NASA Contractor Report 159091

ERECTABLE SPACE PLATFORM FOR SPACE SCIENCES
AND APPLICATIONS

ROCKWELL INTERNATIONAL
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FOREWORD

The Satellite Systems Division of Rockwell International has been conducting a study of an erectable space platform as a host vehicle for space sciences and applications payloads. This work was performed under Contract NAS1-15322, Task Assignments Nos. 3 and 4, for the Langley Research Center, National Aeronautics and Space Administration.

This report documents the findings of that study, discussing the concept for the orbiting host vehicle, the requirements that gave rise to its configuration, and the programs of technology that are suggested as leading toward its eventual development.

The work was conducted under the direction of F. A. Zylius, Rockwell's Program Development Manager for Advanced Space Base Programs, and J. A. Boddy, Study Project Engineer. The following made major contributions of this study:

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1.0 INTRODUCTION AND SUMMARY

Over the last 12 to 18 months, the advanced planning studies for space science operations in earth orbit have been sponsored by the NASA Offices of Space Sciences (OSS), Aeronautics and Space Technology (OAST), and Space Transportation (OST). The NASA studies have indicated the feasibility and advantages of platforms providing services to multiple payloads for longer-duration (greater than 7 to 30 days) Shuttle orbiter supported missions.

The Space Shuttle now makes it reasonable to consider the emplacement on orbit of a facility which acts as host to the groupings of payloads, providing power, heat dissipation, stability, and data handling services. The facility is also the central location to which the Shuttle can return for multi-payload servicing, modification, exchange, and even operation with manned attendance.

The specific objectives of this study were to (1) identify a viable conceptual design for the service module/platform, (2) assess the technology issues that must be faced in planning development, and (3) prepare an initial plan for bringing critical technologies up to acceptable levels.

The seven tasks involved in this study were structured as shown in Figure 1, and their schedule is illustrated in Figure 2. The primary inputs to the study were the payload and mission data supplied by the OSS and Office of Space and Terrestrial Applications (OSTA), References 1 and 2; the large space structures

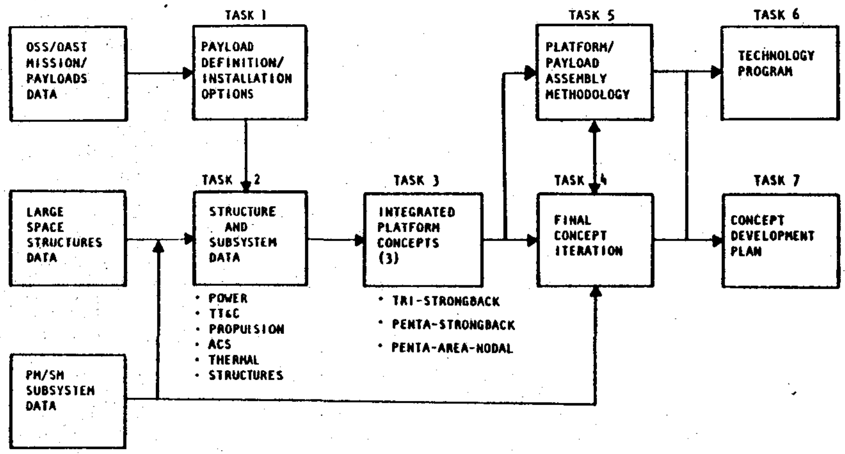


Figure 1. Erectable Platform System Study Flow

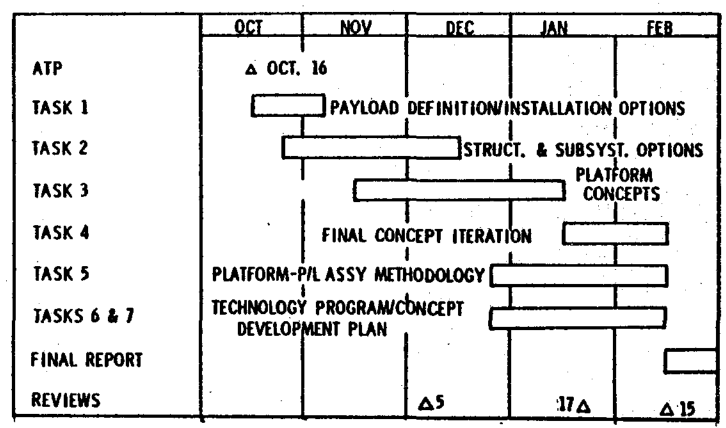


Figure 2. Erectable Platform System Study Schedule

technology data developed previously by Rockwell International, References 3 and 4; and the design and performance data on power modules and orbiting service modules from past and concurrent investigations, References 5, 6, and 7.

Tasks 1 and 2 dealt with the assimilation and analysis of the above data in order to develop alternate platform concepts (Task 3) for servicing the payloads described in the mission model. From an original dozen candidates, three platform concepts were defined and analyzed in some detail; and of these, one was selected for detailed definition (Task 4) to act as a baseline for defining technology issues pertinent to large payload platform development, emplacement in orbit, and operation. Task 5, analysis of platform construction methodology in orbit, was carried on to support Task 4. Finally, a technology program definition (Task 6) was carried out in some detail, and a first, gross development plan was prepared (Task 7).

1.1 PLATFORM REQUIREMENTS

The viable service module/platform concepts were driven by payload and mission data developed as a separate task by OSS and OSTA, and furnished to the study task team as the basis of mission requirements. Other critical guidelines supplied by the NASA which bounded the scope of the study were as follows.

