vernier Rate Limits	---	---
Payload to End Effector Transformation Matrix	(TBD)	(TBD)
Automatic Trajectory Parameters	(TBD)	(TBD)

The RMS initialization parameters (I-Load and Level C) are identified in SSD77-SH-0002A, Level C Functional Subsystem Software Requirements (FSSR) Document. Some of these quantities can be changed on-orbit through GPC keyboard input. To generate the payload dependent RMS software parameters, payload characteristics should be provided approximately one year prior to flight. These characteristics, and their allowable variations, are as follows:

(1) Payload Mass to ±10%.

(2) Payload Center of Mass to ±6 inches, defined in Payload Coordinate System.

(3) Moments of Inertia about Payload Principle Axes to ±10%.

(4) Payload Cross Products of Inertia, to ±10%.

(5) Grapple fixture location(s) and installation orientation, in payload coordinates. If the payload has no preference, NASA will select grapple fixture orientation. The grapple fixture will normally be

located within 5% (of payload length) of the orbiter Y-Z plane of stowed payload center of mass.

4.5.4.2 Downlist Data

A number of RMS parameters are on the GPC downlist. These measurements are signals which are used directly or indirectly to provide data to the flight computers, the RMS operator, the ground mission controllers or flight planners regarding the systems performance, component status, or condition of hardware and/or software elements. Each measurement is given a unique identification number to identify its signal source or location, sample rate, range, and units. The available RMS downlist parameters are listed in SD77-SH-0002A, Level C Functional Subsystem Software Requirements (FSSR) Document.

4.6 PAYLOAD INSTALLATION AND DEPLOYMENT AID

The descriptive material in this section is all abstracted from an undated article by T. O. Ross of the Johnson Space Center, entitled "Payload Installation and Deployment Aid for Space Shuttle Orbiter Spacecraft Remote Manipulator System." The material included herein is selected to emphasize planned capabilities, rather than describe details of how the mechanisms function.

Early developmental testing of the Remote Manipulator System (RMS) revealed that on-orbit handling of various payloads on the Space Shuttle Orbiter Spacecraft may prove to be beyond the capability of the system without the assistance of a handling aid.

An aid concept known as the PIDA (Payload Installation and Deployment Aid) is presented as a way to assist the RMS by relaxing the accuracy required during payload handling in the payload bay. The aid concept was designed and developed to move payloads through a prescribed path between the confined quarters of the payload bay and a position outside the critical maneuvering area of the orbiter.

An androgynous docking mechanism is used at the payload/PIDA interfaces for normal docking functions that also serves as a structural connection between the payload and the orbiter that is capable of being loosened to prevent transfer of loads between a stowed payload and the PIDA structure. A gearmotor driven drum/cable system is used in the docking mechanism in a unique manner to center the attenuator assembly, align the ring and guide assembly (docking interface) in roll, pitch, and yaw, and rigidize the mechanism at a nominal position. A description of the design requirements and the modes of operation of the various functions of the deployment and the docking mechanisms are covered in the following paragraphs.

The PIDA is presently being fabricated as flight-like hardware for engineering development test and evaluation in the JSC Manipulator Development Facility. This effort is intended to develop the PIDA concept to a state of readiness for a minimum lead time for flight hardware and at the same time developing the electromechanical actuator and the docking mechanism for potential use in other applications.

4.6.1 Requirements

The basic requirements that were imposed on the Payload Installation and Deployment Aid concept are:

- o Provide line of sight docking points outside of critical maneuvering area.
- o Utilize single point capture steps as opposed to multi-points requiring simultaneous capture.
- o Use mechanism to move the payload from deployed to stowed position without exceeding a 75mm (3.0 inch) payload clearance envelope.
- o Be controllable by crew from AFD.
- o Accommodate payloads ranging up to 4.57 meters (15 ft) dia. by 18.3 meters (60 ft

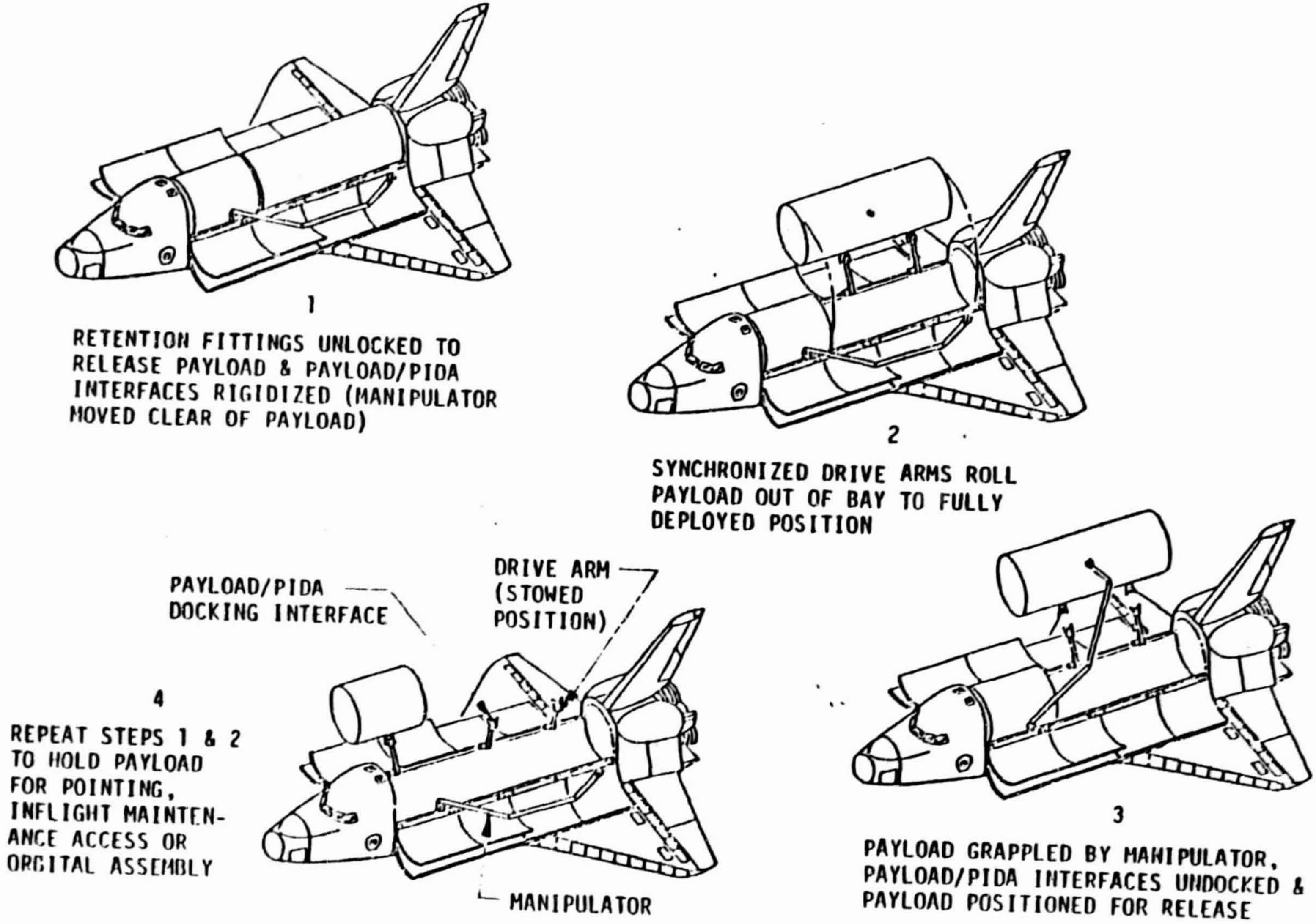


Figure 4.6-1. PIDA Deployment Sequence

- Be controllable by crew from AFD.

- Accommodate payloads ranging up to 4.57 meters (15 ft) dia by 18.3 meters (60 ft) long and 29.484 kg (65,000 lbs) weight.

- Accommodate payload contact velocities up to 30mm/sec (.10 ft/sec) and .011 rad/sec with a lateral mismatch of 150mm (6.0 inches) maximum and angular mismatches of $\pm 15^{\circ}$ in pitch and yaw and $\pm 10^{\circ}$ roll.

- Design the PIDA to stow in a confined space under the closed cargo bay doors outside the payload clearance envelope.

- Utilize existing longeron bridge fitting attachments for structural connection.

4.6.2 PIDA Assembly Description

The PIDA assembly shown in Figure 4.6-2 is made up of a deploy/stow mechanism, an interface mechanism, an electromechanical rotary actuator with its respective electronic controls, and a base, with a jettison interface, that connects the assembly to the orbiter longeron bridge fitting on installation.

The operation of the assembly between the stowed and deployed positions, shown in Figure 4.6-3, is done remotely from the RMS operator's station. The operator can select the degree of deployment desired and monitor its position from a display of the optical encoder data that is used to control the drive motors and keep them synchronized to within one-tenth of a degree. Preprogramming for a specific payload provides the control of the master drive to accelerate and then decelerate the payload to stop at the desired point without overrun or excessive structural loads on the PIDA structure, the payload, or the orbiter longeron attach points. The accuracy provided by the control system offers precise pointing of payloads and opens the possibility of limited tracking using the PIDA drive system with added tracking sensors.

4.6.3 Deploy/Stow Mechanism

The basic purpose of the deploy/stow mechanism is to control the movement of the payload positively and accurately between the stowed and deployed positions and to locate the payload in a deploy position that is away from the orbiter, outside of the critical maneuvering area but with the docking interfaces in the line of sight of the RMS operator. Design guidelines required that the movement between the stowed and deployed positions be provided without exceeding a 75mm (3.0 inches) payload clearance envelope and that the deployed position be located for a minimum clearance of 50cm (19.5 inches) between the payload and the orbiter (or any orbiter appendages). The configuration must permit the mechanism to be stowed in a confined space under the closed cargo bay doors and radiators, yet outside the cargo bay clearance envelope.

The four bar mechanism shown in Figure 4.6-4 has a tabular drive arm member that is connected at one end to the base and the other end to the crank arm on the interface mechanism. A drag link that serves as a tension/compression tie between the base and the end of the crank arm provides the linkage

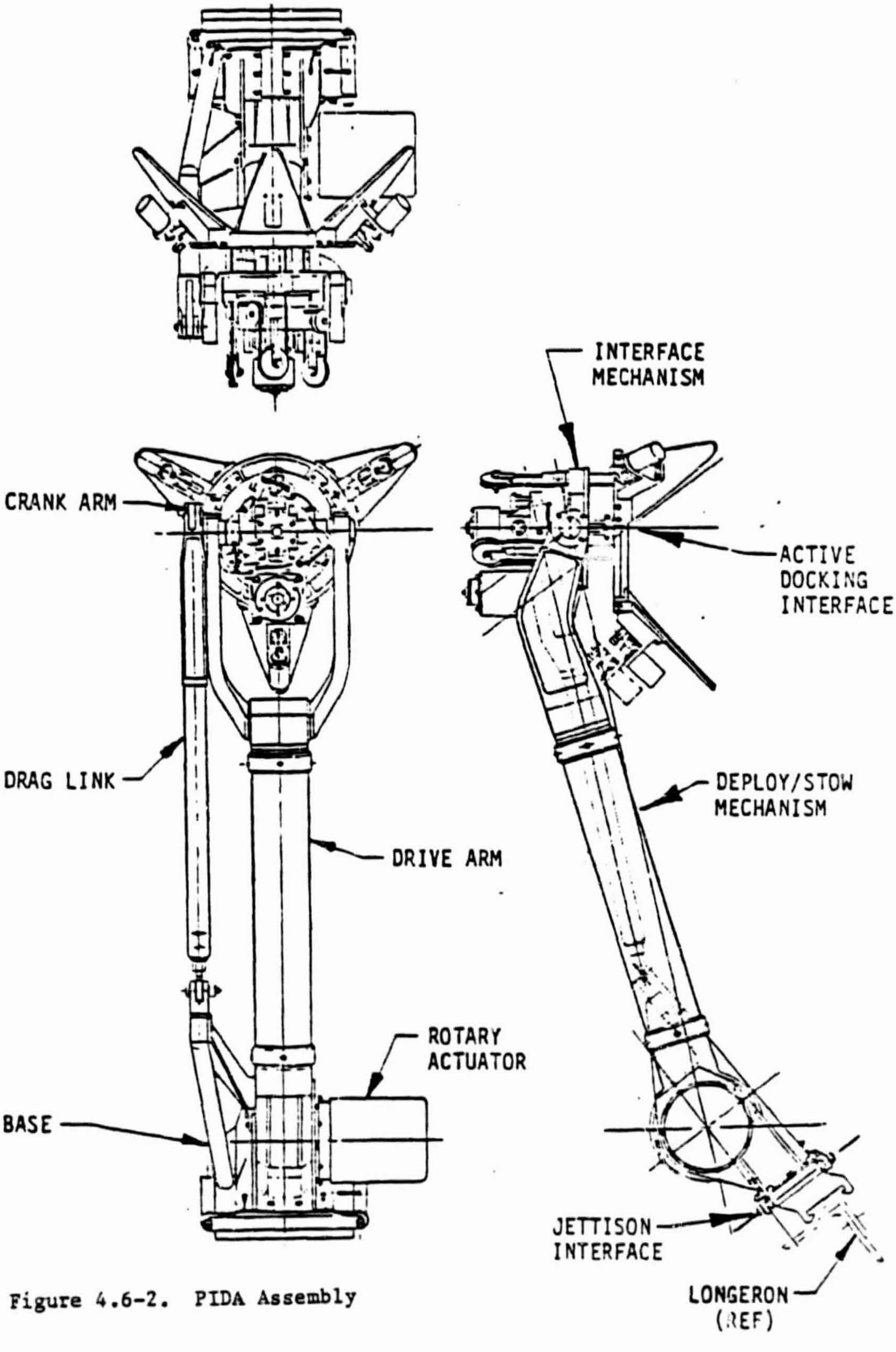


Figure 4.6-2. PIDA Assembly

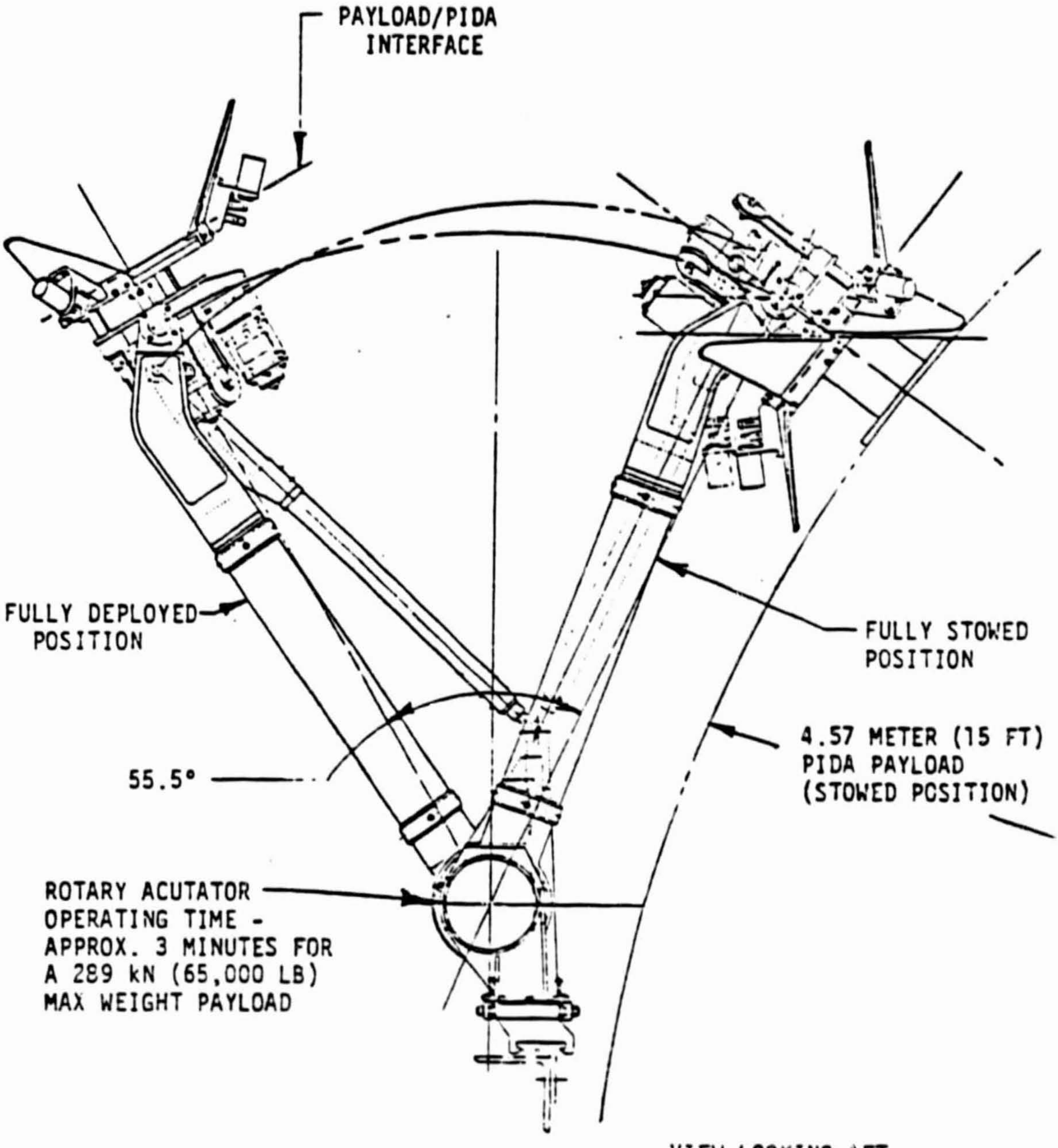


Figure 4.6-3. PIDA Assembly Operating Positions

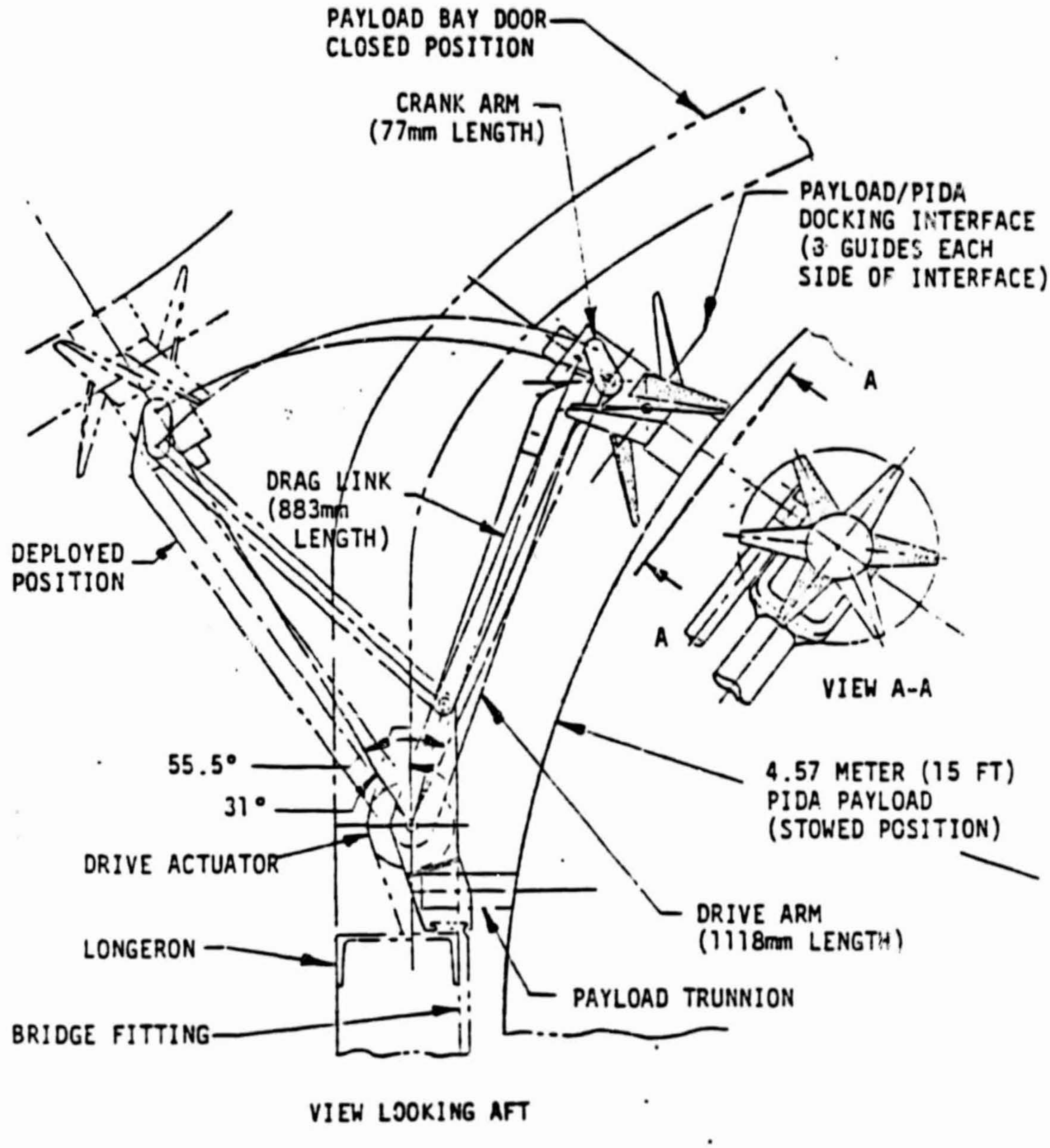


Figure 4.6-4. Deploy/Stow Mechanism

to turn the crank arm as the main arm is driven from one position to another by an electromechanical rotary actuator. As the main arm rotates through an angle of 55°30', the crank arm rotates the interface mechanism 102°37' for an angular displacement ratio of 1.83:1. The total rotation of the payload axis relative to the orbiter axis is the sum of these two angles.

Figure 4.6-5 shows how the initial part of the C.G. path approximates an upward ($+Z_0$ axis) linear withdrawl by a low amplitude sinusoidal movement. The movement of the longeron trunnion next to the mechanism, shown in detail. "Z", provides an upward and outboard movement that is compatible with the deployable retention fittings.

4.6.4 Interface Mechanism

The payload/PIDA interface mechanism, shown in Figures 4.6-6 and 4.6-7, includes a docking mechanism for the RMS operator to connect or disconnect the payload from the deploy/stow mechanism and a structural connection to positively hold the payload during deploy or stow actuation to aid accurate positioning of the payload in the payload bay. After the payload has been placed in the fully stowed position, the structural connection through the PIDA is loosened to provide compliance in order to force the retention fittings to be the primary load paths. The mechanism provides the basic functional modes of docking, such as, compliance, capture, energy absorption, alignment and rigidization in addition to the stowed position compliance.

Docking Compliance

Docking compliance is provided to allow the two mating sides of the interface to align in order that the capture latches can operate. The mechanism on the active side of the docking interface moves as required for alignment except for lateral compliance.

The lateral compliance and attenuation is not an active part of the mechanism, but is accommodated by the dynamics of the orbiter and payload interreactions.

The axial compliance and attenuation, both compression and extension, is furnished by a hydraulic-type attenuator that has internal spring action to return it to a nominal position that is preloaded in both directions.

The roll alignment movement is permitted by the outer part of the ring and guide assembly being free to rotate relative to the center part of the assembly. The two parts are connected through two ball bearings and are spring loaded to a nominal position by the spring preload.

The pitch and yaw compliance is provided by a universal joint located between the center of the interface ring and the attenuator assembly.

Provisions for docking capture include the guides on the interface ring which are sized for 152mm (6.0 inches) lateral misalignment (which includes the mismatch due to $\pm 15^\circ$ pitch or yaw) in combination with a roll misalignment of $\pm 10^\circ$. The guide configuration provides lateral forces to act on the orbiter

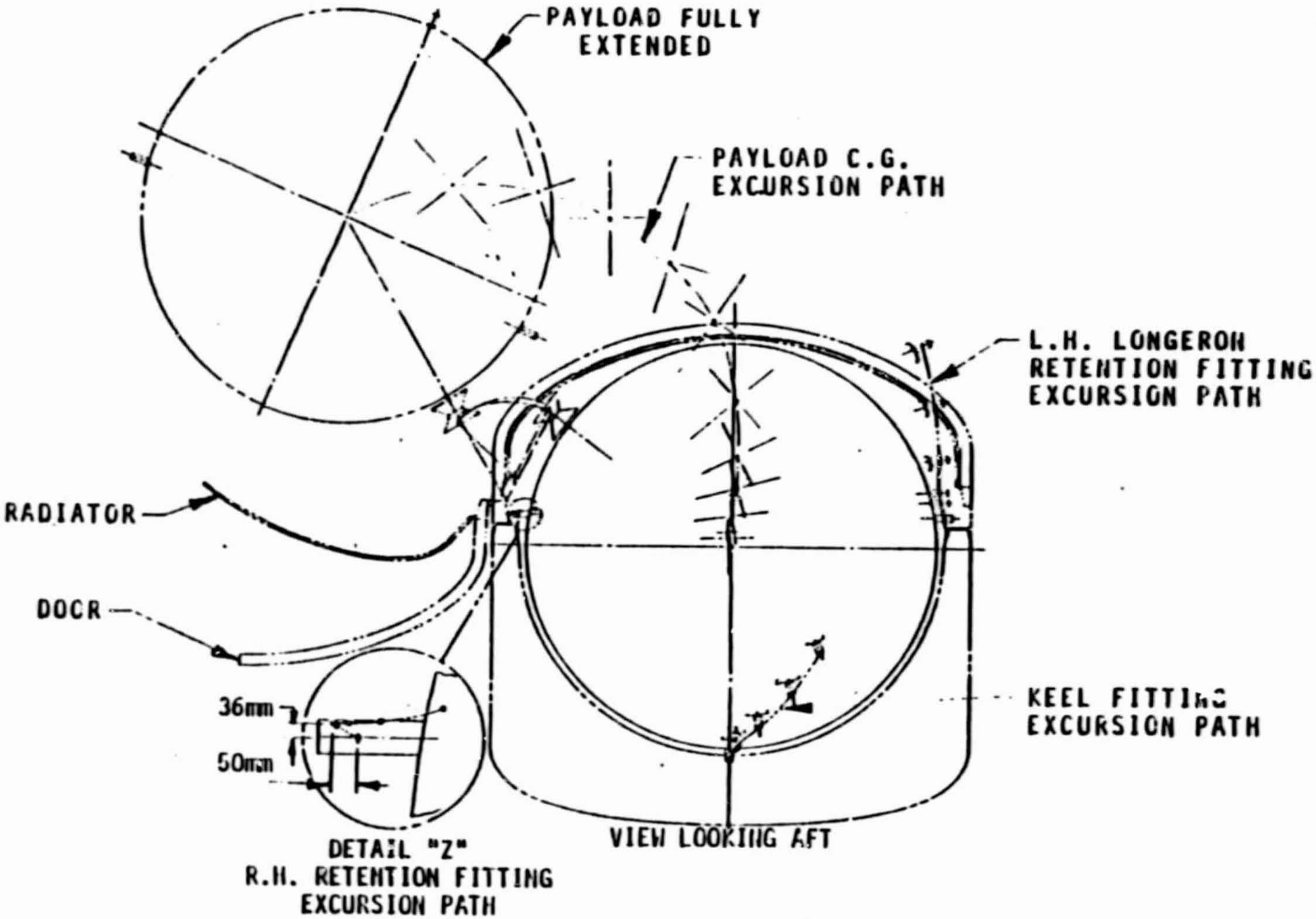


Figure 4.6-5. Single Stage Deploy/Stow Actuation

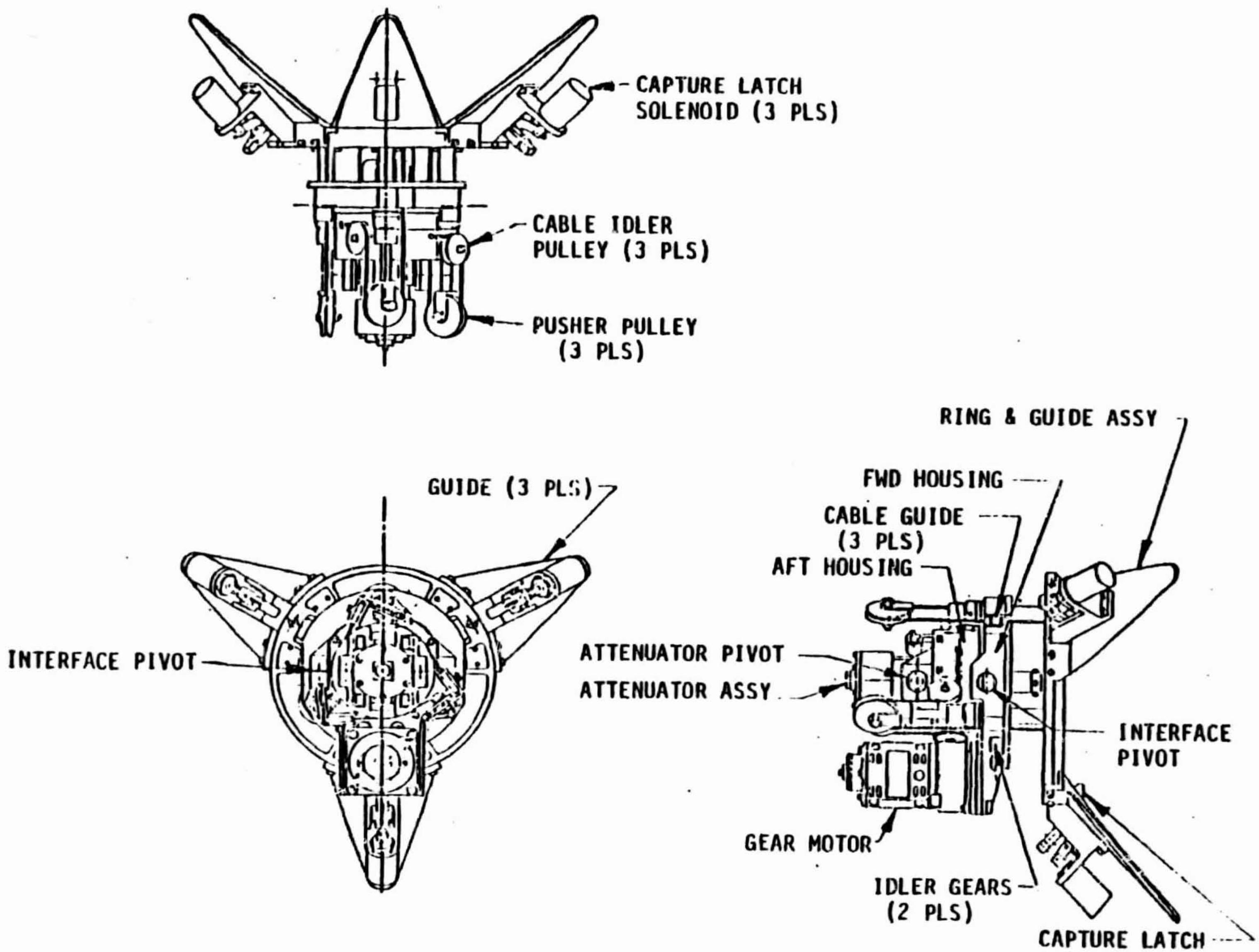


Figure 4.6-6. Interface Mechanism

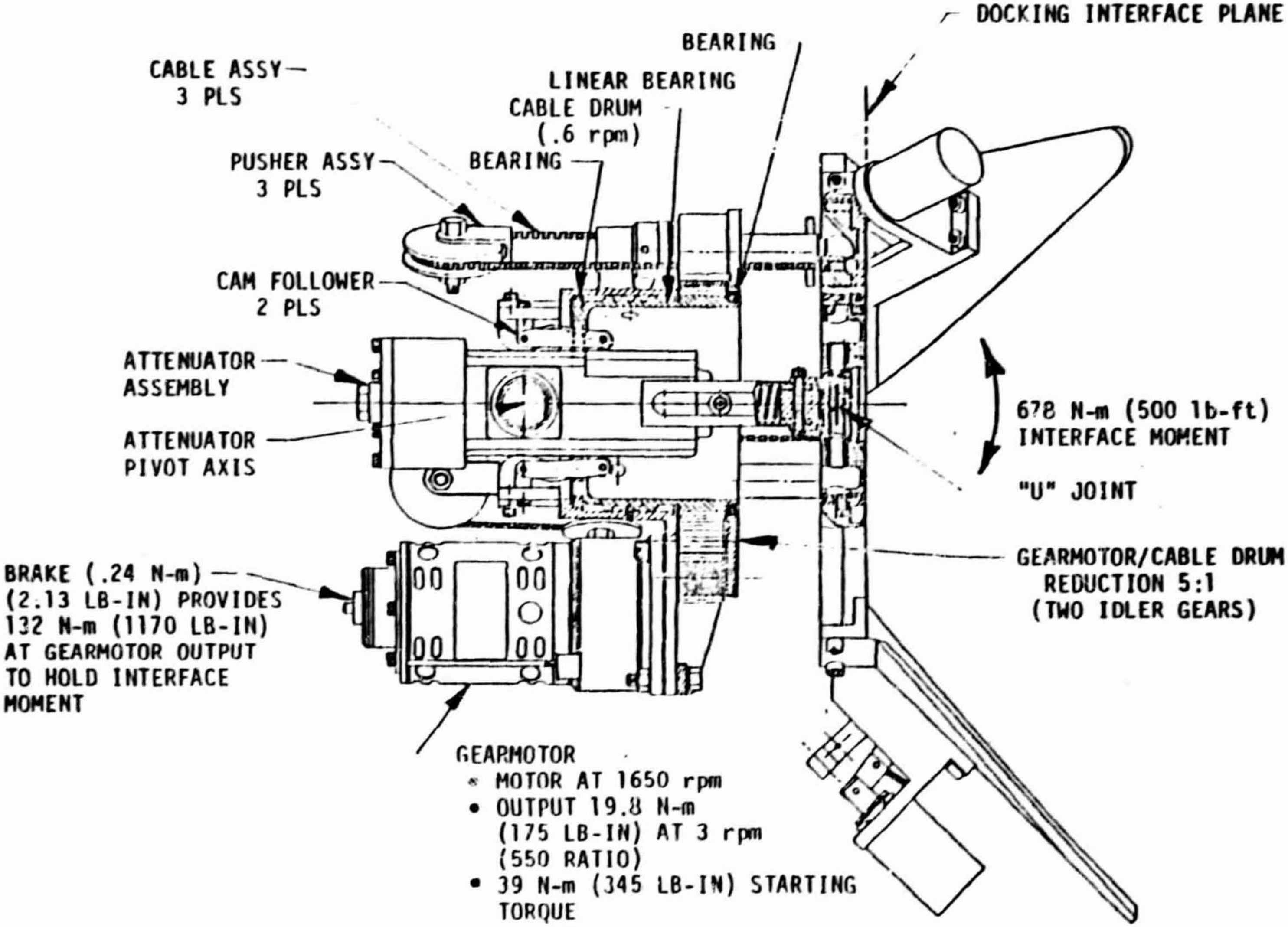


Figure 4.6-7. Interface Mechanism

and payload for dynamic lateral compliance to permit the capture latches to engage. The capture latches are designed such that, if insufficient latches are engaged to react capture loads, none will remain engaged. Any two latches are able to react the capture loads. If only one latch is engaged, the force vectors act in a direction upon the latch during a separation motion such that the toggle linkage of the latch will collapse to allow the two docking surfaces to separate freely. The capture latches serve a dual role inasmuch as they are also used as the structural latches to secure the payload to the orbiter after the docking phase is complete.

Energy absorption capability is also provided. A payload with kinetic energy relative to the orbiter, contacts the docking interface causing the attenuator assembly to be compressed. During this compression stroke, hydraulic fluid is metered from the head end to the rod end of the attenuator. Part of the kinetic energy is dissipated by the fluid metering and the remainder is stored in the attenuator spring as potential energy. At the end of the compression stroke, the spring forces the attenuator to extend toward the nominal position transferring the potential energy back into the payload as kinetic energy. During this extension stroke, the fluid is metered from the rod end to the head end of the attenuator, further dissipating energy. As the attenuator reaches its nominal position the attenuator spring reverses its force direction to once again store the undissipated energy as potential energy. The residual energy is dissipated by the subsequent extension and compression strokes with rapidly decaying amplitude so that ultimately all motion is arrested and the interface returned to the nominal position.

Alignment and rigidization follows capture. Roll, pitch and yaw alignment across the interface is provided by the ring and guide assembly on each side mating with the one on the other side of the interface. This allows a payload to be positioned accurately even in installations employing only one PIDA assembly.

The holding requirement of the mechanism is based on an interface moment of 678 N-m (500 lb-ft) as determined from dynamic analysis of the payload/orbiter system using math modeling.

The inside of the cable drum has two cam surfaces located symmetrically opposite each other to actuate two cam followers, one on each side of the attenuator, to force it to a centered position or free it to allow the attenuator to pivot during the stowed position compliance movement. In the upper half of Figure 4.6-7 the attenuator is held centered and the lower half of the view shows the cam surface away from the cam follower to allow the attenuator to pivot.

4.6.5 Stowed Position Compliance

The payload retention system requires that the payload be permitted to have a three axis movement to accommodate thermal deflections. This necessitated that the PIDA have the same freedom if it is not to act as a primary structural connection for a stowed payload. The x-x axis freedom is provided by floating one of the passive docking interfaces on the payload and it being spring loaded to a center or nominal contact position. The y-y axis and z-z axis movement is provided by retracting the three pusher rods to allow the attenuator to stroke and backing off the two cam followers to permit the attenuator to pivot in the y-z plane.

4.6.6 Electromechanical Rotary Actuator

The electromechanical rotary actuator designed and fabricated to drive the deploy/stow mechanism was sized to provide a maximum torque of 1356 N-m (1000 lb-ft) at a rate of one degree per second. This is accomplished through the use of a gear box with two high ratio planetary drives, a 24/1 input stage and a 32/1 output stage, resulting in an overall ratio of 768/1 for the actuator in conjunction with a 5.4 N-m (4.0 lb-ft) 28 volt direct current electric motor.

4.7 OTHER MAJOR EQUIPMENT OPTIONS

The devices described in this section are generally of a more specialized nature, designed originally for specific shuttle missions in the classes of sortie or deployment/retrieval of free-flyer satellites. However, each has some potential as a means to carry, deploy or handle materials or equipment for assisting large space construction. Although these devices may not be ideal, they may meet a need without requiring a great deal of new engineering development.

4.7.1 Flight Support System for Multi-Mission Modular Spacecraft

The Flight Support System (FSS) is reusable equipment that provides the structural, mechanical, thermal, and electrical interfaces between the Multimission Modular Spacecraft (MMS) and the Space Shuttle for launch, retrieval, and on-orbit servicing missions. The FSS baseline configuration consists of three structural cradles, mechanisms for spacecraft retention and positioning, and avionics. The avionics provide all necessary power, command, control, and data monitoring interfaces to electrically support all operational modes of the baseline MMS from the orbiter. All or parts of the FSS are potentially useful as payload handling aids.

The three major hardware elements are:

- Cradle A - With a remotely operated retention system that provides structural support during launch, reentry, and landing. See Figure 4.7-1.
- Cradle B - A stiffener that bolts to cradle A to increase load capability and that provides additional attach points for optional equipment. See Figure 4.7-2.
- Cradle A'- With the remotely operated Berthing/Positioning System that provides spacecraft positioning capability on-orbit for appendage extention or retraction, inspection, predeployment testing, deployment, servicing, or retrieval berthing. See Figure 4.7-3.

In the baseline configuration, these hardware elements are combined and installed in the orbiter as shown in Figure 4.7-4 and 4.7-5. The FSS can be placed in any of several possible locations in the orbiter cargo bay, facing either forward or aft. In addition, the spacecraft can be placed at any of three preselected positions in the FSS.

The FSS avionics provide for electrical operation of the Berthing and Positioning System mechanisms and electrical support services to the MMS while on the ground and on-orbit. These services, shown in Figure 4.7-6, include operating power, externally applied heater power and serial commands and telemetry relay to the user's Payload Operations Control Center (POCC) via orbiter avionics and the Mission Control Center. Control and monitor of hardline digital and analog signals (not serially encoded data) by the

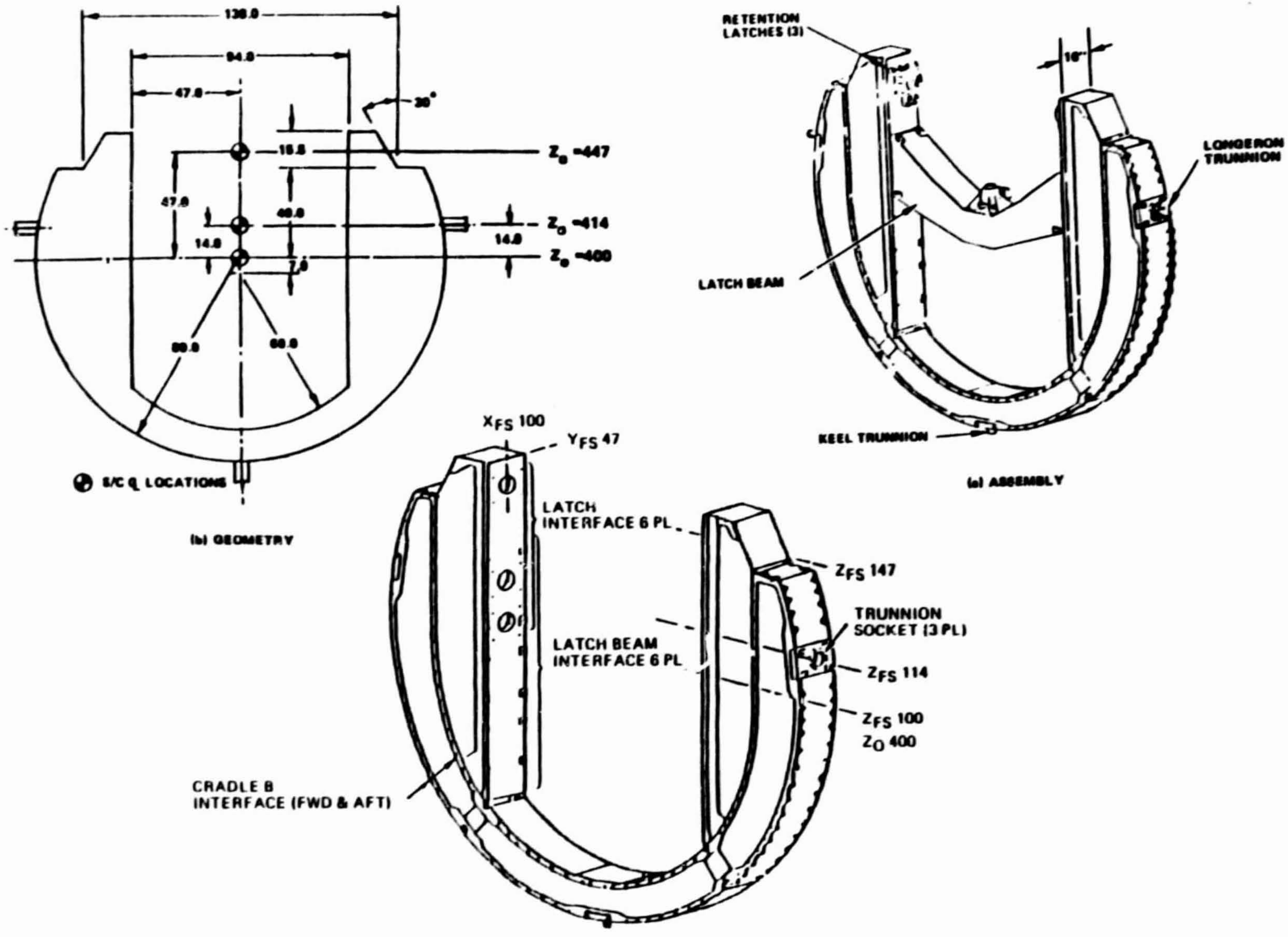


Figure 4.7-1. FSS Components: Cradle A

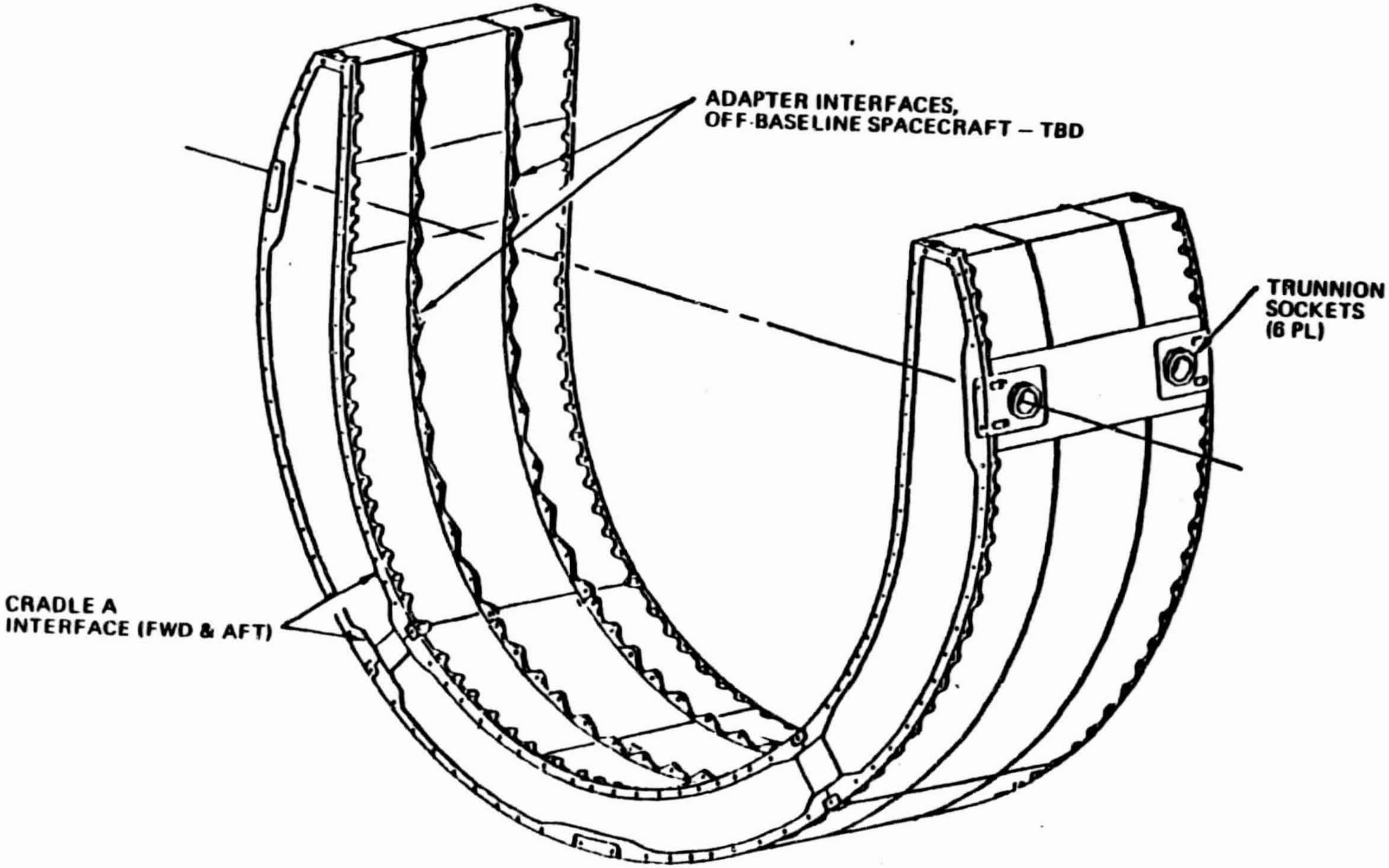


Figure 4.7-2. FSS Components: Cradle B

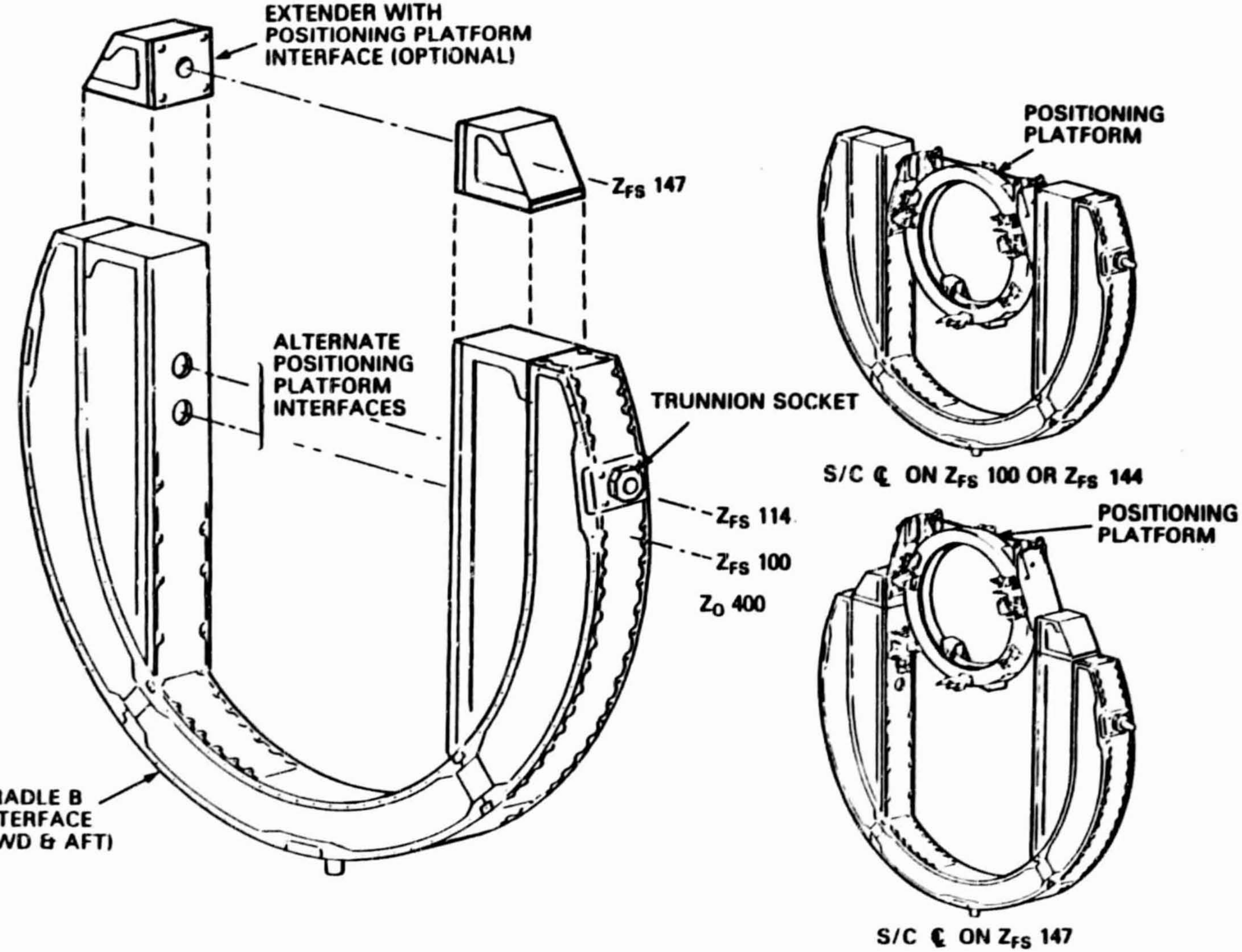


Figure 4.7-3. FSS Components: Cradle A'

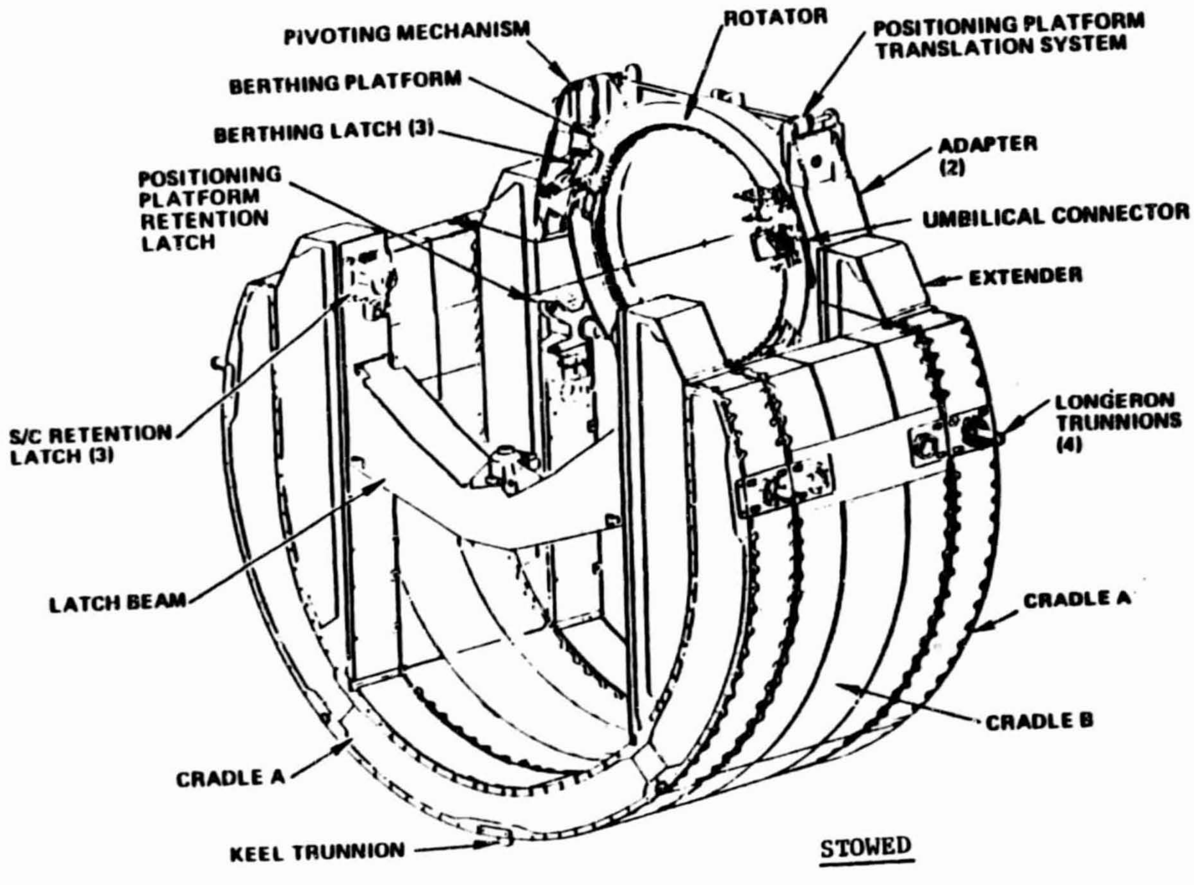


Figure 4.7-4. FSS Baseline Configuration

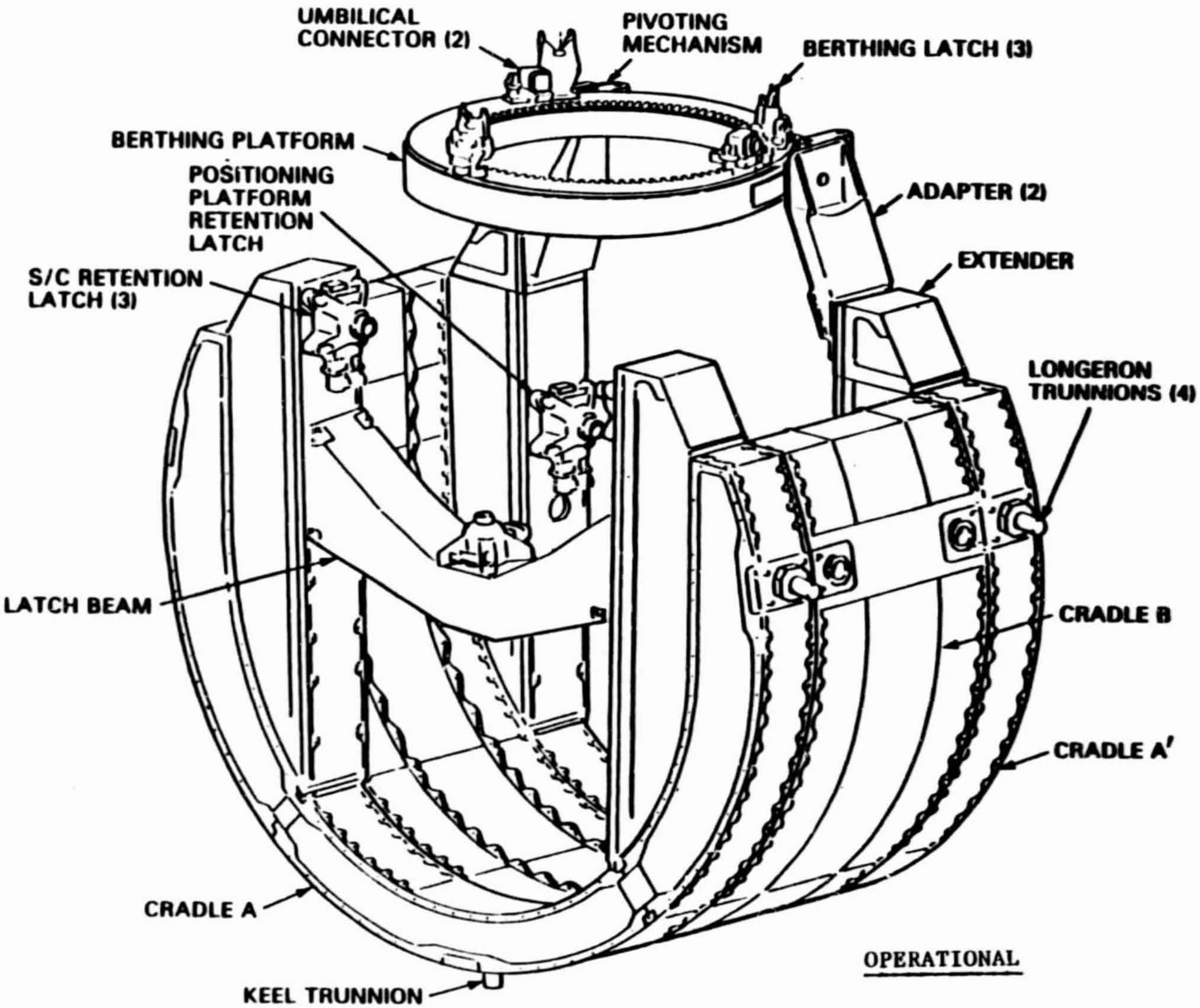


Figure 4.7-5. FSS Baseline Configuration

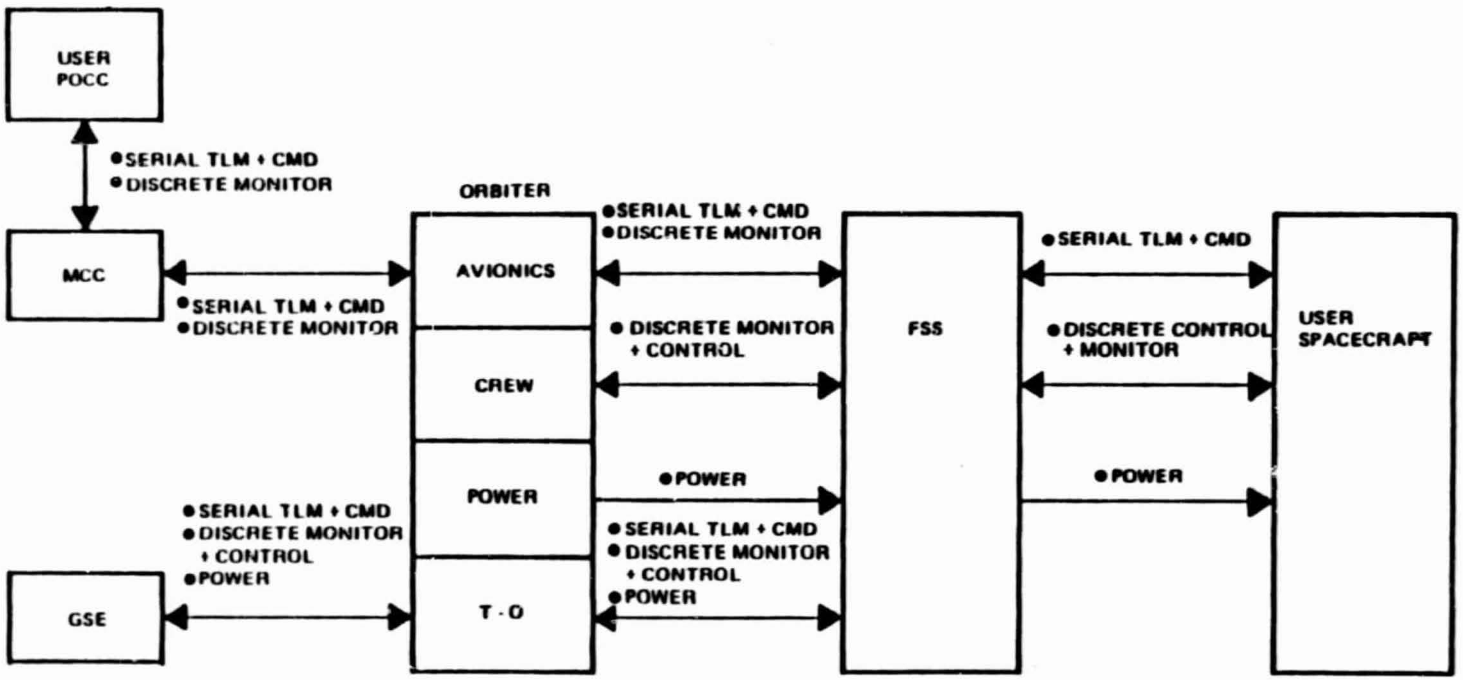


Figure 4.7-6. FSS Electrical and Avionics Services to User

orbiter crew are accommodated at the Aft Flight Deck (AFD) using standard equipment. These signals are also relayed to the user's POCC. These services on the ground are provided by the Ground Support Equipment (GSE) via the orbiter's T-O umbilical (excluding the mechanism drives). When the FSS/spacecraft is not installed in the orbiter, the electrical GSE interface is accomplished by directly connecting to the Orbiter/FSS Interface Unit (OFIU).