- Platforms accommodating these payloads would be able to serve a minimum of 10 to 12 Shuttle-ERNO pallet equivalents.
- Payload weights will average 300 kg with a maximum payload weight of 1000 kg.
- For earth viewing, payloads will be permitted at least a 60° half-zone field of view (FOV) and 2π steradian FOV for space viewing.
- The platform will furnish each payload position with (1) electric power to a maximum of 5 kW, (2) heat rejection, (3) data acceptance and storage, and (4) pointing accuracy of 0.01° at the payloads-platform interface (revised to 0.1° during the study).
- A propulsion system will be included for drag makeup.
- A system will be included to emplace, handle, or replace payloads, either remotely from the orbiter's control station or by EVA.
- The platform will have a 10-year life, with maintenance and element replacement.

It was the specific objective of the payloads analysis to develop groupings of payloads which resulted in requirements that could be met by single platforms acting as host vehicles in low earth orbit. NASA provided data on 65 representative payloads to be flown in the 1983-1989 period. These missions include earth surveillance, solar terrestrial observation, astronomy and cosmology, and space physics. Life science and material processing missions were ground-ruled out of this study. Of the 65 payloads for which NASA furnished

Table 1. Platform Requirements Summary (Circa 1985)

Item	Platform No. 1	Platform No. 2	Platform No. 3
Orbit	400 km, 28°, circ.	575 km, 90°, circ.	400 km, 57°, circ.
Stabilization	Inertial	Inertial, solar pointing	Earth local vertical
Payload quantity	11	7	13
Total weight (kg)	15,300	18,500	10,105
Total average power (kW)	14.5	6.6	16.4
Standard IPS*	5 payloads	2 payloads	4 payloads
Super IPS	2 payloads	None	None
Platform spaces (pallet equivalents)	13	12	11
Cryogens	3	TBD	2

Notes:

1. Assumes compromises can be made to accommodate extraordinary orbit, viewing, and replenishment requirements
2. Based on optimum grouping of payloads through 1985

*Additional IPS (Instrument Pointing System) or gimbals may be required to obtain required field of view.

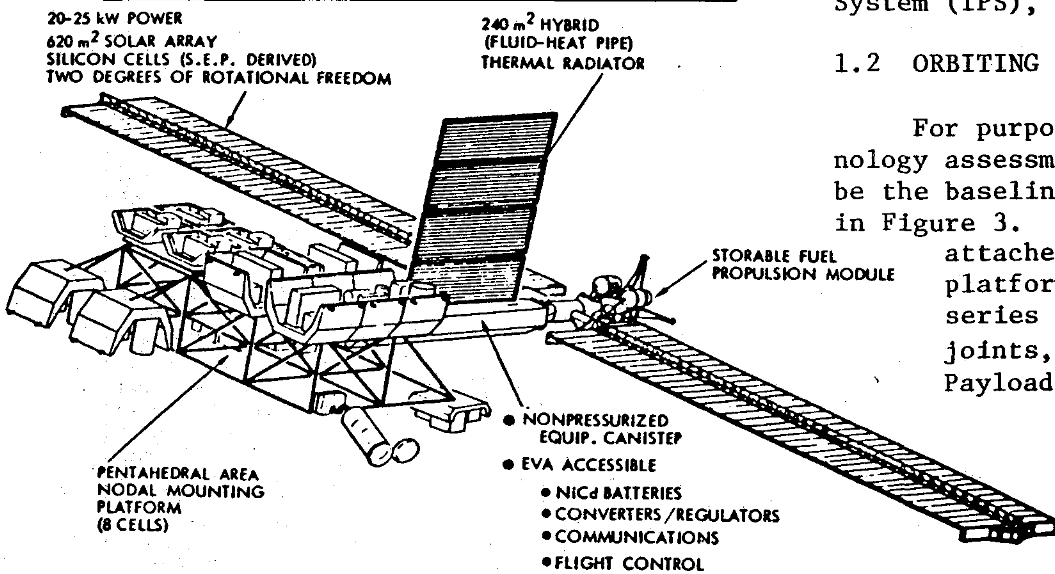


Figure 3. Baseline Configuration—Platform 1 (8 Payload Cells; Pentahedral Area Nodal Mounting)

requirements data, 55 were considered to be acceptable payloads for platform accommodation. Of these, 31 were considered to be accommodated on the platform system by 1985. These 31 formed the baseline requirement for the platform systems and are summarized in Table 1.

The payload model for 1985, then, is accommodated by three platforms: one at 400 km and 28°, the second at 575 km and 90°, and a third at 400 km and 57°. On each, the majority of payloads was assumed to be oriented to the same general direction as the platform itself (i.e., inertial or earth local vertical). Each, however, had more than one exception. Because of the design of individual missions, some payloads required a specific orbit location (e.g., 90° orbit), but needed a payload orientation that differed from the platform (e.g., earth local vertical). For these, a gimbal system was conceived, typical of that used on NASA's Instrument Pointing System (IPS), on which the unique payload could be mounted.