The FSS GSE is incorporated in the MMS Ground Support System (GSS), which is described in the MMS GSS Specification S-700-100. This equipment has the capability to electrically check out and operate the FSS. Mechanical GSE is provided for transporting and lifting the FSS either with or without a spacecraft payload.

4.7.1.1 Alternate Configurations

There are several alternate FSS configurations that can be used for supporting particular missions. For example, if the Berthing/Positioning System is not required, Cradle A' need not be flown. Cradle A can be used alone or in the A and B configuration. Alternately, Cradle A' can be used alone, with the spacecraft loads supported by the orbiter's longeron and keel latches. Cradle B can be used alone in a limited number of locations within the cargo bay.

For missions in which it is necessary to reverse the location of the positioning platform to the opposite end of the FSS, cradles A and A' may be interchanged or the entire FSS can be rotated.

A spacecraft can be installed in Cradle A, Cradle B, or the combination with its longitudinal axis vertical when a payload interfacing platform is provided. This platform is yet to be developed and is not presently funded. Provision has been made for use of a swing-away latch beam (to be developed) that would allow two spacecraft to be carried in Cradles A and B in an over-and-under configuration. The beam is conceptually designed and its implementation is to be studied. It is possible for the FSS to carry a spacecraft in a position over the Spacelab tunnel; however, special ground-handling equipment would have to be developed, and the integration procedure would have to be coordinated with the Kennedy Space Center (KSC). Figure 4.7-7 shows these alternate configurations in concept.

4.7.1.2 Mechanisms

The FSS contains 12 motor driven mechanisms that perform seven separate functions as listed below. Figure 4.7-4 shows these mechanisms.

1) Retention - 3 latches

2) Berthing - 3 latches

3) Umbilical connection - 2 connector devices

4) Rotation - 1 rotator

5) Pivoting - 1 pivoting mechanism

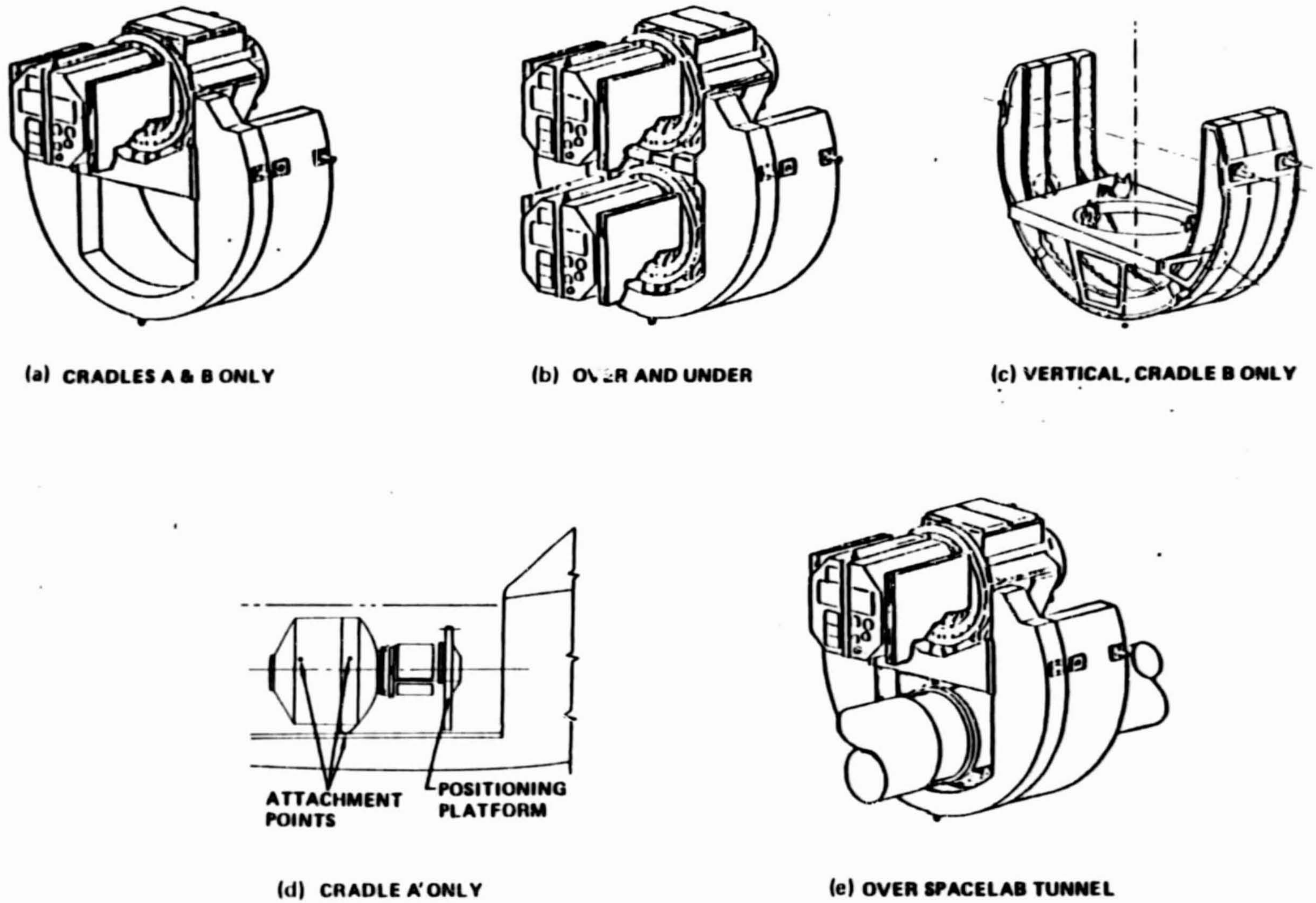


Figure 4.7-7. Several FSS Alternate Configurations

6) Translating - 1 translation system

7) Locking - 1 locking latch

All of the 12 mechanisms are driven by either a high- or low-speed Common Drive Unit (CDU). Each CDU consists of dual three phase 115-V, 400-Hz motors driving through speed reduction gearing and a spur gear.

Further descriptive detail concerning these mechanisms and other system details is provided in the MMS/FSS User's Guide (NASA 1980 (c)).

4.7.2 Spacelab Pallets

Bulk handling of cargo has been shown effective as a means to save time and cost in many industrial and agricultural situations. Therefore, designers of large space structures may well be interested in currently planned cradles and pallets for various Shuttle payloads, devices which potentially could be used for support and bulk handling of equipment and construction materials. One such set of devices is the group of pallets designed for the Spacelab missions, which are described in the Spacelab Accommodations Handbook, ESA SLP/2104.

4.8 EXTRA VEHICULAR ACTIVITY (EVA) CAPABILITIES/CONSTRAINTS AND GUIDELINES

Although a great many large space structures design features will be compatible with remotely controlled or automatic machinery interfaces, direct, "hands-on" processes using space suited crew outside the cabin environment are considered a viable and important technique.

These so-called extravehicular (EVA) methodologies can offer significant cost savings in reducing mechanical and electrical redundancy, avoiding development of new and complex remotely controlled machinery for a few special activities, and providing for on-site visual monitoring and judgments involving assembly and checkout operations and contingency fixes. This section considers the many aspects of EVA, including constraints involving various planned planned and proposed hardware aids applicable to large space structures operations, constraints and guidelines for decisions involving EVA, procedures for accomplishing EVA and provisions required in the orbiter.

Tables 4.8-1 4.8-2, 4.8-3, and 4.8-4 summarize Shuttle constraints and guidelines for design and operations relating to space construction hardware and procedures, together with references to supporting and regulatory documents. Brief descriptive material amplifying these topics follows.

4.8.1 Shuttle Orbiter EVA Definition and Provisions

(Abstracted from Reference NASA 1980 (a)). The term EVA, as applied to the Space Shuttle, includes all activities for which crewmembers don their space suits and life support systems, then egress the orbiter cabin through an airlock into a space vacuum environment to perform operations internal to or external to the cargo bay volume. Further, EVA falls into three basic categories:

(1) Planned - EVA planned prior to launch in order to complete a mission objective.

(2) Unscheduled - EVA not planned, but required to achieve payload operation success or advance overall mission accomplishments.

(3) Contingency - EVA required to effect the safe return of all crewmembers.

Each orbiter mission will provide the equipment and consumables required for three two-man EVA operations. Two of the EVAs will be available for payload operations (e.g., large space construction) and the third retained for orbiter contingency EVA. Additional EVAs are possible if additional comsummables and equipment weights are allotted to the appropriate payload. Large scale space construction operations are likely to benefit from several additional EVA operations, both for planned and unscheduled categories. In providing this capability, the Shuttle program assumes the costs of the development and purchase of the Shuttle EVA systems provisions, standardized EVA support equipment, and EVA crew training.

Table 4.8-1. Checklist: EVA Hardware Constraints for Space Construction

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) Space shall be provided to permit EVA egress from an airlock into the cargo bay. (R-1)	NASA 1980 (a): JSC-07700, Vol. XIV, Para. 9.1.1
. 4 feet from X_o 576 bulkhead is nominally reserved for internal airlock. . Clearance may be provided by payload deployment or jettison. Contingency jettison capability shall always exist. . Additional space may be required for other airlock configurations.	Impacts available stowage volume. See discussion following.
( ) The minimum translation corridor for crew-members in full EVA gear is 101 cm (40 inches) diameter for straight-line translation through hatches and tunnel-like structures, or free-floating without translation aids. Using translation aids (e.g., handholds, space structures) a minimum of 1.1 cm (43 inches) is recommended. Additional volume is required for abrupt turns, generally 48 inches. (R-1)	NASA 1976 (a): JSC-10615, Para. 4.2.1 Crew access space for EVA in space construction. NASA 1977 (a): STS User Handbook
( ) All EVA equipment and space construction interfaces must be designed to be compatible with Shuttle and EVA operations.	NASA 1980 (a): JSC-07700, Vol. XIV, Para. 9.1 NASA 1976: JSC-10615
( ) Handrails and handholds shall have the same standard dimensions: (R-1) . Cross-section: .72 to .78 inches thick, 1.29 to 1.41 inches width . Offset for finger clearance: 2.25 inches to inner surface. . See JSC-10615 for further details, loads.	NASA 1980 (a): JSC-07700, Vol. XIV, Para. 9.1.5 NASA 1976 (a): JSC-10615 Affects weight and detail design of fixtures.
( ) Safety tether attach provisions shall be provided along EVA handrails and handhold pathways, and at workstations. (R-1)	NASA 1976 (a): JSC-10615 Affects weight and detail design of fixtures

Table 4.8-1. Checklist: EVA Hardware Constraints for Space Construction (Continued)

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) Foot restraints shall be provided at EVA work stations, and shall be universal, Skylab type baselined for orbiter and payload applications. (R-1)	NASA 1976 (a): JSC-10615 Affects weight and detail design of fixtures.
( ) All equipment and tools transported or handled during EVA should provide a safety tether attach point. (R-1)	NASA 1976 (a): JSC-10615 Affects weight and detail design of equipment, tools, structure.
( ) Crewman translation provisions (e.g., hand-rails, handholds, mobility aids) in the payload planned EVA work area shall be provided by the payload (space construction project) if requirements exceed orbiter-attached cargo bay handrails. (R-1)	NASA 1976 (a): JSC-10615 Affects design of access to cargo bay stowage by EVA.
( ) The EVA crewman and equipment shall be firmly secured or tethered at all times except during Manned Maneuvering Unit (MMU) operations; in this case, equipment will be secured to the MMU. (R-1)	NASA 1976 (a): JSC-10615 Affects crew timeline, procedures, detail equipment design.
( ) Space construction devices shall provide crew safety from electrical, fluid, radiation, mechanical and other hazards. (R-1)	NASA 1976 (a): JSC-10615 Affects detail design, placement of equipment and handling procedures.
( ) Weight and volume of cargo/equipment transfer aids provided for space construction EVA support is chargeable to the payload carried by the orbiter. (R-1)	NASA 1976 (a): JSC-10615 Affects total weight to orbit and cost of flight.

Table 4.8-1. Checklist: EVA Hardware Constraints for Space Construction (Continued)

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) Weight and volume of portable EVA work stations will be chargeable to the space construction payload carried by the orbiter. NASA plans to make standard devices available with further development cost charges. (R-1)	NASA 1976: JSC-10615
( ) Space construction designs (both spacecraft and construction equipment) shall adhere to criteria limiting sharp edges, corners and protrusions along translation paths and at worksites to avoid possible damage to the EMU. (R-1)	NASA 1976: JSC-10615, Para. 4.4 and Table 4-4. Affects detail design.
( ) Space construction equipment or surfaces sensitive to inadvertant physical damage by an EVA crew member shall be protected or located outside the EVA translation path or EVA work areas. (R-1)	NASA 1976: JSC-10615, Para. 4.4 Affects detail design for EVA.
( ) Crew translation tethers or umbilicals shall be restrained to preclude damage or entanglement and possible damage to surrounding equipment. (R-1)	NASA 1976: JSC-10615 Affects detail design for EVA.
( ) S-band and Ku-band channels will be used for communications between all orbiter crew members and the appropriate grolnd control centers for construction operations transmission to ground. Voice and data can be relayed to ground by the TDRSS (Tracking and Data Relay Satellote System). (R-1) • Reference Figure 4.8-10.	NASA 1980 (a): JSC-07700 Affects selection of support equipment for construction.

Table 4.8-2. Checklist: EVA Hardware Guidelines for Space Construction

( ) GUIDELINES/•QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) EVA work areas and crew interface provisions should be standardized as much as practical to minimize development and testing costs. Equipment should be designed around conventional, well known techniques. (R-1)	NASA 1976: <u>JSC-10615</u> Affects selection of EVA equipment and detail design.
( ) As applicable, EVA support equipment design and lighting will adhere to JSC specifications SC-E-0006 and SC-L-0002, respectively. (R-1) NOTE: A wide variety of special lighting considerations apply to space construction. These are discussed in Section 4.9. Such considerations may alter the above guidelines by means of improved vision apparatus, marker beacons, self-illuminating surfaces and other means to assure adequate <u>vision</u> which is the primary requirement.	NASA 1976: <u>JSC-10615, Para. 4.1.3</u> Affects power required for illumination, weight, volume, and detail design of space construction support equipment and structures/systems. Lighting potentially demands high power, requires careful design analysis.
( ) Handrails shall be colored yellow, Federal Standard 595 No. 23785. (R-1)	Orbiter Engineering Documentation

Table 4.8-3. Checklist: EVA Operations Constraints for Space Construction

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) Scheduled EVA time per day for one crew member in an EMU is 6 hours at 4.0 psi. (R-1)	NASA 1980 (a): JSC-07700, Vol. XIV, Para. 9.1, 9.1.6
. Five hours per day is recommended for repetitive activity, to limit the normal astronaut work day to 10 hours (5 hours is nominal time for pre-breathing and suit donning/doffing). . Advanced EMU developments may increase available work time per crew day.	Affects construction time and crew size. Design of procedures, supplies are related to equipment availability schedule.
( ) Backup IVA crew (for EVA operator rescue) may not pre-breathe 100% oxygen for more than 6 hours. (R-1)	Personal Communication to NASA JSC Medical Operations (J. Walagora)
( ) Provisions are required for EVA crew member backup (rescue) at levels commensurate with job risk. (R-1)	(Safety consideration) . Requires two persons for all EVA.
. Second EVA crew member (buddy system) is most desirable and provides most rapid backup response. . Another option (TBD) is IVA crew member in "pre-breathed" condition, ready to suit up (1½ hour maximum response time) for simple entanglement situations. Consider effect on total EVA capability (6 Hrs. max.)	. May reduce effective EVA time by suit donning and egress time for backup crew, especially last EVA in a series.
( ) EMU (pressure suit) donning and checkout requires 1.5 hours. (R-1)	NASA 1976: JSC-10615
. Based on Shuttle EVA suit system and available documentation. . See Section 4.8.4.	Impacts EVA construction useful work time.

Table 4.8-3. Checklist: EVA Operations Constraints for Space Construction (Continued)

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES IMPLICATIONS
( ) Preparation for EVA requires 3 hours pre-breathing of 100% oxygen. (R-1)	NASA 1976: JSC-10615
. One of these three hours should be preparation devoted to suit donning to minimize preparation time.	Impacts EVA construction useful work time.
. Some IVA crew duties may be performed while wearing O_2 breathing equipment.	. Affects O_2 supplies volume and launch weight.
. See Section 4.8.4.	
( ) EMU doffing requires 1.5 hours. (R-1)	NASA 1976: JSC-10615
. Includes battery changeout.	Affects crew work-rest cycle planning
. See Section 4.8.4.	
( ) EMU battery recharge requires 16 hours. (R-1)	NASA 1976:
	Affects number of batteries and EVA timelines.

Table 4.8-4. Checklist: EVA Operations Guidelines for Space Construction

( ) GUIDELINES/•QUALIFICATIONS	REFERENCES IMPLICATIONS
( ) Consider EVA, or EVA in conjunction with manipulators, for accomplishing a space construction task when: (R-3)	Vought Missiles & Space Co. 1972 P. 25
. Precise feedback is required . The use of small force gradations is required.	Selection of sensors, manipulators and visual devices for space construction.
. Stereopsis is required.	Crew size and schedules.
. Precise placement is required. . Several manipulator terminal devices would be required.	Complexity of manipulators, weight, volume complexity of visual support equipment.
. A wide field of view is required, or a variety of viewpoints will aid rapid assessment of situations.	Safety provisions.
. Access is restricted.	Potential new equipment to extend reach of manipulators.
. No major hazards are present. . Work area is outside manipulator envelope of reach or handling capability.	Construction equipment redundancy, weight and cost.
. Primary means malfunction.	
. On-site supervision, investigation and judgement is required or highly desirable for rapid response and fixes.	
( ) Consider use of EVA to avoid cost of developing specialized, remotely controlled machinery or automatic machinery for non-repetitive tasks, requiring dexterity or judgment. (R-4)	(Summary conclusion from several sources)

Table 4.8-4. Checklist: EVA Operations Guidelines for Space Construction (Continued)

( ) GUIDELINES/•QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) Schedule serial EVA's in close sequence whenever possible to minimize repressurization cycles of the airlock. (R-3) . Minimizes launch weight . Minimizes impact on IVA crew work time. . Minimizes turnaround time. . Helps keep costs lower.	Roebuck, et.al, 1980
( ) Provide adequate scheduled rest periods for EVA work periods. Approximately 10 minutes per hour is recommended for active work periods. (R-4)	Roebuck, et.al, 1980

4.8.1.1 Airlock

The airlock provides the means of transfer from the shirtsleeve environment of the cabin to the vacuum environment of space and contains the pressurization and depressurization systems necessary to effect such a transition. The airlock can be installed in three different orbiter locations, depending upon the payload carried. The baselined airlock location is inside the cabin mid-deck compartment, attached to the forward side of the Xo 576 bulkhead, (Figure 4.8-1). This position allows maximum use of cargo bay volume. However, the airlock may also be rotated 180° and mounted in the cargo bay, attached to the aft side of the Xo 576 bulkhead (Figure 4.8-2). Also, the airlock may be positioned in the cargo bay on top of the tunnel adapter for a habitable payload mission (Figure 4.8-3). The airlock has two D-shaped hatches and provides stowage volume for two sets of crew EVA equipment when not in use.

These airlock arrangements and possible EVA routes are illustrated in Figure 4.8-4. It should be noted that the optional EVA route, shown on Figure 4.8-4 is viable only if a hatch cover is installed at the Xo 576 bulkhead.

4.8.1.2 Tunnel Adapter

The tunnel adapter is provided as a mission kit for missions which contain a habitable payload and do not have a docking requirement. The tunnel adapter is attached to the Xo 576 bulkhead and located in the cargo bay. It is a payload chargeable weight item and is used in conjunction with the airlock (Figure 4.8-3) to provide shirtsleeve access between the orbiter cabin and the habitable payload concurrent with an EVA operation. Any weight in excess of that of an airlock mounted inside the cabin will be allotted to payload(s). The interface between the tunnel adapter and the habitable payload is to be provided as part of the transfer tunnel mission kit and will be bolted directly to the tunnel adapter. Transferable package size limits for suited and unsuited operations through the tunnel adapter are the same as those for the airlock.

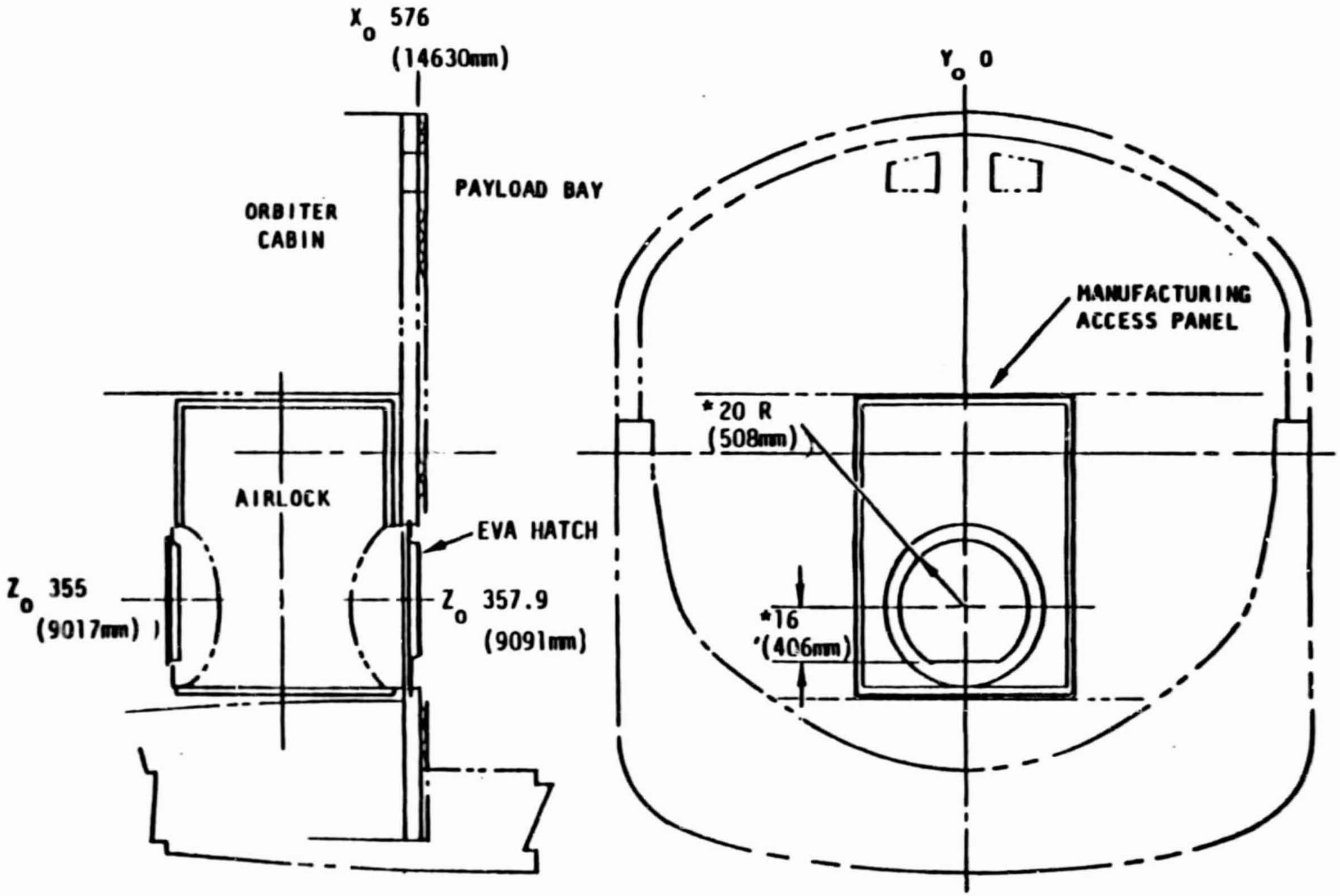
4.8.1.3 Docking Module

The orbiter may be docked to another orbital element by utilizing an extendable docking module (Figure 4.8-5) with an active docking system. The docking module is installed in the cargo bay, attached to the Xo 576 bulkhead and supported from the longeron and keel by payload attachment fittings. The docking module will also serve as an airlock when installed. The nominal protrusion into the cargo bay is to Xo 660. The maximum retracted height of the docking module ring plane is Zo 457, which allows EVA through the upper docking module hatch with the cargo bay doors closed. The docking ring plane will be no lower than Zo 515 when extended. This allows docking to occur 15 inches (380 mm) above the mold line. Transferable package size limits for suited and unsuited operations are the same as those for the airlock.

4.8.1.4 Hatches for Airlock, Tunnel Adapter, and Docking Module

All hatches and hatch components connected to the manufacturing access panel at Xo 576 will be on a Zo 357.9 centerline. All hatches will conform

NASA-S-75-7149 B



NOTE: DIMENSIONS IN INCHES

* THESE DIMENSIONS RELATE TO PASSAGEWAY SIZE

Figure 4.8-1. Internal Airlock (Baseline)

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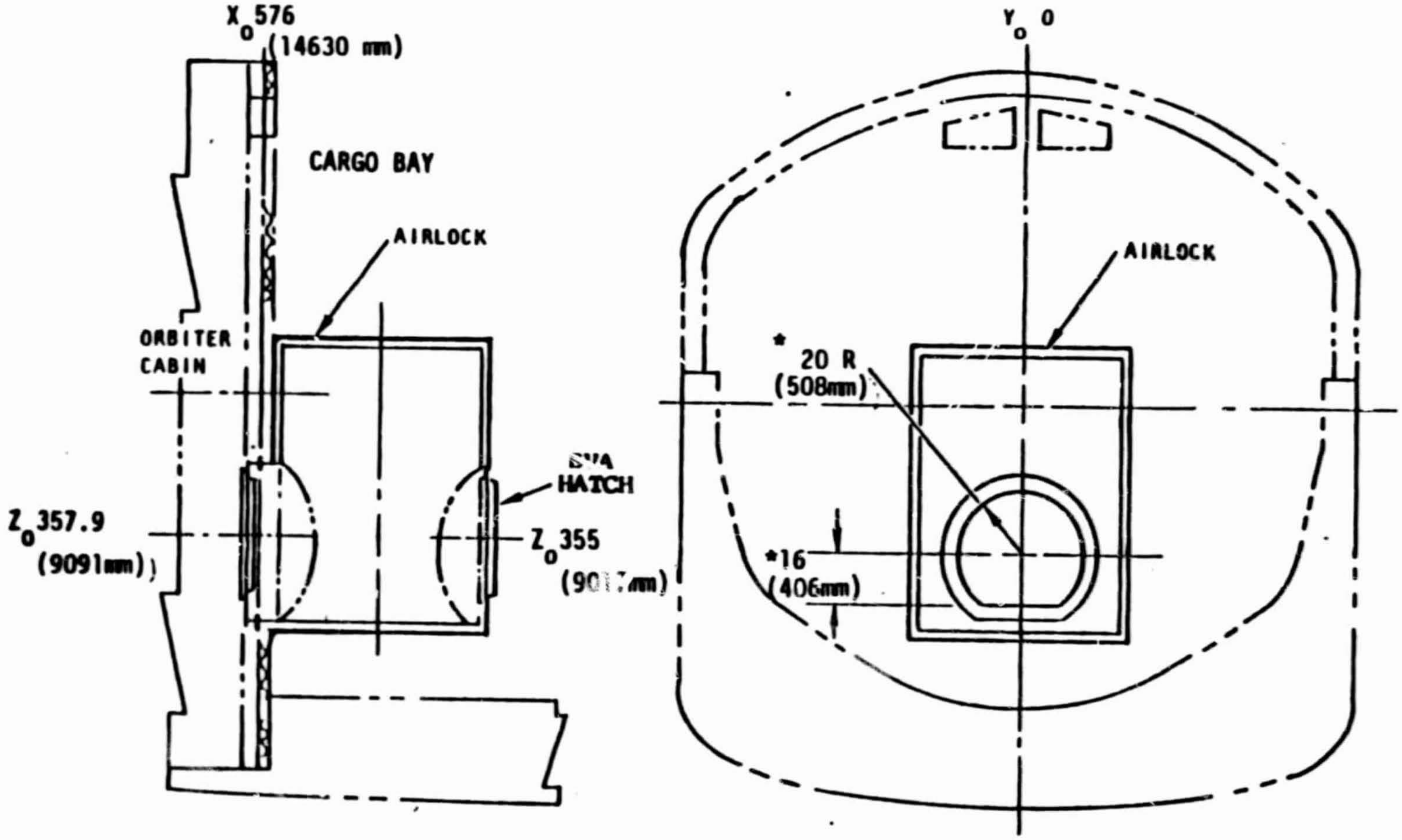


NOTE: DIMENSIONS IN INCHES

* THESE DIMENSIONS RELATE TO PASSAGEWAY SIZE

Figure 4.8-2. Cargo Bay Airlock

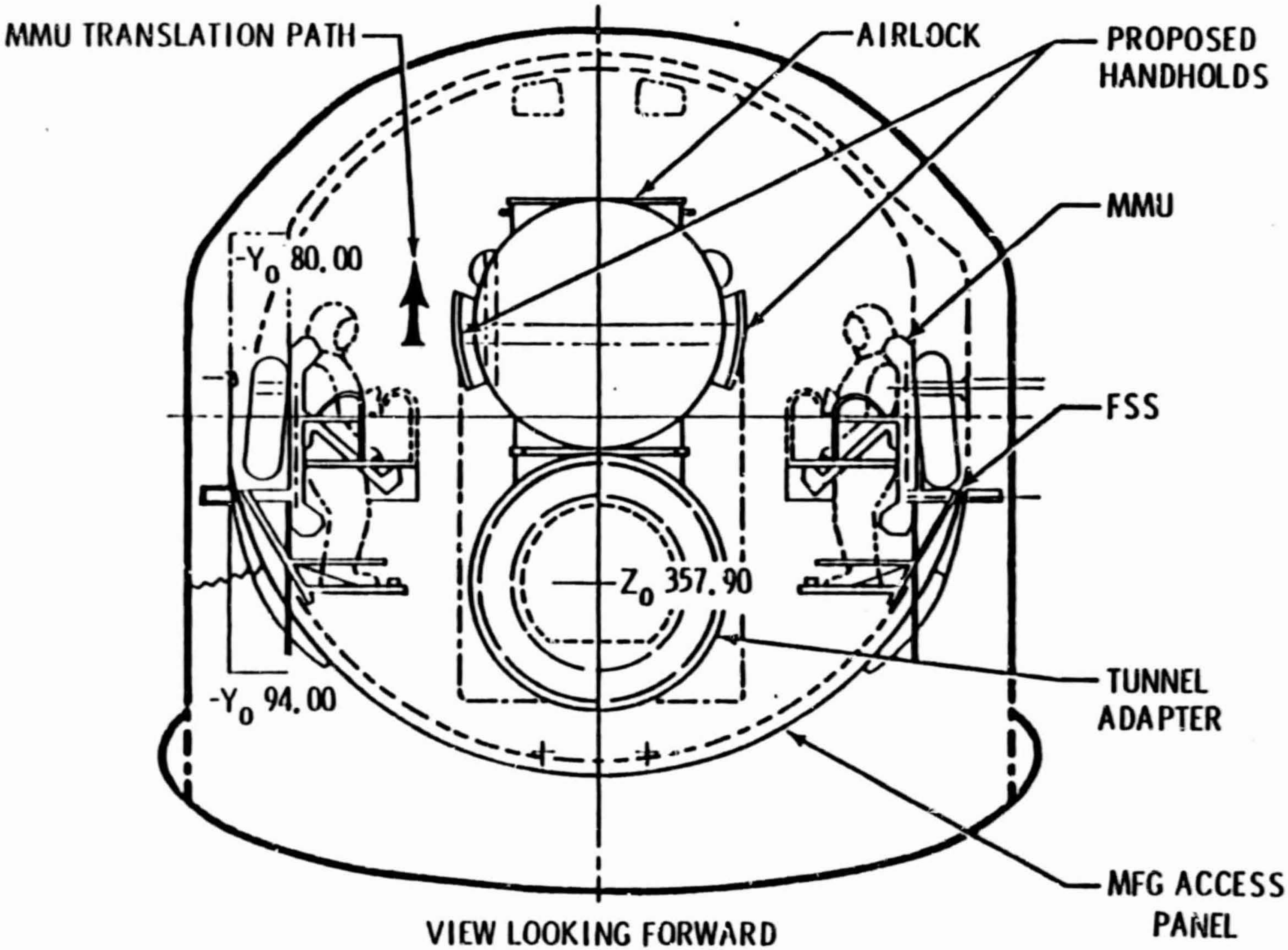
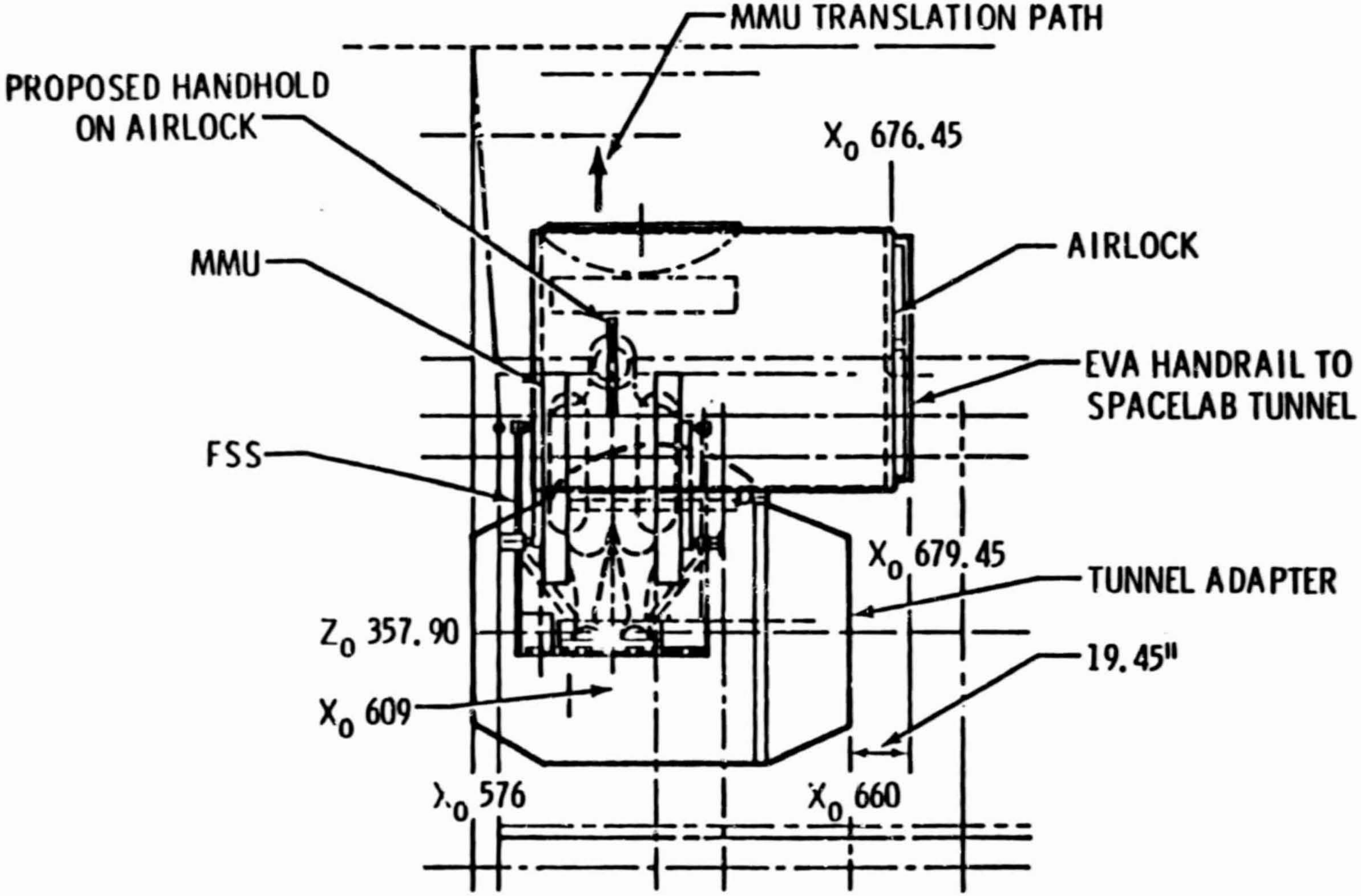


Figure 4.8-3. Cargo Bay Airlock/Tunnel Adapter



VIEW LOOKING INBOARD - LEFT SIDE

Figure 4.8-3A. Cargo Bay Airlock/Tunnel Adapter

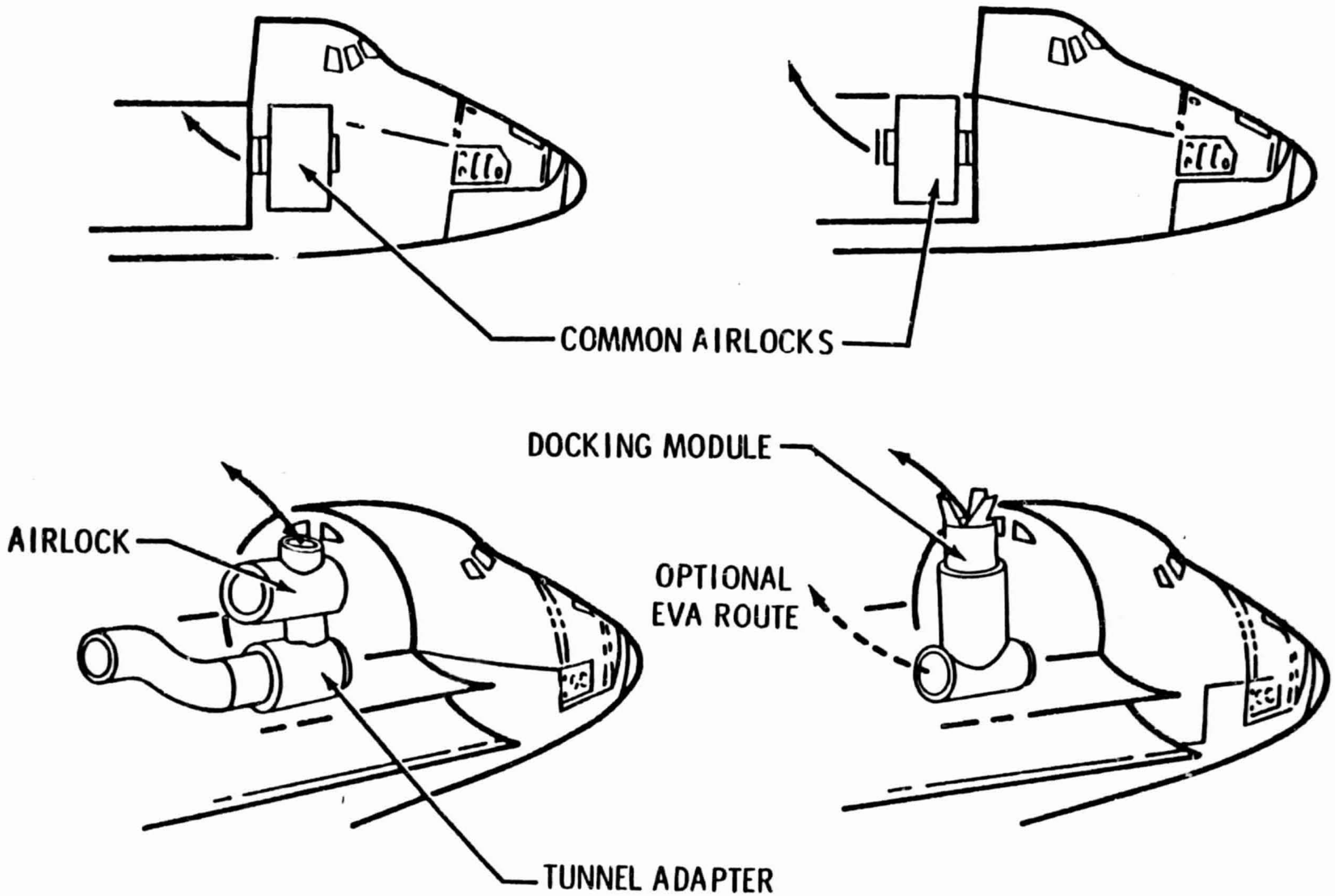


Figure 4.8-4. Possible Airlock Arrangements With EVA Routes

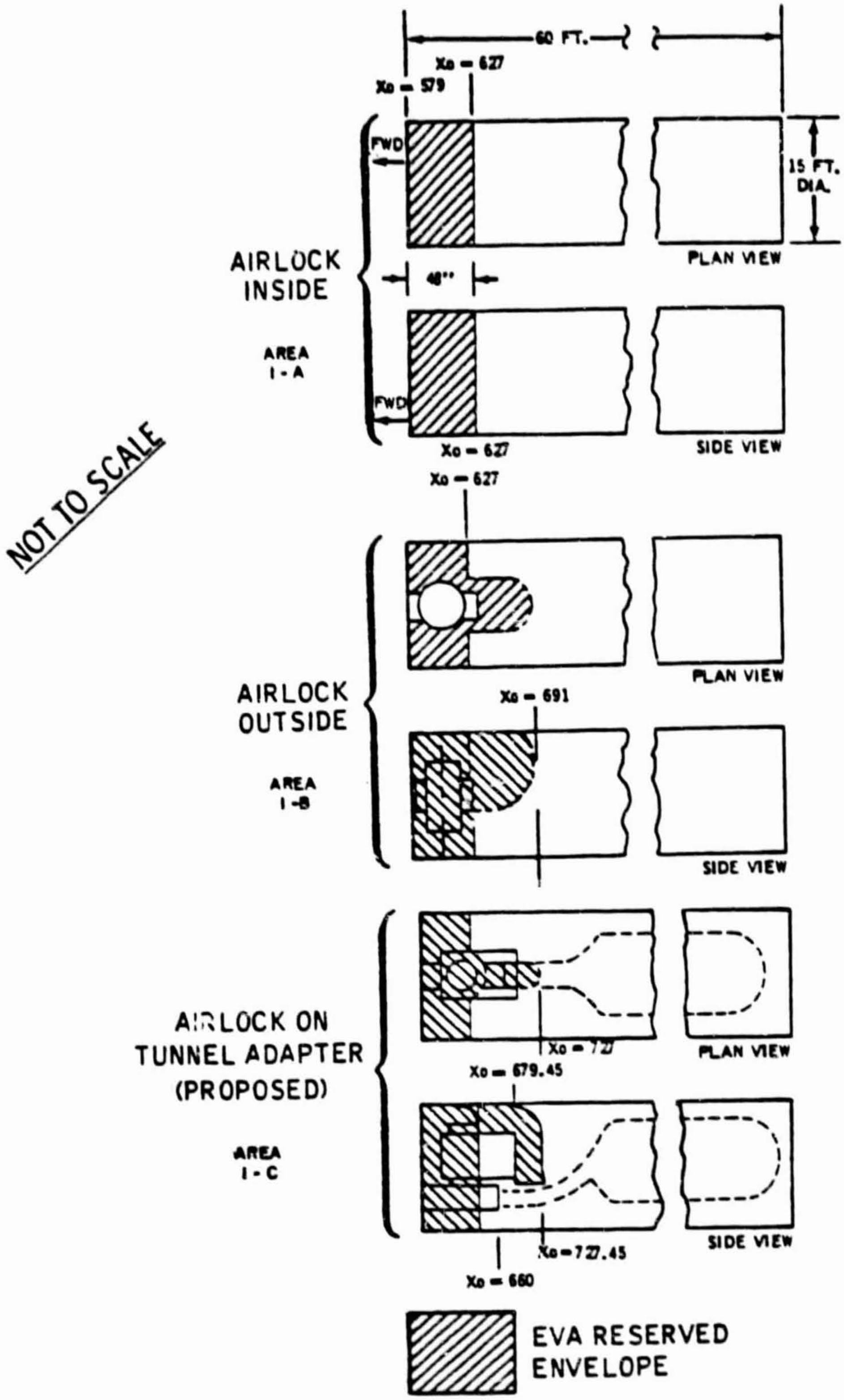


Figure 4.8-4A. EVA Reserved Envelopes for Forward Cargo Bay Area

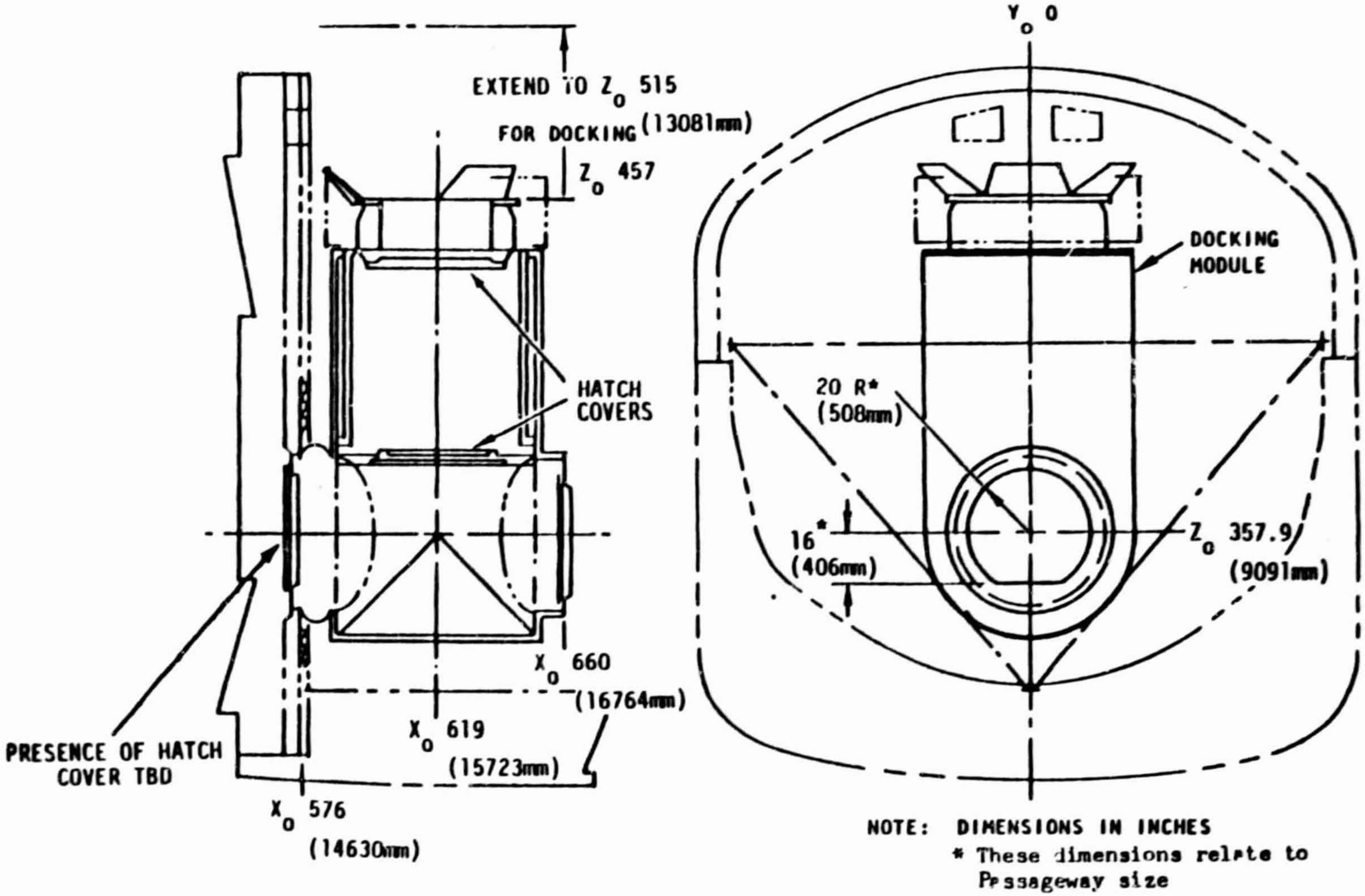


Figure 4.8-5. Docking Module

to Figure 4.8-6 which standardizes hatch opening size, except for the cabin side hatch. All hatch covers, except for the side hatch, will open toward the primary pressure source, the orbiter cabin. They will have six equally spaced latches, a gear box and actuator, and two pressure equalization valves and hatch opening handles, one set on each side. There will be two pressure seals on the hatch-side of the structural interface, one on the hatch cover and one on the structural interface. In addition, there will be pressure relief valves and differential pressure gauges provided for each hatch cover. The guage details and locations are TBD.

4.8.1.5 Handrails, Tethers and Foot Supports

Handrails and handholds are provided to facilitate zero-g crew movement within the orbiter flight and mid-decks, in and on the airlock, and around the periphery of the cargo bay (Figure 4.8-7A). Handrails and handholds throughout the orbiter will have the same standard dimensions for intravehicular and extravehicular use. The cross section outside dimensions of the handrails and handholds are limited to the following minor and major axis range measurements, respectively: .72 to .78 x 1.29 to 1.41 inches (18.29 to 19.81 x 32.76 to 35.81 mm). See JSC-10615 (NASA 1976(a)) for details.

The EVA crew is also provided with safety tethers and associated slidewires on each side of the payload bay. Figure 4.8-7B and 7C illustrate the current concept for stowage and location of these devices on operational vehicles. Note that the starboard slidewire location cannot be used with an optional RMS and could interfere with use of a payload installation and deployment aid (PIDA, Section 4.6), requiring a special new development configuration. Also, the slidewire configuration shown cannot be used on the port side in the presence of an RMS.

The safety tether is partially stowed on a reel assembly located on the X_o 576 bulkhead per Figure 4.8-7B. In the stowed condition, a portion of the tether is extended and temporarily clipped parallel to the handrails of the X_o 576 bulkhead, and from thence route to a fixed attachment at the slide on the forward end of the slidewire. In usage, the EVA astronaut unstows the reel from the stowage box, clips the reel to his suit and permits the tether to be taken up on the reel as he egresses the airlock and travels out toward the cargo bay side handrails. Thus, one end of the safety tether is always attached to the slide on the slidewire, no matter where the EVA crew member is working in the cargo bay. Special new provisions may be required for EVA to construction fixtures and large space structures.

Note: The current baseline also requires that crew members using the MMU will be tethered to the orbiter. Details of these tether arrangements are TBD.

There are no fixed foot supports provided in the cargo bay, except those incorporated in the flight support station for the MMU (see Section 4.8.2.3). A similar set may be provided with the open cherry picker, if installed (see Section 4.8.2.4). A portable foot support system is provided to aid access to cargo bay door latches at the forward and aft cargo bay bulkheads, as described in Section 4.8.2.2.

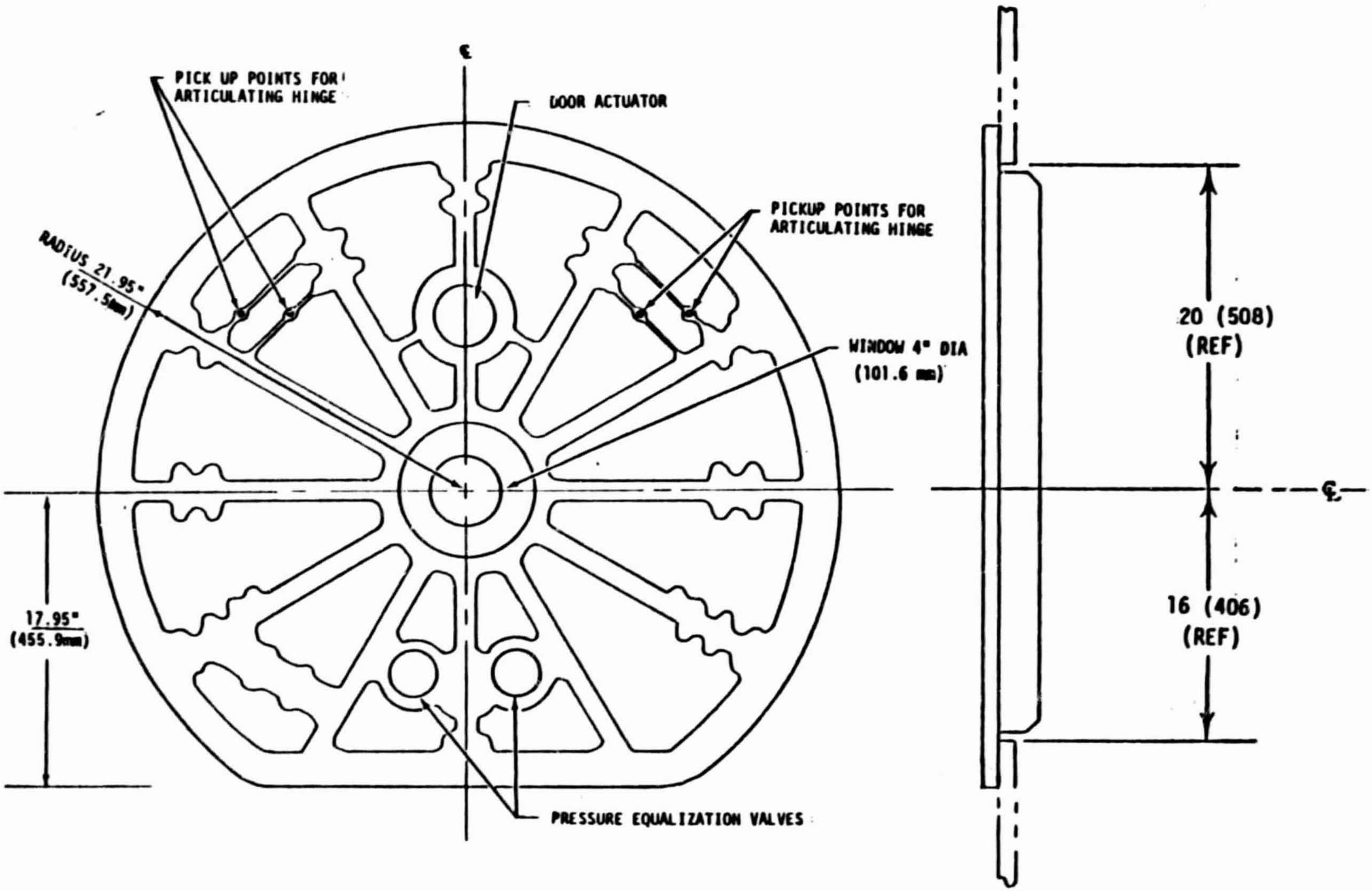
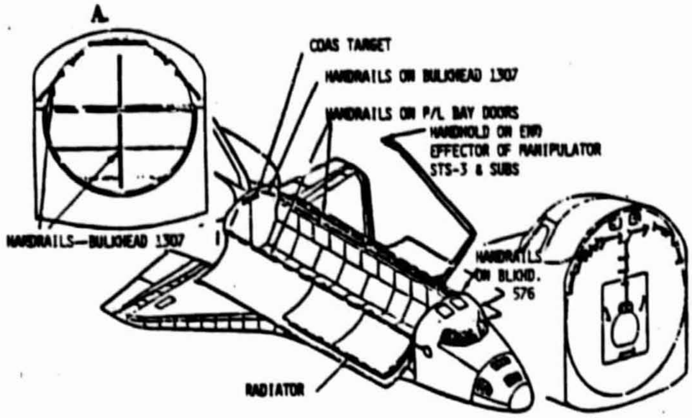
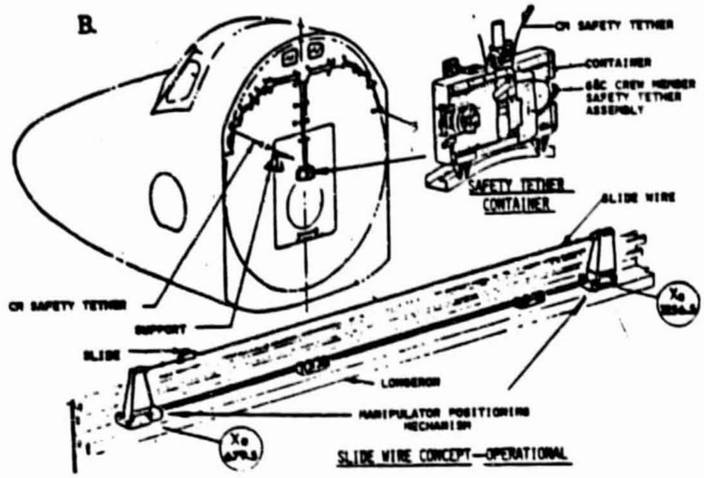


Figure 4.8-6. D-Shaped Hatch Basic Configuration





- Starboard Configuration
- Port (TBD)

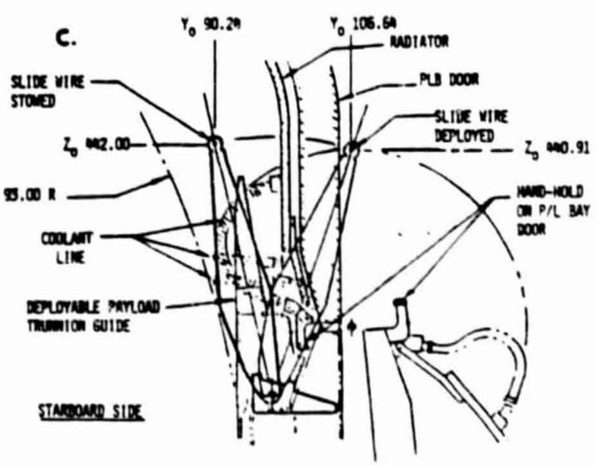


Figure 4.8-7. Payload Bay EVA Handrails and Safety Tethers/Slidewires

4.8.1.6 Extravehicular Mobility Unit

The extravehicular mobility unit (EMU) consists of a self-contained (no umbilicals) life support system and an anthropomorphic pressure garment with thermal and micrometeoroid protection, shown in Figure 4.8-8. It provides a breathing environment at a differential pressure of 4 pounds per square inch (206.8 mm Hg) and incorporates provisions for internal liquid cooling, communications equipment, special EVA helmet visor protection, crew comfort devices and external restraint and tethering fittings.

The unit and associated life-supporting consumables provide for a 6-hour nominal EVA with a subsequent recharge capability for additional EVAs.

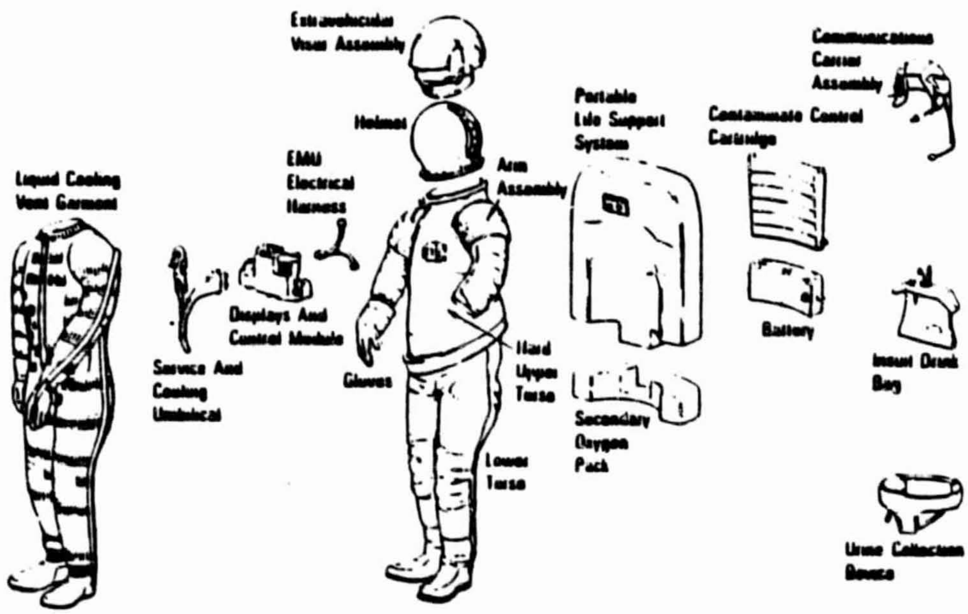


Figure 4.8-8. Space Shuttle Extravehicular Maneuvering Unit (EMU)

4.8.1.7 Communications for EVA

The orbiter provides ultrahigh frequency duplex communications from the flight deck crew members to the EVA crew members and between the latter. S-band and Ku-band channels will be used for communications between all orbiter crew members and the appropriate ground control centers for payload data transmission to ground. Both voice and data can be relayed to ground by the Tracking and Data Relay Satellite System (Figure 4.8-9).

4.8.2 EVA Aids

The following EVA support equipment, may as an option, be available for each operational Shuttle mission.

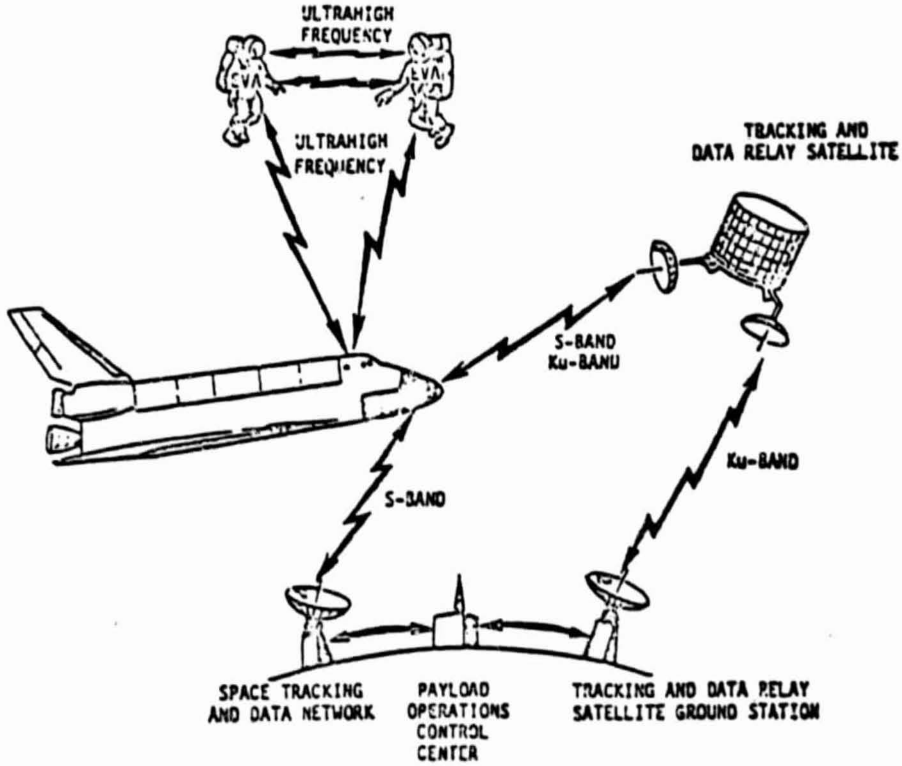


Figure 4.8-9. Shuttle EVA Communications

4.8.2.1 RMS Assistance for EVA

In addition to its capabilities for payload deployment and retrieval operations prior to or subsequent to an EVA, the RMS, within its reach envelope, may perform one or more of the following functions to assist the EVA crewmembers:

1. It can effect multiple transfer of equipment between the EVA work area and the replacement equipment stowage area.

2. It can be used to transfer an EVA crewmember to remote areas on the payload or orbiter by use of a handrail on the end effector or the Open Cherry Picker (see Section 4.8.2.4).

3. The attached lights can be used to supply additional lighting at the work area, and the attached closed circuit TV camera can aid in payload inspection tasks and in tasks involving coordination with the other orbiter crewmembers and with the ground.

The RMS generally can transport modules of large mass faster than can be safely accomplished by EVA methods. Figure 4.8-10 illustrates some tentative results from a comparison of available data for EVA methods and use of the RMS for masses in the range from a few hundred pounds to 32,000 pounds (14,515 kg). Therefore, these data suggest that humans should be utilized for low-mass transport and dextrous operations, and should rely upon the RMS to do large mass transport.

4.8.2.2 Tools, Restraints, Ancillary Equipment

To perform equipment maintenance, repair and replacement, the EVA crewmember requires certain tools, tethers, restraints and portable workstations.

GFE EVA workstations and equipment are available for use by the payloads. When justified, modifications to the NASA workstation hardware can be requested or unique hardware can be provided if it has been designed to appropriate NASA specifications.

An EVA tool kit will be available for use in supporting orbiter and payload EVA operations. These tools will be kept in the NASA ground inventory and flown as required for each payload. Any payload-unique tools required will be furnished by the payload user. A primary design goal, however, will be to design the payload components to be cost effective by allowing standard, NASA-supplied hand tools to be used for their servicing.

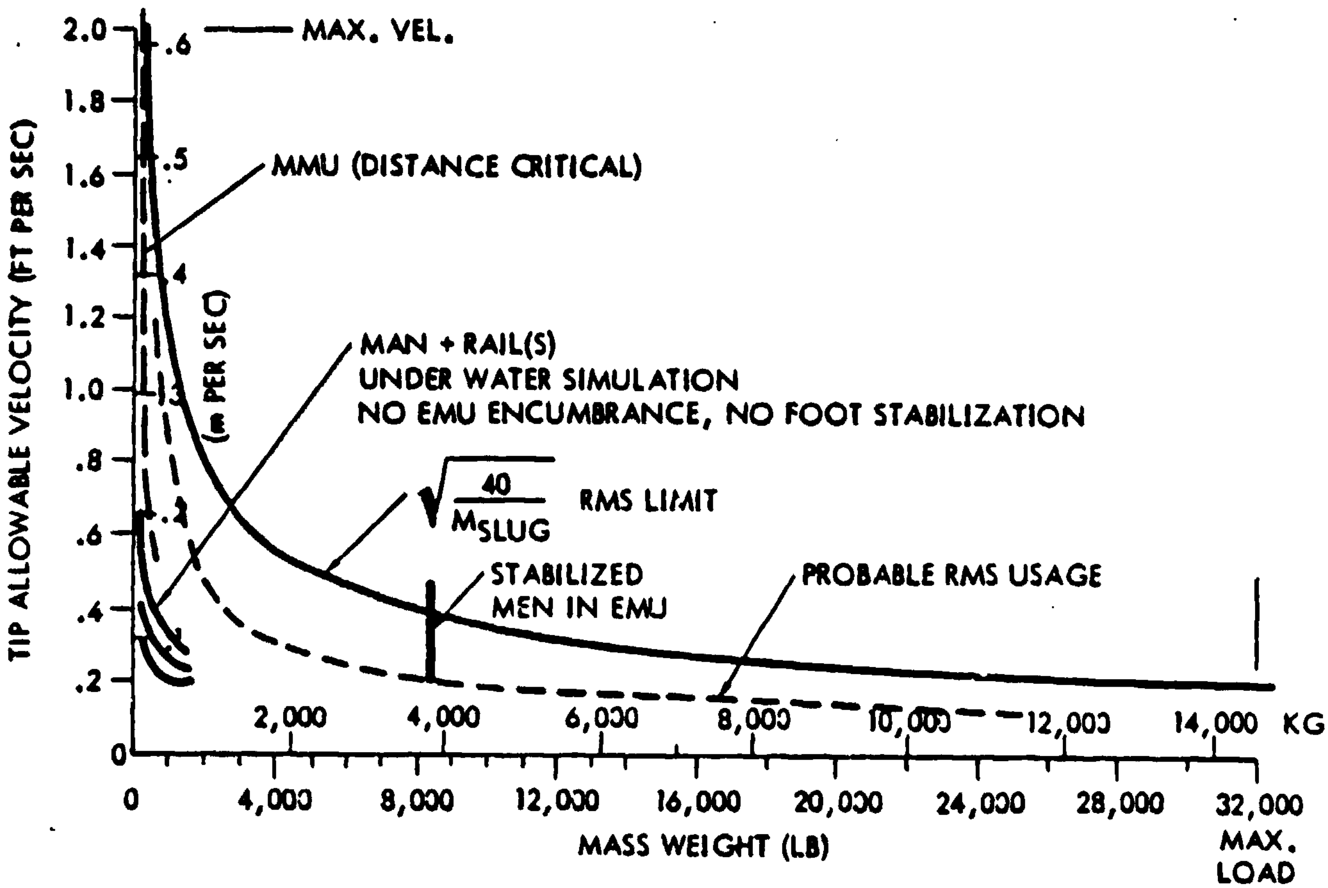


Figure 4.8-10. Human EVA Transport Velocity and RMS Transport Velocity Versus Mass Transported

The current set of EVA tools are pictured in Figure 4.8-11. Their position in relation to a tool caddy for stowage and transport by the crew is shown in Figure 4.8-12. Several of these tools are specifically designed to assist in closing the cargo bay doors and are used in the vicinity of the latches on the forward and aft bulkheads of the cargo bay. Space allocations for use of these tools must be considered as a constraint in designing cargo bay stowage. A particular example of importance is the 3-point latch tool which is used near the longeron and the X_0 576 bulkhead. Specific clearance requirements for this tool usage are TBD.

The portable workstation will be the crewmember's restraining platform while he is performing EVA tasks and will provide stowage for his tools, tethers, portable light and other ancillary equipment. The workstation will be universal in design and may attach directly to the payload, to the orbiter structure or even to the RMS in supporting various EVA tasks.

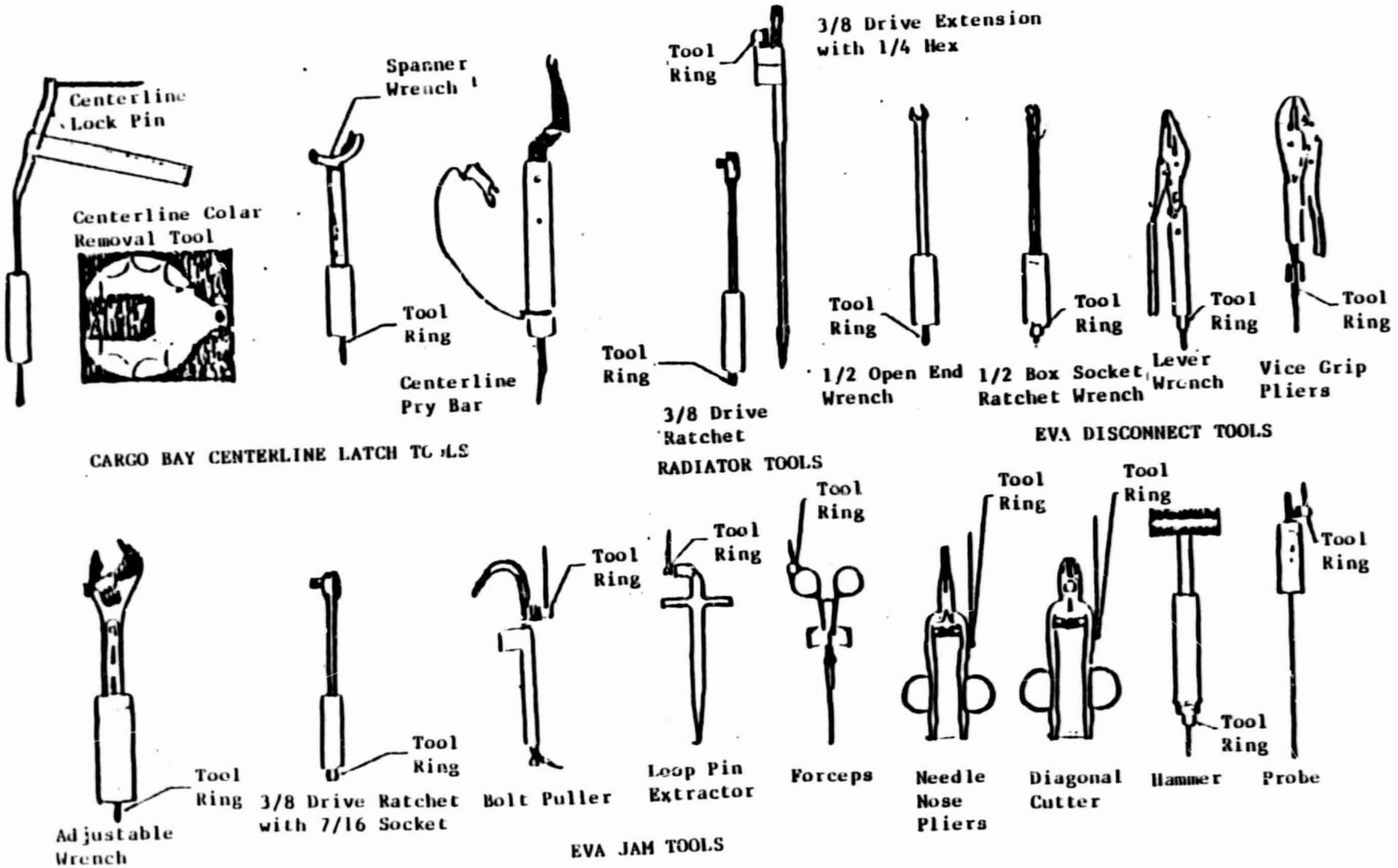


Figure 4.8-11. EVA Tools

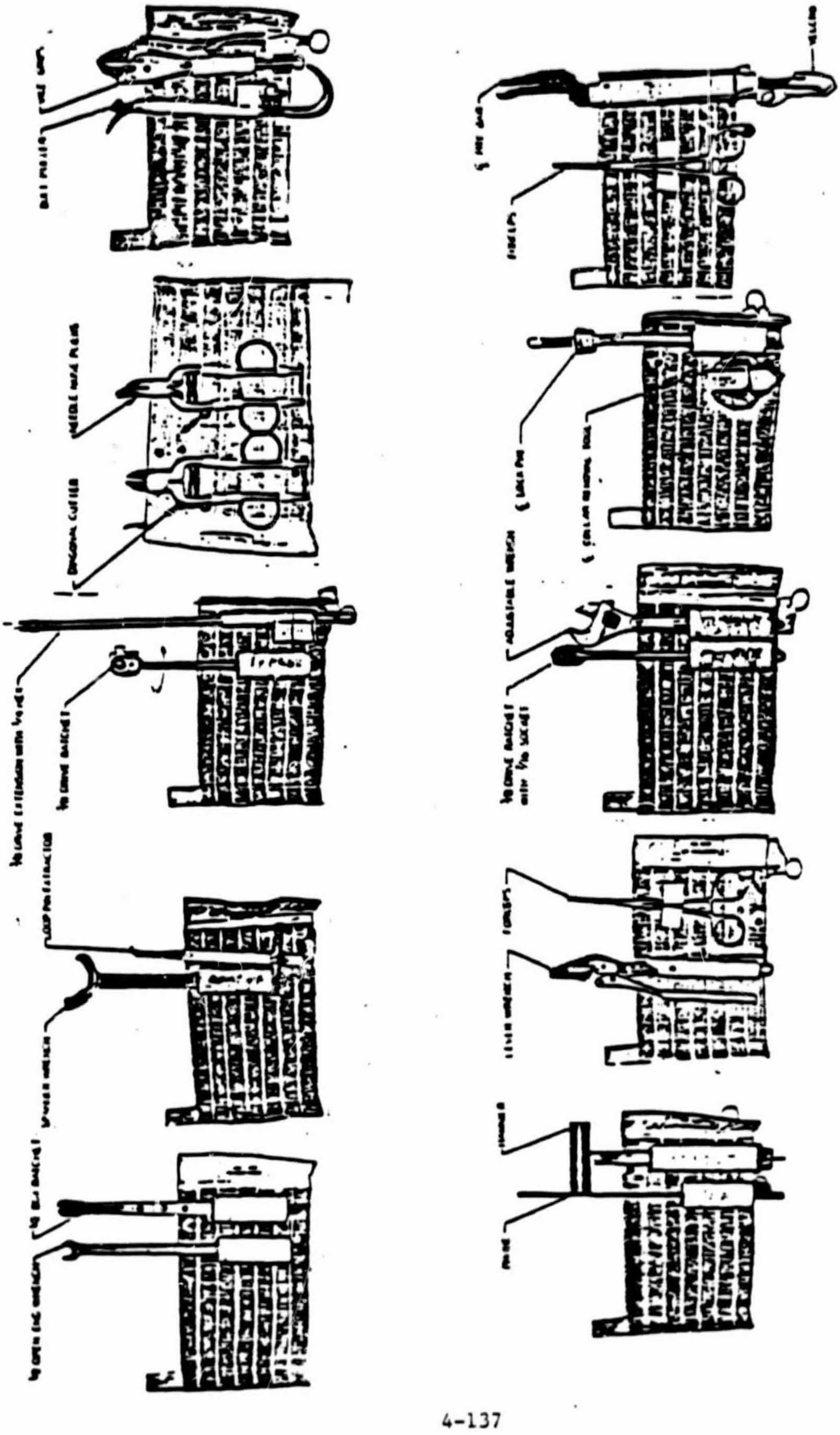


Figure 4.8-12. EVA Tool Caddy Assemblies

All EVA support hardware provided by the payload or user, shall be compatible with EVA operations and approved by NASA. The orbiter will provide EVA translation aids or handrails to support baselined orbiter contingency operations and for general inspection/remedial repair. In addition to the above-described general purpose concept for a portable workstation, NASA will provide a special portable foot restraint device to assist in contingency EVA involving cargo bay door latches. The current concept for this device and its planned usage at the forward bulkhead is shown in Figure 4.8-13. Note that space requirements for using this device may conflict with use of a docking module of the configuration shown in Figure 4.8-5. Special-development hardware would be required to provide for both functions. Other general purpose translation aids will be provided in the GFE baseline support equipment inventory. These aids will provide a means to translate and obtain EVA accessability to various other areas on the orbiter, to payloads installed within the cargo bay, and will, to a limited extent, depending on the payload configuration, provide access to work sites on deployed payloads. EVA crew members restraints and limited supplementary lighting will also be provided as part of the GFE baseline. Where unique translation aids, restraint provisions or supplemental external or internal payload lighting are required, they must be provided by payloads and satisfy NASA design and operational requirements. The handrail on the RMS end effector may also be used to support payload EVA operations.

Provisions will also be available in the cargo bay to plug in a TV camera for EVA coverage.

The payload designer is encouraged to purchase and use standard, NASA-specified "universal" or "multimission" EVA support hardware where possible in order to minimize flight-specific EVA training, operational requirements and cost.

4.8.2.3 Manned Maneuvering Unit (MMU) and Flight Support Station (FSS)

The Manned Maneuvering Unit is a propulsive backpack device which gives the EVA crewmember the capability to reach areas outside the cargo bay, particularly those not accessible by RMS or other means. It also enhances rate of transport for EVA operations which involve relatively long distances. Figure 4.8-14 illustrates the basic MMU in a stowed configuration before crew donning. Figure 4.8-14 illlstrates the nominal servicing position of the EVA crewmenber. Figure 4.8-15 illustrates the donning and egress configurations.

Table 4.8-5 provides operational capabilities and constraints involving use of MMUs.

4.8.2.4 Manned Remote Work Station

At the time of preparation for this document, NASA was considering development of a group of devices with potentially great significance to space operations involving large space structures. The general concept of this group of devices was titled "Manned Remote Work Station". The primary article planned was an open cherry picker (OCP) platform designed by Grumman Aerospace Corporation to attach to the standard end effector of the orbiter.

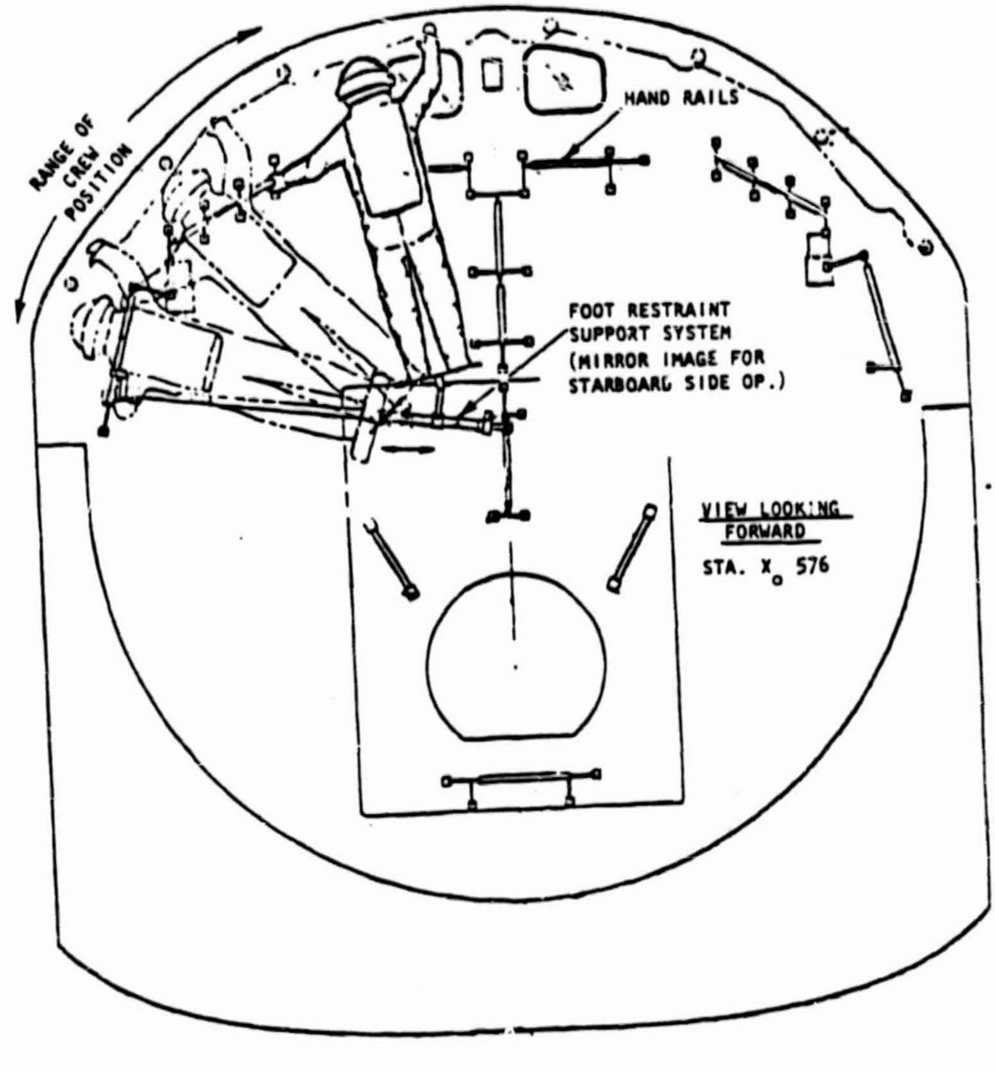
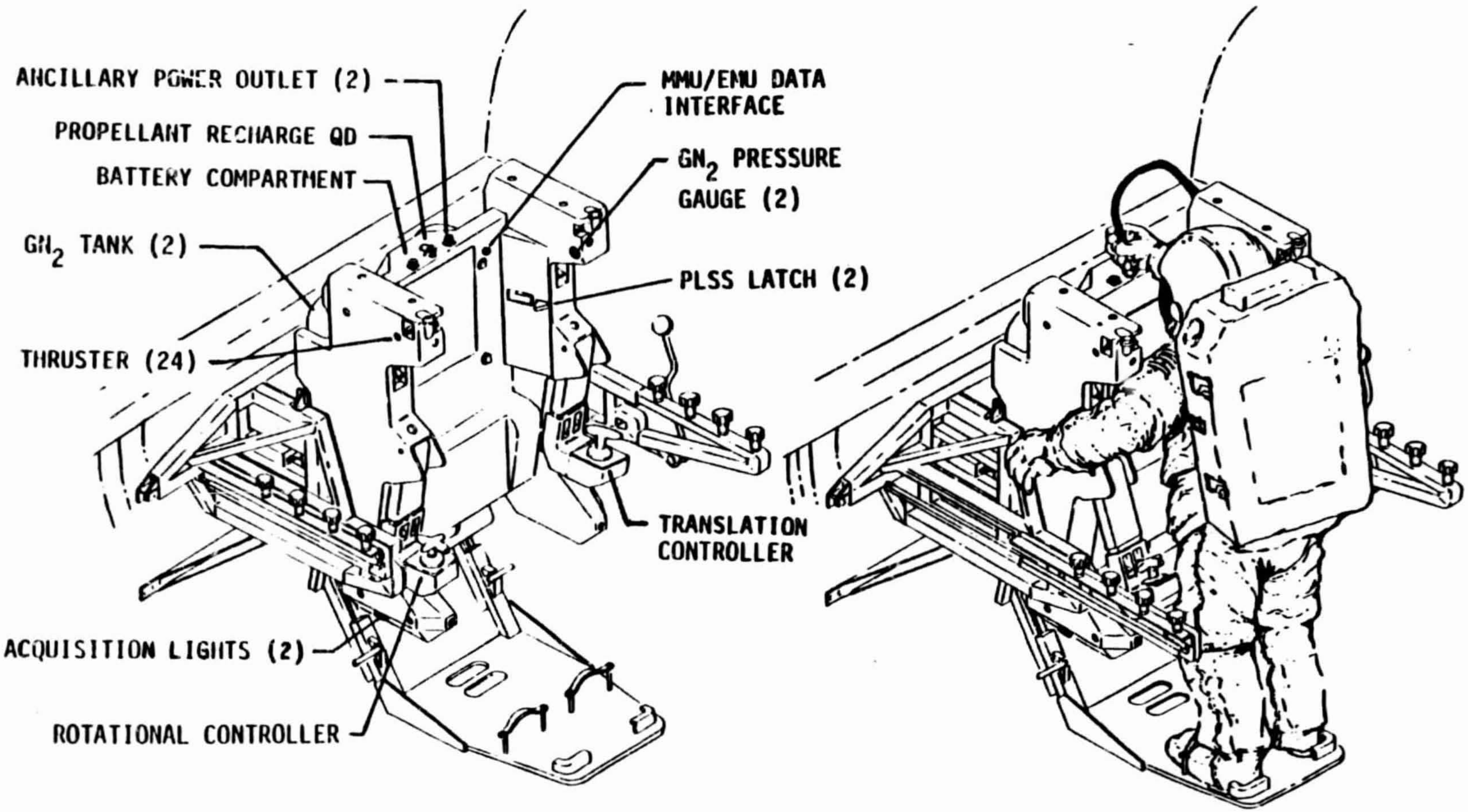


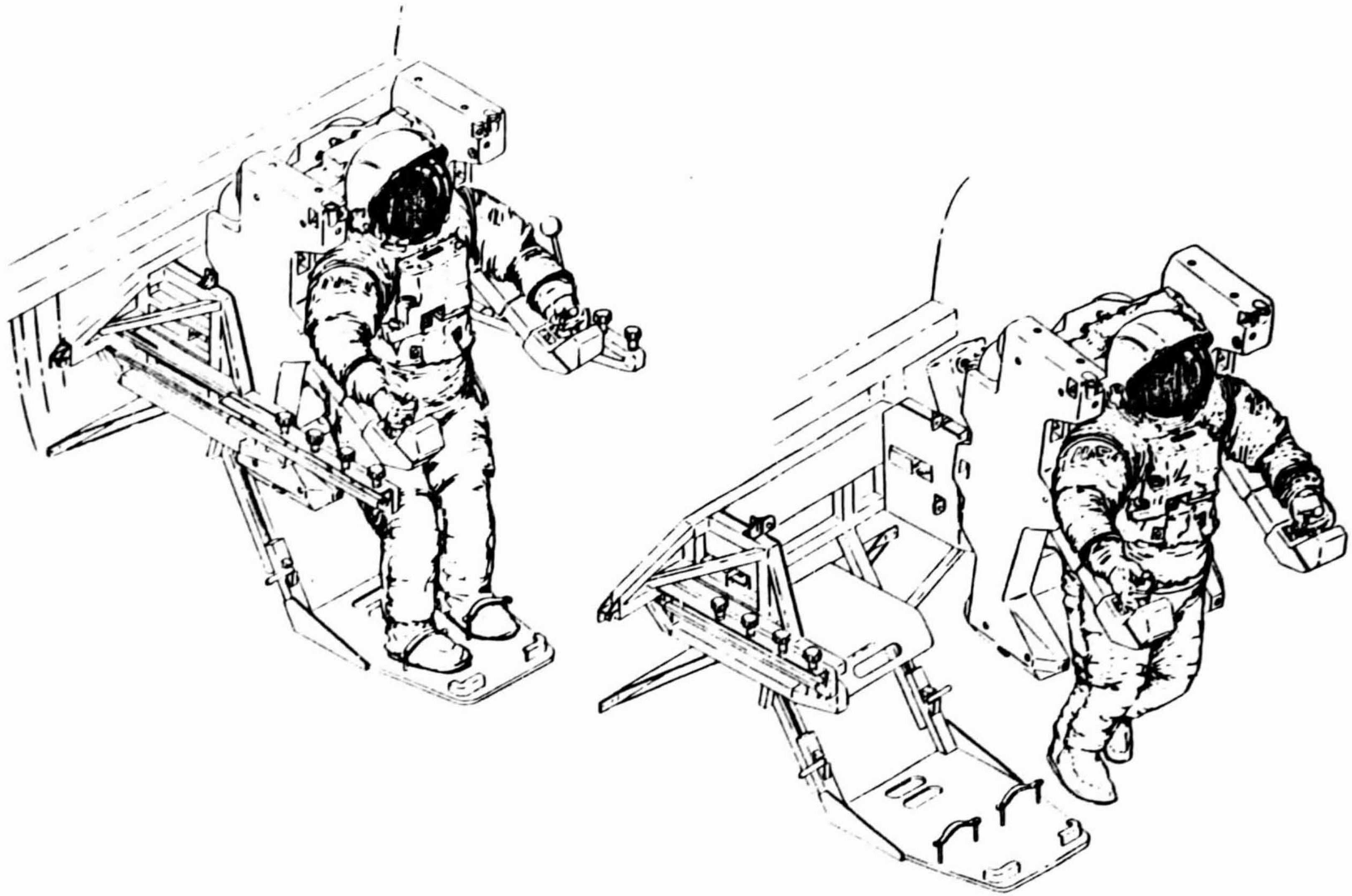
Figure 4.8-13. Crew Foot Restraint Support System in Position to Aid Access to Payload Door Latches



LAUNCH, ENTRY AND ON-ORBIT STOWAGE

SERVICING (PROPELLANT CHARGE)

Figure 4.8-14 MMU Stowed in FSS, and Showing Servicing Position of Crew



DONNING CONFIGURATION

EGRESS/INGRESS

Figure 4.8-15 Crew Donning MMU and Performing Ingress/Egress

Table 4.8-5. Checklist: MMU Operations Constraints and Capabilities

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES IMPLICATIONS
( ) The MMU may be donned, doffed and serviced by one EVA crewmember for use as required during the nominal 6-hour EVA period. (R-1)	NASA 1980 (a): Affects EVA crew work planning.
( ) The MMU has six degrees of freedom control authority and attitude hold capability. (R-1)	NASA 1980 (a): Permits significant versitility for space operations.
( ) The MMU may be used to effect crew transfer in the event of a rescue operation. (R-1)	NASA 1980 (a): Implies need for access during rescue contingencies.
( ) For payloads which are retrieved and berthed on-orbit or for payloads launched and operated on pallets, the Manned Maneuvering Unit provides planned or unscheduled EVA access to instruments on the ends of long booms or appendages extending outside the cargo bay, especially those beyond reach of the RMS. (R-1)	NASA 1980 (a): Significant advantage for large structures.
( ) The unit allows retrieval of small, free flying payloads, including those which may be sensitive to orbiter thruster perturbations and contamination. The dry, cold nitrogen propellant used by the MMU causes no adverse contamination. (R-1)	NASA 1980 (a):
( ) When the MMU and its flight support station are installed in the orbiter to support payloads, the weight and volume for these items will be allocated to appropriate payloads. (R-1) • See Figure 4.8-16 for access volume requirements (preliminary, may be varied to meet specific payload requirements).	NASA 1980 (a): Affects weight allocation and space for large space construction payloads.
( ) The MMU is battery powered. Battery recharge requires 16 hrs. Battery replacement requires 5 min. (R-1)	Lenda 1979 Affects timeline analysis.

Table 4.8-5 Checklist: MMU Operations Constraints and Capabilities (Continued)

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES IMPLICATIONS
( ) The Manned Maneuvering Unit provides two electrical outlets (2 amps at 28 vdc each) for use of ancillary equipment such as electric drills and other power tools, portable lights, cameras, and instrument sensors. (R-1)	<u>NASA 1980 (a):</u> Affects assembly, inspection and repair operations capability.
( ) The MMU can be used to carry cargo of a moderate size and weighing less than 220 lbs. (100 kilograms) such as might be required to service a nearby free-flyer. (R-1)	<u>NASA 1980 (a):</u> Significant capability for construction.
( ) A cargo/worksite attachment device is intended to perform two functions: (R-1) • Transport of large pieces of cargo. • Temporary attachment of crew member to worksite for specific tasks. • See Figure 4.8-17 for cargo/worksite attachment device concept.	<u>Lenda 1979</u> Affects planning for EVA attach to structure.
( ) The cargo worksite attachment device permits limited application of forces and torques because of a slip clutch incorporated into the attachment mechanism. • Force limits: 25 lbs. • Torque limits: 40 ft-lbs	<u>Lenda 1979</u> Affects design of attach points on structure.
( ) Crewmembers typically will translate at velocities of approximately 1% (in feet per second) of the initial separation distance to the destination (in feet). (R-4) • Example: 1 fps for 100 ft separation. Crew accelerates to above velocity, coasts, then decelerates. Since acceleration time is small, time for translation tends to be nearly constant at a little more than 100 seconds per 100 ft flight (103-112 seconds range).	<u>Lenda 1979</u> Affects timeline analysis of crew operations using MMU.

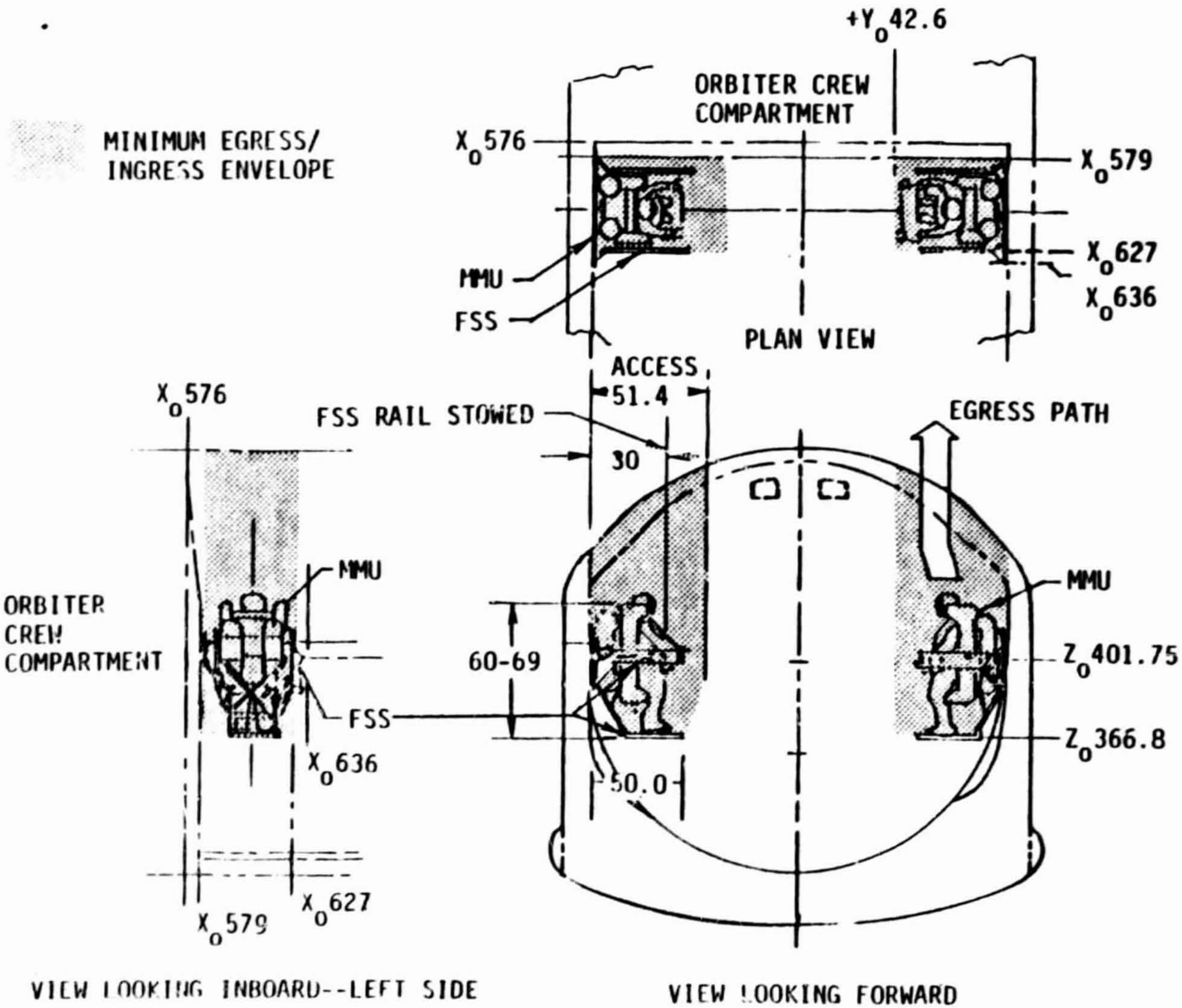


Figure 4.8-16. Manned Maneuvering Unit and Flight Support Station Cargo Bay Envelope

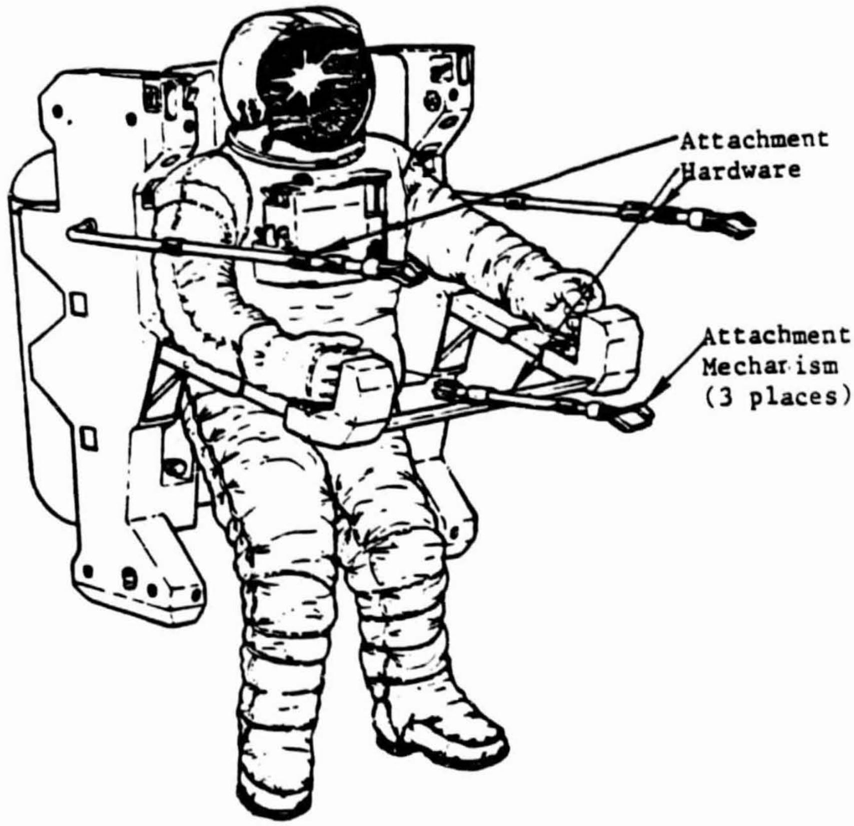


Figure 4.8-17. MMU Cargo/Worksite Attachment Device

Figure 4.8-18 illustrates the general configuration for the development test article of the OCP. The EVA crewmember using the OCP would be restrained at the feet in a standard type of foot restraint which may rotate around the body vertical axis. This crewmember would have control of RMS arm motions from the display and control panel on the OCP, and would have control of a stabilization arm (either remotely or by manual adjustment), which acts like a special end effector of the RMS. Additional provisions include a tool bin and lights (illustrated later in Section 4.9.1).

If developed and utilized for construction and operations involving large space structures, this device could have significant advantages and impacts on design. Such impacts include space for stowage in the orbiter cargo bay, EVA access, RMS access, software provisions, electrical interfaces, lighting provisions, power usage, RMS reach, timelines for preparation, stowage checkout and operations. However, the preliminary status of the development of the OCP at the time of this writing did not appear to warrant extensive discussion of potential details of such impacts on design of large space structures. For further information, the reader is referred to articles, study reports and briefings by Grumman personnel (Nathan 1978, Grumman 1978, Grumman 1979).

4.8.3 EVA Personnel

Commander - The orbiter commander will be in command of the overall space vehicle operations and responsible for the overall orbiter operations, personnel, vehicle safety and will monitor and support EVA as required.

Pilot - The orbiter pilot will be second in command and equivalent to the commander in proficiency. The pilot will be the second crewmember for EVA as required.

Mission Specialist - The orbiter mission specialist will be proficient in payload and experiment operations with knowledge of orbiter and payload systems. The mission specialist is the prime crewmember for EVA operations.

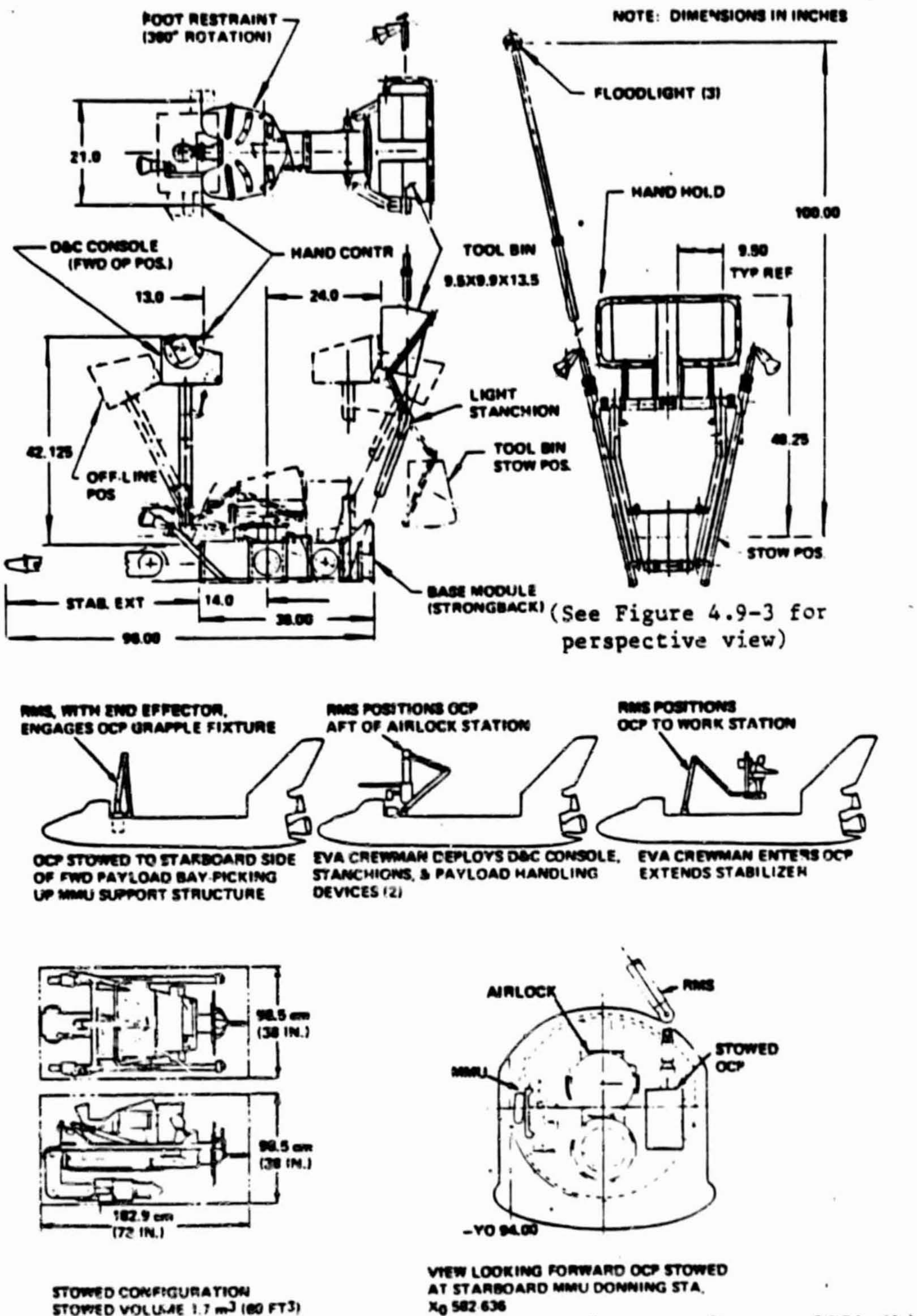
Payload Specialist(s) - Payload specialist(s) will be proficient in the payload and experiment operations, be responsible for the attainment of their objectives and will monitor and support EVA as required.

Construction of large structures in space may be enhanced by utilizing all crewmembers in EVA modes, with multi-shift daily operations. Such operations will require EVA training for all crew members.

4.8.4 EVA Timelines

The EVA period of the mission (Table 4.8-6) is generally considered as comprised of three phases, EVA preparation, EVA operations, and post-EVA activities.

The IVA preparation performed by the EVA crew consists primarily of preparing the EVA equipment and donning the EMU.