1.2 ORBITING PLATFORM CONCEPTUAL DESIGN

For purposes of platform subsystem description and technology assessment, Platform No. 1 of Table 1 was assumed to be the baseline platform whose configuration is illustrated in Figure 3. The system consists of a utilities module attached to a pentahedral matrix payload accommodations platform. The platform is constructed in orbit with a series of cylindrical struts, structural unions or joints, utilities distribution cables, and service lines. Payloads are attached to a variety of mounts, pallets, or equipment platforms. The utilities module has four solar electric propulsion (SEP) arrays which will supply an average of 25 kW to the payload platform. Solar arrays are mounted to the module structure in a manner to permit two-degree-of-rotation freedom, a feature which assures complete flight-mode flexibility. Heat dissipation thermal radiators are rigidly mounted to the the canister structure that contains the energy storage, power control, electronic, and stabilization equipment. Thermal radiators are sized to dissipate all the heat generated by the orbiting complex, including payloads. The propulsion module is used for orbit makeup, orbit transfer, and as a reaction control system (RCS).

From Figure 4, it can be seen that the entire orbiting system is delivered (without payloads) to orbit in a single Shuttle flight. The utility module is deployed first by lifting the large equipment canister with thermal radiators attached from the payload bay, inverting 180° and mounting on an assembly fixture attached to the Shuttle. Then, the solar array and its drives are removed from the bay and mounted to the canister's front face. This is followed by the propulsion module. Finally, all arrays and radiators are deployed. Once the power/utilities unit has been deployed, the payload accommodations platform is erected and attached to the utilities module.

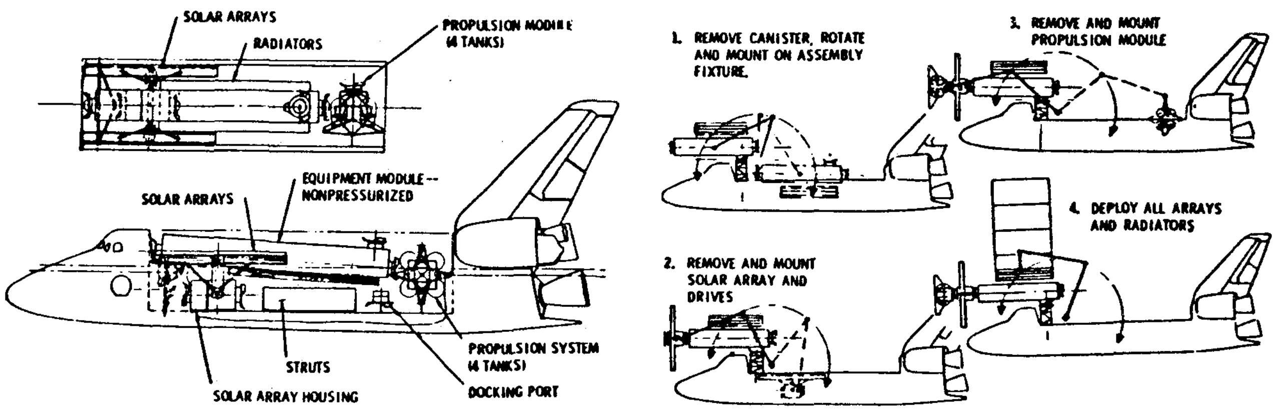


Figure 4. Moderate Size Platform (8 Cells, 4 Arrays) Shuttle Packaging and Deployment

After the platform has been assembled, serviced, and checked out, payloads can be brought to the orbiting complex by subsequent Space Shuttle flights. In these, the Shuttle orbiter is docked to a docking interface provided for that purpose on the utility module's equipment canister. Platform dimensions are such that all payload positions can be reached by a somewhat modified Shuttle remote manipulator system (RMS).

1.3 UTILITIES MODULE

Each major subsystem was analyzed and trade studies conducted among alternate components and concepts. The selected subsystem elements were considered with respect to their technology-level requirements as being (1) enabling, (2) enhancing, or (3) normal engineering development or well within current state of the art. "Enabling" suggests that critical technology advancement activity is required to avoid a potentially serious problem in the course of system development. "Enhancing" suggests that significant program advantages can be realized with properly oriented technology effort.

The overall electrical power subsystem (Figure 5) has each of the four solar arrays associated with a battery and battery charger as a set. Each set is capable of feeding either of two redundant main high-voltage dc (HV dc) buses, but would normally be dedicated to one. Regulators buck the HV dc down to 28 V dc for utility module subsystems and for the orbiter interface. The orbiter interface also allows the Shuttle to provide the utility module initialization power and control. HV dc is the primary supply to the platform, where regulators buck it down to 28 V dc for payload use. Inverters provide 115 V, 400 Hz, 3Ø power for payloads and utility module subsystems (Freon pump, etc.).