(See Figure 4.9-3 for perspective view)

Source: Grumman 1979 (b)

Figure 4.8-18. Open Cherry Picker Deployed and Stowed Configurations

Table 4.8-6. EVA Timeline

START PREBREATHE 3.0 HRS.	2.0 HRS. PERFORM OTHER TASKS	DON PREBREATHE MASKS/START CONTINUE MISSION TASKS WHILE PREBREATHING
FINISH PREBREATHE	EVA PREP 1.5 HRS	UNSTOW/PREPARE EQUIP FOR EVA ENTER AIRLOCK UNSTOW/PREP EMU FOR DONNING DON EMU REMOVE PREBREATHE MASKS DON HELMETS/VISORS DON GLOVES PERFORM O_2 SUIT PURGE INITIATE EMU SYSTEMS/CHECKOUT PERFORM COMM CHECK PERFORM EMU INTEGRITY CHECK DEPRESS AIRLOCK OPEN HATCH, START SUBLIMATOR
	EVA OPERATIONS 6.0 HRS	EGRESS AIRLOCK DON/CHECKOUT MMU (IF REQ'D) PERFORM EVA OPERATIONS DOFF MMU/RECHARGE (IF REQ'D) DEACTIVATE SUBLIMATOR INGRESS AIRLOCK
	POST EVA 1.5 HRS.	CLOSE HATCH REPRESS AIRLOCK REMOVE HELMET/GLOVES DOFF EMU RECHARGE EMU STOW EQUIPMENT INITIATE EMU DRYING

The EVA operations are the actual performance of the required tasks outside the pressurized orbiter cabin.

The post-EVA activities include equipment doffing and stowage as well as life support system recharge for subsequent EVAs.

Since the normal Shuttle EVA crew consists of the mission specialist and pilot, the payload specialist will not normally be involved in EVA preparation and post activities and can perform payload/experiment operations or support EVA related activities, if required, during these periods. However, special situations such as involved in space construction may dictate other duty assignments (see Section 4.8.3).

4.8.4.1 EVA Preparation

Approximately 1 to 2 hours will be required by 1 or 2 EVA crewmembers to complete the preparation activities for an EVA. The activities are preceded by the start of the required three hours of prebreathing, which occurs 3 hours 30 minutes prior to the scheduled start of an EVA. For nearly 2 hours of the 3-hour prebreathe, the EVA crewmembers may perform required EVA or non-EVA related activities.

The EVA crewmember begin the EVA preparation period by unstowing associated EVA equipment. Special EVA equipment, such as cameras, will then be prepared and checked for the EVA and placed in the airlock. The EMU will be unstowed and the crewmembers will don the suits and life support equipment. Ancillary suit equipment, such as the waste management system and the liquid cooling garment, are donned prior to donning the suit.

Approximately 30 minutes prior to the start of the EVA or at the completion of the 3 hour prebreathe period, the crewmembers will doff the prebreathe masks and don the helmet and gloves. Following a suit O_2 purge, the crewmen perform a check of the life support equipment and orbiter/EVA communication systems. Backup communication modes are also checked. The crewmen perform an integrity check of the EMU and begin depressurization of the airlock. After depress, the outer airlock hatch is opened for the start of the EVA.

4.8.4.2 EVA Operations

The EVA tasks are performed outside the orbiter cabin for a maximum duration of 6 hours which is the normal limit of the life support system consumables. EVA task timelines should provide for frequent rest periods when extensive physical activity is involved (typically 10 minutes per hour).

Post-EVA - When EVA operations are completed, the crewmen ingress into the airlock, close the outer hatch, and repressurize the airlock. Helmets and gloves are doffed along with the EMU. The crewmen then perform a recharge of the life support system. Consumables are replenished and the life support equipment is prepared for the next EVA. Loose equipment and donning aids are stowed and suit drying initiated, if required.

4.9 ILLUMINATION AND TELEVISION SUPPORT SERVICES

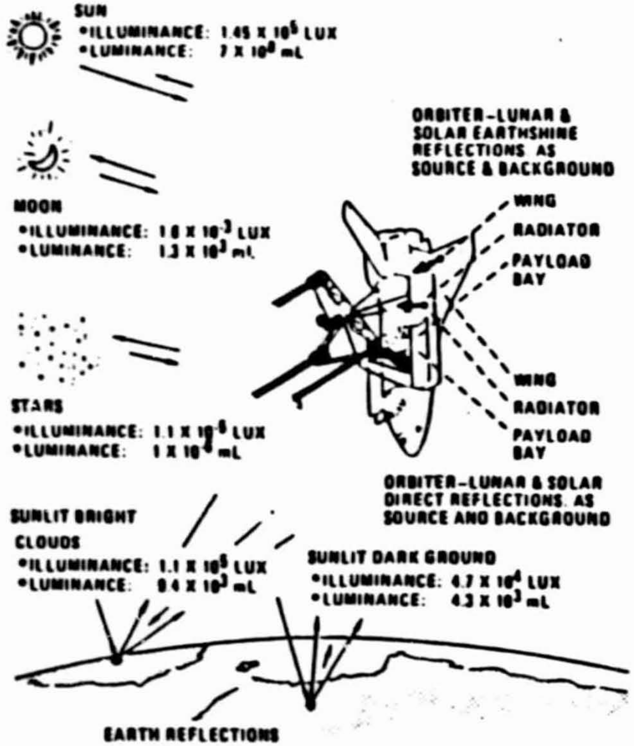
This section discusses Shuttle capabilities/constraints and guidelines dealing with all aspects of crew viewing, including use of windows, TV cameras, orbiter and other Shuttle system lamps, lighting power supplied by orbiter, and orbiter surface finishes and configurations.

4.9.1 Discussion - General Issues in Illumination and TV Services for Space Construction

Space construction, maintenance and systems module changeout considerations for large space structures are strongly dominated by transport, joining, aligning and inspection functions, which require means to accurately sense orientations and alignments, positions (especially critical clearances), relative velocity and condition of deployment. Experience has shown that the use of direct or aided human vision (telescopes, TV, etc.) to perform a majority of these critical sensing operations is generally cost effective and highly reliable when adequate illumination is provided. However, preliminary studies have identified several potentially significant problems associated with assuring adequate illumination in space, whether the light is artificially provided or naturally received. In particular, low-earth orbits (LEO) involve consideration of frequent changes from a very dark environment to a brightly sunlit environment (but with deep shadows in many cases), which in turn create a concern about work interruption due to necessities of adaptation by both man and machine (TV camera). Figure 4.9-1 provides pertinent data on the wide range of illumination fluxes and brightness conditions in LEO.

Construction/maintenance using the Space Shuttle Orbiter typically involves a low earth orbit situation with stark contrasts of deep shadow and bright sunlight in space which require special lighting provisions for vision to be adequate. However, there are differences between the various candidate provisions which affect the required intensity for lighting, the location of the lamps and the resulting power, weight, volume and cost of the lighting and vision aspects. Design of construction and servicing equipment lighting for space structures is strongly affected by the orbiter constraints on available power, type of equipment, and standard locations for lamps and TV cameras. Also, the orbiter configuration and surface finishes influence the visual background and reflections of available light.

Table 4.9-1 outlines several important considerations of the subject. With this context in mind, the following discussion (Sections 4.9.2, 4.9.3, and 4.9.4) describe specific constraints and guidelines of the Shuttle Orbiter equipment which is involved in lighting and viewing. This material is followed by Sections 4.9.5 and 4.9.6, which present more general discussions about potential interactions with orbiting large space structures. Section 4.9.5 considers the dark (eclipse) side of low earth orbit, in which artificial lighting considerations dominate. Section 4.9.6 discusses the sunlit side of orbit, primarily involving direct solar illumination and multiple reflections. The discussions include results from selected example analyses which support general guidelines and recommendations for space construction.



Characteristics Affecting Space Construction

- Wide Range of Brightness - Stars to Full Sun: 10^{10}:1
- Lighting Conditions Continually Changing Angle and from Light to Dark.
- Eye Does Not Fully Dark Adapt During Eclipse Phase of LEO
- Shadows Generally Darker Than in Earth Atmosphere, Structure Outlines May Be Obscured
- Orbiter is Helpful as a Diffuse Light Reflector
- Bright Clouds Reflection and Direct Sun Are Similar in Illumination Intensity But Clouds More Diffuse
- Solar Orientation is Critical - Avoid Looking at it or Avoid Cargo Bays Being Oriented Toward Sun or Both

Figure 4.9-1. Natural Illumination Environment-LEO

Table 4.9-1. Summary of Viewing and Illumination Considerations for Space Construction

Considerations	Applicable Section
o Relatively rapid cycling from bright sunlight to darkness at low-earth orbit.	4.9.6
• Power and lamp cycling questions (life, surge, thermal)	4.9.2
• Dark adaptation considerations of crew eye protection	4.9.5
• TV range of sensitivity (brightness ratios)	4.9.3
o Power requirements for darkside illumination	4.9.5
o Overall lighting versus local lighting	
o Reflective, light colored surfaces	4.9.2
o Portable versus fixed lamps	
o Continuous lighting versus cycled lighting	
o Optimum viewing angles for sunlit side viewing	4.9.6
o Glare avoidance - direct, veiling, contrast	
o Backlighting problems of earth and sun	
o Diffusivity of surfaces versus glare	
o Vision requirements versus thermal requirements	
o Orientation of spacecraft and crew for self-protection from disturbing light input	
o Possible use of shades and diffusers	
o Vision angles required by configuration	4.9.4
o View from crew compartment windows	
o TV camera viewing angles and positions	4.9.3
o EVA viewing positions and angles	4.9.6
• Work interruptions versus productivity	4.9.5
• Dark versus day	4.9.5
• Viewing cutoff by glare, intensity of light, angle, shadows	4.9.6
• Interaction with thermal consideration	4.9.6, 4.13
• Interaction with communications considerations	4.11
o Interaction with stability/control considerations	4.2
o Interaction with orbital plane requirements, launch time, season and relative rate of change of β angle	4.9.6
o Hardware selection and power requirements	4.9.2
o Incandescent lamps	
o Metal Halide	
o Beam versus Flood	
o Reflectors	
o Finishes	
o Portable versus fixed	
• Solar power versus batteries	
o Orbiter fuel cells versus outside sources	
• Assembly sequence versus power requirements	4.4.3
• Setup power module/solar array before construction	4.4.3
• Setup solar array at end of construction	4.4.3
• Schedule operations requiring significant illumination to be performed on sunlit side of orbit.	4.9.5

4.9.2 Shuttle Lighting Systems and Surface Reflective Characteristics

Standard Shuttle provisions for lighting include the several types listed in Table 4.9-2. Locations for the exterior lights (in the orbiter cargo bay and above the cabin) are shown in Figure 4.9-2. Other potential exterior sources of light are the open cherry picker and MMU.

In addition to light sources, visibility of portions of large space structures in the vicinity of the orbiter may be significantly affected by the reflective qualities of the orbiter surfaces. Such surfaces can act as backgrounds for viewing by TV cameras or by EVA crew or crew in the cabin. Also, these surfaces can reflect earth light, sunlight, starlight or moon light. According to their contours and reflective characteristics, they may beneficially diffuse the light or concentrate it in undesirable ways. Such concerns are of particular interest to designers of large space structures because of the typical geometry involved. The structure is typically located in proximity to the orbiter cargo bay, close enough for reach access by the RMS.

Table 4.9-3 lists typical reflective characteristics of orbiter surfaces and selective typical payload surfaces which are likely to be of concern. Further implications of combined effects of reflection are discussed in Section 4.9.6.

4.9.3 Closed Circuit Television (CCTV) System

The Space Shuttle Orbiter has provisions for installation of black and white closed circuit television cameras in the cargo bay and on the remote manipulator system arm. Locations are shown in Figures 4.9-4 and 4.9-5 (also see Section 3.1.2). It is expected that these systems will be fully utilized in most missions involving large space structures. In many cases, additional TV cameras (and lights) will be installed on construction fixtures and assembly machinery for improving visibility of critical large space operations.

There are provisions for two (a) black and white TV receivers, (viewing screens) within the crew cabin, on the port side of the aft flight deck (see Figure 4.8-10). Additional viewers may be incorporated into the payload-dedicated display and control panels if the space is available and the cooling capacity adequate. The latter may be black and white or color systems. Table 4.9-4 summarizes constraints for CCTV systems.

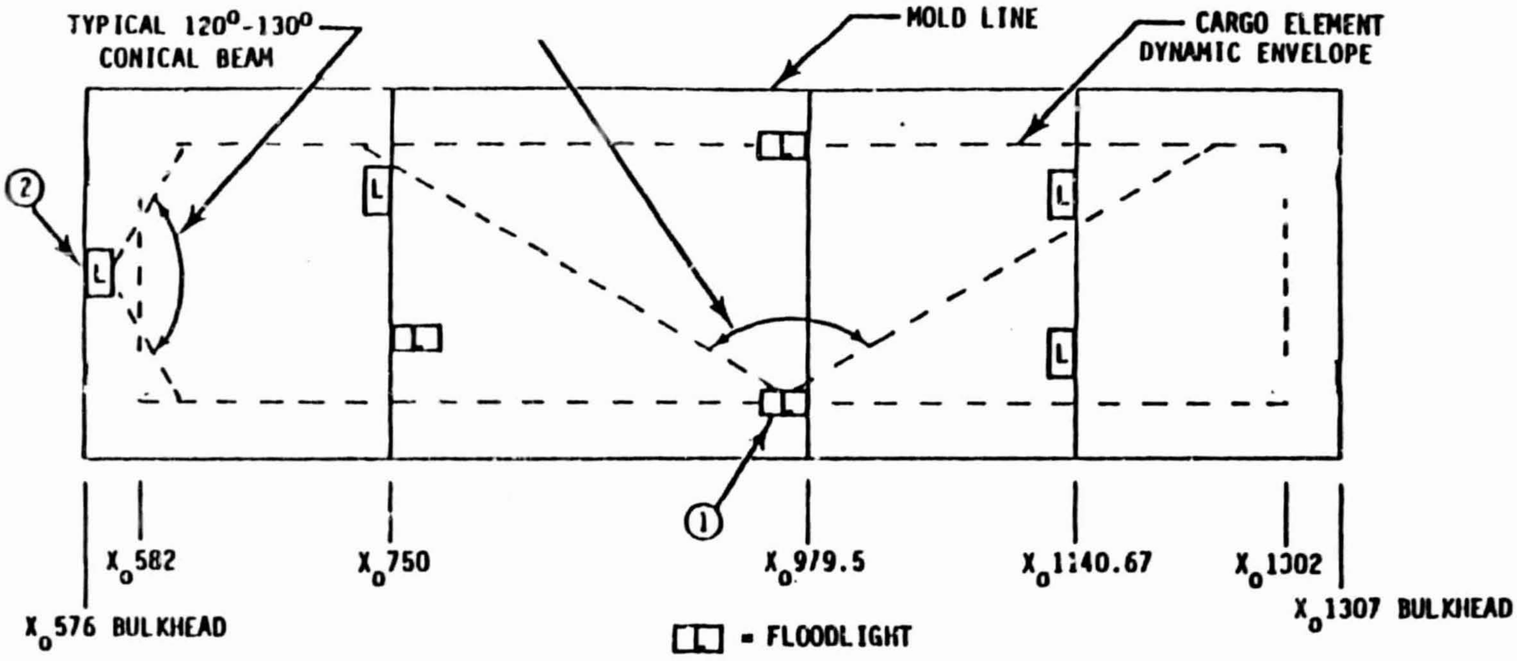
4.9.4 Window Viewing Angles and Transmittances

The following discussion concerns the general constraints on crew viewing through transparencies in the orbiter crew cabin and in the EVA suit helmet.

Figure 4.9-6 describes the nominal constraints on viewing angles for orbiter crew cabin windows, at specified eye positions. These angles are for a so-called "cyclops" single point eye. Actual crew viewing will be different, (usually better) because of the lateral spacing of eyes and the

Table 4.9-2. Orbiter-Provided Lighting for Space Construction

Item	DESCRIPTION					References/Remarks
	Qty	Watts (Each)	Lumens/Watts	Type	Beam	
Cargo Bay Floodlights in Side Walls (R-1) • See Figure 4.9-2	6	200	40 minimum	ARC Discharge (Metal halide)	135° cone or square	NASA 1980 (a): ICD 2-19001/Standard items; may require shrouds to avoid direct view by T.V. cameras
Docking Floodlight on 576 bulkhead, facing aft (R-1) • See Figure 4.9-2	1	200	40 minimum	Arc Discharge (Metal halide)	120° cone	(See Above)
Rendezvous/Docking Light, facing upward (R-1) • See Figure 4.9-2	1	130	12 minimum	Incand.	120° cone	NASA 1980 (a)
RMS Wrist Light (R-1) • See Section 4.5	1 (Per Arm)	150	12 minimum	Incand.	80°	NASA 1980 (a)
EMU Mounted/ Portable Light e (R-2) (R-2)	1 (TBD)		(TBD)	Battery	(TBD)	In development
Manned Remote Work Station Floodlights (R-3) See Figure 4.8-19	3	60 (TBD)	(TBD	Incand.		Grumman, 1979 In development



NOTES:

1. SIX LIGHTS MOUNTED OUTSIDE CARGO ELEMENT DYNAMIC ENVELOPE. (120° MINIMUM CONICAL BEAM POINTED WITHIN APPROX 5° OF NORMAL TO CARGO BAY CENTERLINE)
2. FORWARD BULKHEAD FLOODLIGHT - 120° CONICAL BEAM POINTED IN +X DIRECTION PENETRATES 0.14x5.70 DIA INTO THE CARGO ELEMENT DYNAMIC ENVELOPE)
3. LIGHTING CHARACTERISTICS ARE SPECIFIED IN PARAGRAPH 10.9.1, NASA 1980(A), ICD 2-19001
4. FIGURE NOT TO SCALE

Figure 4.9-2A. Orbiter Baseline External Lighting Locations

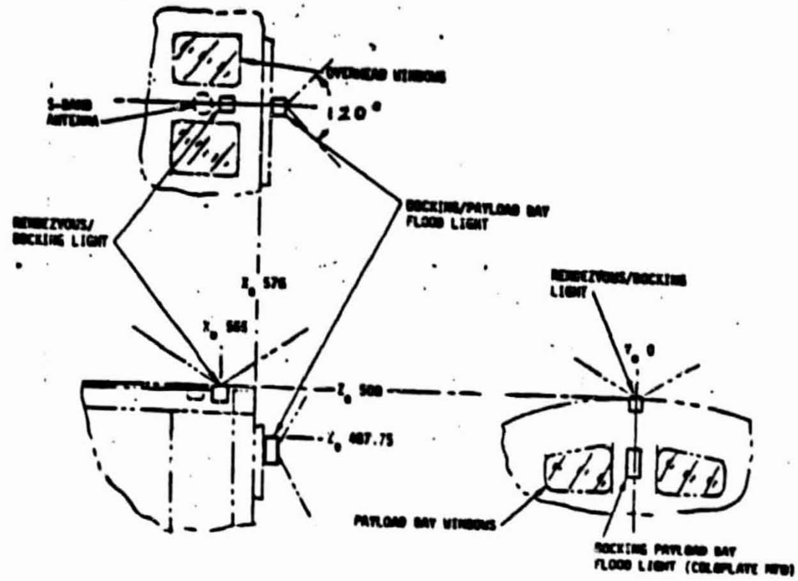


Figure 4.9-2B. Orbiter Baseline External Lighting

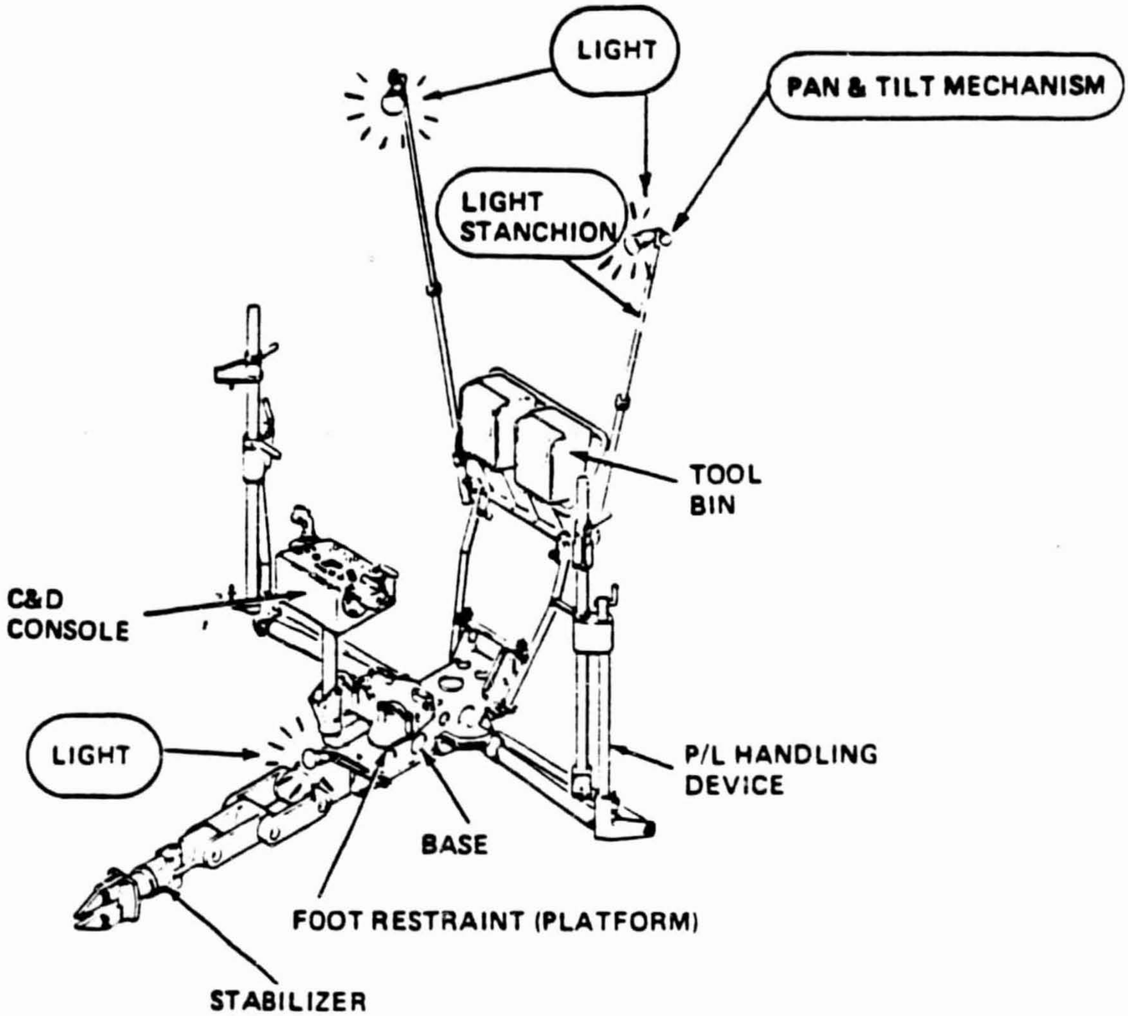
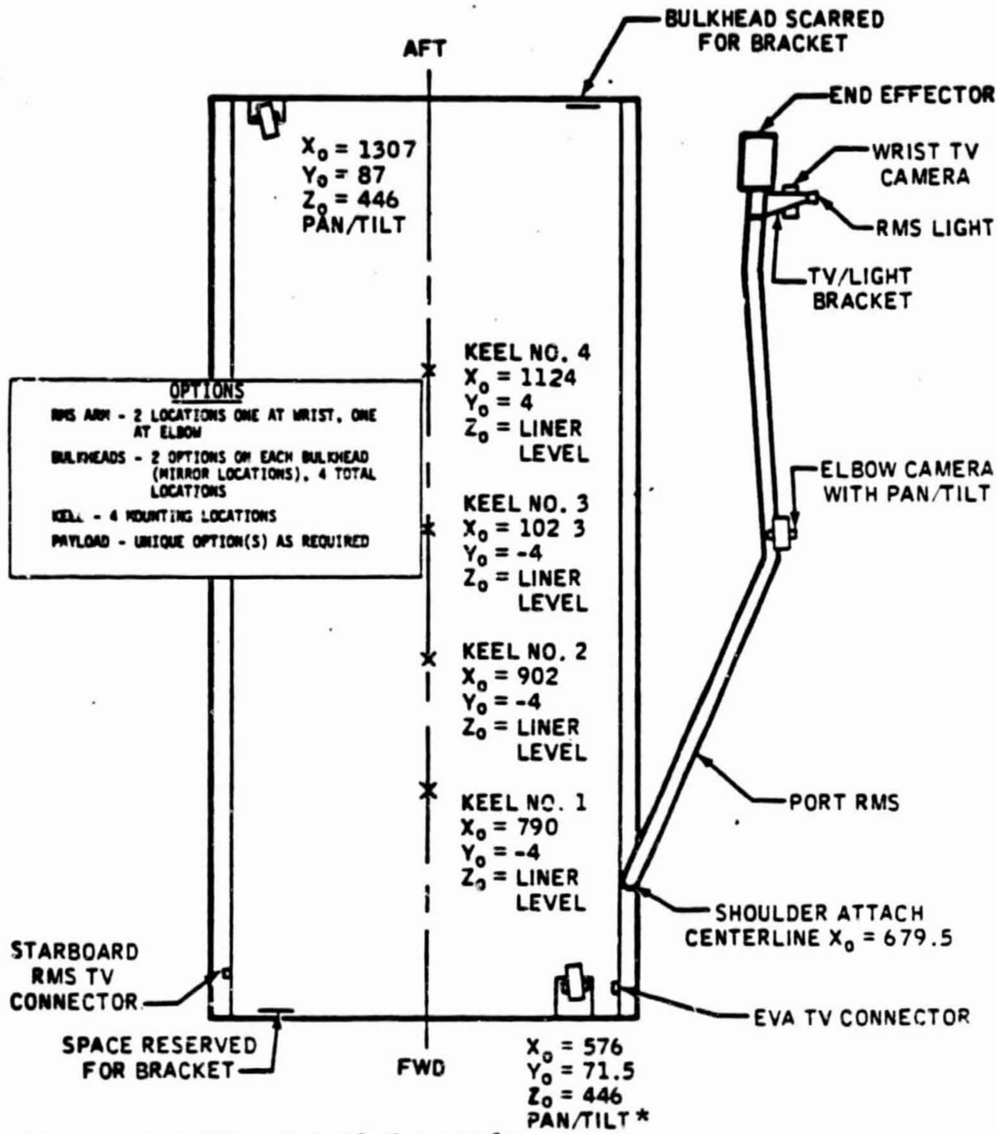


Figure 4.9-3. Lights on Open Cherry Picker Development Test Article

Table 4.9-3. Typical Reflection Characteristics for Orbiter Spacecraft and Construction Equipment

REFLECTIVE SURFACE	REFLECTIVITY
CARGO BAY LINER, TEFLON-IMPREGNATED GLASS FABRIC	85-90% (LOW SPECULARITY)
ORBITER WINGS (UPPER SURFACE) AND BODY	68% (LOW SPECULARITY)
ORBITER RADIATORS	90% (96-100% SPECULAR)
PAYLOADS—PAINTED S13G WHITE	85%
SILVER COATED TEFLON	95%
STRUCTURAL MATERIALS	35-85% (LOW SPECULARITY DESIRABLE)
SOLAR ARRAY (CELLS)	2-40%



*See Figures 3.1-12 & 3.1-13 for angles

Figure 4.9-4. CCTV Camera Mounting Options

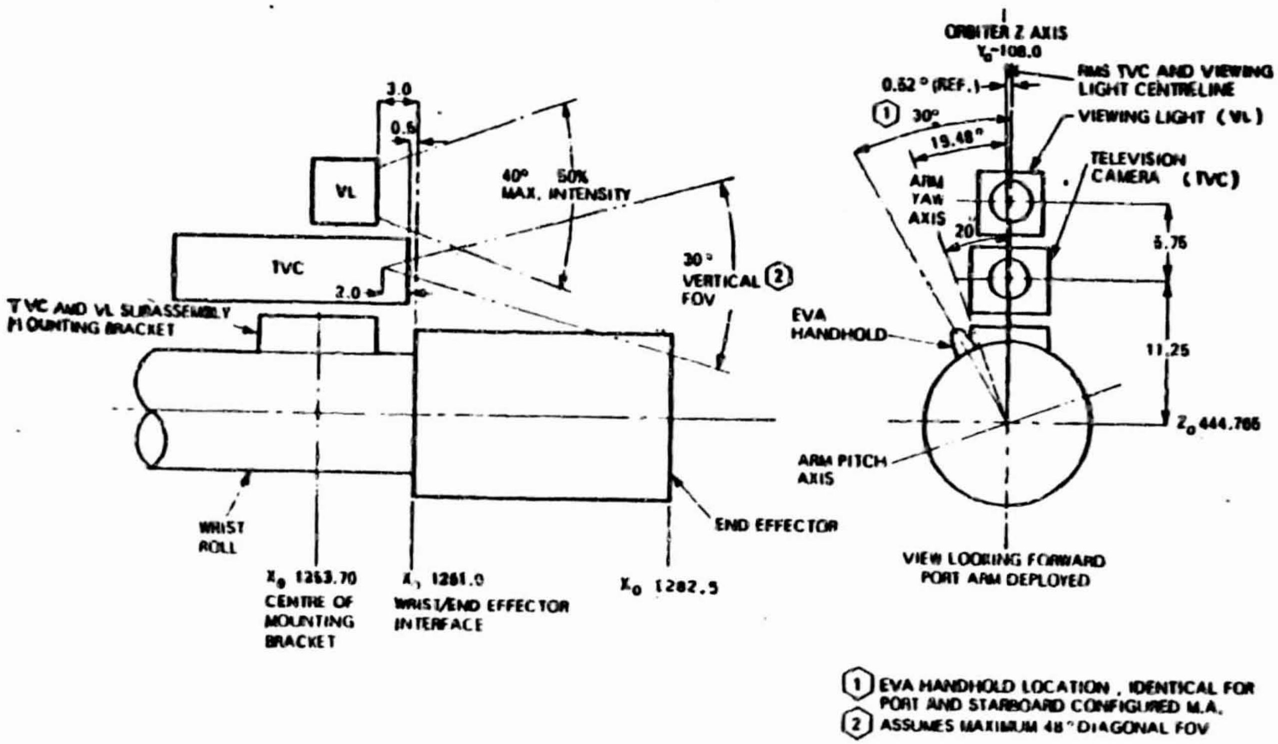


Figure 4.9-5. RMS CCTV Wrist Camera and Light Subassembly Design Configuration

Table 4.9-4. Checklist: Constraints on TV Camera Operations and Locations in Orbiter Payload Bay and Construction Fixtures

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) The orbiter can accommodate up to eight dedicated-cable TV cameras in the cargo bay and payload. (R-1) • Five dedicated cables for cargo bay installation. • Three dedicated cables for payload-mounted cameras.	Personal Communication Affects selection of TV camera locations, costs and schedules.
( ) Additional TV cameras may be utilized by installation of remote switching. (R-1) • Forward bulkhead penetrations dictate number of cables into crew cabin. • Noise and line losses are only practical limits to number of TV cameras.	Personal Communication Affects design of electrical circuits for TV control, costs and schedules.
( ) All CCTV cameras will have zoom and iris control. The iris will automatically close when pointed at the sun, unless manually controlled to open. Normal condition for the iris at start of operation is also the closed position. (R-1)	NASA 1980 (a): JSC-07700, Vol. XIV (Change 31) Affect crew procedures planning.
( ) The forward and aft bulkhead cameras have pan and tilt control with a range of 340°* (±170° from perpendicular to bulkhead), with pan and tilt angles displayed on the CCTV monitors. (R-1)	*Personal Communication Orbiter Engineering Affects viewing analyses.
( ) The TV cameras are capable of accommodating a range of lenses for special payload applications (R-1) • Lenses are removed/replaced prior to flight.	NASA 1980 (a): JSC-07700, Vol. XIV (Change No. 31) Affect viewing analyses.
( ) The field of view for the standard lenses varies from approximately 48° diagonal to approximately 8.5° diagonal, when focused at infinity. (R-1) • Range is 39.2° horizontal and 29.9° vertical to 6.8° horizontal and 5.6° vertical.	Personal Communication Affects viewing analysis, camera location.

Table 4.9-4. Checklist: Constraints on TV Camera Operations and Locations in Orbiter Payload Bay and Construction Fixtures

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) The TV camera is designed for visibility with light levels of 3 foot-candles at 30 feet. (R-1)	Personal Communication Affects lighting design.
( ) The starboard forward camera is nominally dedicated to public relations purposes. It is fitted with a color wheel for downlink color transmission to ground. (R-1) • The crew TV monitor will produce undesirable flickering in using the color mode.	Personal Communication All orbiter crew monitors see black and white.
( ) In general, work should be arranged such that TV cameras do not look directly at a bright light source such as the sun or a lamp in the cargo bay. (R-4) • Adjacent areas may be difficult to see as the iris automatically closes down. • Temporary or fixed shields on lamps may be utilized in some cases.	Rockwell, 1979 (a): SSD 79-0123
( ) The standard RMS payload connection electrical cables will not accept CCTV signals. If it is desired to mount a TV camera on a payload module to aid berthing to another module, a special cable would have to be attached to the RMS arm to provide a hardline signal path from the payload module to the crew cabin. (R-1)	Personal Communication - Orbiter Engineering Affects trades of RF versus hardline signals for payload module mounted TV cameras.

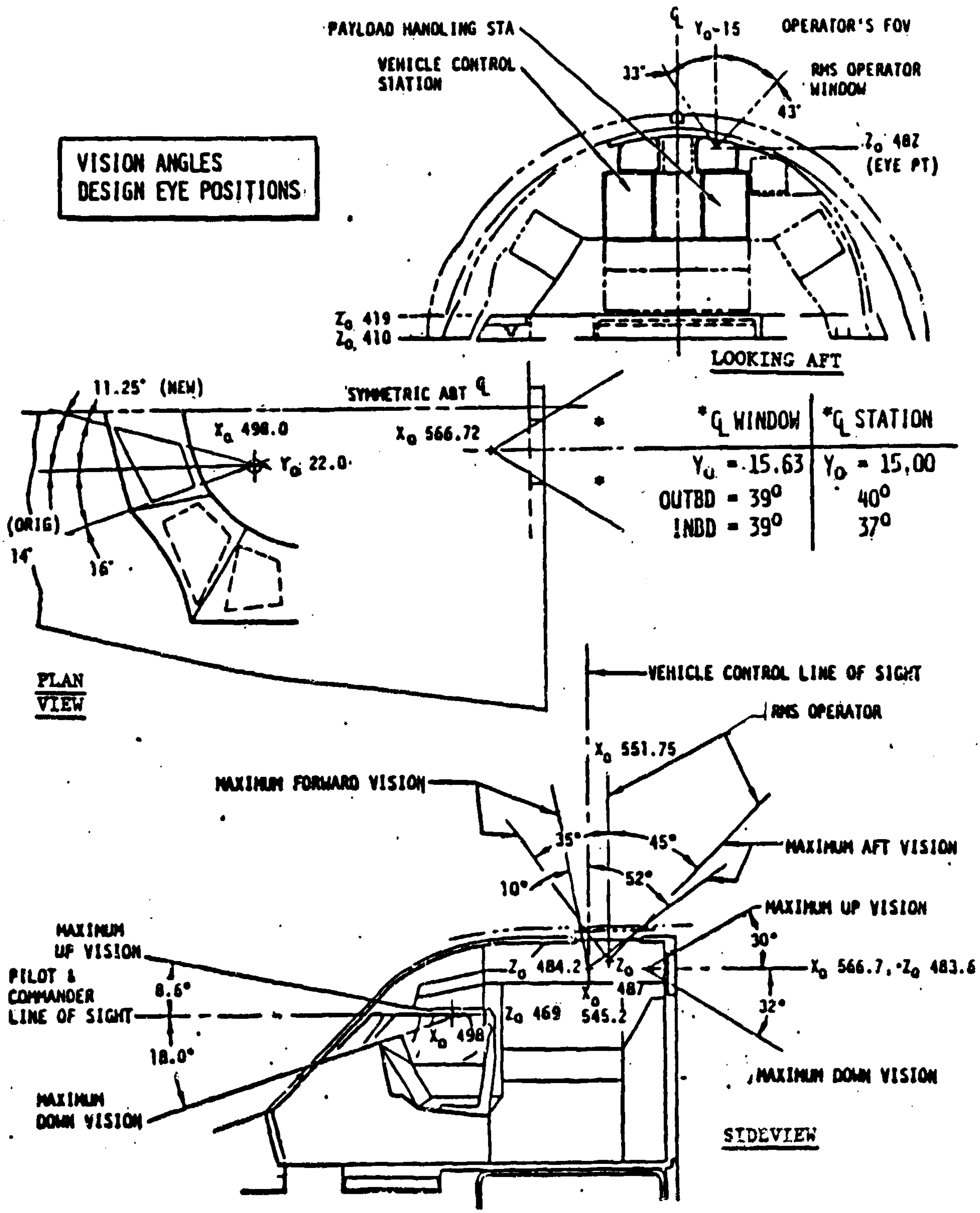


Figure 4.9-6. Orbiter Cabin Window Viewing Angles

capability of the crew to move their heads to achieve different vantage points. Data are not available concerning the full extent of such improvements in viewing.

Table 4.9-5 lists the transmittance capabilities of various transparencies in the orbiter and EVA suit. Note that orbiter windows are provided with removable filters which can be used to control sun glare.

Table 4.9-5. Typical Transparency and Filter Combinations Transmissibilities and Resulting Apparent Illuminances of Sunlit Surfaces

No.	Condition	Transparency/Filter Transmissibility Item	%	Overall Effect-Transmissability	Surface Illuminance Lux	Ft-Candle
0	Sun-Direct	Space	100%	100%	1.46×10^5	13,500
1	Aft Flight Deck Aft Window View 2 Glass Panes	Outer Pane, Inner Pane		88%	1.28×10^5	11,880
2	2 Glass Panes plus filter	Filter, 2 Glass Panes	18%, 88%	15.8%	2.3×10^4	2,133
3	Aft Flight Deck Overhead View 3 Glass Panes	Outer Pane, Middle Pane, Inner Pane		81%	1.18×10^5	10,935
4	3 Glass Panes plus filter	Filter, 3 Glass Panes	18%, 81%	14.6%	2.06×10^4	1,968
5	EVA Helmet Low Density Filter	Lexan Shell, Lo-D Filter	80%, 60%	7.48%	7.00×10^4	6,480
6	High Density Filter Low Density Filter EVA Helmet	Hi-D Filter, Lo-D Filter, Lexan Shell	16+4%, 48%	5.8 to 9.6%	8.46×10^3 to 1.40×10^4	783 to 1,296

4.9.5 Power Requirements for Artificial Illumination and TV on Dark Side of Orbit

Previous studies of space construction have determined that the power requirements for lighting on the dark side of LEO may constitute a significant portion of the total energy requirement for space construction. For example, a study by Grumman (1977) indicated that lighting could require very large amounts of power, from 35 kW to 83 kW, depending on the size of the project. In order to obtain such power, the large solar arrays and stowage batteries would have to be carried up, deployed, and connected to the construction fixture lighting system and other necessary lamps early in the construction sequence. Such solar arrays would need to be oriented so as to face the sun during the sunlit period. Large areas of low mass, such as solar arrays, can cause significant drag on the orbiting construction project at low earth orbit altitudes. This combination of effects is generally undesirable as regards fuel usage for stationkeeping and for pointing on orbit. Also, loads on weakly supported structure during the construction processes can be a significant problem. Some projects require designs where the solar array blankets cannot be deployed until the structure is at least partially built. In such a situation, the lighting power must be limited to that available from the orbiter or from other auxiliary power sources until some of the space construction project blankets are installed and connected to a service power system. As a result of such considerations, it is recommended that lighting systems for space construction should be frugal in power demand.

A question thus arises, "Is it feasible to significantly reduce power demands for illumination?" Examination of the basis for the Grumman study indicates that it is, because the study was based on a relatively generous, general and continuous flood-lighting approach, such as one might use for factory or shipyard construction site illumination. In contrast, a study by Rockwell International (1979 b) took a more austere and limited approach. The concept involved the use of carefully controlled, localized lighting for fine, close work and the use of reflectors, small "running lights", and various portable lamp devices as methods to control and minimize lighting power. The study involved smaller construction projects (two versions of an electronic mail satellite) and a larger number of options in differing types of construction facility and methods (manual versus remote, automatic). Even these austere lighting concepts required power levels from 2.0 kW to 4.8 kW, which represents from 38 to 52 percent of the total system construction power requirements. Such levels of power, when combined with power from normal operations, RMS, etc., seriously tax the Shuttle Orbiter capability and indicate a potential advantage in searching for even more efficient low-power lighting concepts.

Concepts for truly minimal illumination power do exist. A rather extreme, but promising approach is to use electronic amplification of available illumination (natural and artificial), such as employed by low-light level TV cameras and by military night vision goggles and scopes. Carried to their logical extremes, such systems could drastically reduce, perhaps even eliminate the need for artificial lighting altogether. Clearly, some additional development would be required to use such goggles and scopes in space applications. The approach seems promising, since the major electronics feasibility has been proven. However, such developments are not now planned as part of the Shuttle hardware development, and further discussion of such devices is beyond the scope of this document.

The use of light amplification devices during the eclipse phase of orbit brings to the fore a secondary concern, that of orientation considerations for space construction. (In sunlight, orientation is a primary concern, to be discussed later). The naturally available starlight is comfortably diffuse and wide spread, covering over half of the spherical angles surrounding the orbiter/construction project. Moonlight, however, is highly directional and relatively bright compared to starlight, especially at full moon. Shadows in moonlight could become a significant consideration. However, the use of artificial lighting at relatively low levels could practically eliminate any orientation problems for dark side operations. This is in contrast with the sunlit operations, where competing with sunlight is impractical for artificial lighting.

Work Cessation on Dark Side of Orbit

In view of the foregoing complexities, it is appropriate to raise the question, "Why do any construction work on the eclipse side?" If no work is done, no lighting power is required. Closely related to this question is scheduling of unproductive (but necessary) rest periods which can be scheduled for this time, thus mitigating the effects of work cessation or at least minimizing lighting power duration.

Figure 4.9-7 depicts eclipse durations at 463 km altitude versus β angle (angle between orbit plane and sunline). Eclipse durations are about 28 to 36 minutes at 463 km (250 nmi) and a 28.5 degree (52 degree max β angle) inclined orbit. Thus, a ten-minute rest period represents about 36 to 28 percent of the period of darkness. However, the remaining amount (18 to 26 minutes) represents about 19 to 28 percent of the total orbit period. Such percentages are not negligible. EVA operations are especially important as regards maximum use of the available work time, since only about 5 hours out of a 10-hour work day are really useful. The suit donning and doffing time absorb the balance. For efficient use of crew members in the cabin or in EVA, rest periods should be no longer than actually necessary in order to get the maximum benefit of the highly limited work time. On the other hand, a 10-minute rest period could be creatively used for dark adaptation of crew member's eyes in preparation for more efficient vision. Such an adaptation period is generally compatible with effective use of night vision goggles. Alternately, better vision could be obtained for effective use of minimal lighting with the unaided eye. In addition, carefully selected dark goggles can be used on the sunlit side of orbit to reduce the amount of dark adaptation required for eclipse side operations.

TV, Lighting and Intra-Project Geometry Considerations

Certain precautions are necessary in using television cameras for aiding space construction activity on the eclipse side of orbit. A major concern is that the light level not be too bright in the observed scene, whether seen as a reflection or as a source. This leads to the guideline that lights should be arranged to provide indirect or oblique lighting on objects seen by TV cameras.

Table 4.9-6 provides a summary of guidelines for use of lighting, special finishes and procedures on the eclipse side of orbit, based on the recognized limited power capability of the orbiter. The terms "Space Construction" used herein imply all forms of module installation, removal, checkout, inspection and deployment as well as fabrication and assembly of large space structures.

4.9.6 Orientation and Natural Illumination for Sunlit Side of Orbit

Several questions concerning vision and spacecraft orientation on the sunlit side of low earth orbit have been raised during previous space construction studies. For example, General Dynamics analysts recommend against orientations involving viewing conditions in which the earth appears as a brightly lit background to the observed construction activity (General Dynamics, 1978 a). Also, it is a well known fact that direct view of the solar disk is harmful

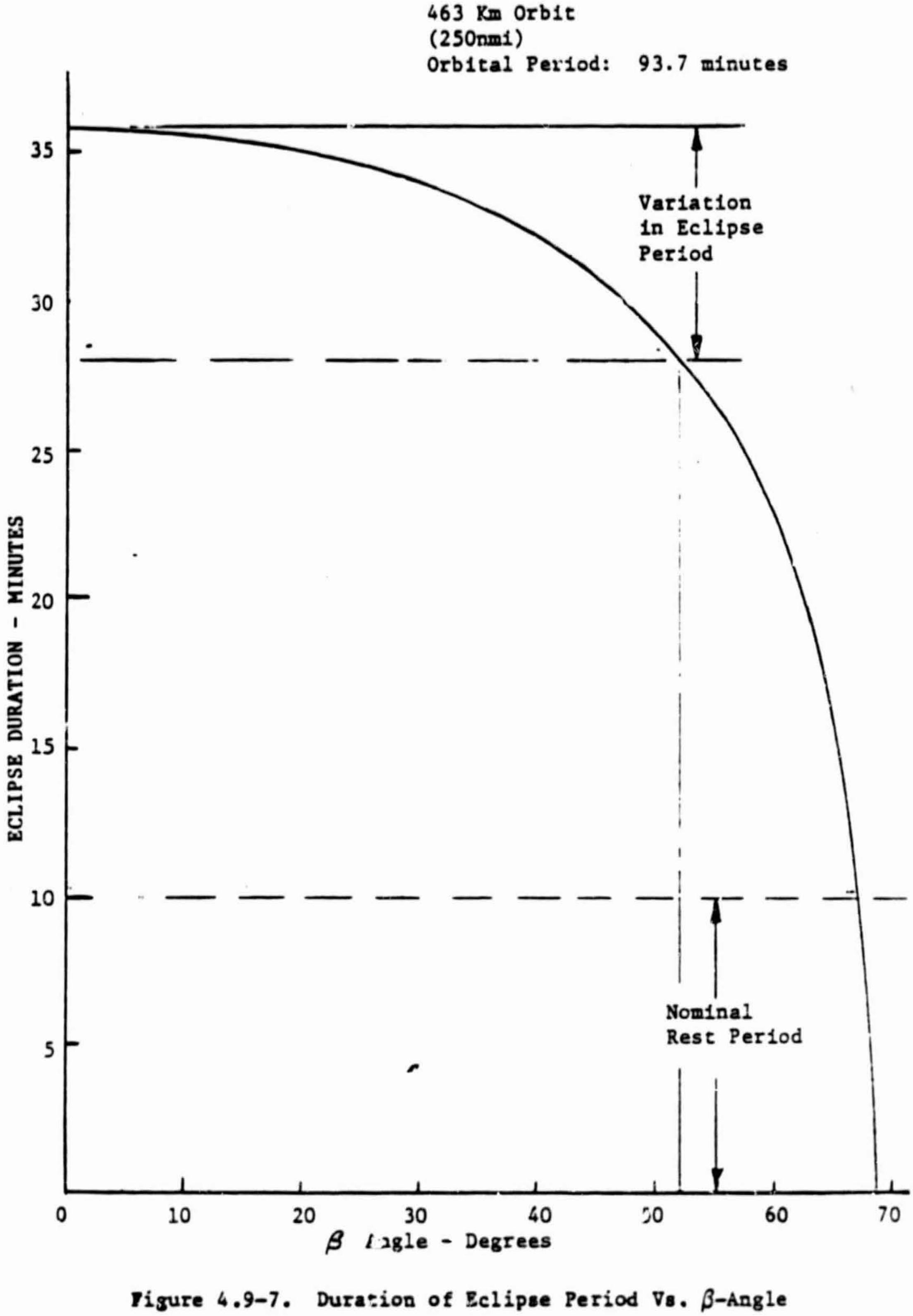


Figure 4.9-7. Duration of Eclipse Period Vs. β-Angle

Table 4.9-6. Checklist: Guidelines for Lighting Provisions in Space Construction on the Dark Side of Orbit

( ) GUIDELINES/•QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) Construction work should be continued during eclipse phase to effectively use crew time. (R-4) • Analysis of specific cases is recommended to investigate alternatives.	Rockwell 1979 (a): SSD 79-0123 Affects duration, power fequirements, lighting designs.
( ) Construction lighting should be held to minimum requirements to avoid excessive use of orbiter power. (R-4) • Less critical if auxilliary power sources are available (Power Module, Solar arrays on construction).	Rockwell 1979 (a): Minimize lighting (See following guidelines)
( ) Deployment of auxilliary solar power arrays for lighting power may cause undesirable high drag in LEO and require solar pointing (attitude control). Affects stress levels in structure, RCS fuel, and/or CMG electrical power requirements. (R-4)	Rockwell 1979 (a): Supports concept of frugality in lighting when construction from orbiter.
( ) Current orbiter lamps are not compatible with frequent on-off cycles.	Rockwell 1979 (a): Minimize on-off operations in orbit. Affects power demand.
( ) Minimize illumination requirements (R-4) • Design for concentrated work space, few lamps. • Install lamps only where and when needed for critical tasks. • Use portable, battery powered lamps for remote work sites. • Provide running lights, flashers at key points. • Provide new lamps: Compatible with frequent on-off cycling, space environment. • Specify light-colored, flat finishes.	Rockwell 1979(a): Affects design of construction fixtures, power supplies, types of lamps, finishes, crew equipment, TV cameras. Affects selection of work/rest cycles and crew procedures.

C-4

Table 4.9-6. Checklist: Guidelines for Lighting Provisions in Space Construction on the Dark Side of Orbit (Cont.)

( ) GUIDELINES/• QUALIFICATIONS	REFERENCES / IMPLICATIONS
• Use retroflectors where appropriate.	
• Provide reflector panels where feasible	
• Checkout equipment in lighted cargo bay.	
• Dark adapt crew at start of eclipse period/rest time.	
• Utilize low-light-level TV for remotely controlled operations.	
• Consider use of night vision goggles and scopes for cabin crew and EVA crew	
( ) Provide indirect lighting for TV cameras wherever possible. (R-4)	Rockwell 1979 (a): SSD 79-0123
• Avoid TV camera viewing of direct reflections of supporting lights in objects handled, especially if objects have specular surfaces.	Affects design of payload modules and lighting locations. Requires analysis of reflections, surface characteristics.

to the human eye, and to most TV tubes. Reflection of the sun from highly specular surfaces, such as the radiators on the orbiter cargo bay doors, also may be harmful to vision and to TV tubes. The latter is particularly troublesome if the solar rays are concentrated by the curvature of the orbiter thermal radiator surfaces. Finally, there are concerns about glare, high contrast and range of brightness (brightness ratios) between sunlit surfaces and adjacent surfaces in deep shadows, a condition which tends to be accentuated in space. Each of these problem areas was investigated during an analysis effort by Rockwell International (1979 a), and pertinant results are reported herein.

High Brightness Background

A large area of cloud cover on the earth can provide a high-brightness background when sunlight is reflected from it. Also, the Shuttle Orbiter has many surfaces with a high reflectivity (ρ = .80 to .95), which could appear as a bright background when observed by a TV camera or EVA crew member looking toward those surfaces. Is this a serious problem? If the surfaces are not specular, the answer is generally "no", for these high-brightness conditions can be reduced to acceptable levels for the human eye by filters of various absorptivities. Table 4.9-5 provides typical transmissability data for various orbiter window, EMU helmet and light filter combinations, as well as the effective reduction in luminance from the sun. For the TV cameras one can use filters, iris diameter control and gain control to reduce undesirable effects of background brightness. In most cases, the more appropriate question is one of the ratio of brightness of the background to the brightness of the observed target (construction project).

Brightness Ratios

In order to see detail contour and marking information on the surfaces of a construction project, it is necessary that the ratio of background brightness to foreground brightness be within the range of capability of the sensors. Whereas the human eye can accommodate to a wide brightness range, TV cameras are usually more limited. Preliminary investigations suggest that TV cameras might accommodate to a range between 30:1 to 50:1. An investigation was performed to evaluate potential problems of brightness contrast in low earth orbit, and calculations were performed to estimate apparent brightness levels and brightness ratios under a wide range of conditions. Initially, information on typical reflected light flux levels in low earth orbit was sought from the literature. Since no such measurement data were found, it was necessary to calculate them.

For purposes of simplifying analyses, it was assumed that all structure would have a coating reflectivity of 0.5. The orbiter spacecraft and construction equipment surface reflectivities used for these analyses appear in Table 4.9-3.

In the space construction environment, many surfaces can act as either a light source (direct or reflected) or as a viewing background as shown previously in Figure 4.9-1. For example, the orbiter payload bay, radiators and wings can act as a source, by reflecting sunlight, earthshine (reflected sunlight from clouds, sea or land), starlight, moonlight, or artificial light onto a construction project in process near the orbiter. Also, these orbiter surfaces may be seen as backgrounds to the construction truss work or modules, when viewed by an EVA crew member or a TV camera looking toward the orbiter.

In order to present succinct data on brightness ratios for many of the possible combinations of direct and reflected lighting on structure and on backgrounds, a series of matrix charts were prepared for this system analysis study. Table 4.9-7 covers cases of a single source of illumination versus a wide range of background conditions. Combinations of two sources of lighting were shown for a given specified background (such as bright clouds, low brightness earth, orbiter payload bay, orbiter radiator or wing) in Tables 4.9-8 to 4.9-13. In these tables, the same direct and reflected light sources are listed horizontally and vertically in order from left to right and top to bottom, respectively. Brightness ratios are listed at intersections of the two conditions. These lists remind us that all natural illumination except starlight comes originally from the sun. However, lighting on the construction project may come directly from the sun or be reflected from sunlight falling on the orbiter or from sunlight which first hits the earth, then is reflected onto the construction work or is reflected onto the various surfaces of the orbiter and then reflected again onto the structure. The large number of potential reflectors creates an extensive analysis problem. However, there are many possible cases which would be either meaningless or trivial, rarely encountered, or known to involve obvious unacceptable viewing conditions. These were noted on the charts by shading or asterisks. It was judged not cost effective to attempt to complete analysis of all possible remaining combinations for purposes of this study project. However, a significant number of important cases were examined to get a sampling of expected conditions.

The results from the approximately 50 cases analyzed showed brightness ratios between structure and background of generally less than 10:1. These analysis results indicate that seeing conditions for construction in low earth orbit are actually often quite favorable. In fact, the multiple diffuse reflections from the wide angle of visible earth surface may provide very good lighting conditions in many cases, particularly by "filling in" the deep shadows which are possible in airless space. The main problem will be work interruption caused by avoiding direct view of the sun (within about 2° of line of sight) or viewing the sun in highly specular reflective surfaces such as the orbiter radiators or glass surfaces on solar arrays at certain unfavorable angles.

Another, more subtle problem, can also occur: that of shifting from viewing a dark structure against a lighter background to viewing a lighter structure against a darker background. At the cross-over time, a lack of contrast may cause problems in discriminating outlines and details of the structure. Again, these conditions are unlikely to occur frequently or to last long, and may be overcome to some degree by color contrast between structure and background, and by distinctive markings on key structural areas where visual alignment is required. However, it is probably valuable to develop methods to predict the frequency and duration of these and other potential work interruption periods related to illumination. In some cases, there may be significant impacts which should be controlled by shading of structure, redesign of construction fixtures or even by the assembly sequence for the project.