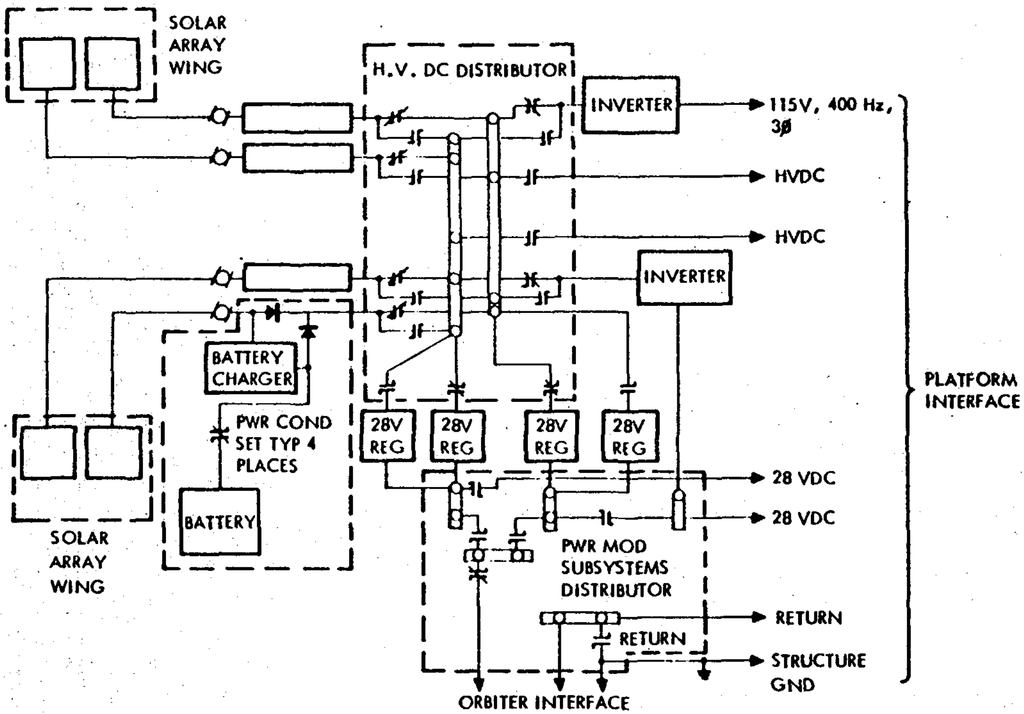


Figure 5. Power Subsystem

The thermal control system, shown in Figure 6, is based on the use of Freon 21 cooling loop using a circulating pump and reservoir for circulating the Freon and then through the heat loads. The radiator bypass valve

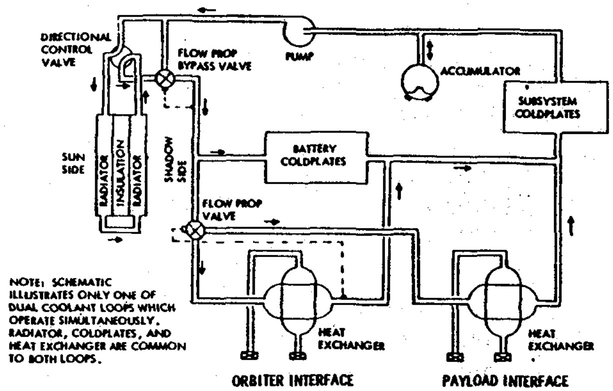


Figure 6. Thermal Control Subsystem

controls the mixed radiator outlet temperature to 5°C. The radiator itself is conceived as double-sided with insulation between sides. A directional control valve with sensors on each side of the radiator measures the incident radiation on each surface and adjusts the control valve so the Freon flows through the hot side first, and then the cold side. This results in about a 25% reduction in radiator size over the conventional double-sided radiator.

A propulsion module may be required for three reasons: to enable makeup of altitude deterioration due to aerodynamic drag, to function as the transportation medium for a platform that must operate at an altitude somewhat higher than that at which the Space Shuttle has an effective payload (e.g., Platform No. 2 that operates at 575 km, but must rendezvous with Shuttle at 400 km for servicing and payload delivery), and to assist in vehicle control if, as a last resort, a propulsion RCS component becomes necessary. The orbit transfer requirements for Platform No. 2 are the most demanding and result in a propulsion module weight of about 2320 kg (5100 lb). The hydrazine propellant was assumed to have a specific impulse of 2255 N-sec/kg (230 sec). If the thrust-to-weight is kept below 0.01, the thrust loads will not impose significant design loads on the flexible structural elements. There is a potential contamination problem of the exhaust plume of the propulsion module with the solar arrays, radiators, and a few potential payload users of the platform.

The communications, command, and control subsystem provides communication links from the platform and its payload to interface with the ground stations directly (S-band) or with relay (S-band or K-band) using TDRS. The unit also provides an interface with the Navstar Global Positioning System (GPS) to receive accurate position, velocity, and timing references for platform experiments. The major equipment items include a K-band system (for high-data-rate transmission), an S-band system (for command, control, and low-data-rate transmission), and an L-band system (to work with GPS); an on-board data recording and playback unit for continuous real-time experiments; and such other equipment as a PCM/encoder, signal conditioning units, Mux/Demux, timers, and data storage elements.

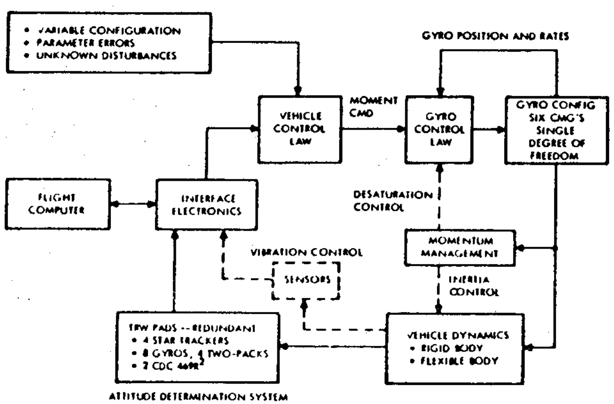


Figure 7. Attitude Control Subsystem Block Diagram

The attitude control block diagram (Figure 7) was used to develop an understanding of the factors that will influence the control of this large, complex, flexible space structure. The principal issues shown on the figure include the problems of plant growth during configuration buildup, modeling uncertainties, use of classic or modern control system theory, and various forms of inertia control, momentum management, and desaturation.

The initial study guideline recommended a pointing and stability accuracy of 0.01° at the payloads/platform interface. Of the 31 payloads (Circa 1985) in the NASA mission model, 19 are satisfied by a pointing accuracy of 0.1°. The remaining payloads require accuracy in excess of 0.01 degree. An analysis of sources of control error buildup indicated that it would be extremely

Table 2. Control Accuracy Budgets

Error Contributions	Design/Mission Solutions	Accuracy Budget (°) Nearest Payload	Accuracy Budget (°) Worst-Case Payload*
Attitude reference determination	Positioning of sensors single or multiple	0.0015	0.0015
Thermal deformation • Short-term transients • Steady state	Material selection Thermal coating Thermal control	0.005	0.016 (0.20)
Manufacturing and assembly	Tolerance buildup On-orbit measurement of bias Figure control	0.012	0.013 (0.010)
Control errors • Bandwidth • Flexible dynamics interaction	Design stiffness Control complexity	0.020	0.020
RSS total error		0.024	0.029 (0.201)

*Payload position furthest away from centralized attitude reference position.
( )Side-mounted aluminum pallet/payloads

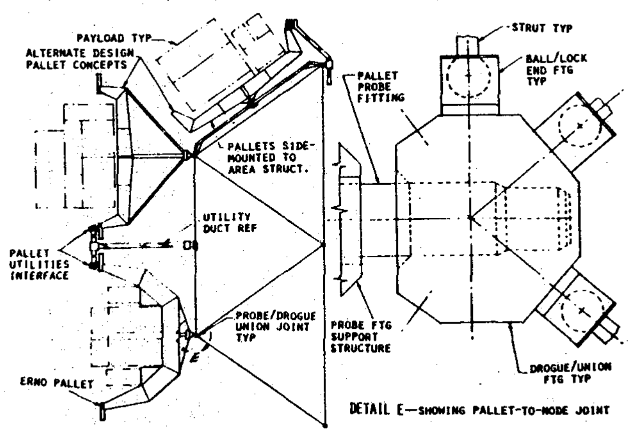


Figure 8. ERNO and Alternate Pallets Node Mounted

difficult to achieve 0.01° (Table 2). Errors resulting from carefully controlled manufacturing and on-orbit assembly could produce about 0.012° error, and the thermal deformation is constrained to 0.01° by resorting to a composite material with a low coefficient of thermal expansion. The obvious conclusion is that ±0.1° is adequate for the platform. Payloads requiring higher levels of accuracy will require a separate reference system like the Instrument Pointing System (IPS).

1.4 ERECTABLE PLATFORM

Several concepts of erectable platforms were developed and analysed in arriving at the configuration depicted in Figure 3. The alternate concepts included (1) triangular strongback, where payload-carrying pallets are rigidly mounted to a spine and rib structure, the structure including utilities channels and the basic units are mounted end to end; and (2) pentahedral cells built up of composite material struts and unions to form a rigid spine and extended rapid angles to form a pair of ribs—payload/pallets are attached at reinforced locations along the pentahedron's side.

The recommended baseline is a pentahedral area, payload nodal mounted configuration. Here, the pentahedral cells are joined in an area matrix with the payloads mounted to the strut/union junctions. Alternative mountings include modified ERNO pallets, mini-pallets, groups of small payload packages placed on common "tables" which in turn were mounted on the platform, payloads mounted to IPS gimbaled subsystems, and large payloads attached directly to the platform. In Figure 8 are shown a group of small mission packages mounted on standard and highly modified Shuttle pallets which are, in turn, mounted at junctions of the platform structure. The primary mount is a probe/drogue fitting illustrated on the right of the figure. Its length is dictated by the 13-inch space available when the pallet is mounted in the shuttle payload bay.

In Figure 9, the system picks up utilities service at the interface junction between the utilities module and payload accommodations. Utility ducts are distributed over the area

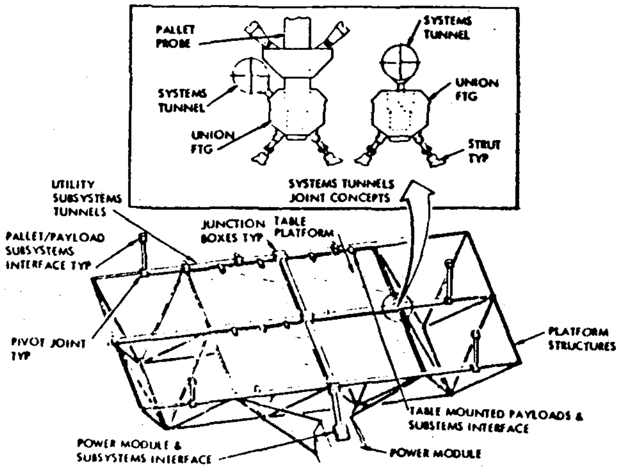


Figure 9. Utility Subsystem Distribution for Front Side

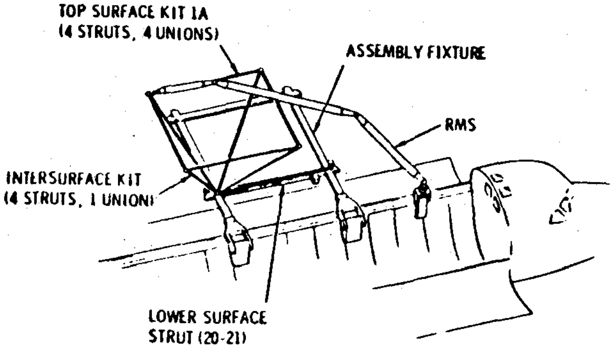


Figure 10. Platform Cell Assembly

structure from that point, with junctions at each payload interface location. Also shown are a pair of utility tunnel mounting concepts.