Table 4.9-7. Space Construction Natural Illumination Analysis, Single-Source Illuminating Structure (Shuttle Orbiter Base)

STRUCTURE ILLUMINATION SOURCES / VIEWING BACKGROUNDS	SUN																
					EARTH, HI-ρ				EARTH, LO-ρ				MOON				
	DIRECT SUN	PAYLOAD BAY	RADIATOR	WING/BODY	DIRECT	P/L BAY	RADIATOR	WING/BODY	DIRECT	P/L BAY	RADIATOR	WING/BODY	DIRECT	P/L BAY	RADIATOR	WING/BODY	STARS
SUN	NOT ACCEPTABLE																
MOON	SHADOWED AREAS												(d)				
STARFIELD	DIFFICULT TO SEE (b)												(d)				OK
EARTH, HIGH-ρ	1.7		(d)		2.3	3.4	OK		RARE				–				
EARTH, LOW-ρ	6.2		(d)		RARE				1.7	1.4	OK						
*P/L BAY, SUNLIT	1.7	2.5	(d)										NOT ACCEPTABLE (c)				
P/L BAY, E.L., HI-ρ	1.7		(d)		1.7	2.5			RARE								
P/L BAY, E L., LO-ρ			(d)		RARE				2.0	2.5							
P/L BAY, MOONLIT					NOT APPLICABLE								(d)				
P/L BAY, STARLIT													(d)				
RADIATOR, SUNLIT (a)			(d)														
RADIATOR, E.L., HI-ρ			(d)		1.8				RARE				NOT ACCEPT. (c)				
RADIATOR, E.L., LO-ρ	1.7		(d)		RARE				2.1								
RADIATOR, MOONLIT	NOT APPLICABLE												NOT USEFUL				
RADIATOR, STARLIT													(d)				
WING/BODY, SUNLIT	1.4		(d)							6.2			NOT ACCEPTABLE (c)				
WING/BODY, E.L., HI-ρ			(d)		1.4				RARE								
WING/BODY, E.L., LO-ρ			(d)		RARE												
WING/BODY, MOONLIT	NOT APPLICABLE												(d)				
WING/BODY, STARLIT													(d)				

NOTES: Numerical values are ratios of brightness, higher illumination level divided by lower.

(a) Reflection of solar disc in radiator is not acceptable viewing condition.

(b) Illuminated structure probably much brighter than shadowed areas. Brightness ratios not relevant (additional reflected light desired).

(c) Illuminated background much brighter than structure. Brightness ratios not relevant (additional reflected light on structure desired).

(d) Complex reflection patterns from specular radiator prevents analysis at this level.

*P/L Bay = Cargo Bay.

Table 4.9-8. Space Construction Natural Illumination Analysis, Double Sources on Structure, Radiator Background Cases (Shuttle Orbiter Base)

RADIATOR AND STRUCTURE LIGHTING SOURCES \ STRUCTURAL ILLUMINATION SOURCES			SUN: DIRECT SUN	SUN: PAYLOAD BAY (a)	SUN: RADIATOR	SUN: WING	EARTH, HIGH ρ: DIRECT	EARTH, HIGH ρ: P/L BAY	EARTH, HIGH ρ: RADIATOR	EARTH, HIGH ρ: WING	EARTH, LOW ρ: DIRECT	EARTH, LOW ρ: P/L BAY	EARTH, LOW ρ: RADIATOR	EARTH, LOW ρ: WING	MOON: DIRECT	MOON: P/L BAY	MOON: RADIATOR	MOON: WING	STARS
SUN		DIRECT SUN (b)	●																
SUN		*P/L BAY (b)		●															
SUN		RADIATOR (a)(b)			●														
SUN		WING (b)			(f)	●													
SUN	EARTH HIGH ρ	DIRECT (c)	1.8	2.0	(f)	1.4	●	2.7	9.5	3.4	RARE								
SUN	EARTH HIGH ρ	P/L BAY (c)	2.7	2.7	(f)	2.7	2.7	●	9.5	3.4									
SUN	EARTH HIGH ρ	RADIATOR (c)			(f)		9.5	9.5	●	3.4									
SUN	EARTH HIGH ρ	WING (c)	1.8	1.8	(f)	1.8	3.4	3.4	3.4	●									
SUN	EARTH LOW ρ	DIRECT (d)	3.6	2.4	(f)	2.1	RARE				●	2.7	9.5	3.4					
SUN	EARTH LOW ρ	P/L BAY (d)	4.6	3.0	(f)	2.7					2.7	●	9.5	3.4					
SUN	EARTH LOW ρ	RADIATOR (d)			(f)						9.5	9.5	●	3.4					
SUN	EARTH LOW ρ	WING (d)	5.7	3.8	(f)	2.1					3.4	3.4	3.4	●					
SUN	MOON	DIRECT (e)	PRACTICALLY SIMILAR TO SINGLE-SOURCE CASES— NOT APPLICABLE												●			OK	
SUN	MOON	P/L BAY (e)														●			
SUN	MOON	RADIATOR (e)														OK	●		
SUN	MOON	WING (e)															(f)	●	
STARS																	(f)		

NOTES: NUMERICAL VALUES ARE RATIOS OF BRIGHTNESS, HIGHER VALUE DIVIDED BY LOWER.

● CASES ARE IMPOSSIBLE OR MEANINGLESS (DUPLICATION OF SOURCES).

(a) REFLECTION OF SOLAR DISC IN RADIATOR IS NOT ACCEPTABLE VIEWING SITUATION.

(b) RADIATOR ILLUMINATED BY DIRECT SUNLIGHT IN THESE CASES (BY ASSUMPTION)

(c) RADIATOR ILLUMINATED BY HIGH-BRIGHTNESS, SUNLIT EARTH IN THESE CASES (BY ASSUMPTION).

(d) RADIATOR ILLUMINATED BY LOW-BRIGHTNESS, SUNLIT EARTH IN THESE CASES (BY ASSUMPTION).

(e) RADIATOR ILLUMINATED BY FULL MOON IN THESE CASES (BY ASSUMPTION).

(f) COMPLEX REFLECTION PATTERN OF POINT SOURCE FROM SPECULAR RADIATOR PREVENTS ANALYSIS AT THIS LEVEL.

*P/L Bay = Cargo Bay

Table 4.9-9. Space Construction Natural Illumination Analysis, Double Sources on Structure, Starfield or Dark Earth Background Cases (Shuttle Orbiter Base)

STRUCTURAL ILLUMINATION SOURCES		SUN																STARS
						EARTH, HIGH ρ				EARTH, LOW ρ				MOON				
		DIRECT SUN	PAYLOAD BAY	RADIATOR	WING	DIRECT	P/L BAY	RADIATOR	WING	DIRECT	P/L BAY	RADIATOR	WING	DIRECT	P/L BAY	RADIATOR	WING	
SUN	DIRECT SUN	●				(DUPLICATES OPPOSITE HALF)												
	*P/L BAY		●															
	RADIATOR	(b)	(b)	●														
	WING			(b)	●													
SUN – EARTH HIGH ρ	DIRECT	1.3		(b)		●												
	P/L BAY	1.5	(d)	(b)		1.5	●											
	RADIATOR			(b)				●										
	WING			(b)					●									
SUN – EARTH LOW ρ	DIRECT	3.6		(b)		RARE (a)				●								
	P/L BAY			(b)							●							
	RADIATOR			(b)								●						
	WING			(b)									●					
SUN – MOON	DIRECT	PRACTICALLY SIMILAR TO SINGLE-SOURCE CASES—(c)												●				
	P/L BAY														●			
	RADIATOR													OK		●		
	WING																●	
STARS																		●

NOTES: NUMERICAL VALUES ARE RATIOS OF BRIGHTNESS, HIGHER VALUE DIVIDED BY LOWER.
DARK BACKGROUND NOT CONSIDERED IN CALCULATING BRIGHTNESS RATIOS, ONLY HIGHER BRIGHTNESS SOURCES.

● CASES ARE IMPOSSIBLE OR MEANINGLESS (DUPLICATION OF SOURCES)

(a) PRESENCE OF LOW-BRIGHTNESS BACKGROUND USUALLY IMPLIES LOW-BRIGHTNESS LIGHTING ON ALL ORBITER SRUFACES.

(b) COMPLEX REFLECTION PATTERN FROM SPECULAR RADIATOR PREVENTS ANALYSIS AT THIS LEVEL.

(c) LUNAR LIGHT ESSENTIALLY INVISIBLE IN PRESENCE OF BRIGHT SUNLIGHT REFLECTIONS. BRIGHTNESS RATIOS GREATER THAN 10^4.

(d) UNLIKELY BOTH SOURCES CAN ILLUMINATE STRUCTURE DUE TO CONFIGURATION AND EARTH-SUN RELATIONSHIP.

*P/L Bay = Cargo Bay

Table 4.9-10. Space Construction Natural Illumination Analysis, Double Sources on Structure Sunlit Bright Cloud Background Cases (Shuttle Or (Shuttle Orbiter Base)

STRUCTURAL ILLUMINATION SOURCES		SUN: DIRECT SUN	SUN: PAYLOAD BAY	SUN: RADIATOR	SUN: WING	EARTH, HIGH ρ: DIRECT	EARTH, HIGH ρ: P/L BAY	EARTH, HIGH ρ: RADIATOR	EARTH, HIGH ρ: WING	EARTH, LOW ρ: DIRECT	EARTH, LOW ρ: P/L BAY	EARTH, LOW ρ: RADIATOR	EARTH, LOW ρ: WING	MOON: DIRECT	MOON: P/L BAY	MOON: RADIATOR	MOON: WING	STARS
SUN	DIRECT SUN	●																
SUN	* P/L BAY	2.6	●							(DUPLICATES OPPOSITE HALF)								
SUN	RADIATOR	(b)	(b)	●														
SUN	WING	1.7	2.7	(b)	●													
SUN: EARTH HIGH ρ	DIRECT	2.3	3.4	(b)		●												
SUN: EARTH HIGH ρ	P/L BAY	3.4	(d)	(b)		3.4	●											
SUN: EARTH HIGH ρ	RADIATOR	2.3	2.6	(b)				●										
SUN: EARTH HIGH ρ	WING			(b)					●									
SUN: EARTH LOW ρ	DIRECT					RARE (a)				●								
SUN: EARTH LOW ρ	P/L BAY										●							
SUN: EARTH LOW ρ	RADIATOR											●						
SUN: EARTH LOW ρ	WING												●					
SUN: MOON	DIRECT	PRACTICALLY SIMILAR TO SINGLE-SOURCE CASES— (c)												●				
SUN: MOON	P/L BAY														●			
SUN: MOON	RADIATOR															●		
SUN: MOON	WING																●	
STARS																		●

NOTE: NUMERICAL VALUES ARE RATIOS OF BRIGHTNESS, HIGHER VALUE DIVIDED BY LOWER.

● CASES ARE IMPOSSIBLE OR MEANINGLESS (DUPLICATION OF SOURCES)

(a) PRESENCE OF HIGH-BRIGHTNESS BACKGROUND USUALLY IMPLIES HIGH-BRIGHTNESS LIGHTING OF ALL ORBITER SURFACES.

(b) COMPLEX REFLECTION PATTERN OF POINT SOURCE FROM SPECULAR RADIATOR PREVENTS ANALYSIS AT THIS LEVEL.

(c) LUNAR LIGHT ESSENTIALLY INVISIBLE IN PRESENCE OF BRIGHT SUNLIGHT REFLECTIONS. BRIGHTNESS RATIOS GREATER THAN 10^5.

(d) UNLIKELY THAT BOTH SOURCES CAN ILLUMINATE STRUCTURE DUE TO CONFIGURATION OR EARTH-SUN RELATIONSHIPS.

*P/L Bay = Cargo Bay

Table 4.9-11. Space Construction Natural Illumination Analysis, Double Sources on Structure, Low-Brightness Earth Background (Sunlit) Cases (Shuttle Orbiter Base)

STRUCTURA ILLUMINATION SOURCES			SUN																STARS
							EARTH, HIGH ρ				EARTH, LOW ρ				MOON				
			DIRECT SUN	PAYLOAD BAY	RADIATOR	WING	DIRECT	P/L BAY	RADIATOR	WING	DIRECT	P/L BAY	RADIATOR	WING	DIRECT	P/L BAY	RADIATOR	WING	STARS
SUN		DIRECT SUN	●				(DUPLICATES OPPOSITE HALF)												
		*P/L BAY	6.2	●															
		RADIATOR	(b)	(b)	●														
		WING			(b)	●													
	EARTH HIGH ρ	DIRECT	RARE (a)				●												
		P/L BAY						●											
		RADIATOR							●										
		WING								●									
	EARTH LOW ρ	DIRECT	6.2		(b)		RARE (a)				●								
		P/L BAY	6.2	(d)	(b)							●							
		RADIATOR			(b)								●						
		WING			(b)									●					
	MOON	DIRECT	PRACTICALLY SIMILAR TO SINGLE-SOURCE CASES— (c)												●				
		P/L BAY														●			
		RADIATOR															●		
		WING																●	
STARS																			●

NOTES: NUMERICAL VALUES ARE RATIOS OF BRIGHTNESS, HIGHER ILLUMINATION LEVEL DIVIDED BY LOWER.

● CASES ARE IMPOSSIBLE OR MEANINGLESS (DUPLICATION OF SOURCES)

(a) PRESENCE OF LOW-BRIGHTNESS BACKGROUND USUALLY IMPLIES LOW-BRIGHTNESS LIGHTING ON ALL SURFACES.

(b) COMPLEX REFLECTION PATTERN OF POINT SOURCE FROM SPECULAR RADIATOR PREVENTS ANALYSIS AT THIS LEVEL.

(c) LUNAR LIGHT ESSENTIALLY INVISIBLE IN PRESENCE OF BRIGHT SUNLIGHT REFLECTIONS.

(d) UNLIKELY THAT BOTH SOURCES CAN ILLUMINATE STRUCTURE DUE TO CONFIGURATION OR EARTH-SUN RELATIONSHIP.

*P/L Bay = Cargo Bay

Table 4.9-12. Space Construction Natural Illumination Analysis, Double Sources on Structure, Cargo Bay Background Classes (Shuttle Orbiter Base)

STRUCTURAL ILLUMINATION SOURCES / PAYLOAD BAY AND STRUCTURE LIGHTING SOURCES			SUN																STARS
							EARTH, HIGH ρ				EARTH, LOW ρ				MOON				
			DIRECT SUN	PAYLOAD BAY	RADIATOR	WING	DIRECT	P/L BAY	RADIATOR	WING	DIRECT	P/L BAY	RADIATOR	WING	DIRECT	P/L BAY	RADIATOR	WING	
SUN		DIRECT SUN	●				(DUPLICATES OPPOSITE HALF)												
SUN		* P/L BAY	2.6	●															
SUN		RADIATOR	(b)	(b)	●														
SUN		WING			(b)	●													
SUN	EARTH HIGH ρ	DIRECT	2.3	(e)	(b)	(e)	●												
SUN	EARTH HIGH ρ	P/L BAY		(d)	(b)	(e)		●											
SUN	EARTH HIGH ρ	RADIATOR		(e)	(b)	(e)			●										
SUN	EARTH HIGH ρ	WING		(e)	(b)	(e)				●									
SUN	EARTH LOW ρ	DIRECT		(e)	(b)	(e)	RARE (a)				●								
SUN	EARTH LOW ρ	P/L BAY		(d)	(b)	(e)						●							
SUN	EARTH LOW ρ	RADIATOR		(e)	(b)	(e)							●						
SUN	EARTH LOW ρ	WING		(e)	(b)	(e)								●					
SUN	MOON	DIRECT	PRACTICALLY SIMILAR TO SINGLE-SOURCE CASES— (c)												●				
SUN	MOON	P/L BAY													OK	●			
SUN	MOON	RADIATOR															●		
SUN	MOON	WING																●	
STARS																			

NOTES: NUMERICAL VALUES ARE RATIOS OF BRIGHTNESS, HIGHER ILLUMINATION LEVEL DIVIDED BY LOWER.

● CASES ARE IMPOSSIBLE OR MEANINGLESS (DUPLICATION OF SOURCES).

(a) PRESENCE OF LOW-EARTH BRIGHTNESS ILLUMINATION USUALLY IMPLIES LOWER LEVELS OF ILLUM. ON ALL SURFACES.

(b) COMPLEX REFLECTION PATTERN OF POINT SOURCE FROM SPECULAR RADIATOR PREVENTS ANALYSIS AT THIS LEVEL.

(c) LUNAR LIGHT ESSENTIALLY INVISIBLE IN PRESENCE OF SUNLIGHT REFLECTIONS.

(d) UNLIKELY THAT BOTH SOURCES CAN ILLUMINATE STRUCTURE.

(e) UNLIKELY COMBINATION OF ILLUMINATION SOURCES AND VIEWING CONDITIONS.

*P/L Bay = Cargo Bay

Table 4.9-13. Space Construction Natural Illumination Analysis, Double Sources on Structure, Wing/Body Surface Background (Shuttle Orbiter Base)

WING-BODY AND STRUCTURE LIGHTING SOURCES \ STRUCTURAL ILLUMINATION SOURCES			SUN																STARS
							EARTH, HIGH ρ				EARTH, LOW ρ				MOON				
			DIRECT SUN	PAYLOAD BAY	RADIATOR	WING	DIRECT	P/L BAY	RADIATOR	WING	DIRECT	P/L BAY	RADIATOR	WING	DIRECT	P/L BAY	RADIATOR	WING	STARS
SUN		DIRECT SUN	●				(DUPLICATES OPPOSITE HALF)												
SUN		*P/L BAY	1.4	●															
SUN		RADIATOR	(b)	(b)	●														
SUN		WING			(b)	●													
SUN	EARTH HIGH ρ	DIRECT					●												
SUN	EARTH HIGH ρ	P/L BAY		(d)	(b)		2.1	●											
SUN	EARTH HIGH ρ	RADIATOR			(b)				●										
SUN	EARTH HIGH ρ	WING			(b)					●									
SUN	EARTH LOW ρ	DIRECT	3.0		(b)		RARE (a)				●								
SUN	EARTH LOW ρ	P/L BAY		(d)	(b)							●							
SUN	EARTH LOW ρ	RADIATOR			(b)								●						
SUN	EARTH LOW ρ	WING			(b)									●					
SUN	MOON	DIRECT	PRACTICALLY SIMILAR TO SINGLE-SOURCE CASES — (c)												●				
SUN	MOON	P/L BAY														●			
SUN	MOON	RADIATOR															●		
SUN	MOON	WING																●	
STARS																			●

NOTES: NUMERICAL VALUES ARE RATIOS OF BRIGHTNESS, HIGHER ILLUMINATION LEVEL DIVIDED BY LOWER.

CASES ARE IMPOSSIBLE OR MEANINGLESS (DUPLICATION OF SOURCES)

(a) PRESENCE OF LOW-EARTH BRIGHTNESS ILLUMINATION USUALLY IMPLIES LOWER LIGHTING LEVELS ILLUMINATING ALL SURFACES.

(b) COMPLEX REFLECTION PATTERNS OF POINT SOURCE FROM SPECULAR RADIATOR PREVENTS ANALYSIS AT THIS LEVEL.

(c) LUNAR LIGHT ESSENTIALLY INVISIBLE IN PRESENCE OF SUNLIGHT REFLECTIONS.

(d) UNLIKELY THAT BOTH SOURCES CAN ILLUMINATE STRUCTURE.

*P/L Bay = Cargo Bay

In summary, the foregoing problems may be anticipated in some degree and avoided in the following ways:

(1) Multiple TV camera locations providing a selection of viewing angles.

(2) Shifting the viewing location (for EVA crew).

(3) Providing localized shade panels or diffusers.

(4) Work scheduling.

(5) Construction project re-orientation.

The last listed option has been noted as generally undesirable because of stress loads and fuel economies. It is probably not necessary or cost effective in most cases. The constantly changing orbital conditions will tend to revise viewing angles in relation to the solar rays within a short time, especially if there is no attempt to maintain a solar inertial attitude control.

Attitude Orientation Considerations

Foregoing considerations of vehicle and construction work orientations have dealt mainly with general problems of light levels and specularity associated with differing orientations rather than specific angular limits and specific tasks involving illumination and viewing with TV. Another consideration is the geometric interaction of construction activity viewing requirements and the critical combinations of light directions impinging on the construction. Previous discussion has shown that the direction of the sun's rays is the single most critical consideration affecting work interruption on the sunlit side of orbit. A method is needed to analyze and summarize typical viewing directions for a specific project and relate them to probable directions of the sun's rays, based upon stability analyses of the orbiting construction projects. The method recommended here for the summary presentation of viewing angles is the single-point globographic diagram. In such a presentation it is assumed that all viewing directions originate from a common point in space. All of the critical viewing angles are plotted as points or areas on a sphere surrounding the common point, in terms of angles of longitude and latitude. With respect to the sun's nearly parallel rays of light, such an approximation is quite reasonable as a summary method. Of course, in the actual physical situation there may be a portion of the orbiter, construction project or space base between the observer and the sun, such that there is no problem of viewing the solar disc by an observer. Such obstructions can also be plotted for a specific eye or camera position.

To develop a meaningful example which illustrates the method and presentation, a brief analysis was performed on a space fabricated tri-beam advanced communication platform project. Figure 4.9-8 illustrates several key construction operations and probable viewing directions (indicated by arrows and TV camera/light sets) which were identified and tabulated during the analysis. The orbiters relation to structure and a summary of viewing directions are shown at the lower right.

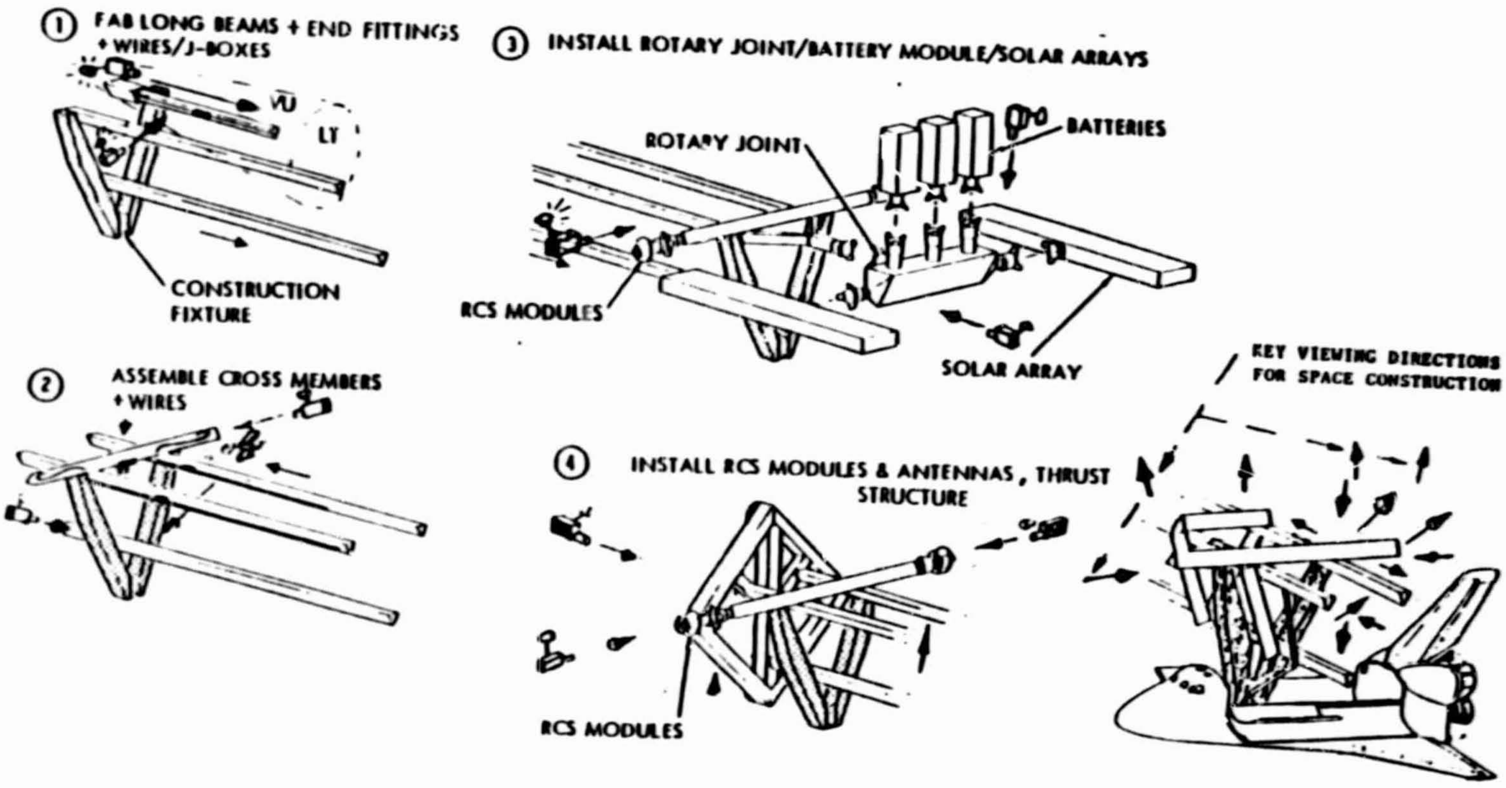


Figure 4.9-8. Key Construction Processes and Associated Viewing Directions for Example Study

Two views of the globographic summary of viewing angles for this project are shown in Figure 4.9-9. The views are a rear quarter view and front quarter view with respect to the orbiter. On the theoretical globe graduated in degrees elevation and azimuth, 0° of elevation and azimuth correspond to the orbiter's -X axis. Areas (spots) marked with shading slanted upward to the right represent an estimated range ($\pm 5^{\circ}$) about the nominal key viewing angles required for TV cameras or EVA crew during construction processes. The directions indicated by arrows shown in Figure 4.9-8 would intersect the globe at the center of these shaded areas. The dashed lines which surround the shaded areas suggest a wider range of angles where sunlight impingement on a TV camera lens would be undesirable, and where some special shielding may be required to reduce viewing angles. Areas shaded with lines slanting upward to the left, covering the top and a portion of the rear of the globe around the orbiter, are also shown. These indicate probable required viewing angles from the windows of the crew cabin on-orbit stations and the payload bay TV cameras. Such angles would be used for erection operations of the construction fixture, general observations of construction processes, and occasionally for selected critical transport and assembly operations.

Knowing the probable viewing angles, one can consider selection of initial orientations and launch parameters for favorable average viewing conditions, specially where some orientation control is achievable by gravity gradient methods. Some of the more obvious orientations options which were identified are briefly evaluated and summarized in Table 4.9-14. It appears that the solar-fixed inertial attitude gives the best control of natural illumination, and is desirable if a favorable attitude can be maintained. However, any unfavorable aspects would also be maintained continuously throughout the sunlit side of the orbit. In contrast, the earth-oriented and the drifting modes have the potential advantage (as well as problems) of variety; any problem which turns up will likely pass away shortly afterward. More important and specific to the selection of orientations are the problems of fuel requirements and power to maintain positive attitude control, potential loads on the structure during the vulnerable period of construction, and the added complexity and cost of stabilization and control systems which are designed only for the construction period. The free-drift, semi-gravity gradient stabilized orientations tend to reduce such fuel costs and complexities.

Based on the foregoing analyses, the following general guidelines were derived for favorable construction orientation selection in relation to the sunlit side of earth orbit:

- Orient the cargo bay toward the earth or toward the horizon such that at least half the earth disc can reflect diffuse sunlight reflections from earth toward the structure and also can be reflected by the orbiter surfaces. This attitude range tends to avoid solar reflections from the orbiter radiators.

- Alternatively, orient the orbiter and construction work so as to reflect sunlight from the wings, radiator and cargo bay toward the construction work, but from behind the majority of TV cameras and EVA work stations.

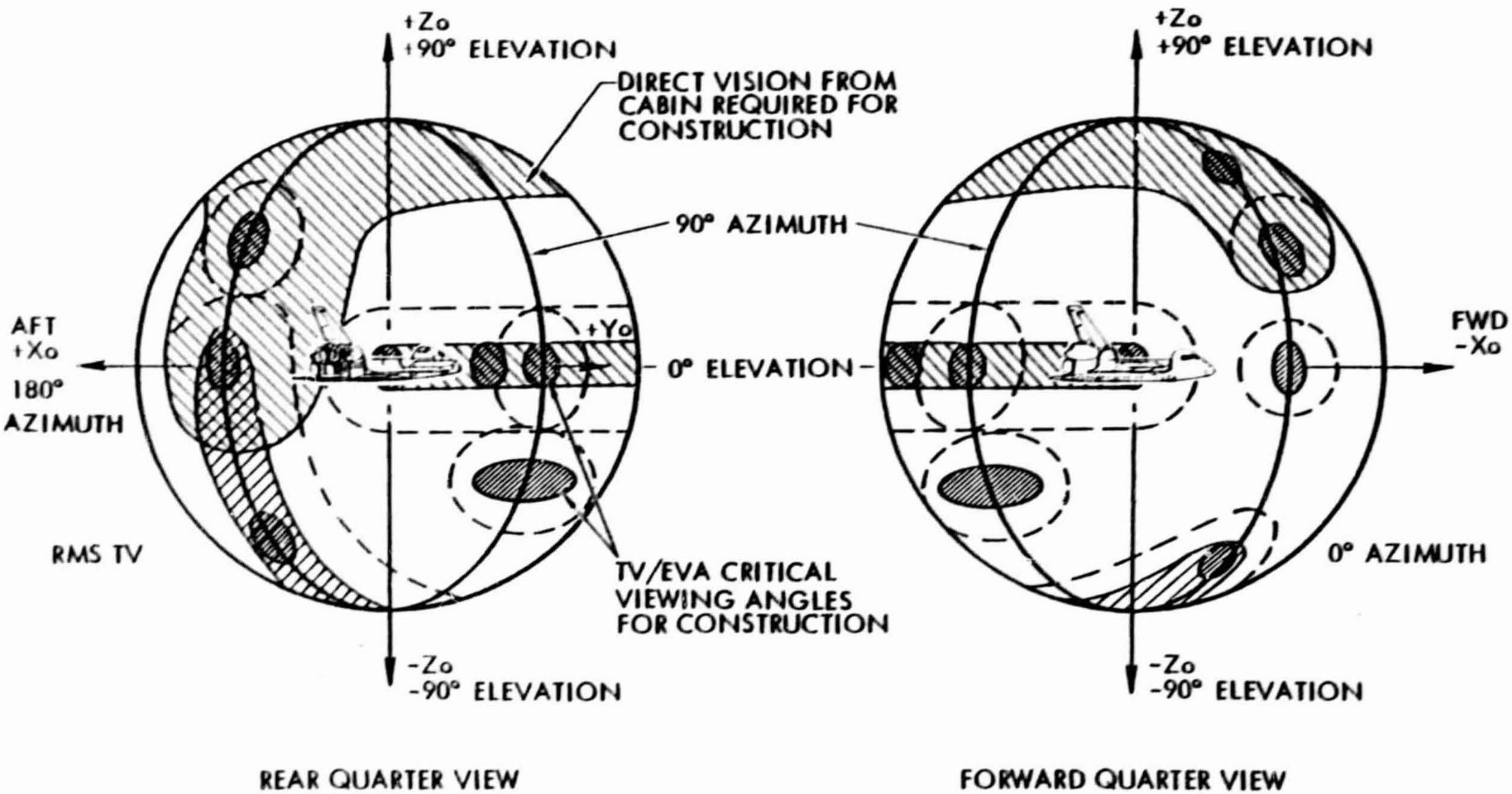


Figure 4.9-9. Globographic Viewing Angle Summary-
Space Fabricated Tri-Beam Communications Platform

Table 4.9-14. General Considerations for Selecting Space Construction Orientation with Respect to Viewing from Orbiter Aft Flight Deck

ATTITUDE	CONSIDERATIONS FOR VIEWING AND THERMAL RADIATION
Solar-fixed inertial:	
o Nose toward sun	Construction is always well illuminated if above cargo bay or to the side of Orbiter. No problems of veiling luminance from sun, radiator reflections, or direct vision of sun, but some concern for bright background from earth-cloud cover. Favorable for thermal radiation.
o Nose toward sun pitch down $45^{\circ} \pm 10^{\circ}$	Construction is well illuminated by direct sunlight and wing/body reflections. Some bright reflections from cargo bay, moderate concern for reflections from radiators. Reduced glare from earth cloud cover. Acceptable for thermal radiation. No solar veiling luminance.
o Nose 90° from sun,	One side of construction and assembly fixture appears in solar shadow, but strong reflection from earth on opposite side helps, especially during middle of sunlit period. Favorable for thermal radiation. Few problems with reflections from radiators.
Earth-fixed relationship	
o Wings toward nadir Nose toward pole (South or North as function of time of year, time of launch) (XPOP, YLV, ZVV)	o General Dynamics (1978) considers favorable to limit earth viewing and sun viewing. Some problems with sun background when coming around limb of earth. Configurations requiring overhead viewing have part-time viewing directly at sun. Solar shadows over cargo bay mitigated by earth reflections when belly of orbiter facing sun. Generally favorable, but variable thermal radiation effects.
o Cargo bay toward earth (XPOP, ZLV, YVV)	o Bright solar reflections off cargo bay at dawn and dusk of orbit. Earth background at all times, very bright at orbital noon, but in shadow of sun; good viewing. Unfavorable for thermal radiation.

Table 4.9-14. General Considerations for Selecting Space Construction Orientation with Respect to Viewing from Orbiter Aft Flight Deck, (Continued)

ATTITUDE	CONSIDERATIONS FOR VIEWING AND THERMAL RADIATION
Earth-fixed relationship, (Cont.)	
o Payload bay away from earth (XPOP, -ZLV, YVV)	o Very bright reflections from sun in cargo bay. Sun background overhead unacceptable during large part of orbit, unfavorable for TV & EVA looking toward orbiter radiators. Unfavorable for maximum heat rejection near solar noon.
Random - free drift	
o Rotating mode - undamped	o Wide variety of angles are not predictable. Optimum approach for vision would be to erect a diffuser/reflector curtain over the cargo bay prior to starting construction work, provide artificial illumination. However, radiator heat rejection is adversely affected.
o Gravity gradient stabilization only. Limited oscillations (ZPOP, -XLV, YVV)	o Attitudes under these conditions must be calculated. Shade or diffuser may be desired if attitude unfavorable. Location is critical to radiator heat rejection. Some of earth-fixed relationships may be applicable, depending on configuration, construction strategy.

- When possible, orient the orbiter and/or construction base (if used) so as to shield viewing sites from direct solar impingement when work is going on at the work site. The latter guideline suggests that the sunlight should illuminate the belley of the orbiter rather than the cargo bay and radiators.

Example of Solar Viewing Angle Analysis

Applications of these guidelines are illustrated by a specific example. A preliminary stability analysis of the tri-beam, space fabricated, advanced communication platform during the construction process has indicated the probable orientation conditions, illustrated in Figure 4.9-10 and further examined as follows:

(1) The longitudinal axis of the project (orbiter Y axis) will point toward the center of the earth (along the local vertical) and will drift approximately $\pm 15^{\circ}$, leading or lagging the orbiter, in the plane of orbit. The period is about 1.7 cycles per orbit. The orbiter wings will approximately lie in the plane of the orbit.

(2) The construction project will oscillate approximately $\pm 10^{\circ}$ about the local vertical in directions perpendicular to the plane of the orbit.

(3) The orbiter should be oriented with tail leading (ZPOP, YLV, XVV), but nose leading (ZPOP, -YLV, -XVV) is acceptable. It will tend to rotate about the orbiter Y axis, approximately $\pm 10^{\circ}$, in a plane approximately perpendicular to the orbiter plane.

Figure 4.9-10 provides an example of how the previously described globographic presentation can be used to describe sun impingement angles with respect to the orbiter: Since the theoretical globe is considered to be firmly attached to the orbiter, a viewer on the orbiter sees the sun trace a different path along the surface of the theoretical globe on each orbit. The path is a function of the β angle of the orbit plane (angle relative to sun) and the attitude drift angle status of the orbiter. View A-A in Figure 4.9-10 shows how the average elevation of the sun's path (relative to the orbiter) may vary during the winter solstice (December 21). The orbital plane β-angle may vary from -52° to $+5^{\circ}$, depending on time of launch during the day. The shaded area on the theoretical globe represents the entire range of possible sun positions in angular coordinates. Figure 4.9-11 indicates how such potential sun viewing angles can be plotted on the globographic presentation (shown as horizontal shaded area) and compared to the critical viewing angles for construction, as shown in Figure 4.9-9. The resulting comparison in Figure 4.9-11 indicates that looking at the earth (starboard) and northward (overhead, away from cargo bay) are favorable directions to avoid a direct view of the sun for conditions stated and the project being constructed. The orientation also satisfies one of the guidelines for orientation; that is, the cargo bay generally faces the northern horizon, so that considerable reflected light from the earth illuminates at least one side of the structure mounted "above" the cargo bay. For those orbits having β-angles of approximately 20° to 52°, the orbiter structure provides considerable shielding against viewing the sun, since the belly is oriented somewhat toward the sun. The globographic diagram can also be used to illustrate the favorable range of angles to shield a specific viewing site from the sun. For example, Figure 4.9-12 compares the benefits of orbiter structure shielding to the possible range of solar viewing angles previously described. In the example, two different work sites are assumed. One

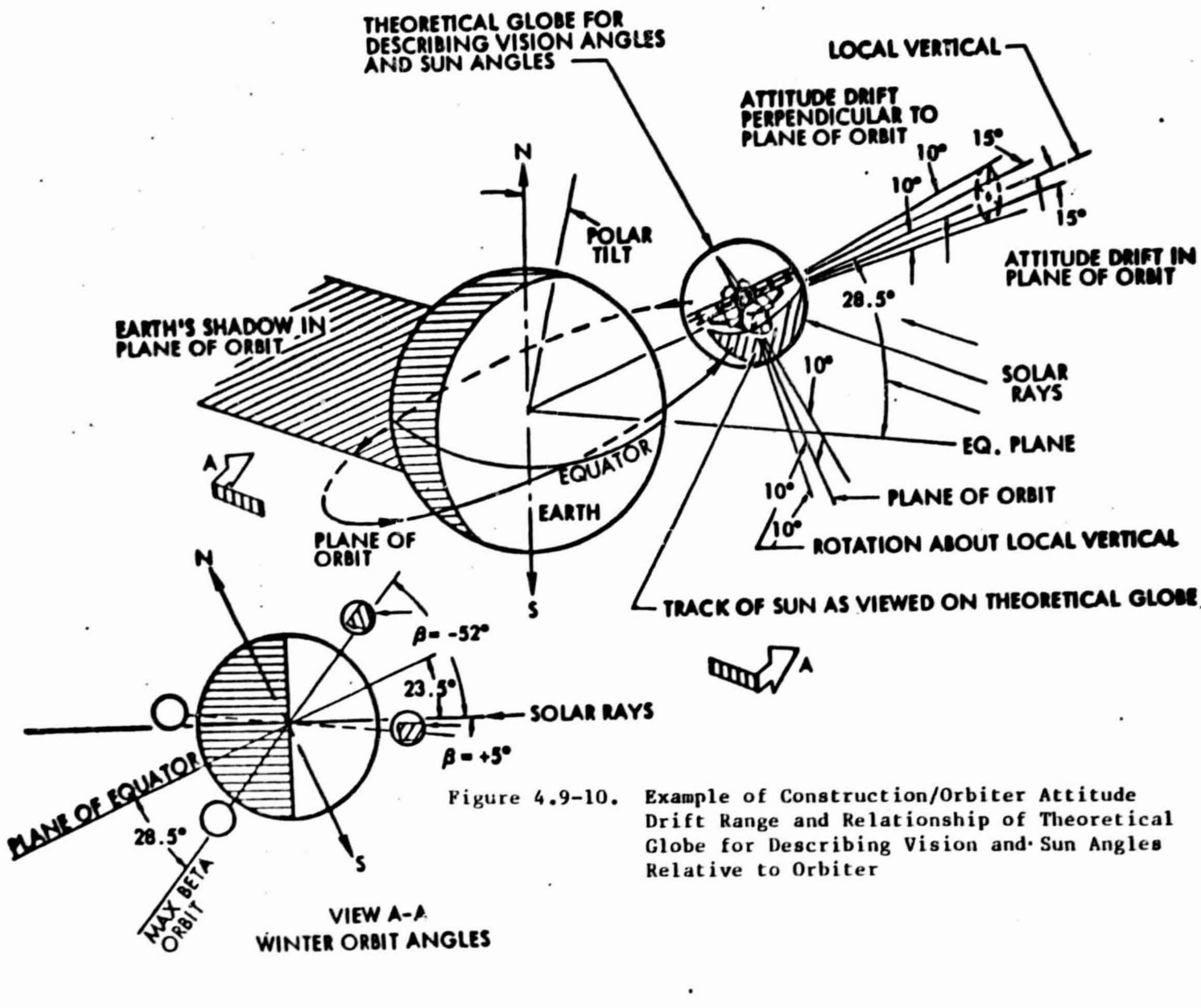


Figure 4.9-10. Example of Construction/Orbiter Attitude Drift Range and Relationship of Theoretical Globe for Describing Vision and Sun Angles Relative to Orbiter

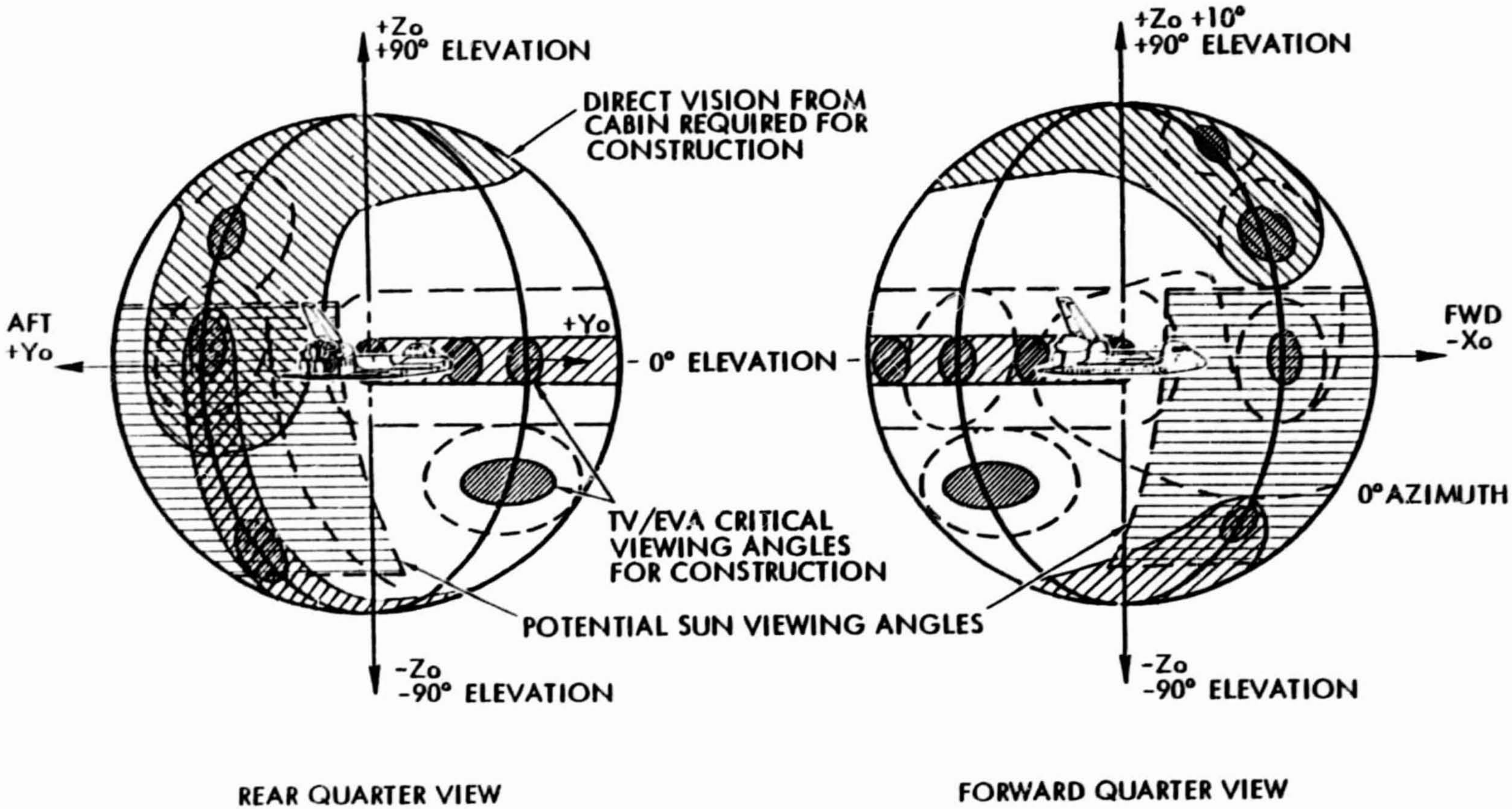
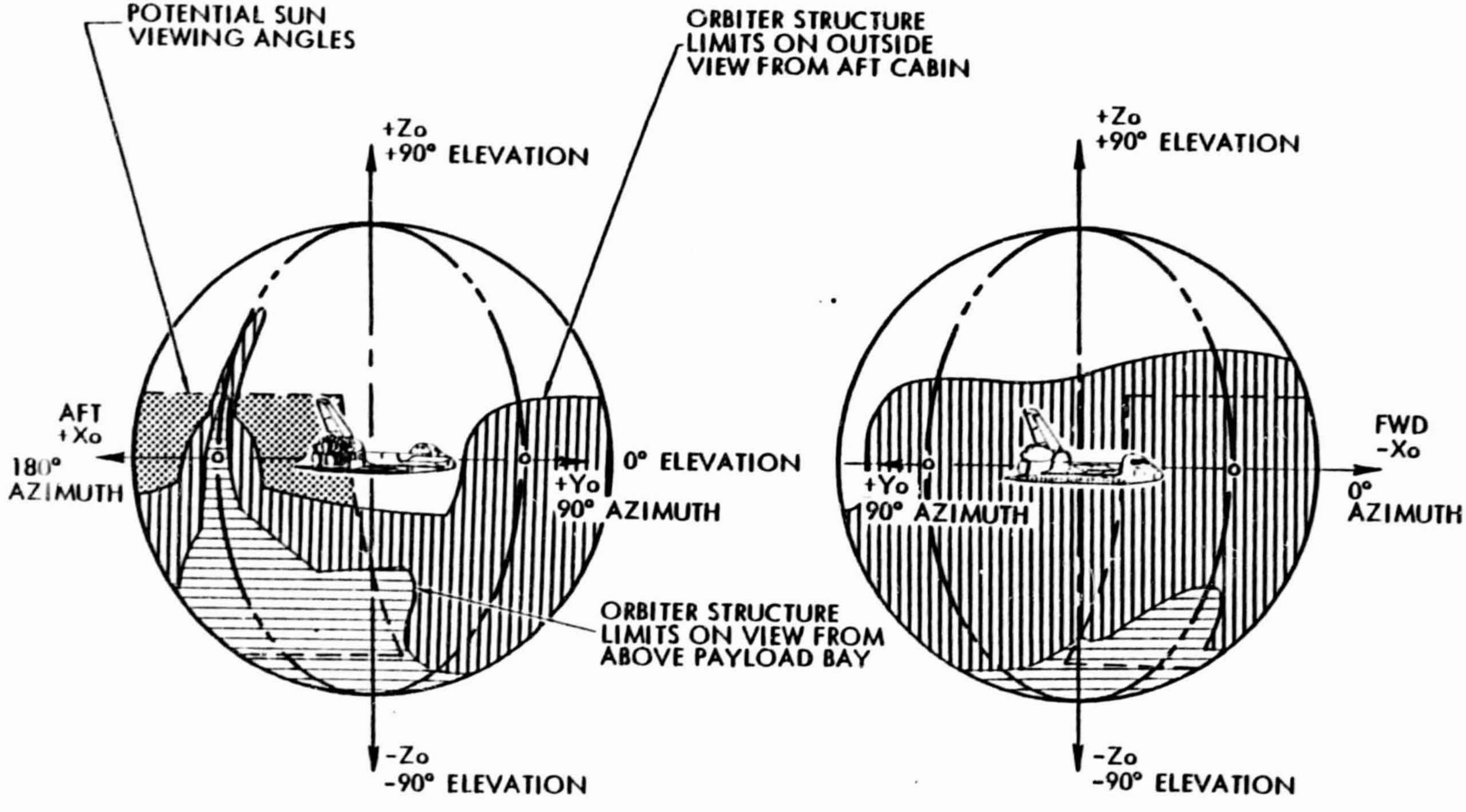


Figure 4.9-11. Solar Viewing Angles Compared to Key Viewing Angles for Construction



REAR QUARTER VIEW

FORWARD QUARTER VIEW

Figure 4.9-12. Solar Viewing Angles Compared to Orbiter Shading Potential From Two Viewing Sites site is the orbiter aft cabin, where considerable head movement is assumed possible so as to increase the field of view out the windows beyond that usually shown. The visually obstructed area is shaded with vertical lines. The other viewing site is a fixed point about 20 meters above the center of the cargo bay. The resulting visual obstruction area is shown shaded with horizontal lines. Figure 4.9-12 also shows the potential range of sun viewing angles outlined by the dashed line. A relatively small portion of this sun-viewing angle range is not shielded from the crew in the cabin (the area shaded by dots). However, the view from the site above the cargo bay has a much larger chance of sunlight viewing. If this site represents a TV camera which is primarily oriented toward the cargo bay and is shaded to prevent an excessively wide range of viewing angle, then it could be largely protected from sun viewing by the orbiter body.

Rotational Rate Considerations in Natural Illumination

Some concern may arise relating to a constantly changing visual environment during space construction. Possible problems are simple annoyance, inconvenience, confusion of shapes or directions, disorientation and even motion sickness. Most of these problems relate to the rates or motion experienced. In the specific example studied, the rotational rates were relatively slow, as shown in Table 4.9-15. Experience shows that such rates would generally have negligible effects on the crew as concerns disorientation or motion sickness. Large rates could be experienced if the construction proceeds at a faster rate, or other balance conditions produce instabilities. The rates shown in Table 4.9-15 are not to be considered either average nor limiting conditions. However, even if the rates were ten times those listed, there is little chance of visual disorientation for the crew.

Table 4.9-15. Rotational Rates During Space Construction of Tri-Beam, Space Fabricated Communications Platform

(Angles Relative to Sun -- One Example Case)

Motion		Rates
Satellite Axes	Orbiter Axes	(Degrees/Sec)
Roll	Roll	-.07 +.05
Pitch	Yaw	+.06 -.06
Yaw	Pitch	+.075 -.070

Finish Coating: Colors and Specularity

Non-specular (flat) finish coatings are highly desirable for viewing on the sunlit side of orbit as well as on the eclipse side. However, there is strong reason for somewhat darker (non-white) hues with lower reflectance on the sunlit side than on the dark side. This type of finish would give greater assurance that structure could be seen against bright white clouds or the white wing or cargo bay surfaces or the orbiter. C. Wheelwright of NASA, Johnson Space Center, recommends colors in the yellow, yellow-green or brown range, with the yellow-green color of zinc oxide primer considered as the ideal. Rather than widely different hues, he recommends varied purity and chroma for contrast in differing structures and modules.

Table 4.9-16 summarizes the foregoing discussion into guidelines for use of the orbiter and systems in viewing and illumination on the sunlit side of low earth orbit.

Table 4.9-16. Checklist: Guidelines for Use of Orbiter and System in Viewing/Illumination on Sunlit Side of Orbit

( ) GUIDELINES / • QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) Design large space structures to minimize requirements for direct viewing of sun or its reflection in orbiter radiators and other specular surfaces. (R-4)	Rockwell 1979 (a): SSD 79-0123 Affects location with respect to orbiter, attitude control, configuration.
( ) If possible, orient viewing angles and orbiter to take advantage of reflected light from earth. (R-4) • Applies to both direct viewing and TV camera viewing. • Free-drift condition is probably acceptable in most cases - undesirable viewing conditions are short-lived. • Earth viewing (as background) may be acceptable with adequate window filters, dark glasses, helmet filters.	Rockwell 1979 (a): SSD 79-0123 Affects analysis of gravity gradient or active control of attitude. Affect selection of TV camera locations, EVA workspace locations, orientations. Affects crew procedures and equipment selection.
( ) Provide flat (non-specular) finishes, light colors on all space construction equipment and structures, to the degree feasible. (R-4)	Affects selection of materials, manufacturing processes, finish selection.

4.10 ELECTRICAL ORBITER POWER SUBSYSTEM

The electrical power subsystem (EPS) consists of the equipment and reactants required to supply electrical power to the electrical buses. The general arrangement of electrical power system components is illustrated in Figure 4.10-1. Electrical power distribution and conditioning equipment beyond the power generation equipment terminals is not considered a part of this subsystem. Power is supplied to fulfill all orbiter requirements when it is not connected to GSE power.

The EPS can be functionally divided into two major subsystems:

1. Power generation subsystem: fuel cell powerplants (FCP).

2. Power reactant storage and distribution (PRSD) subsystem.

The EPS supplies power during the peak, average, and minimum load periods of the mission. It supplies oxygen to the environmental control and life support subsystem (ECLSS). The peak power and average power requirements are supplied by the three FCP's with each FCP connected to one of the three main dc buses. The minimum power requirements are supplied by two of the three FCP's with one FCP connected to two buses. Under certain conditions, namely, a +40°F environment, the third FCP can be shut down and disconnected from the bus and can be restarted and reconnected to the bus within 15 minutes as required to support higher loads. For conditions where the environment will be lower than +40°F, the third FCP can be placed on standby and disconnected from the bus and can be reconnected as required. Excess heat from the fuel cells is transferred to the Freon cooling loop through heat exchangers.

Key constraints of concern to typical construction operations for large space structures are the total energy, the maximum continuous power and the peak power availability to payloads. The maximum power limitation typically constrains combined use of the RMS, lights on the orbiter and construction fixture and possibly heaters for construction machinery. However, some projects may find constraints on power usage for ascent and descent to be of significance.

Table 4.10-1 summarizes the key numerical constraints on the above mentioned quantities. Details of the orbiter capability and descriptions of the subsystems follow.

Figure 4.10-2 summarizes the power and energy availability of the orbiter as a function of the number of tank sets installed. The nominal complement is three (3) tanks sets (all below the payload bay liner).

If mission energy requirements exceed that provided by the three baseline PRSD tank sets, additional tank sets can be installed with cost and weight penalties. The additional sets are called kits, with kit one being the fourth PRSD tank set. The baseline PRSD system is

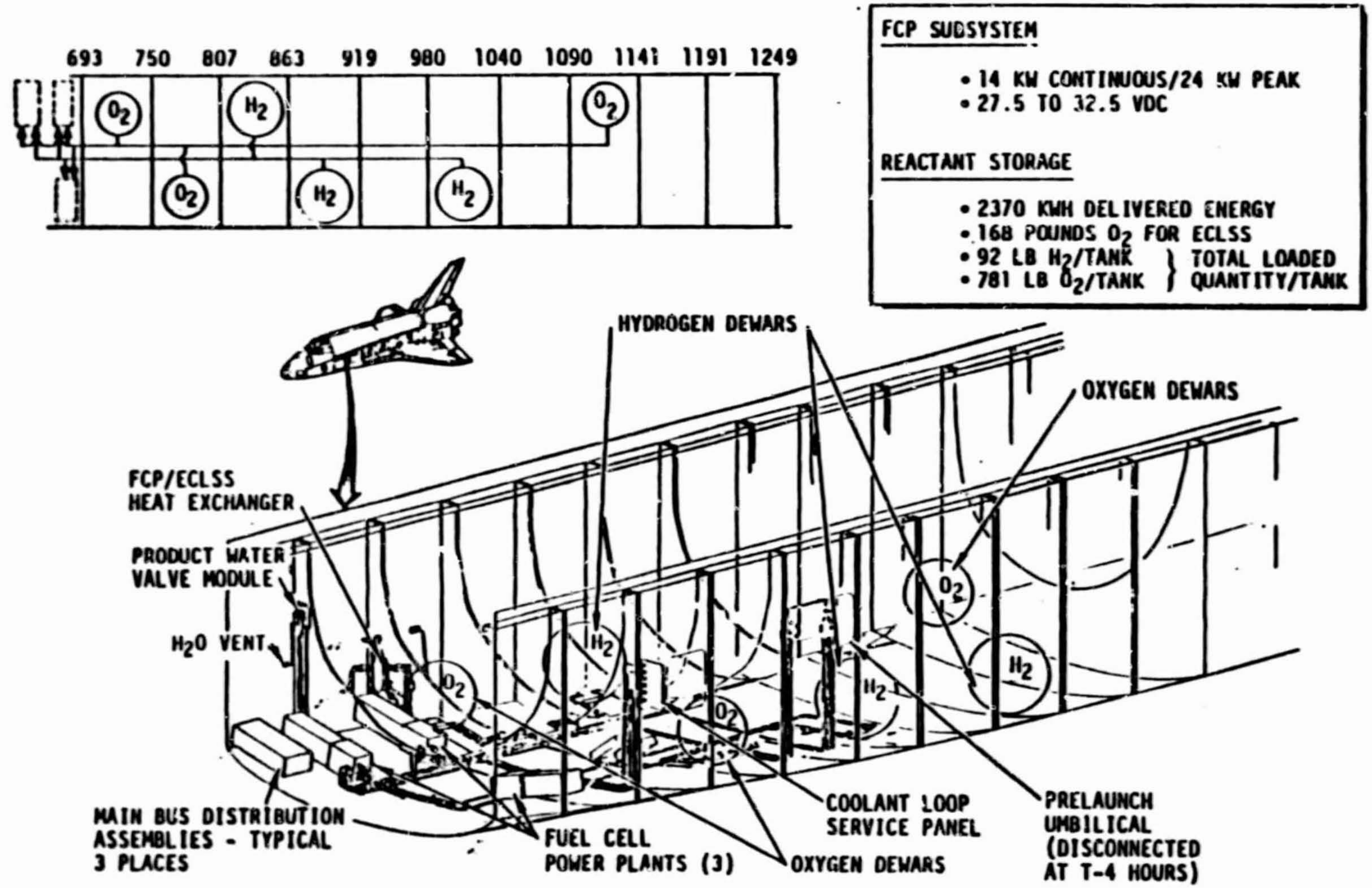


Figure 4.10-1. Electrical Power Subsystem

Table 4.10-1. Checklist: Orbiter Conntraints on Use of Electrical Power for Large Space Structure Operations

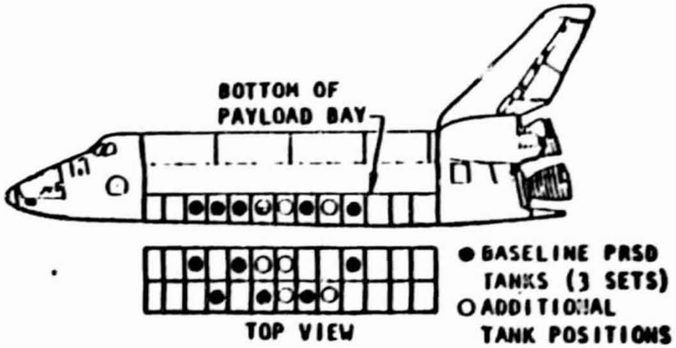
( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) During 10 minute launch-to-orbit phase and the 30-minute deorbit-to landing phase; 1000 watts average to 1500 watts peak are available to payloads in the cargo bay (R-1)	Rockwell, 1980 (b): SSV80-1 NASA, 1980 (a); JSC-07700, Vol XIV
( ) During payload equipment operations on orbit (e.g., large space construction) the average power demand by payloads shall not exceed 7,000 watts (with radiator kits installed), and peak power demand shall not exceed 12,000 watts for more than 15 minutes every three hours. (R-1)	Rockwell, 1980 (b): SSV80-1 NASA, 1980 (a): JSC-07700 Vol XIV Affects scheduling of high-power units for construction or checkout operations.
• Maximum Power usage is constrained by orbiter heat rejection capability (See Section 4.13). • Power may be drawn from mid-bay connector or/and aft bulkhead connectors, but maximum power cannot be exceeded. • If single connector, only the mid bay connector is capable of full load.	Orbiter orientation, radiator structure blockage by large space structure and heat rejection required to cool payloads all affect maximum power usage.
( ) Without the payload radiator kit available, power is 6000 watts maximum continuous, with 12,000 watts peak for 15 minutes every three hours. (R-1)	Rockwell, 1980 (b): SSV80-1
( ) Each of the two aft bulkhead interfaces can provide up to 1500 watts continuous and 2000 watts peak power (R-1)	Rockwell, 1980 (b): SSV80-1
( ) Buses must be isolated on the payload side of the interface except for the auxiliary buses. (R-1)	Rockwell, 1980 (b): SSV80-1
( ) If a fuel cell failure occurs during on-orbit operation, the cargo power will be supplied from the orbiter bus and time-shared with orbiter subsystems at a reduced level of 5 kw continuous and 8 kw peak (R-1)	Rockwell, 1980 (b): SSV80-1

Table 4.10-1 (continued) Checklist:

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) A payload may experience a total loss of power if operating on a dedicated fuel cell which fails, or if the fuel cell powering the orbiter bus connected to the payload fails. (R-1)	Rockwell, 1980 (b): SSV80-1
• Power is manually restored to the bus by the orbiter crew within a maximum of five minutes.	
( ) The nominal, or baseline fuel tanks complement is three tank sets, and the corresponding nominal payload energy available is 40kwh for the standard payload user charge. Equivalent mission duration is about 6.8 days. (R-1)	Rockwell, 1980 (b): SSV80-1 Additional cryo tank sets may be required for sufficient energy to construct large space structures (see Text).
• Duration is dependent on basic orbiter power requirements	
( ) As payload energy allocation is increased, mission duration declines to about 4.4 days if the payload uses 7 kw continuously (R-1)	Rockwell, 1980 (b): SSV80-1 Implies trade of energy /power usage vs mission duration
• Duration is dependent on basic orbiter power requirements	

ENERGY AVAILABLE WITH THREE PRSD TANK SETS*:	2370 kWh
NOMINAL ASCENT, DESCENT & DISPERSION ENERGY REQD:	270 kWh
NOMINAL ON-ORBIT ORBITER ENERGY REQUIRED/DAY**:	300 kWh
ENERGY AVAILABLE FROM EACH ADDITIONAL PRSD KIT:	840 kWh

*AFTER ACCOUNTING FOR 96-HOUR CONTINGENCY RESERVES
**SORTIE TYPE MISSION



KIT WEIGHTS (LB)		
KIT NO.	1	2
LANDED EXPENDED	473 844	959 844
TOTAL	1317	1803

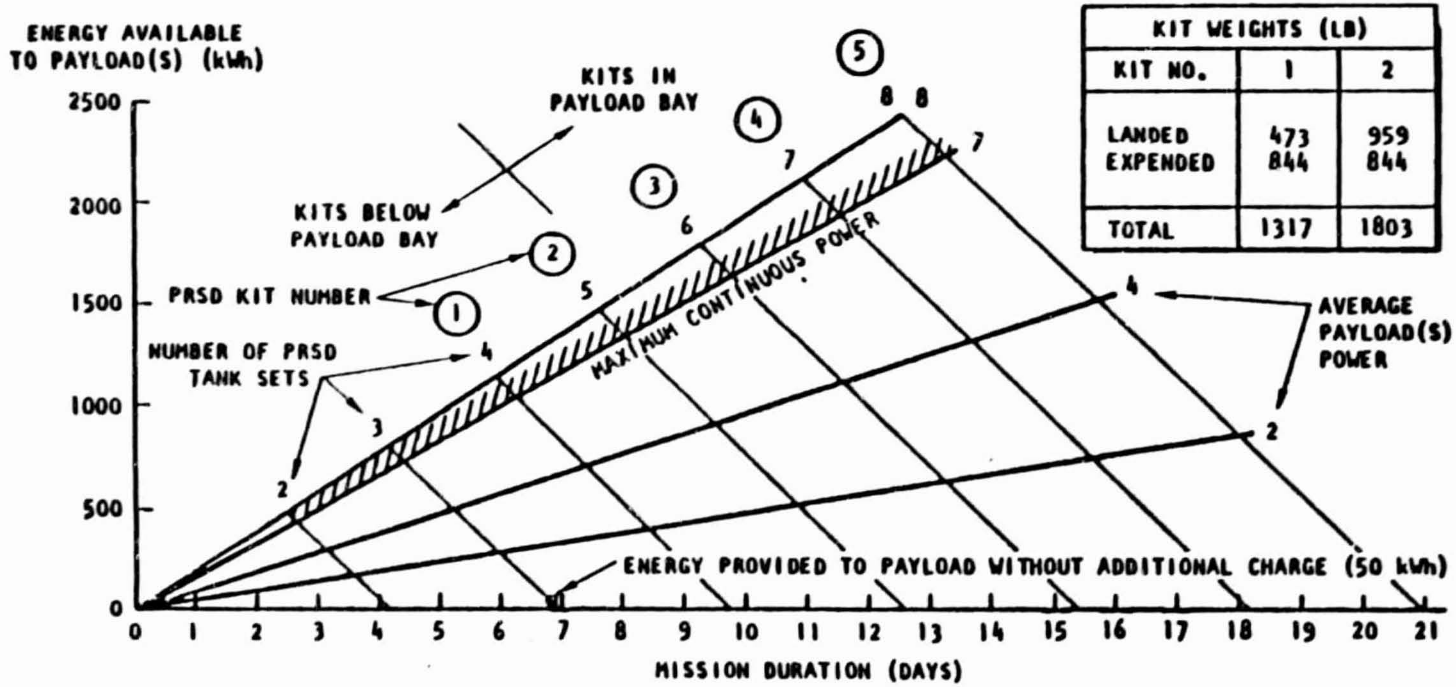


Figure 4.10-2. Energy/Power Available to Payload(s)

designed to accommodate kit one. The addition of tank set five and beyond will require new development.

The number of tank sets that can be accommodated beneath the cargo bay is limited to five. However, current conceptual design studies suggest that when more than four PRSD tank sets are needed, the additional sets should be placed in the cargo bay in clusters of up to four sets each. Penalties for this approach to energy extension include reduction of useful cargo bay length as well as ascent and descent payload weight and cost.

Another concept for extending orbiter energy is through the addition of a solar array that can be deployed on-orbit to supplement fuel cell/PRSD delivery energy. Orbiter flight durations with such a system could exceed 30 days, provided the mission is flown at high β angles. The concept under study is called Power Extension Package (PEP).

4.10.1 EPS - Fuel Cell Subsystem Schematic

Three low-temperature fuel cell powerplants (FCP), generating power through the electrochemical reaction of hydrogen and oxygen, supply the primary in-flight electrical power used by the Shuttle. The fuel cell schematic is shown in Figure 4.10-3. Each powerplant is capable of providing direct current energy at 27.5 to 32.5 volts over a power range of 2 to 12 kilowatts.

Heat generated by the electrochemical conversion of the reactants is removed from the FCP's by heat exchangers in the ECLSS. Potable water, produced by the reaction, is also used by the ECLSS.

4.10.2 Power Reactant Storage and Distribution Subsystem

The power reactant storage and distribution subsystem (PRSD) contains the components which supply oxygen and hydrogen to the fuel cell powerplants (FCP) and oxygen to the ECLSS during all mission phases. The Power Reactant Storage and Distribution Subsystem is shown in Figure 4.10-4. The cryogenic reactants are stored in a supercritical condition in insulated double-walled tanks with a vacuum annulus. The PRSD supplies oxygen at a nominal pressure of 900 psia and hydrogen at 250 psia, in a single-phase gaseous state, to the FCP which regulates pressures to a normal range of 55 to 65 psia. Automatic controls, activated by pressure, energize internal tank heaters which add heat energy to the reactants to maintain pressure during depletion. The heaters are controlled by the instrumentation and control subassembly to provide equal depletion of the first and second oxygen and first and second hydrogen tanks. Each tank has relief valves to prevent overpressurization from abnormal operating conditions.

Each tank has the following distribution components: a filter, check valve,solenoid valves, and relief valves that are mounted as modules on two panels adjacent to each tank. Redundancy is provided by having two components

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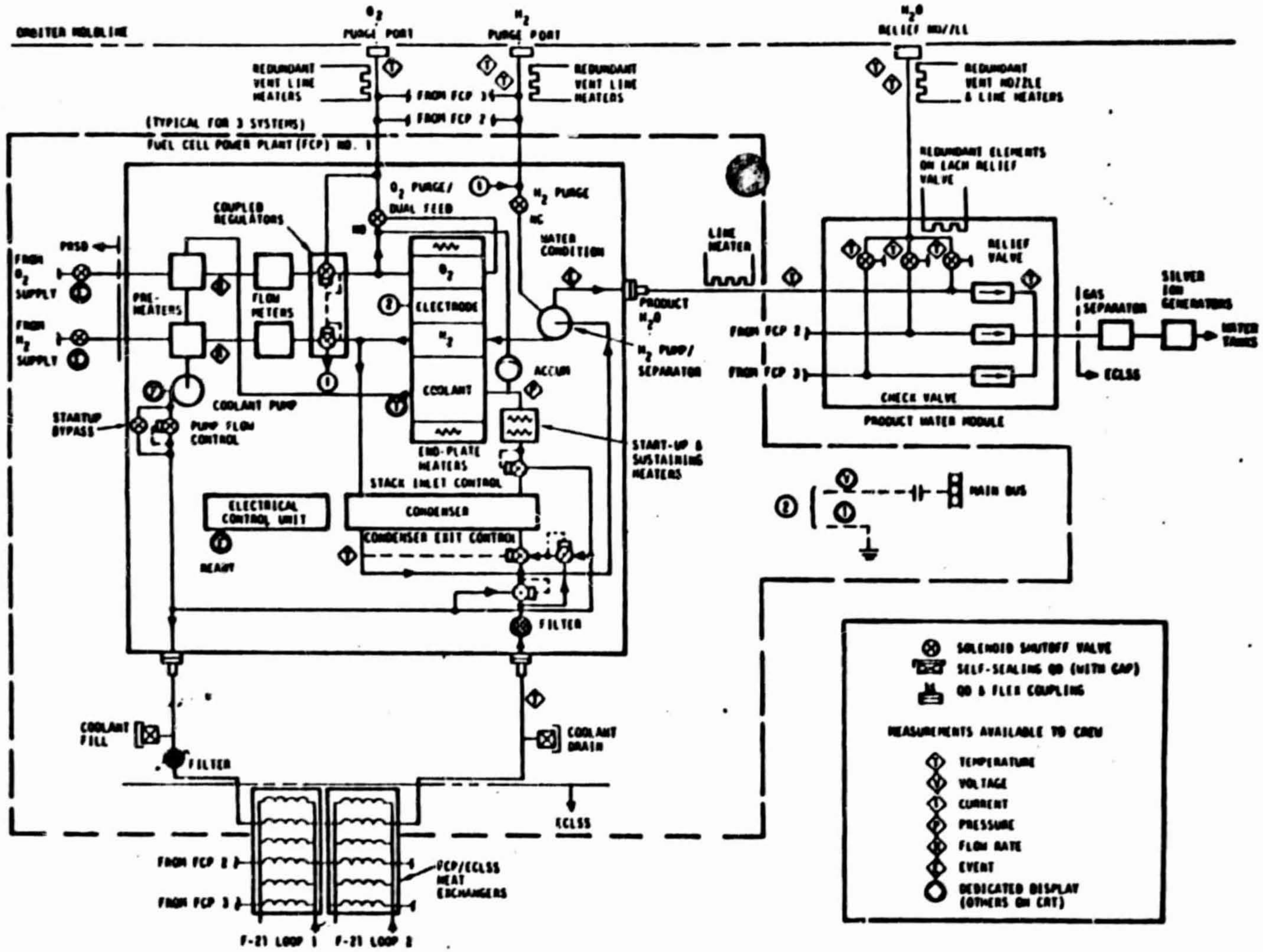


Figure 4.10-3. EPS - Fuel Cell Subsystem Schematic

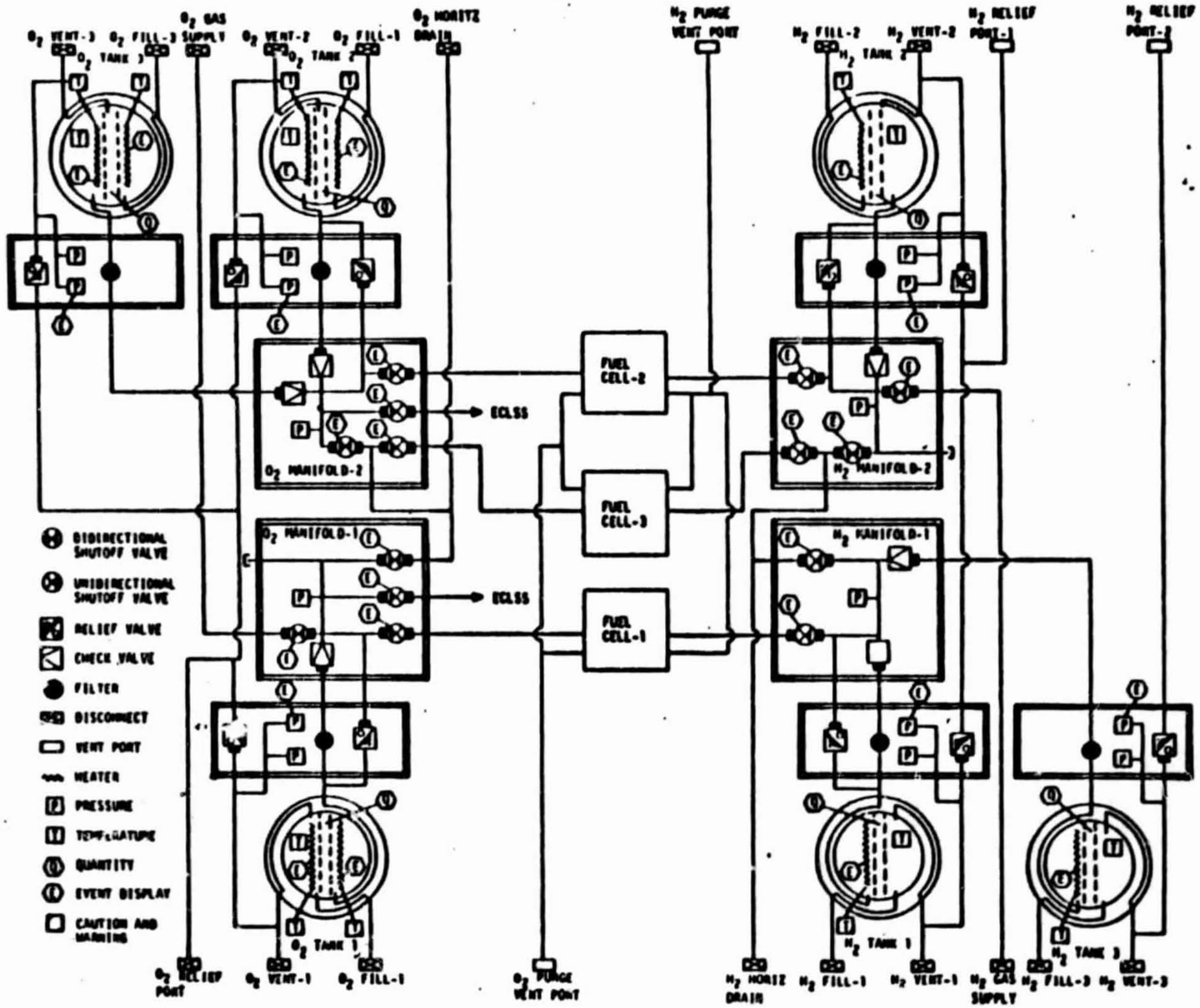


Figure 4.10-4. Power Reactant Storage and Distribution Subsystem Schematic

for each major function or by providing manual override for the automatic controls. The distribution shutoff valves are arranged so that any tank can be used to feed any of the subsystems, or, in case of failure, can be isolated at the distribution manifolds.

Approximately 276 pounds of hydrogen and 2343 pounds of oxygen are stored in the three sets (one hydrogen and one oxygen tank per set) of storage dewars. Each of the oxygen dewars carries, in addition to the fuel cell energy requirement, 36 pounds of oxygen for the orbiter ECLSS. The tanks also carry a reserve for emergency requirements.

4.11 SHUTTLE COMMUNICATIONS, TRACKING, AND DATA MANAGEMENT

Voice, television, and data-handling capabilities of the Orbiter support onboard control of the payload, or, when desirable, remote control from the ground. The Orbiter communications and tracking subsystem provides links between the Orbiter and the payload. It also transfers payload telemetry, uplink data commands, and voice signals to and from the space networks. The provisions in the Orbiter for communications, tracking, and data management are flexible enough to accommodate most payloads.

Links through the Orbiter are outlined in table 4.11-1.

Table 4.11-1

Orbiter avionics services to payloads

Function	Direct or through Tracking and Data Relay Satellite		Hardline		Radiofrequency link	
	Payload to ground via Orbiter	Ground to Payload via Orbiter	Orbiter to attached payload	Attached payload to Orbiter	Orbiter to detached payload	Detached payload to Orbiter
Scientific data	x	x		x		
Engineering data	x	x		x		x
Voice	x	x	x	x	x	x
Television	x		x	x		
Command		x	x		x	
Guidance, navigation, and control		x	x	x	x	
Caution and warning	x			x		x
Master timing			x			
Rendezvous					x	x

Data Source: STS User Handbook (NASA 1977)

The data processing and software subsystem of the Orbiter, described further in Section 4.12, furnishes the onboard digital computation necessary to support payload management and handling. Functions in the computer are controlled by the mission specialist or a payload specialist through the main memory loads from the tape memory. The stations in the Orbiter aft flight deck for payload management and handling are equipped with data displays, CRT's, and keyboards for onboard monitoring and control of payload operations.

4.11.1 Communications and Tracking Subsystem

A schematic of the Communications and Tracking Subsystem is shown in Figure 4.11-1. On operational missions, the Communications and Tracking (C&T) Subsystem can perform the following functions.

1. Selects and maintains operationally required RF communication links to support orbiter missions and processes and distributes command signals and data received for use by the supporting avionics.

2. Coherently returns RF communications link carriers for two-way doppler velocity tracking by ground stations, recovers received carrier signals for one-way doppler extraction, and provides a turnaround ranging tone modulation to the ground during ascent, reentry, and landing operations.

3. Provides for location, operation, and interfacing of government-furnished equipment (GFE) decryptors and encryptors for processing (a) voice and data from and to ground stations directly for DOD missions and (b) data received from and transmitted by detached DOD payloads.

4. Generates RF NAVAID information and air traffic control (ATC) voice for atmospheric flight.

5. Provides audio-voice communications among crew stations within the orbiter, to attached manned payloads via hardwire, to ground stations, and to manned released payloads via RF links.

6. Generates, transmits, and distributes closed-circuit television (CCTV) and generates and transmits color TV or CCTV to the ground via the RF link.

7. Acquires and tracks passive and cooperative targets for supporting rendezvous.

8. Acquires, tracks, receives data from, and transmits data to, NASA tracking and data relay satellites (TDRS).

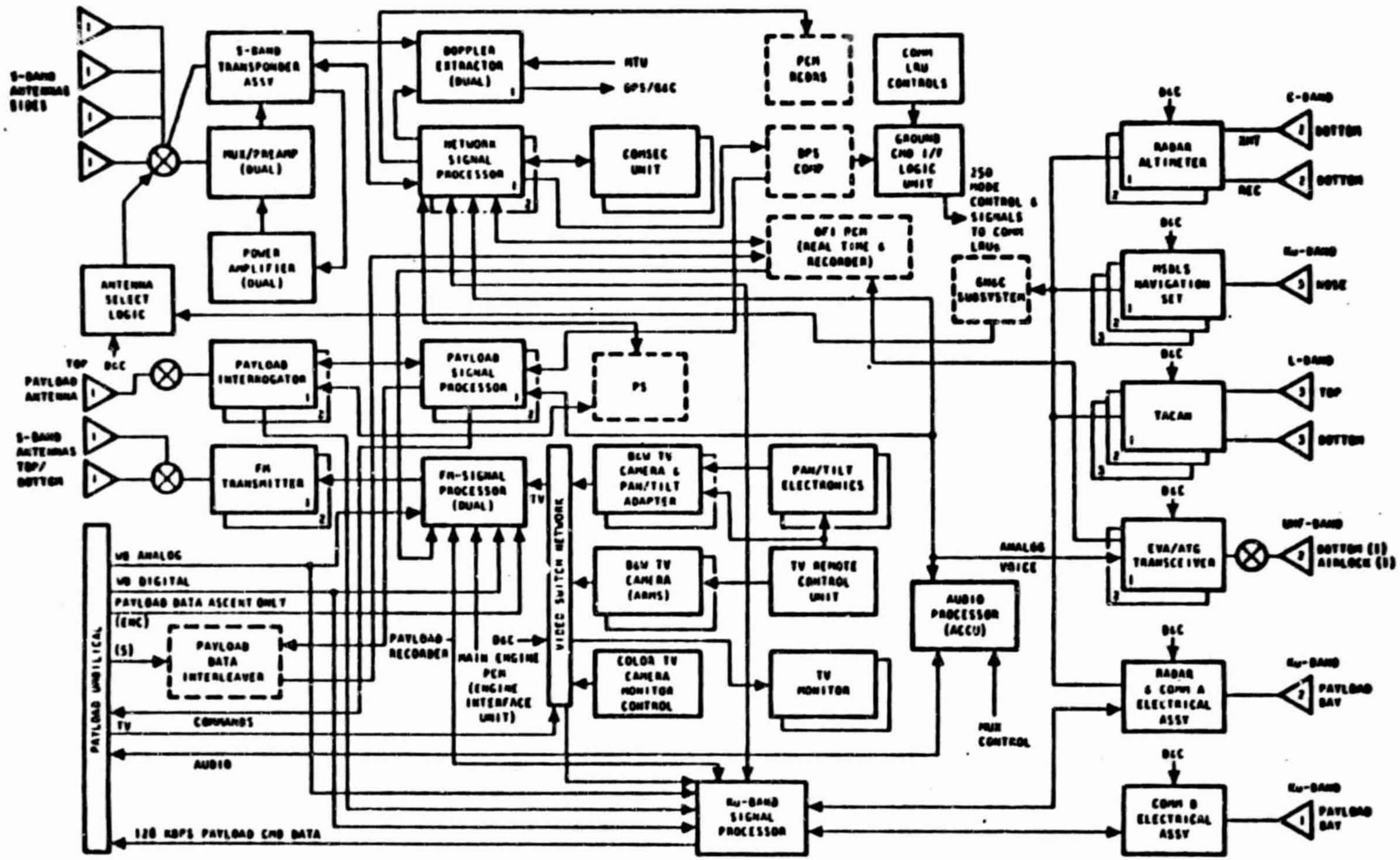


Figure 4.11-1. Communications and Tracking Subsystem Schematic

4.11.2 Radio Frequencies

The orbiter carries up to 23 antennas for communication with ground stations, detached payloads, and crewmen undertaking extravehicular activities (EVA). Through these antennas, information is both transmitted and received at S-, Ku-, L-, C-, and P-band frequencies, see Table 4.11-2.

Phase-modulated (PM) data can be transmitted to STDN or SCF ground stations on two S-band carrier frequencies. Two additional S-band carrier frequencies are employed for ground-to-orbiter transmission of FM data. In addition, frequency-modulated (FM) signals from the orbiter to ground go on an S-Band carrier, as does the FM data from the development flight instrumentation (DFI) carried aboard the Shuttle test flights. Multichannel two-way communication between the orbiter and attached or detached payloads is accomplished within the S-band regime. The intelligence carried on these channels includes scientific and engineering data, commands, digital voice, C&W and performance monitoring information, and video signals.

The Ku-band link between ground stations and the orbiter via the tracking and data relay satellite system (TDRSS) carries the same kinds of intelligence as the S-band subsystem, but at wider bandwidths and higher data rates. The orbiter rendezvous radar and the microwave scan beam landing system (MSBLS) also work in the Ku-band.

During orbiter approach and landing phases of a mission, standard L-band TACAN units will be employed, as well as C-band radar altimeters and P-band analog voice links for air traffic control (ATC). EVA voice communications will also be at P-band frequencies.

4.11.3 S-Band Information Transmission Combinations

Various combinations of voice, commands, and telemetered data as shown in Table 4.11-3 can be sent to or from the orbiter over S-band transmission links. Phase-modulated (PM) signals, beamed directly to the orbiter from STDN or SCF stations or relayed through the TDRS, can be transmitted at two different bit rates. In the high bit-rate mode, two digital voice channels at 32 kbps per channel, and 8 kbps of command data, are interleaved into a 72-kbps digital data stream. The low bit-rate mode consists of one 24 kbps digital voice channel plus the 8 kbps of command data. Transmission through the TDRS are convolutionally encoded and have a spread spectrum signal with a rate of 11.2 megachips.

Two bit-rate modes are available for the transmission of PM carrier signals from the orbiter directly to SCF or STDN ground stations or by relay through the TDRS. The high bit-rate mode will accept two channels of digital voice at 32 kbps per channel, interleaved with 128 kbps of telemetered information to form a 192-kbps digital data stream. Up to 64 kbps of the telemetry data can be from a payload. In the low bit-rate mode, one channel of digital voice plus 64 kbps of telemetry are interleaved for transmission. Transmissions through the TRDS are convolutionally coded.

Table 4.11-2 Radio Frequencies

FUNCTION/SYSTEM	ORBITER TRANSMIT	ORBITER RECEIVE
STDN PM-1	2287.5 MHz	2106.4 MHz
STDN PM-2	2217.5 MHz	2041.9 MHz
STDN/FM	2250.0 MHz	NONE
DFI FM	2205.0 MHz	NONE
NASA PAYLOADS	2025.0 TO 2120.0 MHz	2202.5 TO 2297.7 MHz
DOD PAYLOADS	1760.0 TO 1843.0 MHz	2202.5 TO 2297.7 MHz
EVA COMMUNICATIONS	296.8 MHz	259.7 MHz
RENDEZVOUS (RADAR)	13.679 - 13.887 GHz	13.679 - 13.887 GHz
Ku-BAND COMM	15.0034 GHz	13.775 GHz
RADAR ALTIMETERS	4.3 GHz BAND	4.3 GHz BAND
TACAN	1025 TO 1150 MHz	962 TO 1213 MHz
ATC VOICE	296.8 MHz	259.7 MHz
MSBLS	Ku-BAND	Ku-BAND

Table 4.11-3. S-Band Information Transmission Combinations (KBPS)

RF LINK	VOICE 1	VOICE 2	** CMD	** TLM	TOTAL UNCODED	CHANNEL ENCODING	TOTAL CODED	SPREAD SPECTRUM
TO ORBITER FROM GROUND								
STDN DIRECT								
HIGH BIT RATE	32	32	8	-	72	NO	-	NO
LOW BIT RATE	24	-	8	-	32	NO	-	NO
STDN RELAY - TDRS								
HIGH BIT RATE	32	32	8	-	72	YES	216	YES
LOW BIT RATE	24	-	8	-	32	YES	96	YES
SCF DIRECT				-				
HIGH BIT RATE	32	32	8	-	72	NO	-	NO
LOW BIT RATE	24	-	8	-	32	NO	-	NO
TO ORBITER FROM PAYLOAD								
UNMANNED NASA	-	-	-	16	16	NO	-	NO
UNMANNED AF	-	-	-	16	16	NO	-	NO
FROM ORBITER TO GROUND								
STDN DIRECT								
HIGH BIT RATE	32	32	-	128	192	NO	-	NO
LOW BIT RATE	32	-	-	64	96	NO	-	NO
FM	-	-	-	*	*	NO	-	NO
STDN RELAY - TDRS								
HIGH BIT RATE	32	32	-	128	192	YES	576	NO
LOW BIT RATE	32	-	-	64	96	YES	288	NO
SCF DIRECT								
HIGH BIT RATE	32	32	-	128	192	NO	-	NO
LOW BIT RATE	32	-	-	64	96	NO	-	NO
FM (ENCRYPTED)	-	-	-	256	256	NO	-	NO
FROM ORBITER TO PAYLOAD								
UNMANNED NASA	-	-	2	-	2	NO	-	NO
UNMANNED AF	-	-	2***	-	***2	NO	-	NO

*4.5 MHz, MAXIMUM **MAXIMUM RATE ***TERNARY FORMAT

The orbiter can also transmit frequency-modulated (FM) S-band signals directly to the ground. Payload and orbiter data with bandwidths up to 4.5 MHz can be transmitted to STDN stations. The data can include recorded voice, PCM and main engine data, real time CCTV, and digital or wideband analog data from payloads. On DOD missions, FM transmissions to SCF ground stations can consist of recorded voice and encrypted PCM data, 256 kbps of encrypted payload data, and unencrypted main engine data.

The orbiter can transmit or relay a 2-kbps command signal to attached or detached NASA payloads.

Commands to free-flying DOD payloads are sent at a 1 or 2 kiloband rate by using a ternary frequency shift keyed (FSK), amplitude modulated signal. A 500 or 1000 Hz synchronization signal is provided as the amplitude modulated signal.

Attached or detached payloads can transmit encrypted or unencrypted data to the orbiter.

4.11.4 Ku-Band Radar/Communication Subsystem

The orbiter Ku-band radar/communication subsystem is packaged in two sets of assemblies. One set, Radar/Comm A, is carried aboard the orbiter as standard equipment. Radar/Comm A consists of a deployable antenna assembly and an electrical assembly located in avionics Bay 3A. The antenna is mounted on the starboard payload bay door longeron at Station X_o 589. During ascent, the antenna is stowed in the space between the payload bay door radiator panels and the 15-foot-diameter payload bay clear volume. The antenna is deployed outboard of the orbiter mold line after the payload bay doors have been opened. See Section 4.4.1 for clearance envelopes. Figure 4.11-2 summarizes key information concerning Ku-band antennas. Further discussion follows:

In the radar mode, the Ku-band subsystem can detect, acquire, and automatically track at a range of 19 kilometers passive targets with equivalent radar cross-sections of one square meter and Swerling Case 1 scintillation characteristics. The maximum tracking range increases to 560 kilometers when the target is equipped with an appropriate beacon transponder. The radar can acquire a target in 60 seconds, or less, after being directed along the expected target vector, and can provide line-of-sight (LOS) range to the target, range rate, angles relative to the orbiter rendezvous axis (-Z axis), and angle rates, from the maximum range down to a minimum range of 30 meters.

The Comm A Ku-band communications unit shares the radar antenna and pedestal. Two different modes are available for the tranmission of data to the ground through tracking and data relay satellites (TDRS). In Mode 1, up to 50 megabits per second (mbps) of wideband data from an attached payload, plus up to 2 mbps of operational,stored, or experiment data, and 192 kbps operational data can be transmitted. Mode 2 transmissions can consist of 4.5 mbps of analog (TV) data from either the

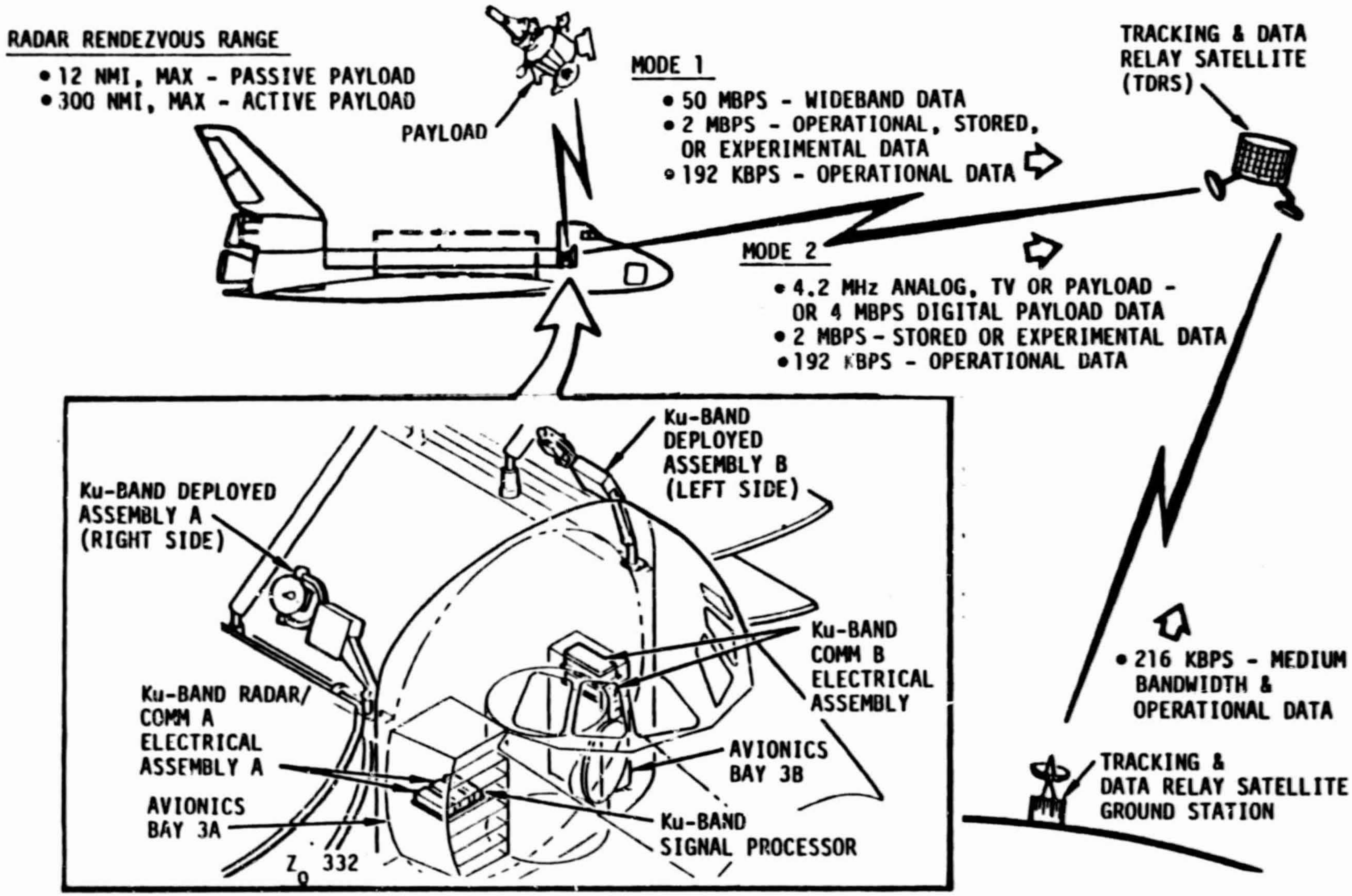


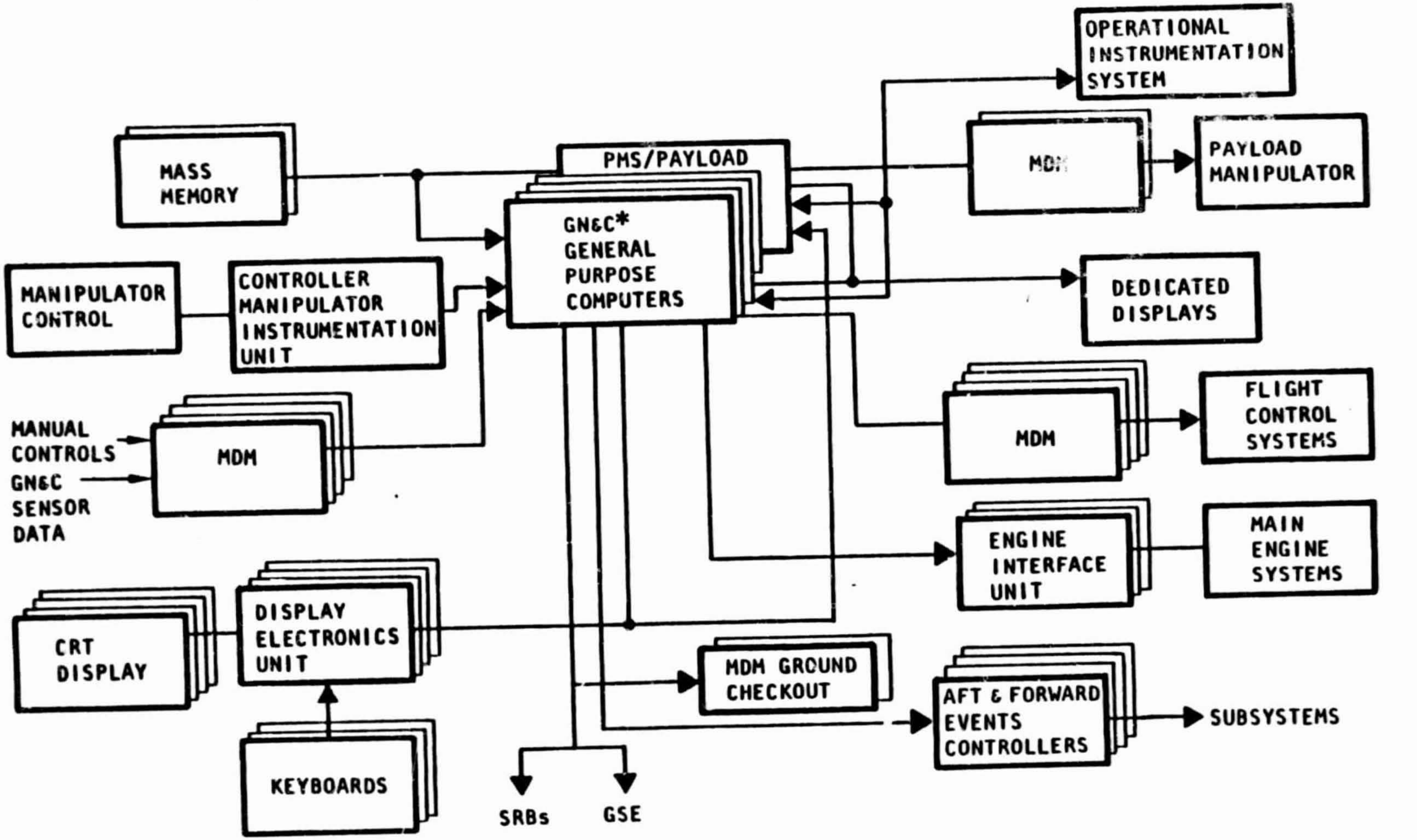
Figure 4.11-2. Summary of Ku-Band Communications

orbiter or an attached payload, or 4 mbps of payload digital data, plus up to 2 mbps of stored or experiment data and 192 kbps of real-time operational data from the orbiter. The forward (ground-to-orbiter) link can carry 216 kbps of data, consisting of 72 kbps of operational data for the orbiter, 128 kbps for the payload, and 16 kbps of overhead. The forward link can also have 72 kbps of encrypted data convolutionally coded at a 3 times rate as a special mode of operation.

If continuous Ku-band data transmission is required over more than 40 percent of an orbiter orbit, then a second set of Ku-band communication assemblies is required. This assembly kit (Comm B) consists of a deployable antenna mounted on the port payload bay door longeron, plus electrical units housed in avionics Bay 3B. This 263-pound kit is part of the payload weight allowance. With this kit, the orbiter can communicate with both satellites, sequentially, without disrupting the flow of data during the satellite handover sequence.

4.12 DATA PROCESSING AND SOFTWARE SUBSYSTEM

The orbiter data processing and software subsystem (DP&S) provides data processing capabilities for guidance, navigation, and control (GN&C); communications and tracking (C&T); displays and controls (D&C); system performance monitoring; payload management; payload handling; subsystem sequencing; and selected ground functions with sufficient provision so as not to limit the performance of the subsystems which it supports. The DP&S accepts input commands and/or data from the crew, on-board sensors, and external sources; performs computations and processing; and generates output commands and data as necessary to accomplish the requirements specified for GN&C, C&T, D&C, instrumentation, electrical power distribution and control, computers, performance monitor function, and payload handling and management. A schematic of the subsystem is shown in Figure 4.12-1.



*FOUR COMPUTERS DEDICATED TO GN&C DURING CRITICAL FLIGHT PHASES. ONE OR MORE CAN BE RECONFIGURED FOR OTHER USES DURING NONCRITICAL FLIGHT PHASES

Figure 4.12-1. Data Processing and Software Subsystem Block Diagram

The DP&S equipment configuration is organized around a computer complex consisting of five general-purpose computers which are interconnected so that they may be operated in redundant groups for critical services. Memory capacity of each computer is 104,000 thirty-two bit words. Additional storage of programs and fixed data is provided by two mass memory units having a data capcity of 134 megabits each.

Data transfer between the computer complex and data users employs a data bus network composed of serial, half-duplex data channels operating at one megabit per second.

Interface adaptation between the data bus network and the orbiter subsystems is accomplished by multiplexer/demultiplexer (MDM) units. These units provide signal conversion capability, digital-to-analog as well as analog-to-digital, in addition to the multiplexing/demultiplexing function.

Engine interface units provide operational control of the main engines from GN&C commands or, during ground checkout, via ground checkout MDM's, and provide response data back to the GN&C. The units also provide main engine data for recording, telmetry, or GSE.

4.12.1 Caution and Warning

The key constraint in regard to caution and warning (C&W) concepts for large space structures is that items which affect orbiter safety must interface with the orbiter computer, as well as provide signals to the crew.

The baseline C&W and performance monitoring interface beween a payload and the orbiter is described in Figure 4.12-2. Further descriptive material is provided below from Rockwell (1980 b). "Up to five warning parameters are hardwired from sensors on board the attached payloads to the orbiter C&W electronics. Movement of any of these parameters outside the limits pre-set in the C&W electronics will activate the master alarm tone, master alarm lights (4), and the payload warning annunciator on the C&W annunciator matrix. Up to 50 caution parameters (25 each, analog and digital) are hardwired to an MDM for limit sensing by a general-purpose computer (GPC). Up to five hardwire and up to 36 computer-controlled safing commands are provided to safe payload components."

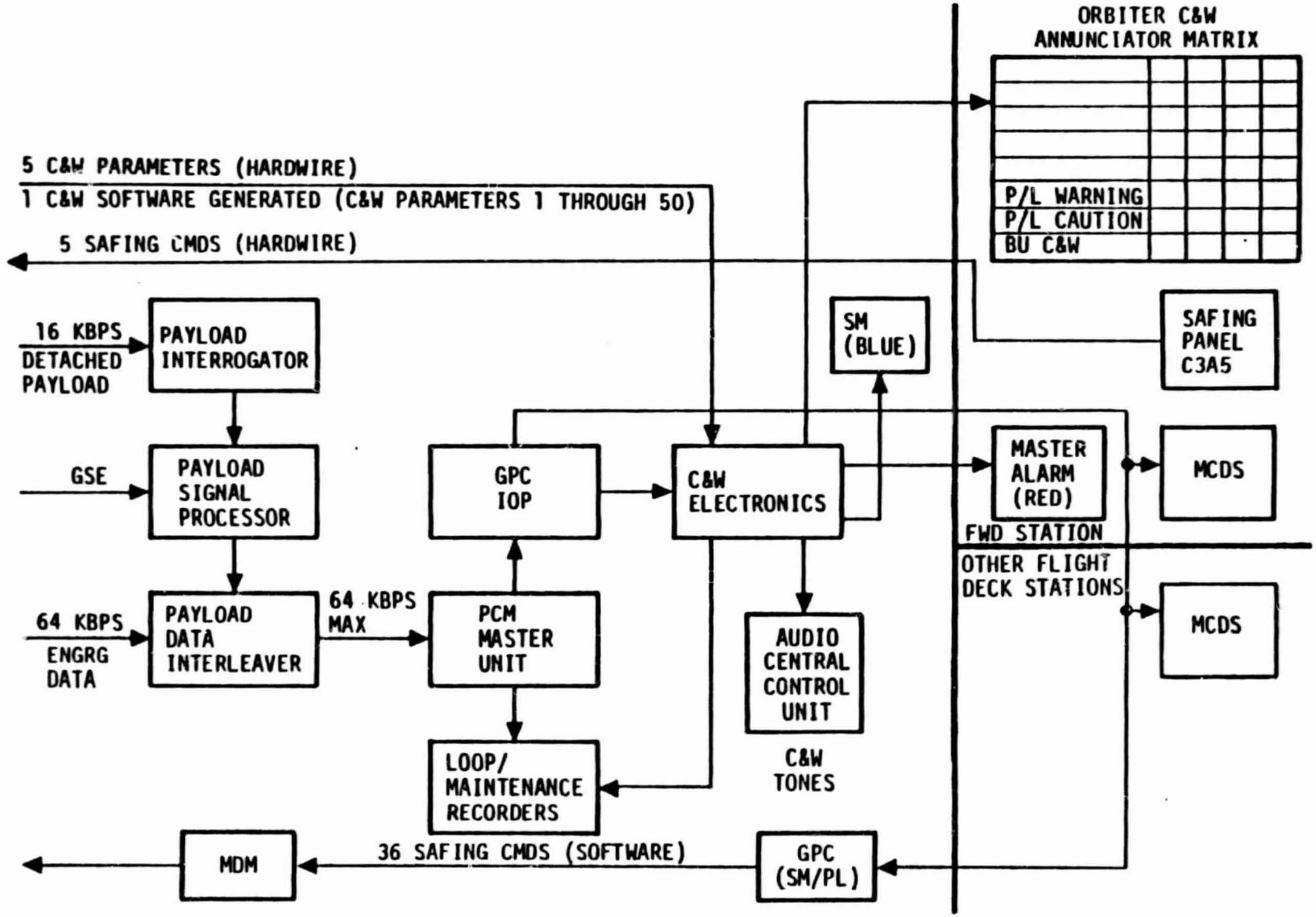


Figure 4.12-2. Orbiter/Payload C&W and Performance Monitoring Interface

"Detection by the general-purpose computer (GPC) of the violation of a pre-set limit for any of the 50 parameters will result in the issuance of a discrete output to the orbiter C&W electronics. Detection of the presence of the GPC-issued discrete signal will activate the master alarm tone, master alarm lights (4), and the payload caution annunciator on the C&W annunciator matrix. Data on these same parameters form part of the digital data stream from the payload to the orbiter performance monitoring function (PMF). Detected out-of-tolerance conditions will cause the backup caution and warning annunciator to illuminate. Potentially hazardous conditions in the attached payload should be evident from both the hardwired/software C&W parameters and the backup from the PMF; i.e., both the dedicated C&W annunciator and the backup annunciator will illuminate. Health and status information from additional payload sensors can also form a part of the telemetry data stream to the orbiter PMF. Unfavorable conditions appearing in any of these housekeeping data are signaled by an illumination of the system management (SM) indicator. The C&W tones, from the audio central control unit (ACCU), and the master alarm visual annunciator, via a switch closure, can be made available to an attached, manned payload. The PMF can also be employed to monitor the status of detached payloads."

"The capability for in-flight reset of a trip threshold for any hardwired C&W parameter is provided at the mission specialist station (MSS) using the C&W status board. A similar capability for the software backup is provided at the multifunctional cathode ray tube display system (MCDS). Payload C&W shall be limited to out-of-tolerance measurements which may, if uncorrected, lead to the loss of the orbiter vehicle and/or crew. Those payload parameters that cannot have an impact upon the orbiter or crew, even though the effect may result in compromising the payload, are not presently candidates for the orbiter C&W system."

4.12.2 Software Interfacing Concept

Development of software concepts for interfacing large structures to the orbiter has just entered the conceptual stage. However, a few guidelines have been suggested by inferences derived from other payload integration studies. These preliminary general guidelines are listed in Table 4.12-1.

Table 4.12-1. Checklist: Guidelines for C&DH Software

( ) GUIDELINES/QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) A principal criterion for development of payload software is to design for a central C&DH computer which handles all routine construction/payload operations/housekeeping functions, and works with orbiter or other computers to the minimum degree feasible when the orbiter is berthed to the large space structure. (R-4)	Affects design of lectronic data handling systems and software development for large space structure construction and operations.
( ) The principal construction/project or orbiter computer should act as a "traffic cop" for complex project/payload systems which have their own mini-computers. (R-4)	
( ) Interdependence of software between orbiter and construction/operation of large space structures should be minimized to avoid costs of changes in the orbiter software. (R-4)	

4.13 ORBITER THERMAL REQUIREMENTS AND ENVIRONMENTS

The Shuttle orbiter requires an effective heat rejection capability in order to generate electrical power and possibly to dissipate payload heat. The primary path for heat rejection while on orbit is by way of the radiators mounted on the inner surface of the payload bay doors, as illustrated in Figure 4.13-1. The figure also shows a simplified radiator subsystem schematic. Unfortunately, the most readily reachable and visible volumes for space construction are right above the radiators, where significant blockage of the heat radiation is a potential problem. Therefore, the designer of large space structures must evaluate the possible loss in orbiter heat rejection capability due to the large space structure and determine if other means for heat rejection must be introduced. Possible examples of other means for heat rejection could be water boiling (a weight penalty) or auxiliary, separate radiators on payloads with consequent significant design, integration, and weight impacts on the construction project.

Another related consideration is the orbiter/construction orientation with respect to the earth, sun, and deep space. Obviously, if the radiators can be facing space rather than earth, they are more effective as heat rejectors. However, this requirement may conflict with other orientation requirements discussed in Section 4.2.

Table 4.13-1 summarizes the heat rejection capability available to payloads, as described in the STS User Handbook (NASA 1977 (a)).

Note that prelaunch and post-landing thermal control is provided by ground support systems. The payload heat exchanger is designed so either water or Freon 21 can be selected as a cooling fluid, according to the needs of the payload. The payload side of the heat exchanger has two coolant passages; either or both can be used. Coolant is provided to the payload at 40° to 45°F (278 to 280 K). Fluid circulation through the payload side of the heat exchanger must be supplied as part of the payload. A water flash evaporator is used to supplement the radiator cooling capacity. During ascent and descent, when the cargo bay doors are closed and the radiators are ineffective, cooling is provided by the water boilers at altitudes above 140,000 ft. During orbiter entry and post-landing, orbiter cooling requirements below 100,000 ft altitude are satisfied by an ammonia boiler until GSE cooling is connected.

The orbiter also provides 48 ft^3/min of conditioned air for removal of metabolic sensible and latent heat and CO_2 of four crew members while working in a manned payload module.

A portion of the orbiter cabin heat load may be contributed by payload-supplied displays and controls mounted on the port side (as shown in Figure 3.2-4). Such heat rejection is accomplished by drawing cabin air through the avionics boxes. This heat rejection capability at the payload station is limited to a maximum of 750 W (2560 Btu/hr) average, and 1000 W (3413 Btu/hr) peak for 15 minutes once every three hours.

Table 4.13-2 summarizes the above concerns in terms of general payload heat rejection capabilities and guidelines. The following discussion presents some representative magnitudes of heat rejection due to blockage of structures above the payload bay.