The platform was conceived to be erected in orbit with the help of the orbiter's remote manipulator system (RMS) and assembly fixtures deployed from the cargo bay. Platform components consist of two different length struts and unions, some preassembled in various cell kits. On-orbit assembly is accomplished off the side of the orbiter (Figure 10) where kits are deployed from the cargo bay, installed onto the assembly fixture and connected to existing platform cells.

Similar steps and requirements have been identified for completing the structure, its mounting to the utilities module, and for the installation of the utilities ducts between the utilities module and the payload positions on the payload accommodations platform.

Each step in the construction operation was examined for the requirements that step imposes on some piece of equipment, machinery, procedure, or structural item itself. The operational analysis of the platform erection process uncovered technology issues to which attention should be directed.

1.5 TECHNOLOGY PLANNING

The analysis of the erectable platform subsystem by subsystem resulted in the identification of numerous technology items that need to be resolved for platform development. The primary objective of the study was to identify the platform technology requirements, and recommend technology development for the deficiency areas. Of the approximately 100 technology items identified, technology advancement plans were required for one half of the items (Table 3). Twenty items were judged to be in the enabling category (i.e., the science and applications platform development program would proceed at higher levels of risk without significant advances in these technology areas).

Table 3. Technology Planning Summary

Subsystem	Technology Items Identified	Technology Advancement Plans	Enabling Items	Examples of Enabling Technologies
Power	12	5	1	Solar array deployment
Thermal	9	3	0	
Data, communication, control	6	2	0	
Structures	5	2	2	Joints and joining
Propulsion	7	4	2	Contamination-free propellant
Stabilization/control	12	10	5	• Model uncertainties • Control system modeling
Payload accommodations	11	4	2	• Structural mounting • Load-carrying disconnect fitting
Utilities distribution	7	2	2	Reliable remote electrical and fluid connectors
Platform assembly/operations	23	8	4	• RMS end effectors • RMS collision avoidance software
Rendezvous/docking	5	4	4	Rendezvous approach
Total	97	44	22	

Several technology candidates within each subsystem area were resolved with advancement plans. Typical plans for three types of technology issues are shown in Figures 11, 12, and 13, with their technology status, approach, and expected results, and also a suggested schedule and budget (not shown).

PLATFORM TABLES

OBJECTIVE

- PRODUCE STRUCTURAL MOUNTING TABLE FOR ERECTABLE PLATFORM-LIGHT WEIGHT AND LOW COST

TECHNOLOGY ASSESSMENT

- ERNO PAYLOAD PALLETS DESIGNED PRINCIPALLY FOR ORBITER USE

APPROACH

- DESIGN LIGHTWEIGHT TABLE FOR PLATFORM INSTALLATION-TABLE TO BE FOLDED FOR PACKAGING DURING TRANSPORTATION IN ORBITER
- DEVELOP AND VERIFY CONCEPTS FOR MULTIPLE PAYLOAD INSTALLATIONS AND REMOVALS ON ORBIT

EXPECTED RESULTS

- LOW-COST/LIGHTWEIGHT PLATFORM TABLES DESIGNED TO SUPPORT PAYLOADS ON THE ERECTABLE PLATFORM

LOAD-CARRYING DISCONNECT FITTING

OBJECTIVE

- DEVELOP STRUCTURAL FITTING CAPABLE OF BEING CONNECTED OR DISCONNECTED IN ORBIT REMOTELY, RMS, OR EVA ASSIST

TECHNOLOGY ASSESSMENT

- ERNO PALLET SUPPORT TRUNNIONS
- SHUTTLE PAYLOAD CRADLES FOR MMS, IUS, ETC., ATTACHMENT & RELEASE MECHANISM

APPROACH

- DESIGN & DEVELOP SIMPLIFIED FITTING FOR STANDARD APPLICATION BY PAYLOAD USERS
- PERFORM RMS SIMULATOR GROUND TEST TO DEMONSTRATE & VERIFY ACCEPTABLE DESIGNS

EXPECTED RESULTS

- STANDARDIZED STRUCTURAL FITTINGS FOR PAYLOADS AND PALLETS WITH EASY DISCONNECT FEATURES

Figure 11. Payload Accommodations Technology Candidates

RMS COMPUTER SOFTWARE

OBJECTIVE

- PROVIDE COMPUTER AIDED ASSISTANCE IN PREVENTING RMS-PLATFORM/ORBITER COLLISIONS DURING PLATFORM ASSEMBLY

TECHNOLOGY ASSESSMENT

- REQUIRED COMPUTER/SOFTWARE NOT PRESENTLY AVAILABLE

APPROACH

- ANALYZE COLLISION AVOIDANCE REQUIREMENTS FOR PLATFORM ASSEMBLY AND MISSION OPERATION MANEUVERS PERFORMED BY RMS
- DETERMINE APPROPRIATE COMPUTER FOR APPLICATION
- DEVELOP SOFTWARE FOR COLLISION AVOIDANCE PROGRAM AND INTEGRATE WITH RMS CONTROL SYSTEM

EXPECTED RESULTS

- GREATER RELIABILITY AND SAFETY OF PLATFORM ASSEMBLY OPERATIONS

RMS END EFFECTOR (STRUT/UNION JOINING)

OBJECTIVE

- PROVIDE RMS END EFFECTOR WITH OPTIMIZED EFFECTIVENESS FOR STRUT/UNION JOINING

TECHNOLOGY ASSESSMENT

- AN RMS END EFFECTOR THAT CAN SUPPORT THE STRUCTURAL UNION WHILE STRUT/UNION JOINT CONNECTION IS BEING MADE IS NOT AVAILABLE BUT MAY INCREASE ASSEMBLY PRODUCTIVITY

APPROACH

- COMPARE TYPICAL ASSEMBLY TIMELINE ESTIMATES WITH/WITHOUT SPECIALIZED END EFFECTOR
  PERFORM GROUND TESTS OF MODEL
- DEVELOP THE MORE CRITICAL END EFFECTORS REQUIRED

EXPECTED RESULTS

- END EFFECTOR COMBINATION TO OPTIMIZE ERECTABLE PLATFORM-TYPE OPERATIONS

Figure 12. Platform Assembly Technology Candidates

MODEL UNCERTAINTIES

OBJECTIVES

- INVESTIGATE AREAS OF MODEL UNCERTAINTIES ARISING FROM MODEL SIMPLIFICATION
- ISOLATE MODEL CHANGES ARISING FROM STRUCTURES/MATERIAL AGING--PAYLOAD BUILDUP

TECHNOLOGY ASSESSMENT

- STRUCTURE WITH SEVERAL JOINTS--50% MODES UNCERTAIN
- MULTI-JOINT/HIGHLY FLEXIBLE STRUCTURE WITH SLACK DEADBANDS; ANALYSES NEED DEVELOPMENT

APPROACH

- IDENTIFY THE SOURCES, MAGNITUDES, AND EFFECTS OF STRUCTURAL MODEL UNCERTAINTIES ON CONTROL RESPONSE

EXPECTED RESULTS

- RANGES OF MODEL BEHAVIOR WITH MISSION WHICH REQUIRE CONTROL MODELING

CONTROL SYSTEM MODELING

OBJECTIVES

- TO IDENTIFY STRUCTURAL FREQUENCY WITHIN CONTROL BANDWIDTH
- DEVELOP CONTROL DESIGN MODEL (MODAL REDUCTION)

TECHNOLOGY ASSESSMENT

- CURRENT CONTROL BANDWIDTH--1 ORDER SEPARATION FROM STRUCTURAL FREQUENCY
- FILTERING OUT OF 1 OR 2 LOWER STRUCTURAL FREQUENCIES
- ADVANCED TECHNIQUES NEEDED FOR MANY FREQUENCIES WITHIN CONTROL BANDWIDTH

APPROACH

- IDENTIFY MODEL STRUCTURAL FREQUENCIES
- ESTIMATE CONTROL BANDWIDTH NEEDED TO MEET ACCURACY REQUIREMENT

EXPECTED RESULTS

- DEGREE OF OVERLAP BETWEEN CONTROL BANDWIDTH AND STRUCTURAL FREQUENCIES
- INSIGHT INTO ACCEPTABLE CONTROL THEORY

Figure 13. Control Subsystem Technology Candidates

2.0 MISSION AND PAYLOADS ANALYSIS

The following section describes the results of a requirements analysis performed on the mission and payload model provided by the NASA for this study.

Payloads Analysis

For purposes of this analysis, the following definitions apply:

- *Mission*. General scenario defining operating conditions; synonymous with "payload" below platform level.
- *Platform*. Orbiting facility providing services and positioning to a group of payloads.
- *Pallet Equivalent*. Number of pallet spaces occupied by payload in orbiter or on platform.
- *Payload*. Unique grouping of instruments having a common set of data requirements.

A total of 65 payloads derived from OSTA, OSS, and OAST requirements were provided. These were screened and grouped to establish platform ission requirements based on payload capture potential. A summary of payload requirements was prepared and is included as an appendix. Each payload was arbitrarily assigned a number for easy reference. It is recognized that there will be other payloads for consideration; however, those provided were taken as representative for study purposes.

Criteria for screening included the elimination of geosynchronous, manufacturing/processing, and life science missions. Geosynchronous missions were beyond the capability or interest of the platforms and delivery systems being considered. Manufacturing/processing payloads often were massive, had large power requirements, and were of short duration. These did not integrate well with the mix of other experiments, most of which were scientific and long term in nature. There were no purely life science experiments in the 65 listed payloads. The life science payloads listed were integrated with manufacturing/processing experiments.

Geosynchronous and manufacturing combined to eliminate seven payloads. An additional seven were eliminated as unsuitable or unlikely. These had booms, tethers, or large appendages such as antennas which made them difficult or impossible to accommodate on a multi-payload platform. After screening, 51 of the original 65 payloads remained. Four payloads are duplicated in different orbits, however, making a total of 55 in all.

PAYLOADS ANALYSIS

OBJECTIVE

- DETERMINE PAYLOAD GROUPINGS WHICH RESULT IN PLATFORM REQUIREMENTS GIVING A REASONABLE COMPROMISE IN CAPTURE POTENTIAL, QUANTITY, AND TYPE OF PLATFORMS.

APPROACH

- SCREEN (65 POTENTIAL PAYLOADS PLUS MULTIPLES) TO ELIMINATE COMMUNICATIONS, MANUFACTURING, AND LIFE SCIENCE MISSIONS. IDENTIFY "TALL POLES" AS UNLIKELY CANDIDATES FOR PLATFORM ACCOMMODATION.
- ESTABLISH GROUPINGS IN TERMS OF CRITICAL REQMTS.
- DEFINE PLATFORM REQUIREMENTS FOR PAYLOADS AVAILABLE THROUGH 1985.