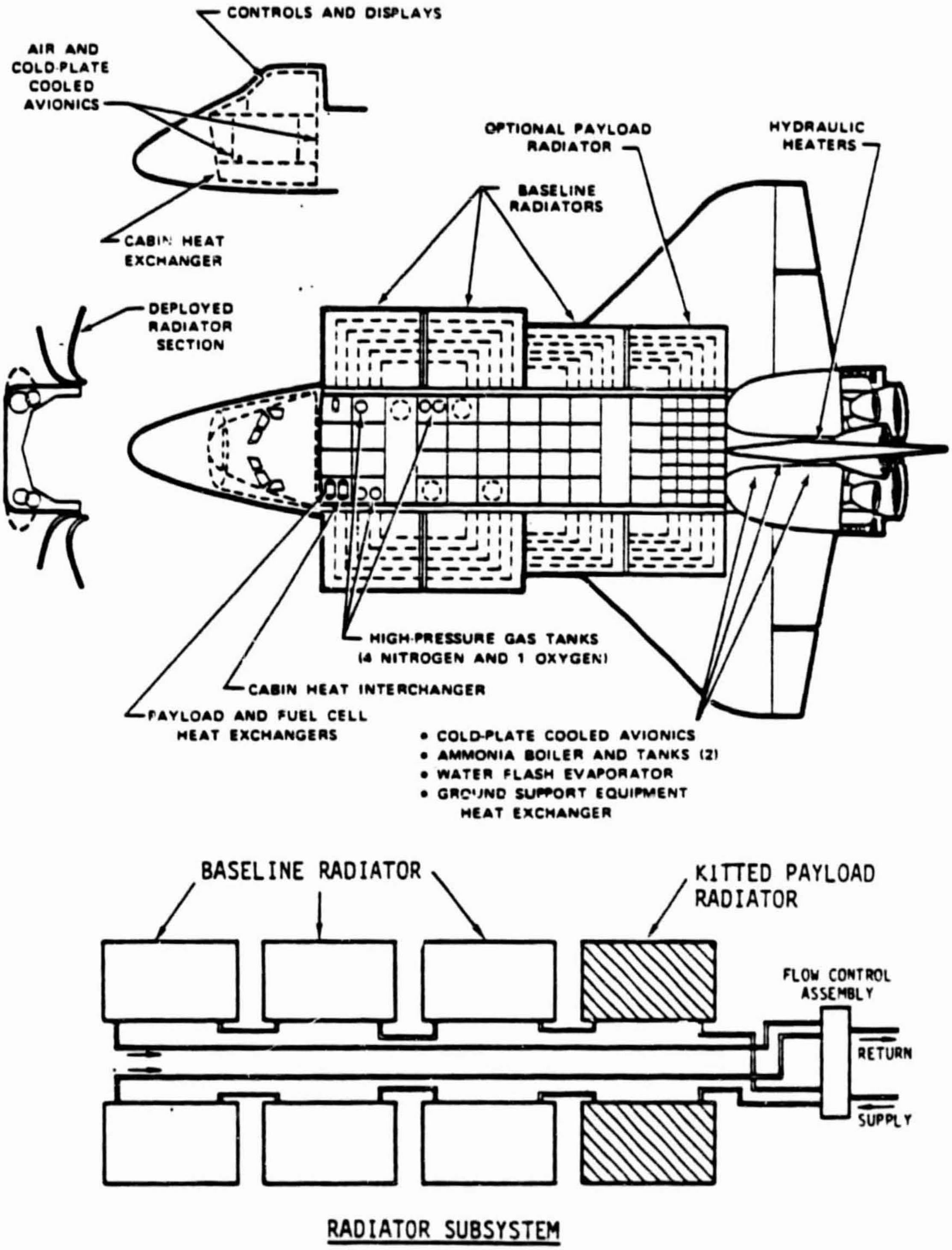


Figure 4.13-1. Orbiter Environmental Control Subsystem

Table 4.13-1. Payload Heat Rejection Available

Flight Phases	Capability (kW)		Description
	Aft Flight Deck	Cargo Bay	
Prelaunch, ascent, descent, post-landing (cargo bay doors closed)	0.35 0.42	1.52 NA	Average 2-min peak
On orbit without radiator kit (cargo bay doors open)	0.75 1.00 0.35	5.90 5.65 6.30	Average Peak for 15 min each 3 hr Minimum for aft flight deck, maximum for cargo bay
On orbit with radiator kit (cargo bay doors open)	0.75 1.00 0.35	8.10 7.85 8.50	Average Peak for 15 min each 3 hr Minimum for aft flight deck, maximum for cargo bay

Table 4.13-2. Checklist: Guidelines for Optimizing Orbiter Heat Rejection During Operations near Large Space Structures

( ) GUIDELINES/• QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) Large space structures should be designed and constructed in locations such as to minimize blockage of orbiter radiators on the payload bay doors. (R-1) • Minimize solid surface area, use trusswork instead of solid webs. • Locate blocking surfaces as far above radiators as feasible (see Section 4.13.1). • Orient surfaces perpendicular to radiators rather than parallel to them.	Jelinek 1977 (b): ECLSS-190-77-008 Affects design of construction fixtures, structure, and orientation on orbiter.
( ) Where feasible, orbiter radiators should face deep space in preference to earth or sun. (R-4)	Affects attitude control of orbiter/space structure combination. Affects natural lighting on structure.
( ) If heat rejection by standard orbiter radiators on payload bay doors is inadequate, cooling can be accomplished by boiling water. (R-1) • Additional water is chargeable to orbiter payload weight. • Additional moisture in vicinity of large space structures hould be evaluated for potential contamination effects.	NASA 1980 (a): JSC 07700, Vol. XIV Affects design trades of payload weight and volume, rate of energy in construction.
( ) The orbiter has limited capability to reject payload generated heat. A separate, payload-mounted radiator may be more effective as a means to reject such heat than using orbiter radiators. (R-1)	NASA 1980 (a): JSC 07700, Vol. XIV See Section 3.9 for fluid interfaces.

4.13.1 Loss in Orbiter Radiator Heat Rejection caused by Construction Above Payload Bay

The following is a summary of analyses performed by Jelinek 1977 (b) to determine the loss in orbiter heat rejection capability due to construction above the payload bay and radiator. A set of preliminary curves were plotted to give the designer a feel for the impact of positioning beams or platforms above the payload bay in the process of construction. The curves included the shape of a rectangle going from an aspect ratio of 1, 5, 10 to 100. This rectangular shape should simulate a range of objects from platforms to long, narrow beams. Other variables included in the curves were the distance from the orbiter to the beam, the beam's emissivity, and its temperature. No attempt was made to evaluate the integrated effect of the space construction as a matrix of beams at this time, but to consider the construction as separate rectangels. To use this information, it would be necessary to evaluate the largest beams separately and total their individual effects or to assume a conglomerate area approximating the integrated effect.

Radiation

The net radiation heat transfer from the orbiter radiator is the heat radiated by the radiator less than the total heat absorbed.

$$\frac{Q_i}{A_i} = \sigma_i \alpha T_i^4 - \alpha_i \sum_{j=1}^{N} \sigma \epsilon_j F_{AiAj} T_j^4 \qquad (4.13\text{-}1)^*$$

The total heat absorbed is the second term in the equation and represents all heat radiated from all surrounding surfaces such as the beams, platforms, solar arrays, and modules plus the direct solar radiation, earth albedo, and earth emission. Equation (4.13-1) is a simplified version that assumes that there are no reflections between surfaces. This condition would exist for a black body surface that radiates as a perfectly diffuse surface. The surfaces emit equally in all directions but do not reflect at all. This assumption simplifies the calculations and reduces the effort to obtain an early solution. The solution will be on the conservative side of the more exact solution assuming reflecting surfaces requiring a matrix solution of the multiple reflections.

Form Factors

The form factor is normally calculated from the representation of surfaces in Figure 4.13-2.

$$F_{A_iA_j} = \frac{1}{A_i} \int_{A_i} \int_{A_j} \frac{\cos\theta_i \cos\theta_j}{\pi \ell_{ij}^2} \, d A_i \, d A_j \qquad (4.13\text{-}2)$$

---

*Nomenclature for this analysis is contained in Table 4.13-3.

Table 4.13-3. Thermal Analysis Nomenclature

Symbol		Subscripts	
a	Beam length, ft	A	Orbiter radiator
A	Area, ft^2	B	Beam rectangle
b	Beam width, ft	i	First surface
F	Geometric form factor	j	Second surface
ℓ, L	Length, ft		
Q	Total heat transfer, Btu/hr		
R	Aspect ratio, a/b		
t	Temperature, degrees Fahrenheit		
T	Temperature, degrees Rankine		
X,Y,Z	Rectangular coordinates		
Greek Symbol			
α	Absorptivity to radiation		
ε	Emissivity		
σ	Modified Stefan-Boltzmann constant, 0.1713 Btu/($hr \times ft^2 \times R^4$)		
θ	Angle, degrees		
η	Normal to surface		

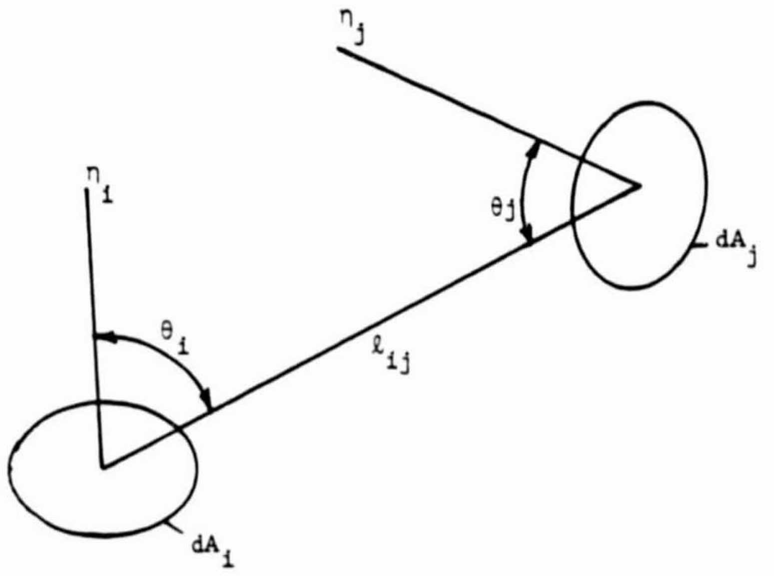


Figure 4.13-2. Surface Representation Factors

Here, η_i and η_j are lines normal to surfaces dA_i and dA_j. The analysis was accomplished using a computer program based on a Stoke's theorem transformation of the view factor area integral, Equation (4.13-2), to an equivalent contour integral. Data input for the program requires values for the X, Y, and Z coordinates of the surfaces.

X, Y, Z Coordinates

To calculate the form factors from the orbiter radiator to a beam or rectangular shape above the payload bay, it is necessary to determine the X, Y, and Z coordinates of the surface. Figure 4.13-3 establishes a coordinate system above the orbiter radiator. It appeared convenient to represent the shape

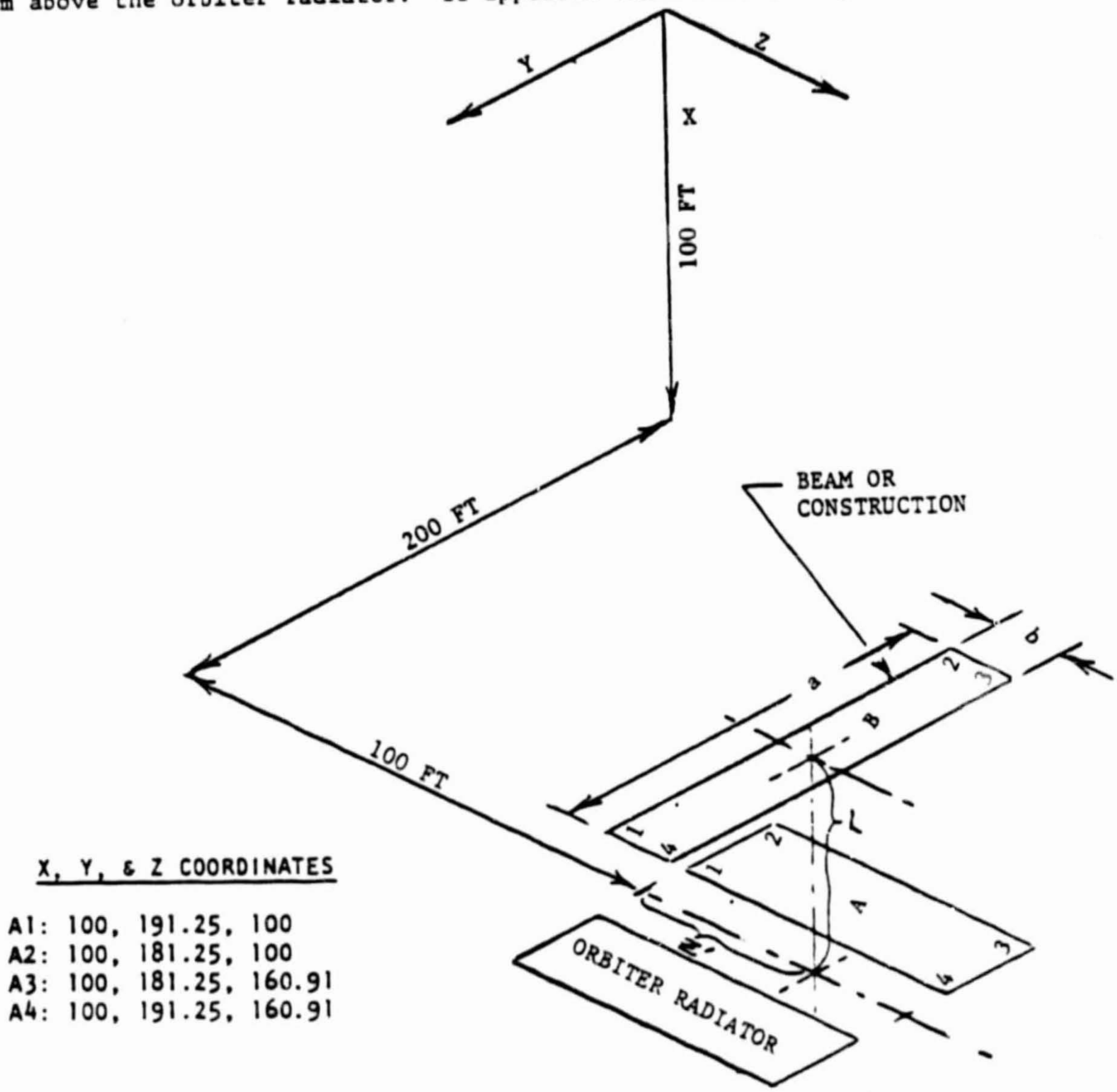


Figure 4.13-3. Coordinate System for Construction Above Orbiter Radiator

of a rectangle by using the aspect ratio of length to width ($R = a/b$). Given a value of area and aspect ratio, the dimensions of the rectangle are:

$$a = \sqrt{AR}$$

$$b = \sqrt{A/R}$$

The coordinates of area B can then be calculated with the equations for points 1, 2, 3, and 4, as listed in Table 4.13-4.

Table 4.13-4. Equations for Calculating Boundary Points of Blocking Surface

Point No.	X	Y	Z
1	100-L	200 + a/2	100 + Z' - b/2
2	100-L	200 - a/2	100 + Z' - b/2
3	100-L	200 - a/2	100 + Z' + b/2
4	100-L	200 + a/2	100 + Z' + b/2

To keep the problem simple, the value of Z' was chosen at the center of the payload bay, so Z' = 30.45 ft.

Loss in Heat Rejection

The X, Y, and Z coordinates were calculated using the computer program. These points were then entered into the form factor program and the results are shown in Table 4.13-5 as functions of area, aspect ratio, and distance to orbiter radiator. The loss in heat rejection capability was calculated from the second part of Equation (4.13-1) and the results listed in Table 4.13-6 for various values of construction temperature and emissivity. The results are shown plotted in Figure 4.13-4 for parameters of area, aspect ratio, emissivity of construction and its temperature for a distance of L = 25 ft. The form factors of F_{A-B} were calculated for distances of L = 10, 25, and 50 ft in Figure 4.13-5. These charts may then be applied to Figure 4.13-4 to determine the loss in orbiter heat rejection.

To determine the temperature of the space structure, an approximation may be made by using Figure 4.13-6 which was copied from Jelinek (1977 (a)). It was designed to give the maximum temperature with known values for emissivity rear, solar absorptivity rear, solar absorptivity front, and emissivity front. Most space construction will probably be in the range of 50 to 100°F. Figure 4.13-6 is also applicable to "blankets" or solar arrays which may be mounted on large space structures.

The orbiter radiator heat rejection capabilities are shown in Figure 4.13-7 for three attitudes in earth orbit. Data originally appeared in Jaax (1977). The curves did not include any estimated losses for solar arrays or other objects near by. The space construction heat losses may then be subtracted from the orbiter heat rejection capability to determine the net capability due to space construction. Heat rejection for other orbiter attitudes may be found in reference NASA (1977 (a)).

Table 4.13-5. Geometric Form Factors from Orbiter Radiator to a Rectangular Shaped Construction Element

A (FT^2)	Z' (FT)	R	F_{A-B} L = 5 (FT)	L = 10	L = 25	L = 50
100	30.46	1	0.0110	0.0224	0.0213	0.0096
		5	0.0210	0.0293	0.0214	0.0095
		10	0.0284	0.0338	0.0212	0.0093
		100	0.0137	0.0181	0.0134	0.0071
500	30.46	1	0.1365	0.1487	0.1043	0.0467
		5	0.1764	0.1651	0.0984	0.0442
		10	0.1136	0.1268	0.0852	0.0408
		100	0.0312	0.0411	0.0321	0.0195
1000	30.46	1	0.3942	0.3398	0.2007	0.0900
		5	0.2959	0.2594	0.1690	0.0811
		10	0.1845	0.1864	0.1337	0.0704
		100	0.0449	0.0582	0.0455	0.0282
1500	30.46	1	0.6691	0.5169	0.2866	0.1302
		5	0.3734	0.3241	0.2212	0.1123
		10	0.2460	0.2312	0.1685	0.0937
		100	0.0560	0.0714	0.0558	0.0347

Table 4.13-6. Loss in Orbiter Radiator Capacity due to Construction above Radiator

T_B (F)	(1) $\varepsilon_A \sigma (T_B/100)^4 A_A$	(2) $\varepsilon_B F_{A-B}$	Q_{LOSS} = (1) × (2) (Btu/hr)
150	201,995	0.05	10,100
		0.15	30,299
		0.25	50,499
		0.35	70,699
100	143,474	0.05	7,174
		0.15	21,521
		0.25	35,869
		0.35	50,216
50	98,697	0.05	4,935
		0.15	14,804
		0.25	24,674
		0.35	34,544
0	65,321	0.05	3,266
		0.15	9,798
		0.25	16,330
		0.35	22,862
-50	41,225	0.05	2,061
		0.15	6,184
		0.25	10,306
		0.35	14,429
-150	24,504	0.05	1,225
		0.15	3,676
		0.25	6,126
		0.35	8,576

ε_A = 0.76 A_A = 1,120.6 ft^2

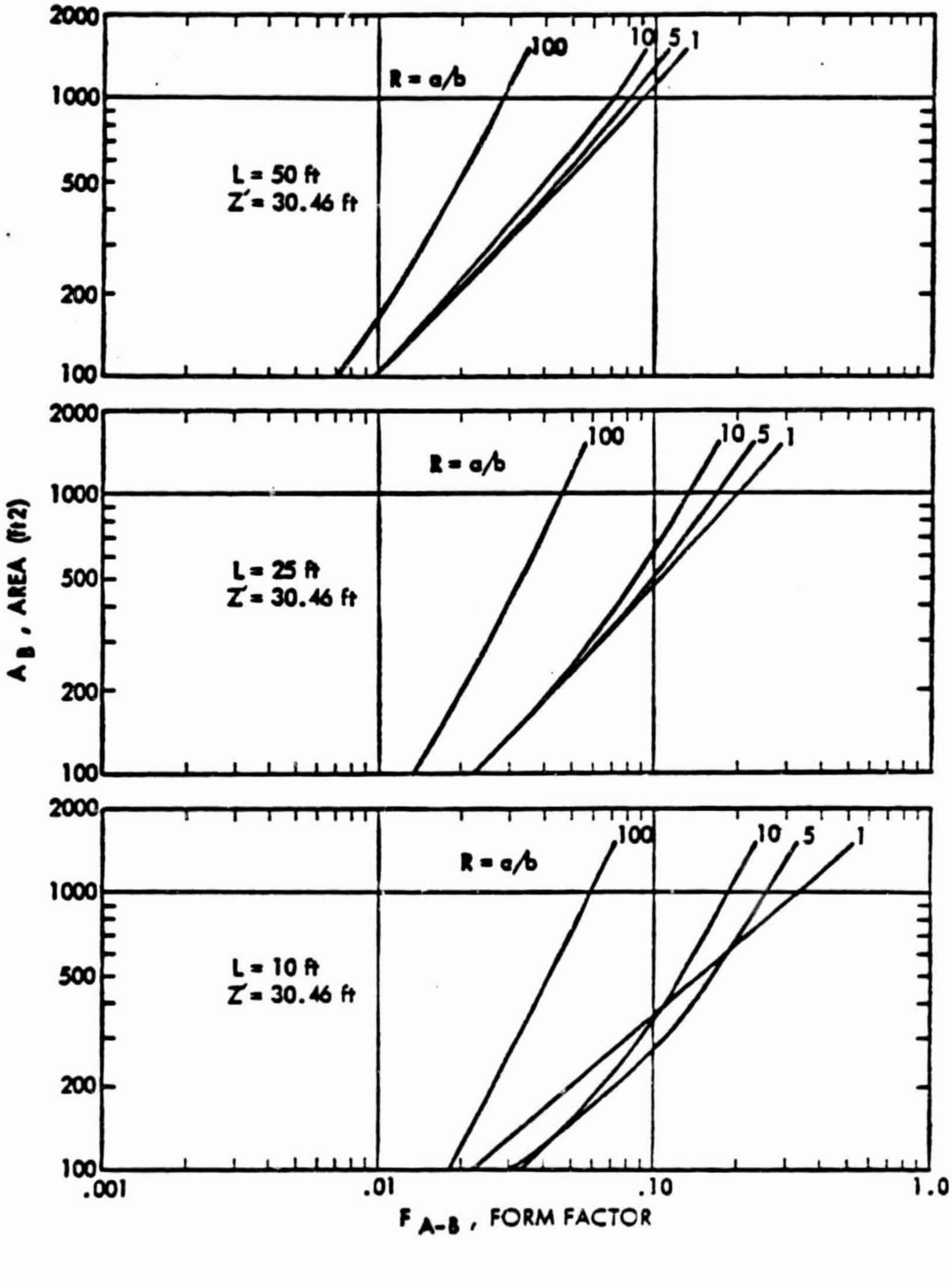


Figure 4.13-3. Loss In Orbiter Heat Rejection for Construction Above Payload Bay

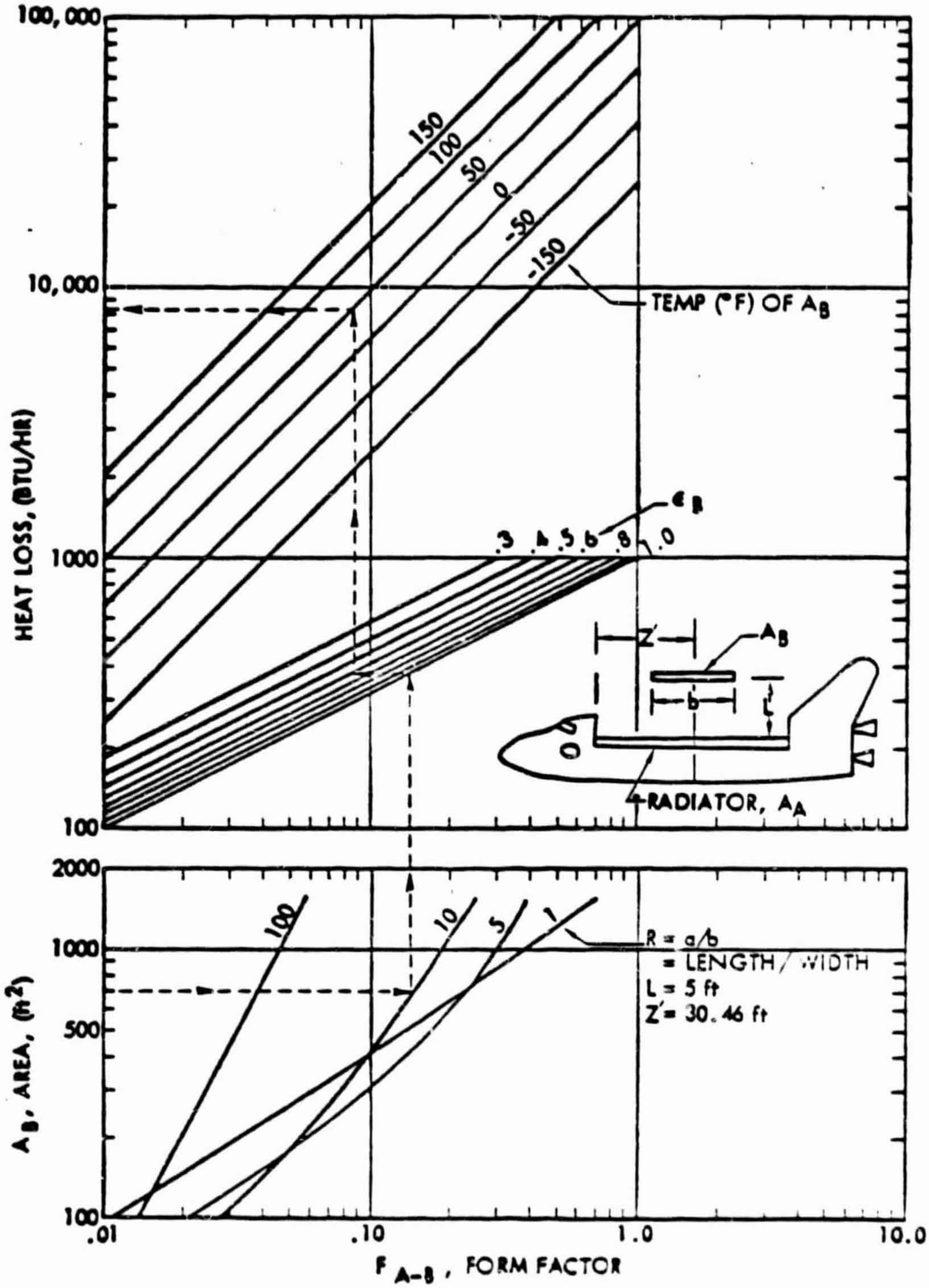


Figure 4.13-4. Form Factors for Construction Areas Above Payload Bay

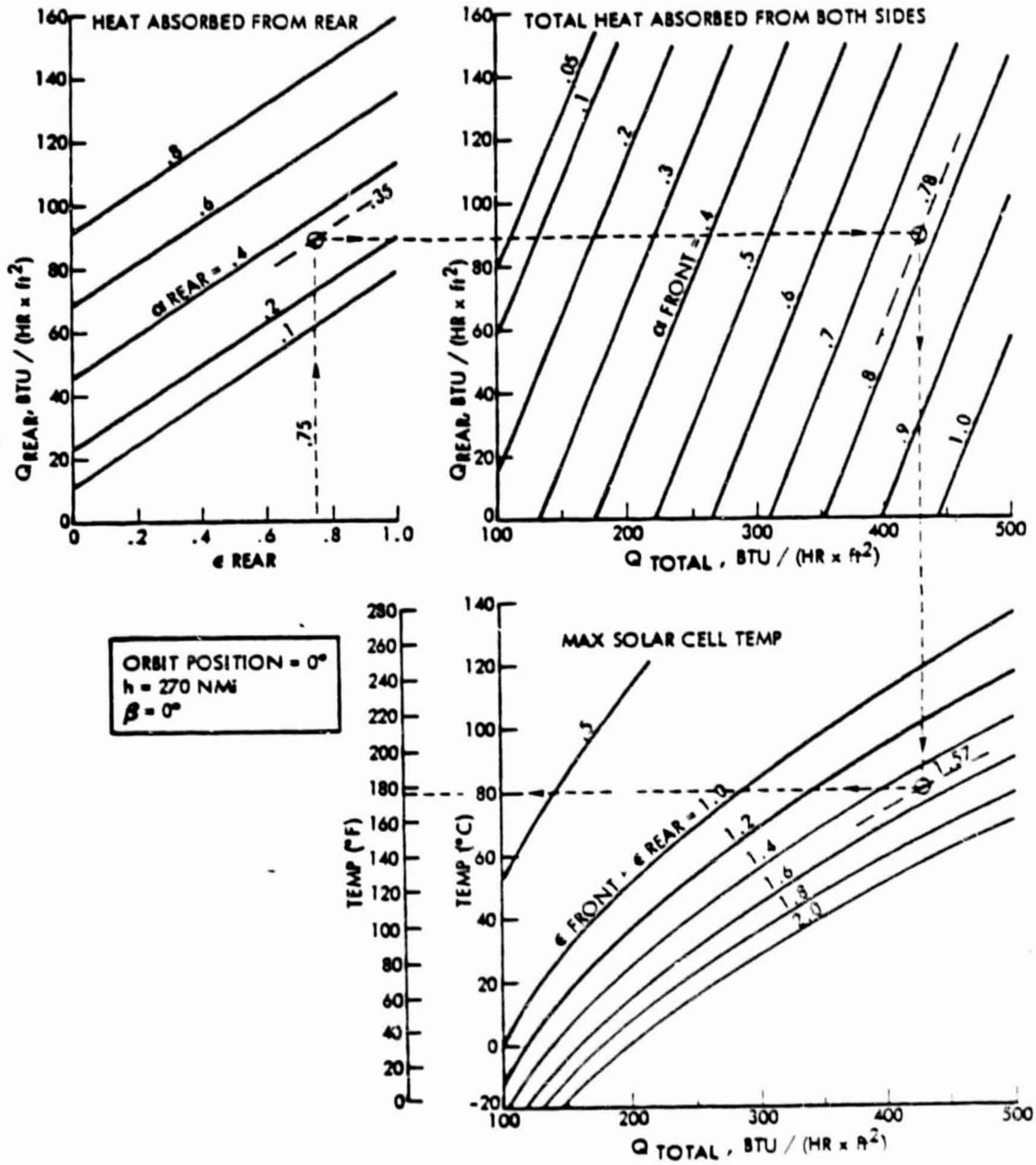


Figure 4.13-5. Maximum Structure Temperature

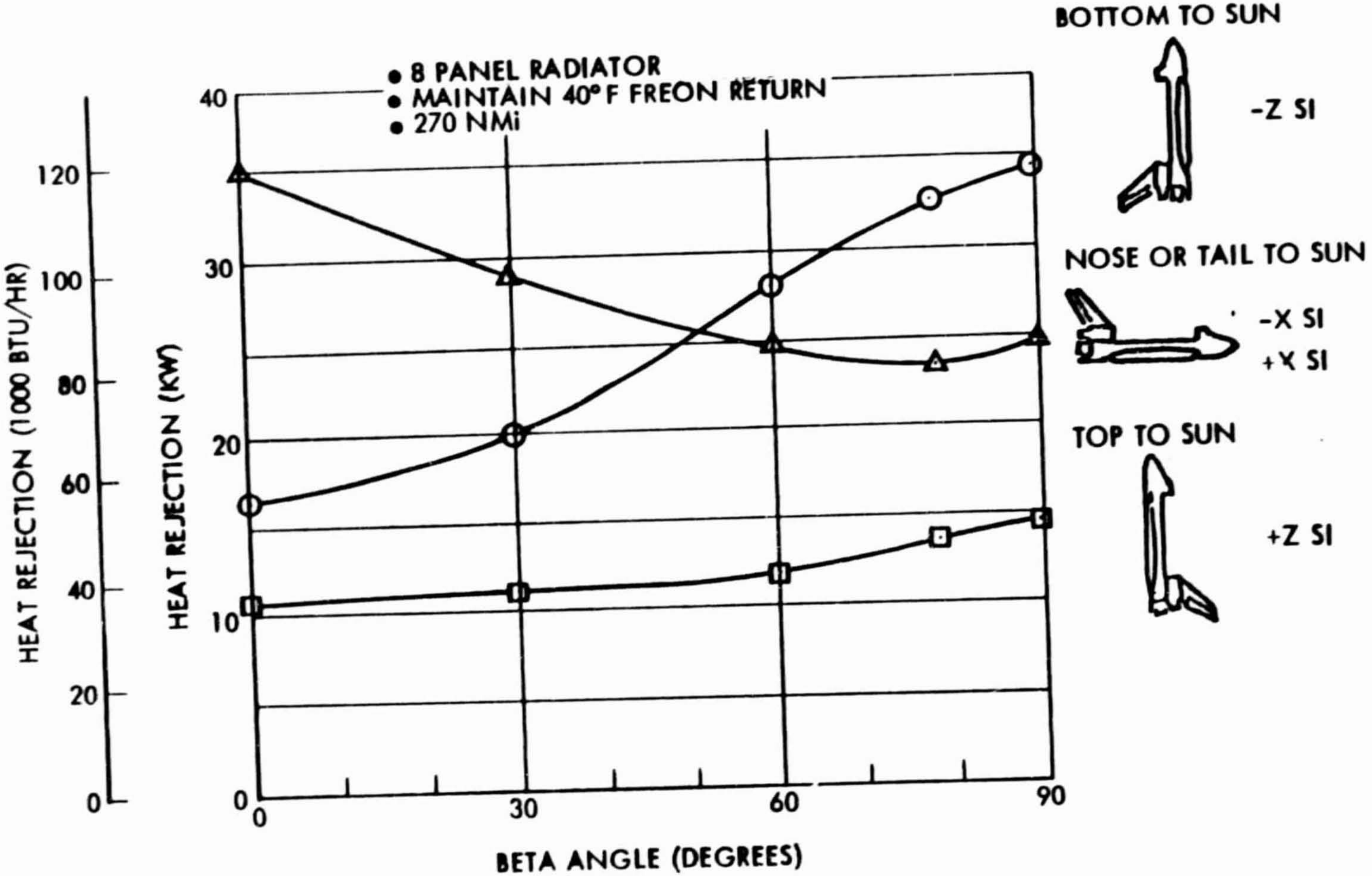


Figure 4.13-6. Orbiter Radiator Capability in Solar Inertial Attitudes

The traditional method of plotting from factors from a point to a rectangle uses ratios of each dimension of the rectangle to the distance from the point such as a/L and b/L. This procedure generally covers all values of the length, L, and the form factors are the same as long as the ratios are the same. This reduces the number of curves required. In the case of the radiator or rectangle-to-rectangle the form factors are not the same for different values of length, L, when the ratios of a/L or b/L are the same; therefore, it required separate curves for different values of length, L.

4.13.2 Thermal Environment Effects

In addition to the foregoing major concerns regarding heat rejection capability of the orbiter, there are secondary concerns involving effects of the rejected heat,reflections and shadowing by the orbiter. These are briefly noted in Table 4.13-7. For the most part, the potential impacts of these constraints are not yet defined, and will be different for each specific large space structure design. Analysis is recommended for each case.

Table 4.13-7. Checklist: Additional Orbiter Thermal Constraints affecting Large Space Structures Design

( ) CONSTRAINTS/•QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) The orbiter payload bay has moderately specular, curved surfaces which can concentrate solar energy on structures and equipment stowed in the bay when facing the sun. (R-1) • See Tables 4.13-8 and 4.13-9 for representative examples. • Analysis is recommended for specific cases.	Rockwell, 1979 (a): SSD 79-0123 Affects insulation, materials, coatings, thermal conditioning provisions and attitude control during construction.
( ) The orbiter can experience shadow in the payload bay and affect portions of large space structures during orbital operations. (R-1) • See Tables 4.13-8 and 4.13-9 for examples. • Analysis is recommended to determine possible range of structural distortions.	• Cold-soak worst-case design conditions • Potential distortion due to local cooling on sunlit side of orbit.

Table 4.13-7. Checklist: Additional Orbiter Thermal Constraints affecting Large Space Structures Design (Cont.)

( ) CONSTRAINTS/QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) The Shuttle orbiter radiators on the payload bay doors act as heat sources which can radiate toward nearby space structures and construction support equipment. (R-1) • Maximum (return fluid) temperature in radiators is approximately 140°F (40°C)	Rockwell, 1980 (a): SSV 80-1 • Possible uneven heating, thermal distortion of large space structures. Affects material selection. • Possible heating of equipment; reduction of power for thermal conditioning or work interruption due to overheating.
( ) The Shuttle orbiter radiators on the payload bay doors have highly reflective, curved surfaces which can focus solar rays to create local zones of heating of structure located above the radiators at selected solar angles (R-1) • Analysis is recommended to quantify conditions for any specific structure • Figure 4.13-10 illustrates an example of reflected solar energy concentration by orbiter radiators • Sunlight input toward radiators can be shadowed by structure or controlled by orbiter attitude relative to the sun.	Rockwell, 1979(a) & Rockwell, 1980(f) Affects attitude control, selection of materials and coatings, location of structure and equipment. The RMS may be affected locally. Work may be interrupted.

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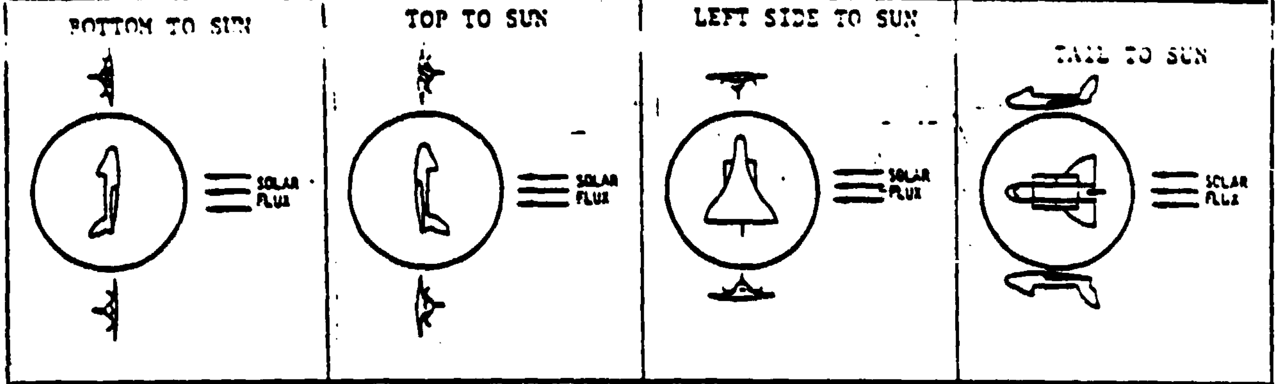
Table 4.13-8. Typical Temperatures at End of Six-Hour Hold, Payload Bay Doors Open

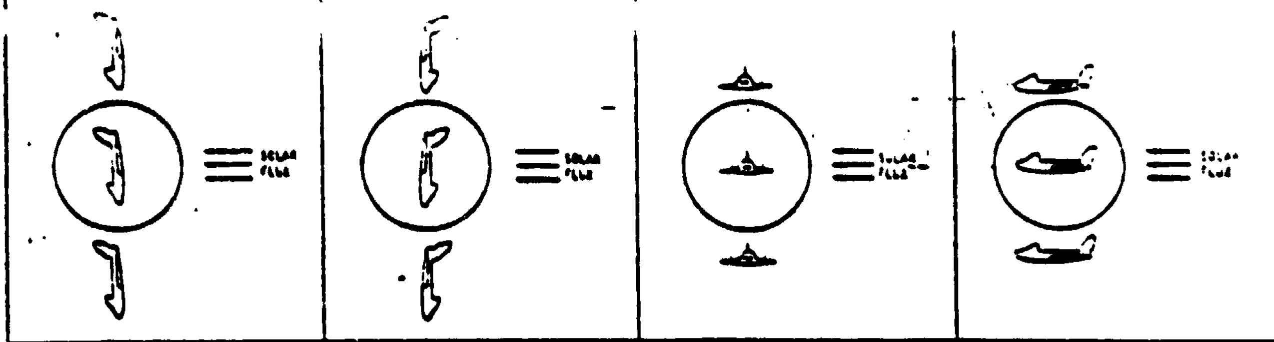


Data Point Location	Predicted Temperature (degrees F)* Bottom to Sun	Top to Sun	Left Side to Sun	Tail to Sun
Payload Bay Wall (Liner), X_o 582 to 760				
Left side, near longeron	(-) 101	143	(-) 5	(-) 188
Bottom, left side	(-) 102	196	5	(-) 194
Bottom, right side	(-) 69	189	5	(-) 194
Right side, near longeron	(-) 43	130	(-) 3	(-) 188
Longeron, X_o 582 to 760				
Left	13	144	40	(-) 53
Right	20	147	4	(-) 53
Radiator, X_o 582 to 760				
Left	(-) 14	21	29	(-) 77
Right	(-) 12	44	(-) 12	(-) 77
Orbiter Exterior Surfaces				
Left Side				
X_o 400 to 576, Z_o > 400	(-) 9	0	97	(-) 99
X_o 400 to 576, Z_o < 400	15	(-) 46	104	(-) 65
X_o 576 to 760	67	32	58	(-) 31
Right Side				
X_o 400 to 576, Z_o > 400	(-) 106	60	(-) 43	(-) 49
X_o 400 to 576, Z_o < 400	(-) 77	34	(-) 47	(-) 65
X_o 576 to 760	55	94	(-) 25	(-) 31

*Analytical tolerance for all temperatures is plus or minus 30 degrees F.

Table 4.13-9. Typical Temperatures at End of 80-Hour Solar Inertial Hold, Payload Bay Doors Open

BOTTOM TO SUN	TOP TO SUN	LEFT SIDE TO SUN	TAIL TO SUN



Data Point Location	Predicted Temperature (degrees F)*			
	Bottom to Sun	Top to Sun	Left Side to Sun	Tail to Sun
Payload Bay Wall (Liner), X_o 582 to 760				
Left side, near longeron	(-) 77	137	(-) 84	(-) 82
Bottom, left side	(-) 89	193	(-) 81	(-) 83
Bottom, right side	(-) 88	194	(-) 80	(-) 82
Right side, near longeron	(-) 76	137	(-) 83	(-) 82
Longeron, X_o 582 to 760				
Left	12	149	19	(-) 55
Right	13	150	(-) 55	(-) 55
Radiator, X_o 582 to 760				
Left	8	32	1	(-) 47
Right	8	32	(-) 50	(-) 47
Orbiter Exterior Surfaces				
Left side				
X_o 400 to 576, Z_o > 400	(-) 62	29	94	(-) 77
X_o 400 to 576, Z_o < 400	(-) 41	(-) 11	109	(-) 68
X_o 576 to 760	113	82	96	(-) 58
Right side				
X_o 400 to 576, Z_o > 400	(-) 61	28	(-) 77	(-) 77
X_o 400 to 576, Z_o < 400	(-) 40	(-) 10	(-) 77	(-) 68
X_o 576 to 760	58	83	(-) 58	(-) 58

*Analytical tolerance for all temperatures is plus or minus 30 degrees F.

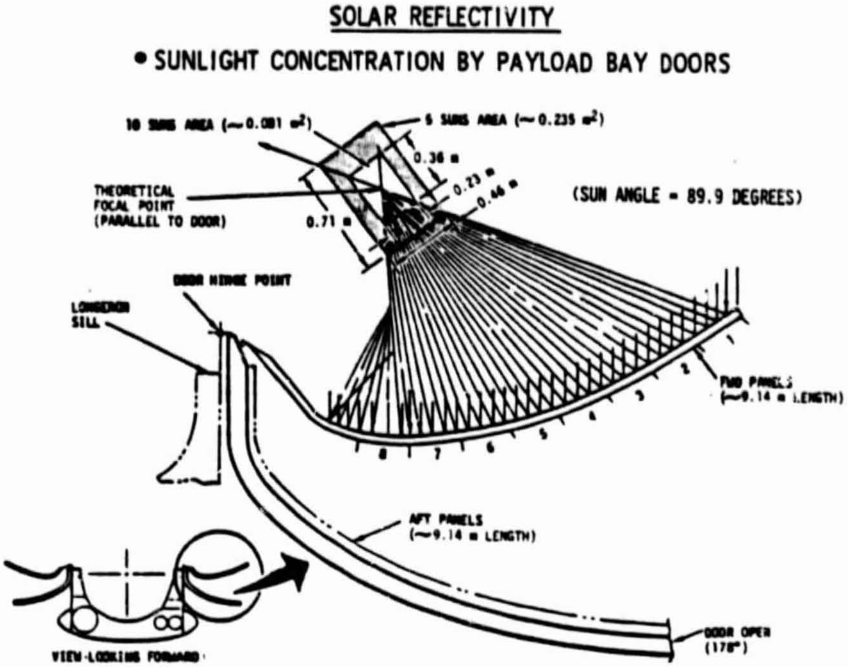


Figure 4.13-7. Example of Possible Solar Energy Concentration by Radiators on Cargo Bay Doors

4.14 ORBITER CONTAMINANT ENVIRONMENTS

The Shuttle orbiter can constrain cleanliness of the local (on-orbit) environment due to its effluents. In addition, certain effluents or contingency spillage concerns of fuels or materials used in space transportation might create hazards to EVA or other orbiter operations. Both aspects are described in the following sections.

4.14.1 Orbiter Effluents

Normal on-orbit operations with the Shuttle orbiter are accompanied by various trace effluents of gases from the orbiter materials, cabin leakage, orbiter RCS, and flash evaporator. These effluents are summarized in Table 4.14-1, in terms of predicted column density and return flux (per NASA 1977 (a) STS User Handbook). See Section 4.1.6 for RCS plume effluents.

Table 4.14-1. Predicted Column Density and Return Flux

Source	Number column density, molecules/cm^2			Return flux, molecules/cm^2/sec		
Outgassing (a) and (b)	$< 10^{12}$ after 10 hr			$< 10^{12}$		
				Values at 235 n. mi. (435 km)		
Vernier RCS	Aft -Z	Aft Y	Forward X/Z	Aft -Z	Aft Y	Forward Y/Z
(a)	4.4×10^{14}	2.0×10^{14}	3.9×10^{12}	7.6×10^{12}	3.4×10^{12}	6.6×10^{10}
(b)	1.8×10^{14}	8.1×10^{13}	2.7×10^{12}	3.2×10^{12}	1.4×10^{12}	4.6×10^{12}
Flash evaporator				378 n.mi. (700 km)	235 n. mi. (435 km)	108 n. mi. (200 km)
(a)		5.6×10^{12}		8.4×10^{8}	2.4×10^{12}	1.3×10^{12}
(b)		5.6×10^{12}		8.5×10^{8}	2.4×10^{10}	1.3×10^{12}
Leakage						
(a)		2.2×10^{13}		1.2×10^{10}	3.7×10^{11}	1.9×10^{13}
(b)		3.5×10^{13}		2.0×10^{10}	5.6×10^{11}	3.1×10^{13}

[a]Zero degree line-of-sight (in the $+Z_o$ direction) originating at X_o 1107.

[b]50° off of +Z towards $-X_o$ (forward) originating at X_o 1107.

4.14.2 Contaminant Control Options

Contamination control systems, as well as various techniques to eliminate or minimize contamination, are provided by the orbiter design and standard flight plans. The sensitivity of most payloads to contamination is recognized and each mission can be tailored to meet specific requirements. Before lift-off and after landing, the cargo bay is purged and conditioned as specified in the description of thermal controls. At launch and during early ascent, the cargo bay vents are left closed to prevent exhaust products and debris from entering the bay. During final ascent and through orbit insertion, the cargo bay is depressurized and the payload is generally not subjected to contaminants.

The payload vent sequencing is shown in Section 2.6. During deorbit and descent, the cargo bay vents are closed to minimize ingestion of contaminants created by the orbiter systems. During the final phase of reentry, the vents must be opened to repressurize the orbiter. To help prevent contamination during this phase, the vents are located where the possibility of ingestion is minimal.*

At this stage in development of large space structures concepts, there are very few recognized concerns for contamination. Exceptions may apply to specific sensors attached to large structures during the process of construction and checkout/maintenance. However, the subject does not yet warrant detailed definition of orbiter constraints, in the context of this document.

*Data Source — NASA 1977(a): STS User Handbook

5.0 CREW PRODUCTIVITY AND SAFETY CONSIDERATIONS

The contrast to the foregoing relatively independently considered specific hardware and related operations for payload handling, crew operations must consider elapsed time throughout an entire mission. For example, crew sleeping and eating periods normally must be introduced at certain intervals, starting from preparations for boarding the orbiter on the ground. Otherwise, performance may degrade and costly errors may be committed. Crew operations throughout the mission are here considered as a whole, rather than allocating such considerations into the foregoing sections (2.0, 3.0, and 4.0). It is recognized that some aspects of crew accommodations are specifically related to mission phase. A major example is the location of seating provisions. Seats are attached in place on the deck (aft flight deck or mid-deck) for ascent and descent, but they are taken up and stowed during orbital operations. Conversely, the on-orbit station is not operated at all during ascent or descent. The following discussion separately considers crew accommodations, work/rest cycles, and orbiter housekeeping requirements.[1]

5.1 CREW CABIN ACCOMMODATIONS AND SUPPLIES

The orbiter crew accommodations constraints which affect design of large space structures in a major way relate primarily to the configuration of the aft flight deck, the airlock, the sleeping space allocations, and the overall size of the habitable volume.

The aft flight deck configuration limits viewing of large space operations as shown in Section 4.9.6, and effectively limits the number of usable crew stations to four as shown in Figure 5.1-1 (upper right). Of the four stations, the two on the starboard side are largely dedicated to flight phases (ascent, rendezvous, docking/berthing, and descent). Those on the port side are most useful in space construction and other operations related to payloads. Figure 5.1-2 depicts the baseline configuration of the aft flight deck for on-orbit operations (seats removed).

The mission station is located aft of the pilot's station on the starboard side. This station contains the displays and controls required to manage orbiter-to-payload interfaces and payload subsystems that are critical to the safety of the orbiter. An auxiliary caution and warning (C&W) display is provided at this station to detect and alert the crew in cases of critical malfunctions in the payload system. The station provides for on-orbit housekeeping functions and for the management of nonflight-critical orbiter subsystems functions, which do not require immediate access. All coutrols are hardwired, as are the critical payload measurements for display and C&W. A CRT display and keyboard are located at this station for monitoring payload interfaces and for orbiter subsystem performance monitoring. Payload conditions critical during ascent and entry can also be displayed at the forward flight stations via C&W and CRT displays.

[1] "Caution: Not a controlled document! See appropriate reference documents for current data."

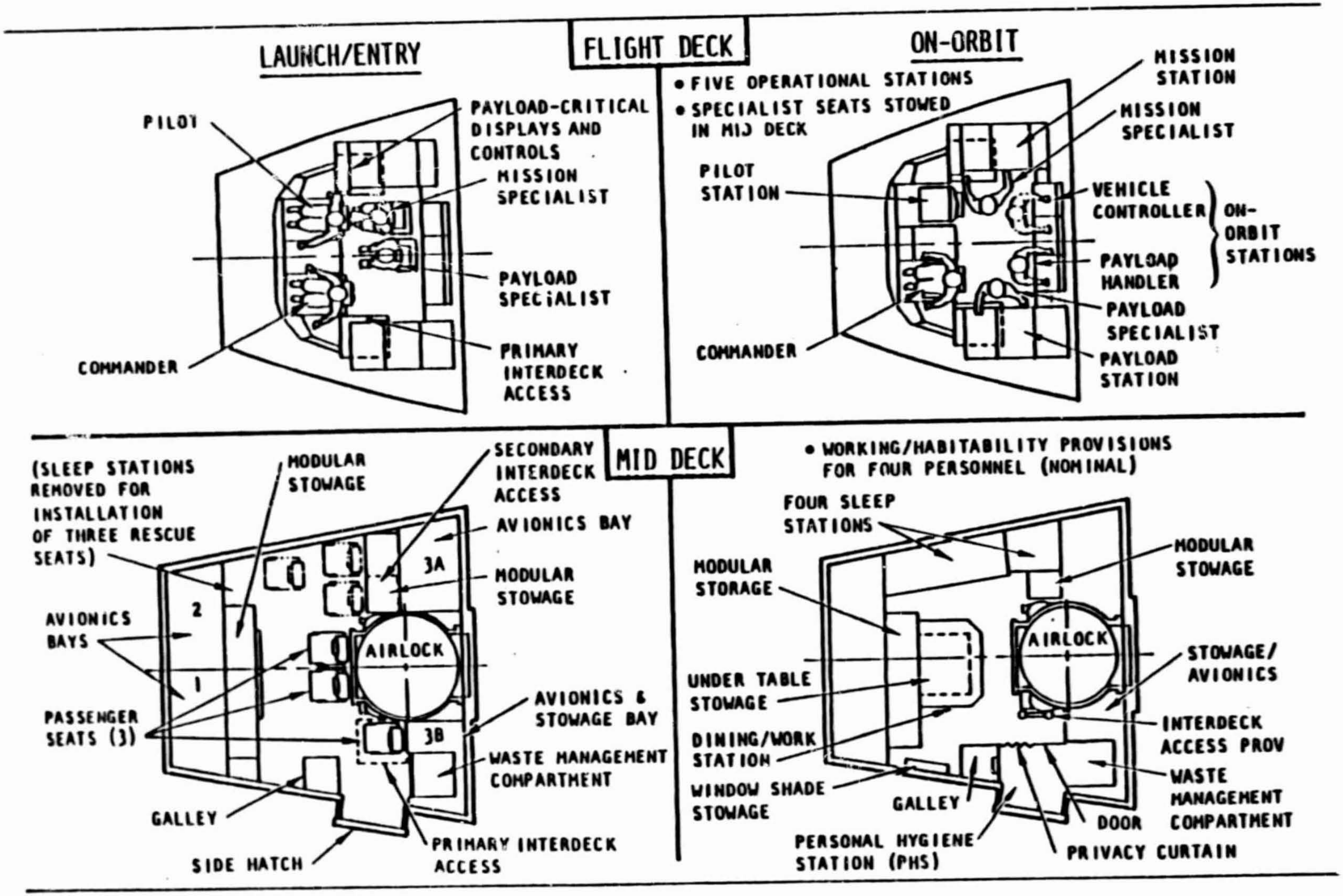


Figure 5.1-1. Crew Cabin Arrangement and Crew Functions

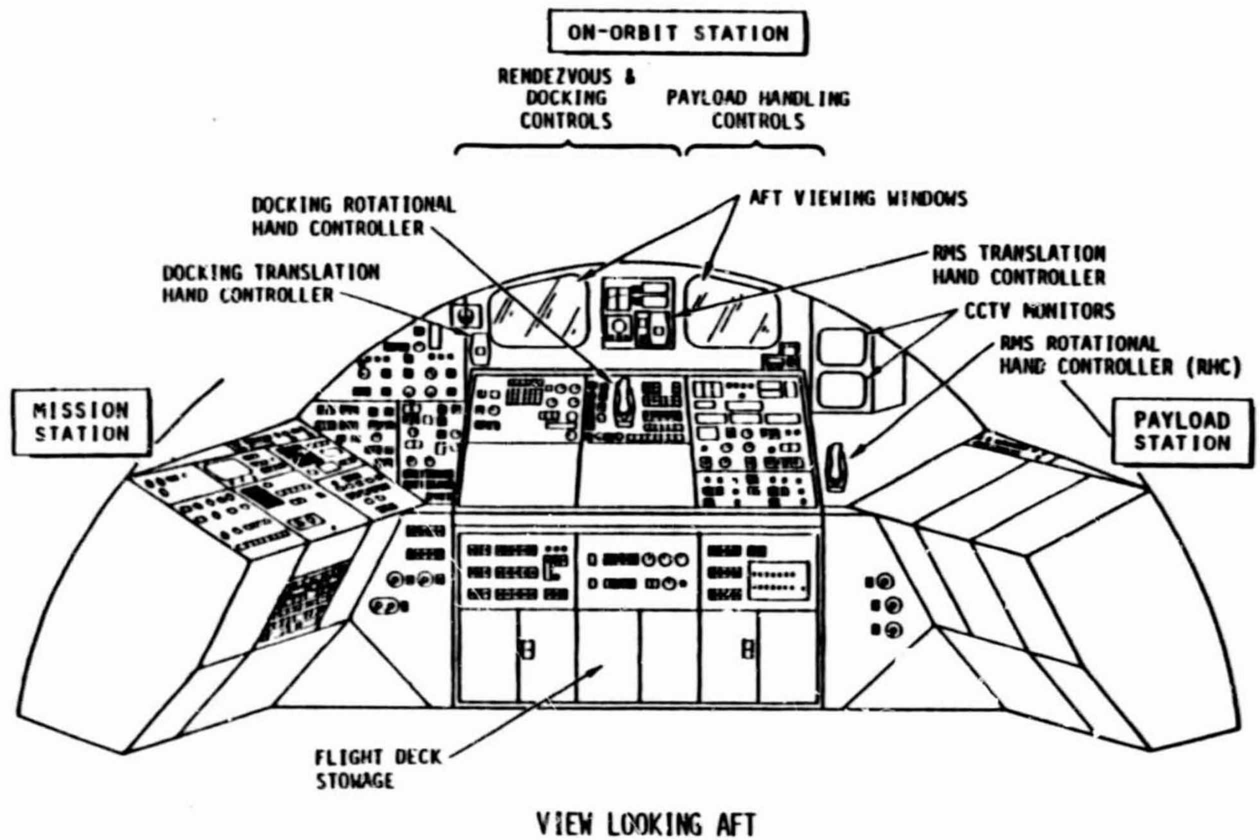


Figure 5.1-2. Aft Flight Deck

The rendezvous and docking portion of the on-orbit station contains the displays and controls required to execute orbiter attitude/translation maneuvers for terminal-phase rendezvous and docking. Rendezvous radar controls and displays and cross-pointer displays of pitch and roll angles and rates are provided at this station, as well as rotation and translation hand controllers, flight control mode switches, and attitude direction indicators.

The payload handling portion of the on-orbit station contains those displays and controls required to manipulate, deploy, release, and capture payloads. Displays and controls are provided at this station to open and close payload bay doors; deploy radiators; deploy, operate, and stow the payload manipulator arm; and operate payload-bay-mounted lights and TV cameras. Two closed-circuit TV monitors display the payload bay video pictures of payload manipulation operations.

The payload station is located aft of the commander's station on the port side. This incorporates standardized provisions for installation of payload-unique GFR displays and controls required for monitoring and operating the various payloads. A CRT display and associated keyboard may be added, as an optional provision, for communication with payloads via the orbiter data processing subsystem. Standardized electrical interfaces are provided for payload power, monitoring, command, and control.

Note that the on-orbit station really consists of two crew stations: (1) one for rendezvous and docking, and (2) the other for payload handling (RMS control). The latter is of major interest for operations related to large space structures. Figure 5.1-3 illustrates the key crew working positions for design of these stations. A significant concern of these stations is that when crew members are using the RMS controllers or orbiter controllers and COAS, they are closely constrained in body position and, thus, in range of vision through the windows.

Figure 5.1-4 illustrates the proximity of the payload specialist station to the payload handler. In fact, these stations are so close that each crew person must watch for the others' movements to avoid interference. In some space construction operations it is likely that both these stations will be controlled by one operator on an alternating basis. Some changes in body restraints may be required to enhance this capability.

Figure 5.1-1 shows the locations for seating during launch and entry and the nominal four sleep stations for on-orbit operations (lower left and right, respectively). Also shown is the internally located airlock which appears most likely to be used in conjunction with construction of large space structures. Figure 5.1-5 illustrates nominal eating space for four crew members on the mid-deck.

Table 5.1-1 summarizes key constraints related to crew accommodations, crew equipment and supplies stowage, which may impact operations involving large space structures. Of particular concern are weight and volume of crew supplies, and stowage for additonal EMU's (space suits—for EVA). Stowage for additional seats, suits, and other gear must compete for space with food, personal supplies, LiOH, and other crew needs. Stowage volume usage should

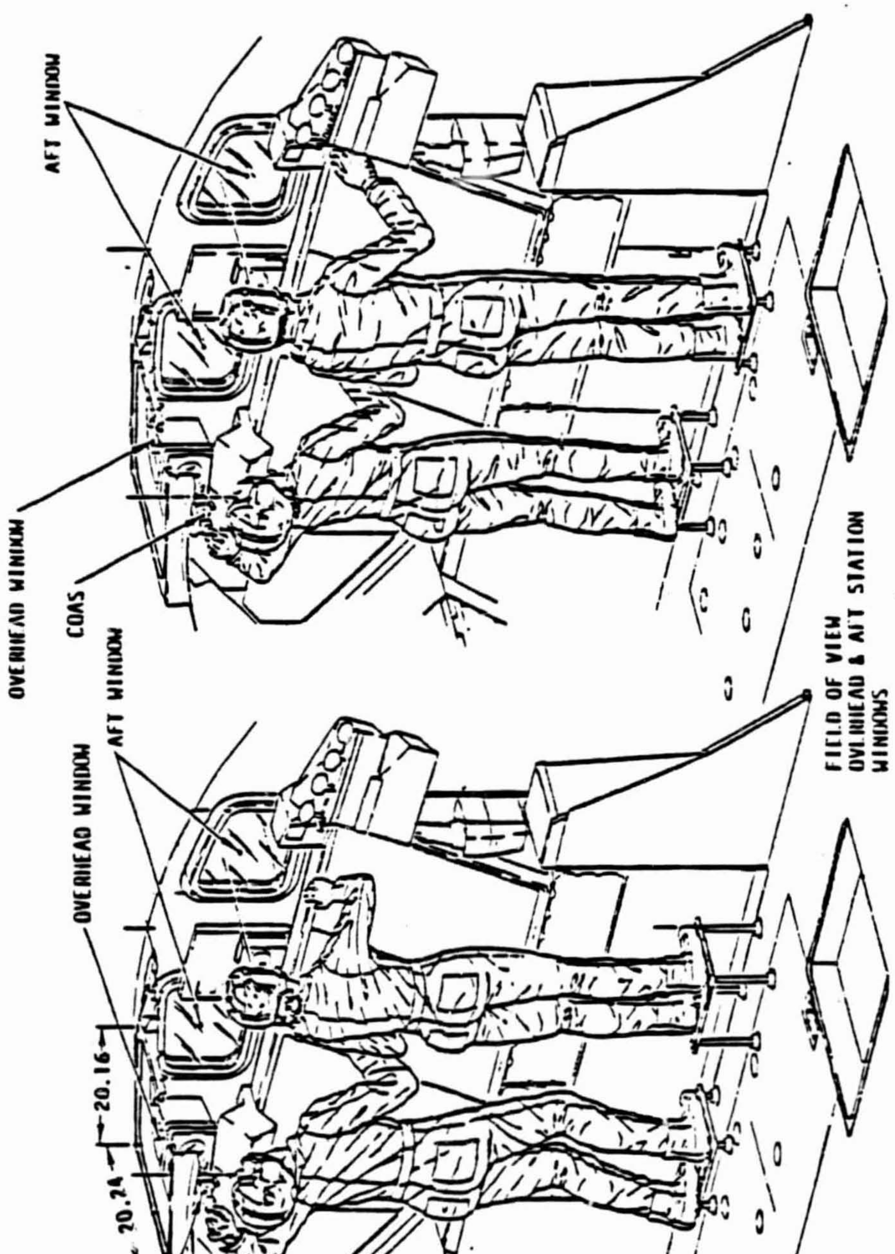


Figure 5.1-3. Crewpersons at On-Orbit Station

• REQUIREMENT - DOCKING, PAYLOAD HANDLING OPERATIONS

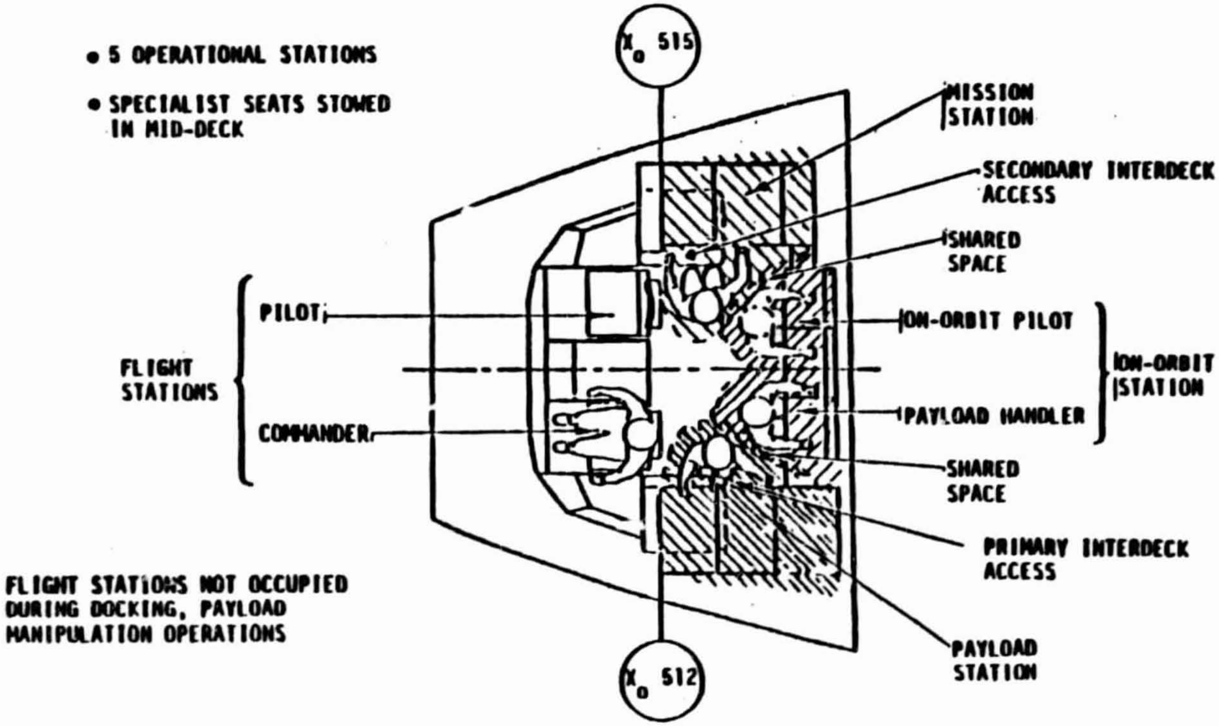


Figure 5.1-4. Workspace used for Aft Flight Deck

- REQUIREMENTS
  - 4 PERSONS EATING SIMULTANEOUSLY
  - 3 PERSONS EATING, 4 SLEEPING SIMULTANEOUSLY

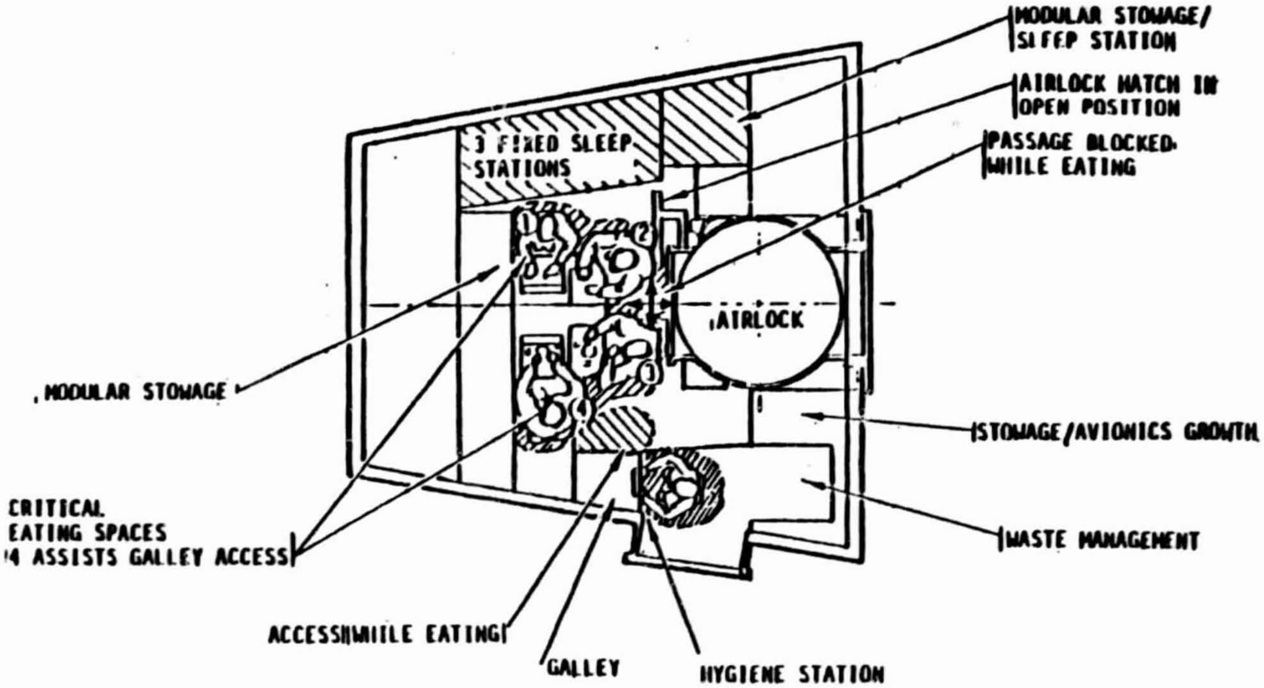


Figure 5.1-5. Space Utilized for Simultaneous Eating and Sleeping

Table 5.1-1. Checklist: Constraints on Crew Operations Related to Crew Cabin Accommodations and Supplies

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) For normal operations, crew cabin seating is limited to seven (7) persons. (R-1) • Three additional seats may be added (primarily intended for contingency rescue modes, since the usual four sleep stations are pre-empted) • See Figure 5.1-1 lower left	Rockwell, 1980 (b): SSV80-1 Constrains number of crew available for construction.
( ) For normal operations, crew sleeping accommodations are limited to four spaces. (R-1) • Additional spaces for sleeping bags may be available	Rockwell, 1980 (b): SSV 80-1
( ) Space suit (EMU) stowage is normally limited to two suits, located in the airlock. Additional space suit stowage volumes are not defined (R-1) • For particular missions, special stowage provisions may be provided as payload chargeable special equipment (e.g., in sleep station volume during launch and entry (TBD) (R-4) • See also Section 4.8.1.1	NASA 1980 (a): JSC-07200, Vol. XIV Impacts consideration of additional EVA capability (over two crew).
( ) The airlock can accommodate a limit of two crew during one depressurization on repressurization period (R-1) • Following egress of two crew, two crew on outside may ingress and repressurize airlock. A third set of two crew may later don suits and egress. Timeline constraints for such operations are (TBD).	NASA, 1980 (a): Impacts number of crew which can exit to EVA work simultaneously: affects crew work rest cycles, work schedules.

Table 5.1-1 (continued) Checklist:

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) Stowage volume for crew personal equipment & supplies is limited to approximately (TBD) cubic feet. (R-4) • Trades of time on orbit, equipment volume requirement and operational capabilities must be performed to determine optimum balance.	

be considered for the orbiter cabin as a whole, and is negotiated for each flight. Although crew cabin storage volume may be a small portion of the total payload carried for space construction missions, it is possible that this volume could effectively limit tolerable time for habitation on orbit and/or the number of crew who can be effectively and safely utilized for on-orbit operations.

5.2 CREW WORK/REST CYCLE CONSTRAINTS

Table 5.2-1 lists selected constraints on crew work/rest cycles which impact space construction and operations related to large space structures.

A key concern in setting up crew work/rest cycles is the question of adjustment of sleep periods. Pertinent considerations are described in the three paragraphs below, largely quoted from JSC-10541 (Freeman and Scales, 1978).

"Normally, each crew member will simultaneously be scheduled 8 hours of sleep following a 16-hour awake period. This 8-hour sleep period may require adjusting in real time. Four sleep stations are located in the mid-deck area ..... For operational flights, rigid sleep stations will be available." (Sleeping bags may be considered for special flight modes, if required, to permit additional stowage.) (R-4).

"Each crew member will wear ear muffs while sleeping to provide noise attenuation. If all crew members are sleeping simultaneously, at least one crew member must wear a Comm Carrier Assembly connected to a mid-deck comm outlet to assure reception of ground calls and caution and warning alarms. Another crew member will wear a dedicated headset for caution and warning (only has audio capability).

"The sleep period should be scheduled every day to start at the same local time. Variations should be limited to 2 hours daily for circadian rhythm purposes. The entry day should be limited in duration from 6 hours minimum to 12 hours maximum. This provides time to perform the deorbit preparation and limits the maximum duration to ensure an alert crew for entry. Therefore, the initiation of the sleep period is generally dictated by the landing time of the flight."

Another major concern in crew scheduling involves the question of number of shifts and their overlap during a crew work day. In general, space vehicle crews have preferred to sleep simultaneously in order to minimize disturbances in the relatively small volume of the crew cabin. The orbiter situation may be better—but not much different. However, enhancing space construction productivity tends to favor multi-shift operations (Roebuck, 1980). Therefore, it is expected that efforts will continue to find ways to minimize sleep disturbances of light, sound, motion, etc., by non-sleeping crew members. Suggested timelines for one-shift and two-shift operations with up to four crew members are described by Freeman and Scales (1978).

When there are more than four crew members on a flight, two-shift (or more) operations must be scheduled, since there are only four sleep stations

Table 5.2-1. Checklist: Constraints on Crew Work-Rest Cycles Affecting Large Space Structures

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) For pressure suit operations at 4-psi, prebreathing of 100% O_2 shall be required for three hours prior to egress (R-1) • See Section 4.8.4	NASA 1976 (a): JSC-10615
( ) For pressure suit operations at 8-psi (or approximately half the cabin atmosphere pressure) prebreathing of 100% O_2 is not required. (R-4) • 8-psi suits are not currently available. Research and development studies are continuing toward that goal.	Roebuck, 1980
( ) Crew members required to prebreath oxygen will be supplied with walk-around oxygen equipment permitting voice communication and minimal interference with orbiter housekeeping duties. IVA operations of RMS or control/display panels can be performed while prebreathing. (R-1)	Roebuck, 1980
( ) For long term space operations, the nominal crew work day is ten hours (R-1) • For short-term (1-3 days) operations up to 11 hours per day may be acceptable • Crew metabolic rates should be considered • EMU donning and doffing are included in the above work day period.	Freeman and Scales 1978: JSC-10541
( ) Space suit (EMU) donning and doffing require approximately 5 hrs per day. (R-1) • See Section 4.8.4 for detail timelines	NASA, 1976 (a): JSC-10615
( ) Current Shuttle 4-psi suits permit a maximum of 6 hours EVA work time (R-1) • The EVA work time includes necessary rest periods for heavy physical labor, travel from airlock to work site and return.	NASA, 1976 (a): JSC-10615

Table 5.2-1 (continued) Checklist: Constraints on Crew Work-Rest Cycles Affecting Large Space Structures

( ) CONSTRAINTS / • QUALIFICATIONS	REFERENCES / IMPLICATIONS
( ) The crew will follow a 16-hour awake/8-hour sleep cycle, per 24 hour period (R-1) • Adjustments can be made to accommodate circadian shift requirements and particular activities, such as EVA, which may impact work load (See Text) ( ) There will be 3/4 hour per day schedule for pre-sleep and post-sleep activities (1-1/2 hours total). (R-1) ( ) The crew meal periods will be scheduled for one hour at least twice per day (R-1) ( ) No scheduled exercise is planned for flights less than one week's duration (R-1) ( ) Orbiter housekeeping duties shall be scheduled for crew work periods (R-1) • See Section 5.3	Freeman and Scales 1978: JSC-10541

available in the orbiter. The following comments from Freeman and Scales (1978) are pertinent to planning concerns.

"1. It is recommended that activity plans should minimize the number of mid-deck activities as much as possible when another crew is sleeping. This is desirable because of the proximity of the sleep stations to the mid-deck stowage lockers, meal preparation/eating area, personal hygiene station, waste management compartment, and airlock. Activities such as MEAL, CO_2 ABSORBER REPLACEMENT, and PSA's can interfere with the shift that is sleeping.

2. For two-shift operations, significant problems occur when scheduling on-orbit STS activities on the launch and entry days. It is assumed that the CDR is on one shift and the PLT is on the other. On launch day, one shift's sleep period has to be scheduled too soon after orbital insertion so that the other shift's sleep period can begain at a reasonable time (not too late).

3. On entry day, the problem is more severe. For this situation one shift will be awake for more than 12 hours prior to the actual landing time so that the other shift can be awake for the minimum 6 hours prior to entry.

4. Problems also arise when scheduling extravehicular activities (EVA) during two-shift operations. The EVA itself would more than likely require personnel from both shifts. In addition, the EVA preparation would take place in and near the airlock which is located in the mid-deck. These activities would interfere with another shift's sleep period. Generally speaking, EVA is not compatible with a two-shift operation."

Freeman and Scales conclude that: "The issue of scheduling two-shift operations as part of the STS work day can greatly impact current crew activity planning ground rules, especially for launch, entry, and EVA operations."

On the other hand, systems analysis of space construction studies by Rockwell International (Roebuck, 1980) have shown the strong desirability of up to three-shift operations per day as a means to control launch weight, flight costs, and optimize orbiter utilization. Clearly, there needs to be further research directed toward resolving these conflicts in desires and requirements for operations with large space structures, so as to quantify impacts and improve habitability conditions which will result in optimized shift situations for the Shuttle orbiter operations.

5.3 ORBITER HOUSEKEEPING REQUIREMENTS

Crew productivity in the orbiter is affected to some degree by day-to-day housekeeping time constraints on available crew work time. Such time requirements are currently not considered to be a major impact, since they require on the order of two to three man-hours per 24-hour day. Some of the time elements are affected by number of crew (e.g., LiOH filter element changeout) and possibly by other mission functions. Table 5.3-1 (from Freeman and Scales, 1978) summarizes the major housekeeping time constraints, as well as other daily STS on-orbit activities.

Table 5.3-1. Daily STS On-Orbit Activities

C = CDR; P = PLT; M = MS
ALL = CDR, PLT, MS

STS REQUIREMENTS	CMN	TIME	FREQUENCY 1/8 HRS	1/12 HRS	1/24 HRS	REMARKS (All durations are increased to nearest 5 min increment)
SLEEP	ALL	8 hrs			X	
MEAL	ALL	1 hr	X			
PRE SLEEP ACTIVITY	ALL	45 min			X	
POST SLEEP ACTIVITY	ALL	45 min			X	
HANDOVER	ALL	15 min		X		2-SHIFT OPERATION ONLY
WASTE COMPARTMENT SYSTEM CLEANING	P/C/M	15 min			X	
WATER DUMPS	P/C	5 min+		X		FREQUENCY DEPENDENT UPON WATER QUANTITY. PERIODIC CRT MONITORING IS ALSO REQD UNTIL DUMP COMPLETED
CO_2 ABSORBER REPLACEMENT	P/C	5 min			X	FREQUENCY DEFINED FOR 2 CREWMEMBERS ONLY
FUEL CELL PURGE	P/C	5 min	X			
STATUS REPORT	P/C/M	15 min			X	LAST HALF SHOULD COINCIDE WITH GSTDN OR TDRS AOS
IMU ALIGNMENT	P/C	15 min	X			ASSUMES IMU'S AND TWO STARTRACKERS POWERED UP
TRASH STOWAGE	P/C/M	15 min			X	

6.0 SPACE SHUTTLE GROUND OPERATIONS[1]

Constraints associated with shuttle ground operations can be especially important when considering large space structure construction because many large space structures will require more than one shuttle flight to complete. Ground turnaround time between flights must be considered when calculating orbit decay of the partially completed structure left in construction orbit. Turnaround time constraints tend to become important limiting factors for dedicated orbiters which have special provisions for added cryogenic fuel tanks, nitrogen tanks, EVA aids, multi-shift operations, MMU and cherry picker stowage, and payload handling equipment interfaces. Duplication of such provisions on two or more orbiters could add significantly to costs, although launch sequence of successive flights might thereby be reduced.

In addition, ground handling operations affect design of payloads and payload carrier in regard to loads and hard points for lifting and holding during ground handling, clearances for installation, checkout provisions and environmental protection. These general concepts for payload installation were discussed in Section 3.7, and are not repeated here.

A document of particular significance to this area of payload integration concern is the KSC Launch site Accommodations Handbook for STS Payloads, K-STSM-14.1 (NASA, 1978).

6.1 KSC SPACE SHUTTLE SYSTEM GROUND FLOW

Figure 6.1-1 shows the ground flow interrelationship of the Space Shuttle system elements: orbiter, external tank, solid rocket booster, some facilities, and some GSE.

Immediately after landing, the orbiter systems are secured, and the ground cooling equipment is connected. The vehicle is then towed to the orbiter processing facility where safing, payload removal, maintenance, and checkout and premate operations are performed.

In parallel with orbiter operations, the solid rocket boosters (SRB's) are stacked and aligned on the mobile launch platform in the VAB. This is followed by the mating of the external tank (ET) to the SRB's, and then the mating of the orbiter to the ET and SRB's, after which an integrated systems test is performed and ordnance is installed prior to movement of the vehicle to the launch pad. Launch pad operations include a launch readiness verification test, hazardous servicing operations and the launch countdown. Payload changeout can also be performed on the launch pad.

After launch and separation, the SRB's descend via parachutes into the ocean where they are recovered and returned to the launch site for disassembly and return to the manufacturing site for refurbishment.

[1]"Caution: Not a controlled document! See appropriate reference documents for current data."

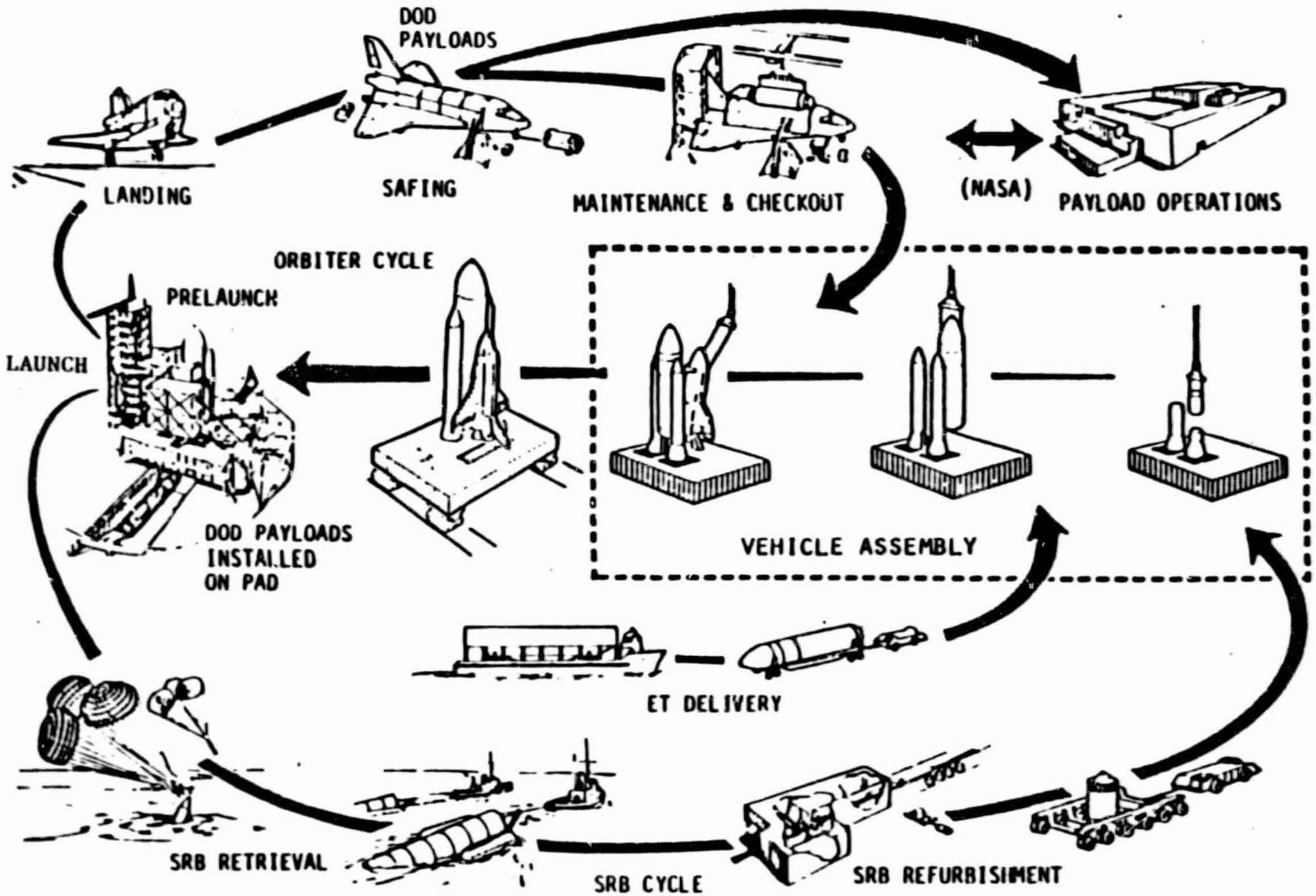


Figure 6.1-1. KSC Space Shuttle System Ground Flow

The external tank is expended on each launch, and therefore does not have a closed reuse cycle like the orbiter and solid rocket booster.

The payloads may be loaded into the orbiter during the maintenance and checkout operations or on the launch pad. Payloads returned from orbit will be unloaded from the orbiter in the orbiter processing facility and sent to payload operations for disposition.

6.2 ORBITER GROUND TURNAROUND

A design goal for the Space Shuttle orbiter and the support system is to be able to relaunch the orbiter 160 working hours after it returns/lands from an orbital mission. These 160 working hours are equivalent to 10 working days (8 hours per shift, 2 shifts per day, and 5 days per week) or 14 calendar days. This short ground turnaround time was in the interest in decreasing the maintenance cost (part of the cost per flight), decreasing the inventory of orbiters and support system elements, and increasing the utilization rate of the orbiter. Gross time allocations for these activities are shown in Figure 6.2-1.

The first actions after the orbiter lands and which require about 1 hour to accomplish are the removal of the flight crew and the attachment of ground cooling and towing equipment. The orbiter is then towed to the orbiter processing facility (OPF) where the vehicle is safed (fuels and oxidizers drained, tanks purged, and ordnance removed). Thereafter, the OMS pods/RCS, payload, and OMS propellant kit are removed, and maintenance activity on the vehicle commences. The OMS/RCS are refurbished and reinstalled, the vehicle is checked out, the payload is installed, and the vehicle/payload interfaces checked. This activity in the OPF consumes the bulk (96 working hours) of the 160-hour goal.

The vehicle is then moved to the vehicle assembly building (VAB) where it is lifted, erected, and mated to the SRB/ET that were stacked and mated while the orbiter was still in the OPF. The interfaces and integrated vehicle are checked, and ordnance is installed. The time allocated in the VAB for the orbiter is 39 working hours. When the Space Shuttle vehicle and the mobile launch platform are ready to be moved, they can be moved or maintained in this configuration for a long period.

The move to the launch pad, connecting interfaces, servicing, checkout, and launch are planned to take a minimum of 24 working hours. The system is designed to be capable of launch within two hours after starting the tanking of fuels and oxidizers.

6.3 VAFB GROUND SUPPORT SYSTEM

Figure 6.3-1 illustrates the ground support system for the STS at Vandenberg Air Force Base (VAFB), California. Most of the operations and equipment will be identical to that used at the Kennedy Space Center (KSC), Florida, but there are some differences due to variations in operating philosophies of NASA and DOD.

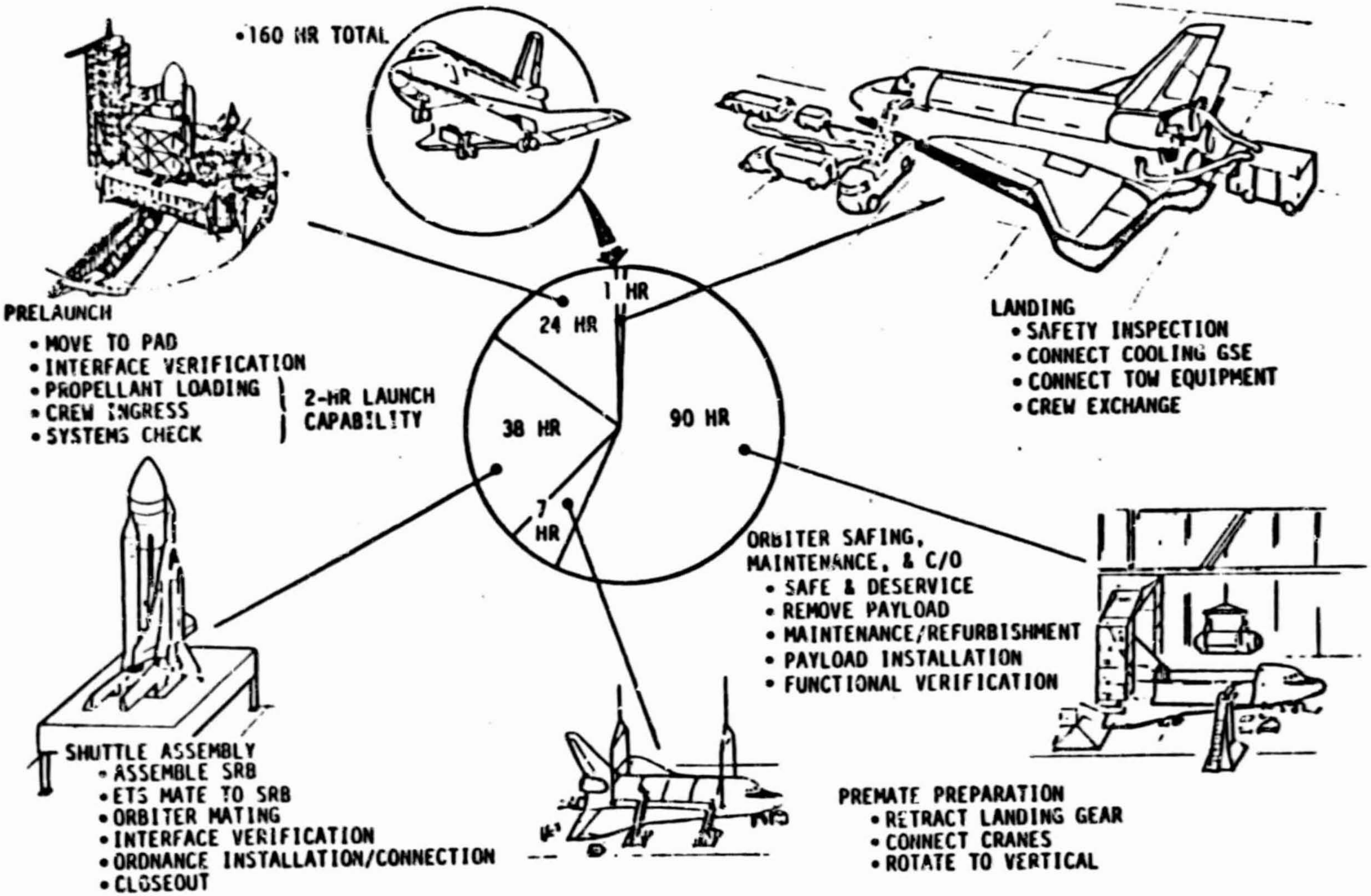


Figure 6.2-1. Orbiter Ground Turnaround

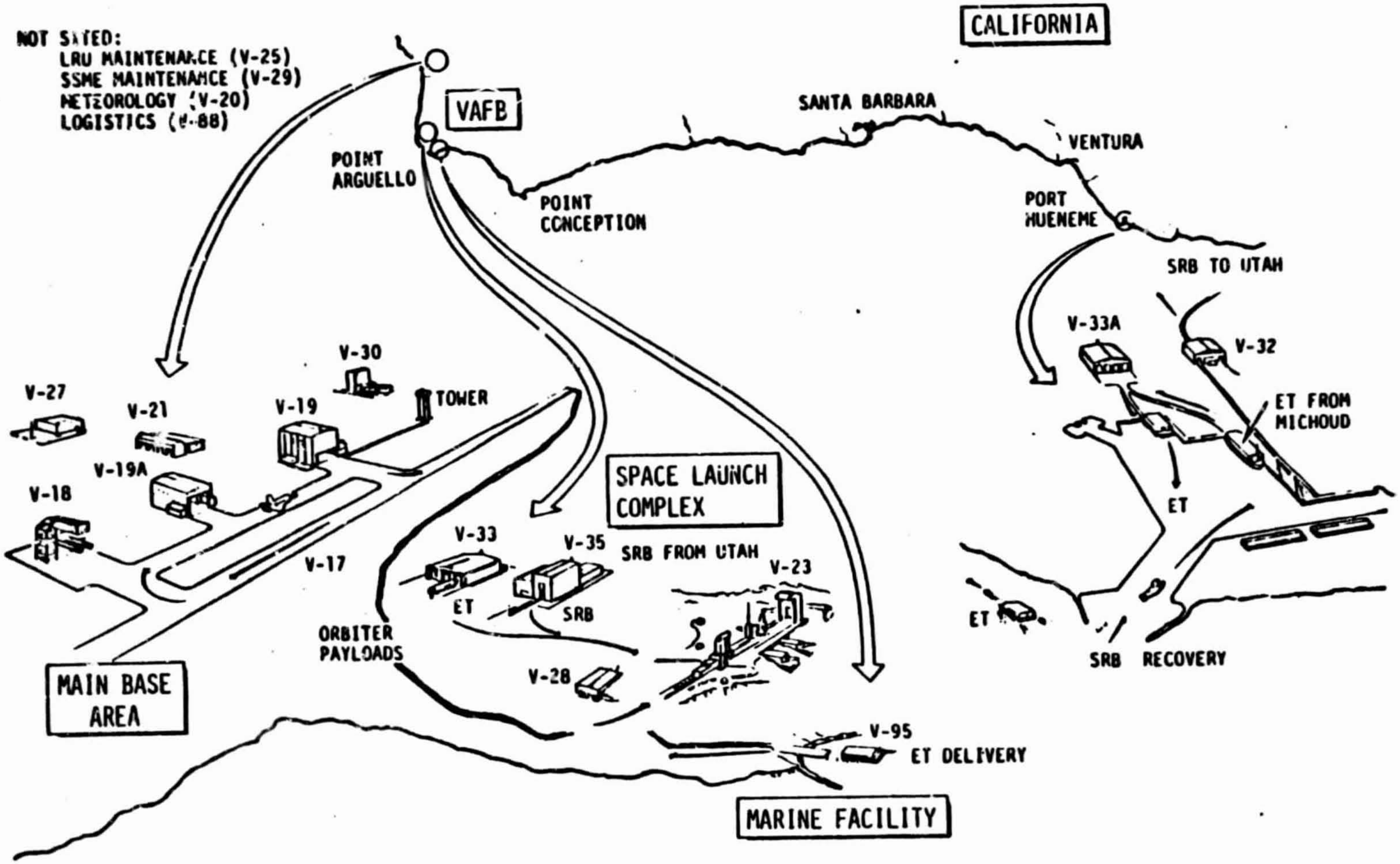


Figure 6.3-1. VAFB Ground Support System

Current plans call for extension of the landing strip (V-17) and the addition or modification of facilities at the main base area to include: orbiter/747 mate/demate device (V-18), orbiter maintenance and checkout (V-19), safing and deservicing (V-19A), hypergolic maintenance and service (V-21), flight crew systems (V-27), and parachute refurbishment (V-30). Orbiters are delivered to the SLC-6 area via a tow-way (V-80).

The Space Launch Complex (SLC-6) area was originally constructed for the Titan/MOL program. It will be modified for the Shuttle. The launch pad (V-23) will consist of a launch pad, mobile service tower, payload checkout room, launch service tower, and payload preparation room. Supporting the launch pad will be an external tank processing and storage facility (V-33), the SRB refurbishment and subassembly facility (V-35), and a launch control center (V-28). The VAFB Marine Facility (V-95) will accept ET's from Port Hueneme.

The Port Hueneme area will receive tug-recovered expended SRB's which are disassembled, and the SRM's are prepared for shipment to Utah for refurbishment with new propellant grains, while other elements of the SRB's are transported to Stations V-30 and V-35 at VAFB for refurbishment. Station V-35 receives all of the SRB elements for reassembly and storage of the SRB subassemblies. The final assembly and stacking the segments/subassemblies take place at the launch pad (V-23).

External tanks (ET's) are manufactured in Michoud, Louisiana, and are delivered four at a time on a ship. At Port Hueneme, the ET's are off-loaded, stored (in V-33A), and loaded on a barge (which carries two ET's) for delivery to VAFB Marine Facility. The ET's are transported to Station V-33 for storage until required at the Launch Site V-23.

APPENDIX A.

ACRONYMS AND ABBREVIATIONS

ACCU	Audio Central Control Unit
AFD	Aft Flight Deck
AOA	Abort to Orbit
APU	Auxiliary Power Unit
ASE	Airborne Support Equipment
ATC	Air Traffic Control
ATO	Abort to Orbit
C&DH	Command and Data Handling
CADSI	Communications and Data Systems Integration
C&T	Communications and Tracking
C&W	Caution and Warning
CCTV	Closed Circuit Television
CDR	Critical Design Review or Commander
CMDS	
CMG	Control Moment Gyro
COAS	Crewman Optical Alignment Sight
CRT	Cathode Ray Tube
DA	Deployed Assembly
DAP	Digital Autopilot
D&C	Display(s) and Control(s)
DFI	Development Flight Instrumentation
DOD	Department of Defense
DP&S	Data Processing and Software System
ECLS	Environmental Control/Life Support
ECLSS	Environmental Control/Life Support Subsystem
EMC	Electromagnetic Compatibility
EMI	Electromagnetic Interference
EMU	Extravehicular Mobility Unit
EPS	Electrical Power Subsystem
ET	External Tank
EVA	Extravehicular Activity
FID	Flight Implementation Directive
FRR	Flight Readiness Review
FSK	Frequency Shift Key
FSS	Flight Support System
FSSR	Functional Subsystem Software Requirements
GET	Greenwich Elapsed Time
GFE	Government Furnished Equipment
GN&C	Guidance, Navigation and Control
GO_2	Gaseous Oxygen

GPC	General Purpose Computer
GSE	Ground Support Equipment
GSS	Ground Support System
HPA	Holding-Position Aid
HQ	Headquarters
ICD	Interface Control Drawing
IH/SR	Integrated Hardware/Software Review
IMU	Inertial Measuring Unit
IVA	Intravehicular Activity
JSC	Lyndon B. Johnson Space Center
KSC	John F. Kennedy Space Center
LEO	Low Earth Orbit
LH_2	Liquid Hydrogen
LiOH	Lithium Hydroxide
LOS	Line of Sight
LOX	Liquid Oxygen
LSA	Launch Services Agreement
LSS	Large Space Structures/Systems
LSSM	Launch Site Support Manager
LV	Local Vertical
MCDS	Multi-functional Cathode Ray Tube Display System
MDM	Multiplexer/Demultiplexer
MECO	Main Engine Cutoff
MESA	Modular Equipment Stowage Assembly
MMS	Multi-mission Modular Spacecraft
MMU	Manned Maneuvering Unit
MPS	Main Propulsion System
MSBLS	Microwave Scan Beam Landing System
MSS	Mission Specialist Station
NASA	National Aeronautics and Space Administration
NAVAID	Navigation Aid
NMI	Nautical Miles
OMS	Orbital Maneuvering System
OPF	Orbiter Processing Facility (at KSC)
ORGS	Organizations
PCR	Payload Changeout Room
PDR	Preliminary Design Review
PDRS	Payload Deployment Retrieval System
PEP	Power Extension Package
PGHM	Payload Ground Handling Mechanism
PIDA	Payload Installation and Deployment Aid
PIP	Payload Integration Plan
PL	Payload
PLT	Pilot
PM	Phase Modulated
PMF	Performance Monitoring System
POCC	Payload Operations Control Center
POP	Perpendicular to Orbital Plane

PRCS	Primary Reaction Control System
RCS	Reaction Control System
RF	Radio Frequency
RHC	Rotation Hand Controller
RMS	Remote Manipulator System
RTLS	Return to Launch Site
SCF	Satellite Communication Facility
SLC	Space Launch Complex
SM	System Management
SMCH	Standard Mixed Cargo Harness
SPIDPO	Shuttle Payload Integration and Development Program Office
SRB	Solid Rocket Booster
SREP	Space Radiation Environments Program
STDN	Space Tracking and Data Network
STS	Space Transportation System
TACAN	Tactical Air Navigation
TAEM	Terminal Area Energy Management
TBD	To be determined
TDRS	Tracking Data and Relay Satellite(s)
TDRSS	Tracking Data and Relay Satellite(s) System
THC	Translation Hand Controller
TPI	Terminal-Phase-Initiate
TVC	Thrust Vector Control
VAB	Vehicle Assembly Building
VAFB	Vandenberg Air Force Base
VRCS	Vernier Reaction Control System
VV	Velocity Vector

APPENDIX B

GLOSSARY

abort The termination of a mission following any STS or payload failure occurring during the mission. This will initiate unscheduled payload ground operations, including removal, disassembly, deservicing, and return to the factory or launch site.

Airborne Support Equipment (ASE) All unique and common payload equipment located in the Orbiter to support the mission. It comprises payload support frames, cables, attachments, panels, etc.

aft flight deck That part of the Orbiter cabin on the upper deck where payload controls can be located.

airlock A compartment, capable of being depressurized without depressurization of the Orbiter cabin, used to transfer crewmembers and equipment. (A similar compartment in the Spacelab module is used to expose experiments to space.)

azimuth True launch heading from KSC or VAFB measured clockwise from 0° north.

barbecue mode Orbiter in slow roll for thermal conditioning.

beta angle Minimum angle between the Earth-Sun line and the plane of the orbit.

capture The event of the remote manipulator system end effector and/or other handling aids making contact with and firmly attaching to a payload grappling fixture. A payload is captured at any time it is firmly attached to the remote manipulator system.

automated payloads Those payloads which are supported by an unmanned spacecraft capable of operating independently of the STS. Automated payloads are detached from the Orbiter during their operational phase of their flights.

cargo The total complement of payloads (one or more) on any one flight. It includes everything contained in the Orbiter cargo bay, plus other equipment, hardware, and consumables located elsewhere in the Orbiter that are user-unique, and are not carried as a part of the basic Orbiter payload support.

cargo bay (or payload bay) The unpressurized mid-part of the Orbiter fuselage behind the cabin aft bulkhead where most payloads are carried. Its maximum usable payload envelope is 15 feet (4.6 meters) in diameter and 60 feet (18.3 meters) long. Hinged doors extend the full length of the bay.

cargo bay assembly The combination of a payload and all its ASE that is installed in the Orbiter cargo bay.

cargo bay liner (or payload bay liner) Protective soft material used to isolate sensitive payloads from the bay structure.

cargo integration review Part of STS planning process that results in a cargo manifest, cost per flight, and billing schedule.

cargo integration test equipment Setup at KSC that can provide testing of both payload-to-payload and cargo-to-Orbiter interfaces.

certificate of compliance Documentation prepared by the user confirming that a payload has successfully completed interface verification.

commander This crewmember has ultimate responsibility for the safety of embarked personnel and has authority throughout the flight to deviate from the flight plan, procedures, and personnel assignments as necessary to preserve crew safety or vehicle integrity. The commander is also responsible for the overall execution of the flight plan in compliance with NASA policy, mission rules, and Mission Control Center directives.

contingency Any deviation from the planned flow and schedule for cargo or payload preparation and launch or on-orbit operations.

crew activity planning The analysis and development of activities to be performed in flight by the crew, resulting in a time line of these activities and reference data for each flight.

deadband That attitude and rate control region in which no orbiter reaction control subsystem or vernier correction forces are being generated.

dedicated cargo mission A flight dedicated to one payload or STS cargo element.

deployment The process of removing a payload from a stowed or berthed position in the cargo bay and releasing that payload to a position outside the cargo envelope, or attached to a large space structure.

external tank Element of the Space Shuttle system that contains liquid propellant for the Orbiter main engines. It is jettisoned prior to orbiter insertion.

extravehicular activity All activities for which crewmembers don their space suits and life support systems (Extravehicular Maneuvering Units), then egress the orbiter cabin (through an airlock, except in emergencies) into a space vacuum environment to perform duties (internal or external to cargo bay volume).

extravehicular mobility unit A self-contained (no-umbilicals) life support system and anthropomorphic pressure garment for use by crewmembers during extravehicular activity. It provides thermal and micrometeoroid protection.

flight The period from launch to landing of an Orbiter - a single Shuttle round trip. One flight might deliver more than one payload; more than one flight might be required to accomplish a single mission.

flight deck assembly The aggregate of the ASE, such as control consoles, that is installed in the Orbiter cabin upper deck.

flight kit Optional hardware (including consumables) to provide additional, special, or extended services to payloads. Kits are packaged in such a way that they can be installed and removed easily.

flight manifest The designation of a flight, assignment of the cargo to be flown, and specific implementing instructions for STS operations personnel.

flight phases Prelaunch, launch, on-orbit, deorbit, entry, landing, and post-landing.

flight types Payload deployment and retrieval, on-orbit servicing of satellites, and on-orbit operations with an attached payload, as suited to the purposes of a mission. A single flight may include more than one of these purposes.

free-flying system (or free-flyer) Any satellite or payload that is detached from the Orbiter during operational phases and is capable of independent operation.

Ground Support Equipment (GSE) All the unique and common payload equipment required to support ground processing operations for a cargo element. It is comprised of flight hardware handling, devices, test sets, fluid transfer and metering systems, etc.

inclination The angle between the plane of the orbit and the equatorial plane.

integration A combination of activities and processes to assemble payload and STS components, subsystems, and system elements into a desired configuration, and to verify compatibility among them.

interface The mechanical, electrical, and operational common boundary between two elements of a system.

interface verification Testing of flight hardware interfaces by an acceptable method that confirms that those interfaces are compatible with the affected elements of the Space Transportation System.

launch pad The pad area from which the Space Shuttle will be launched. The stacked Space Shuttle will undergo final prelaunch checkout and countdown at the launch pad.

launch site support manager Individual at KSC who is the single point of contact with users in arranging payload processing at the launch site.

long duration exposure facility Free-flying reusable satellite designed primarily for small passive or self-contained active experiments that require prolonged exposure to space. It is launched in the Orbiter cargo bay and deployed and retrieved by the remote manipulator system.

manned maneuvering unit A propulsive backpack device for extravehicular activity. It uses a low-thrust, dry, cold nitrogen propellant.

mission The performance of a coherent set of investigations or operations in space to achieve program goals. A single mission might require more than one flight, or more than one mission might be accomplished on a single flight.

Mission Control Center Central area at JSC for control and support of all phases of STS flights.

mission kit Flight kit is the preferred term.

mission specialist This crewmember is responsible for coordination of overall payload/STS interaction and, during the payload operations phase, directs the allocation of the STS and crew resources to the accomplishment of the combined payload objectives. The mission specialist will have prime responsibility for experiments to which no payload specialist is assigned, and/or will assist the payload specialist when appropriate.

mission station Location on the Orbiter aft flight deck from which payload support operations are performed, usually by the mission specialist.

mixed payloads Cargo containing more than one type of payload.

mobile launch platform The structure on which the elements of the Space Shuttle are stacked in the Vehicle Assembly Building and are moved to the launch pad.

mobility aid Handrails or footrails to help crewmembers move about the spacecraft.

module Typically a pressurized manned laboratory suitable for conducting science, applications, and technology activities. Unmanned modular units may also be described by this term.

module exchange mechanism Part of the Multimission Modular Spacecraft flight support system that is used for servicing.

Multimission Modular Spacecraft Free-flying system built in sections so that it can be adapted to many missions requiring Earth-orbiting remote-sensing spacecraft. It is launched in the Orbiter cargo bay and deployed and retrieved by the remote manipulator system.

multiple payloads When more than one payload is carried in the cargo bay. Multiple payloads involve multiple Payload Mission Managers.

multiuse mission support equipment Hardware available at the launch site for handling payloads, or common flight hardware used by various payload disciplines.

nadir That point on the celestial sphere vertically below the observer, or 180° from the zenith.

off-line integration Assembly of payload elements or multiple payloads that does not involve any STS element.

on-line integration Mating of payloads with the Orbiter, Spacelab, or upper stage. Level I is with the Orbiter. Level II is with the Spacelab, upper stage, etc.

operations planning Performing those tasks that must be done to ensure that vehicle systems and ground-based flight control operations support flight objectives.

orbital flight test One of first four scheduled developmental space flights of the Space Shuttle System.

orbital maneuvering subsystem Orbiter engines that provide the thrust to perform orbit insertion, circularization, or transfer; rendezvous; and deorbit.

Orbiter Manned orbital flight vehicle of the Space Shuttle system.

Orbiter Processing Facility Building near the Vehicle Assembly Building at KSC with two bays in which the Orbiter undergoes postflight inspection, maintenance, and premate checkout prior to payload installation. Payloads are also installed horizontally into the Orbiter in this building.

pallet An unpressurized platform, designed for installation in the Orbiter cargo bay, for mounting instruments and equipment requiring direct space exposure.

pallet train More than one pallet rigidly connected to form a single unit.

payload The total complement of specific instruments, space equipment, support hardware, and consumables carried in the Orbiter (but not included as a part of the basic Orbiter payload support) to accomplish a discrete activity in space. Also referred to as a cargo element.

payload changeout room An environmentally controlled room at the launch pad for inserting payloads vertically into the Orbiter cargo bay.

payload discipline training Preparation of a mission or payload specialist for handling a specific experiment. This training is usually the responsibility of the user.

Payload Operations Control Center Central area, located at any of three NASA centers, from which payload operations are monitored and controlled. The user, in many instances, will have direct command of a payload from this control center.

payload preparation room Facility at the Vandenberg Air Force Base launch pad for processing and checking payloads.

payload specialist This crewmember, who may or may not be a career astronaut, is responsible for the operation and management of the experiments or other payload elements that are assigned to him or her, and for the achievement of their objectives. The payload specialist will be an expert in experiment design and operation.

payload station Location on the Orbiter aft flight deck from which payload-specific operations are performed, usually by the payload or mission specialist.

payload supplier Owner/operator of any Space Shuttle payload.

pilot This crewmember is second in command of the flight and assists the commander as required in the conduct of all phases of Orbiter flight.

reactor control subsystem Thrusters on the Orbiter that provide attitude control and three-axis translation during orbit insertion, on-orbit, and reentry phases of flight.

remote manipulator system Mechanical arm on the cargo bay longeron. It is controlled from the Orbiter aft flight deck to deploy, retrieve, or move payloads.

retrieval The process of utilizing the remote manipulator system and/or other handling aids to return a captured payload to a stowed or berthed position. No payload is considered retrieved until it is fully stowed for safe return or berthed for repair and maintenance tasks.

simulator A heavily computer-dependent training facility that imitates flight hardware responses.

solid rocket boosters Element of the Space Shuttle that consists of two solid rocket motors to augment ascent thrust at launch. They are separated from the Orbiter soon after lift-off and recovered for reuse.

scar Additional thickness of material, brackets or stiffeners and possibly holes incorporated in structure (of Orbiter in this context) to provide fastening points for future installations of non-standard equipment, electrical lines or fluid lines.

spacecraft (SC) or satellite The combination of mission equipment and a carrier capable of autonomous operations in space. The spacecraft or satellite may include an integral propulsion system. In this document, the terms "spacecraft", "satellite", and "automated payload" are synonomous, and apply to large space structures and construction fixtures, etc. capable of free-flight during the absence of the orbiter, either in a partially constructed stage or as a fully completed project.

Spacelab

A general-purpose orbiting laboratory for manned and automated activities in near-Earth orbit. It includes both module and pallet sections, which can be used separately or in several combinations.

Space Shuttle Orbiter, external tank, and solid rocket boosters.

space tracking and data network A number of ground-based stations having direct communications with NASA flight vehicles.

Space Transportation System An integrated system consisting of the Space Shuttle (Orbiter, external tank, solid rocket boosters), upper stages, Spacelab, and any associated flight hardware and software.

Space Transportation System (STS) Cargo Element The combination of the cargo bay and flight deck assemblies associated with one payload.

STS cargo The combination of one or more cargo elements and all NASA-supplied mission kits provided for special or extended services that are not carried in the standard baseline Orbiter.

stability rate The maximum angular rate error during steady state limit cycle operation.

stowing The process of placing a payload in a retained position in the cargo bay for ascent or return from orbit.

Tracking and Data Relay Satellite system Two-satellite communication systems providing principal coverage from geosynchronous orbit for all STS flights.

user An organization or individual requiring the services of the Space Transportation System.

utilization planning The analysis of approved (funded or committed) payloads with operational resources, leading to a set of firm flight schedules with cargo manifests.

Vehicle Assembly Building (VAB) High-bay building near KSC launch pad in which the Shuttle elements are stacked onto the mobile launch platform. It is also used for vertical storage of the external tanks.

Western Launch Operations Division NASA operation at Vandenberg Air Force Base.

APPENDIX C

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ALPHABETICAL SUBJECT INDEX

In the following index, subject matter in the document is arranged alphabetically and referenced to the document Section number. It is recognized that this method is less precise than listing page numbers. However, the method greatly facilitates updating the document.