Range of Requirements (55 Remaining Payloads)

The 55 payloads remaining were primarily scientific experiments and appeared to be compatible with the concept of a service platform in low earth orbit. Within the group, however, these possessed a diversity of requirements. The task at hand was to establish payload groups, on the basis of comonality, thereby providing a basis for establishing platform requirements. Platforms could then be defined which, overall, would have the greatest capture potential. Most significant grouping drivers from the platform standpoint are orbit, pointing reference, and accuracy.

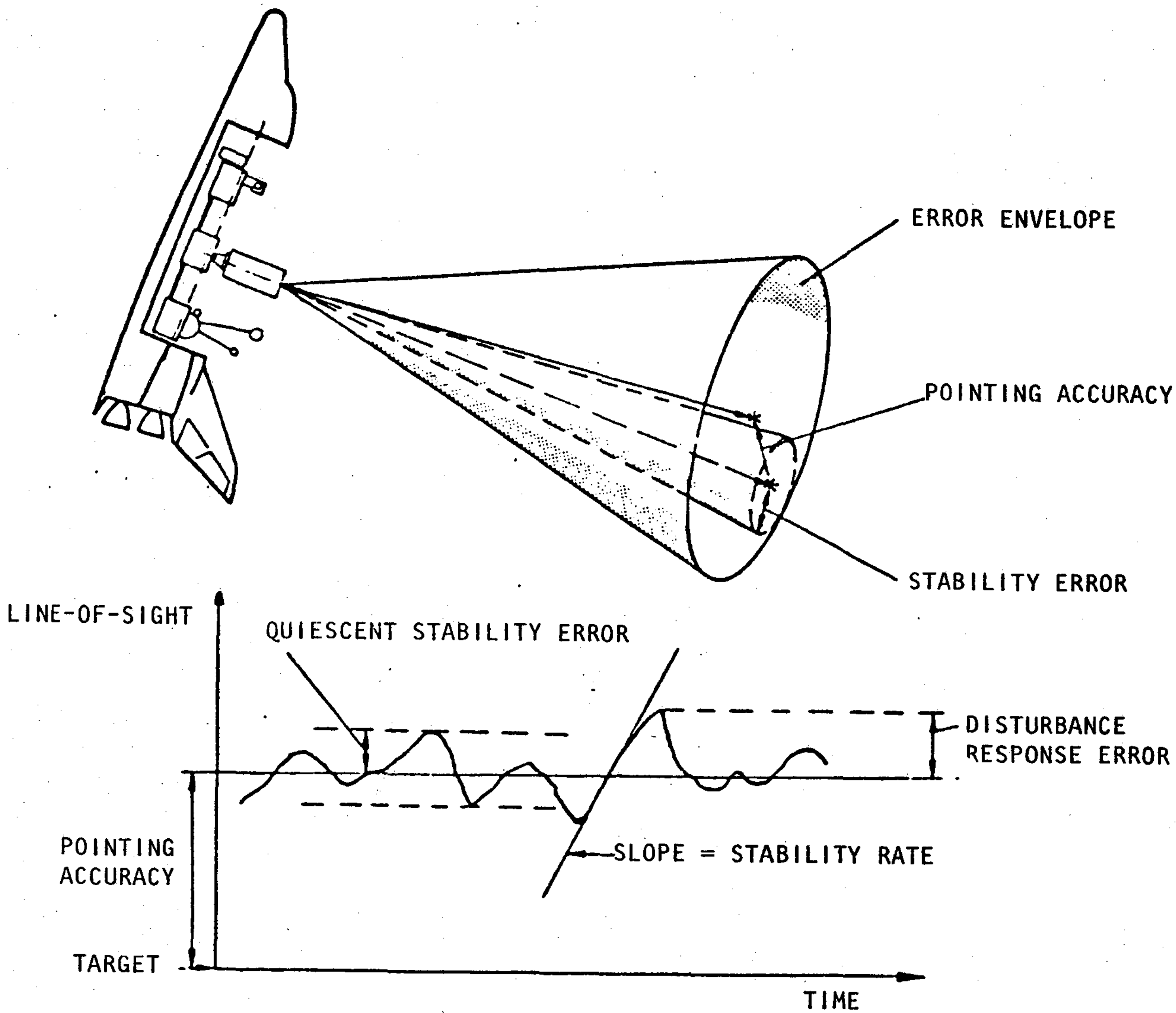
RANGE OF REQUIREMENTS
(55 Remaining Payloads)

- VIEWING—EARTH, SPACE, SOLAR
- POINTING REFERENCE—INERTIAL, SUN, CELESTIAL BODY,
  MAGNETIC LINES, EARTH LOCAL VERTICAL
- POINTING ACCURACY—3 DEG TO 0.001 ARC-SEC
- ORBIT—28° TO 98°, CIRCULAR/ELLIPTICAL, 200 TO 900 KM, SUN SYNC, ETC.
- WEIGHT—75 TO 10,000 KG
- POWER—100 W TO 4 KW (9 KW PEAK)
- SIZE—UP TO 3 PALLETS (EQUIV.)
- DATA—UP TO 3 Mb/s STORE/TRANSMIT, SOME TV AND/OR FILM
- THERMAL CONTROL—5°K TO 25°C
- CRYOGENS—SOME REQUIRE LHe
- REPLENISH—AS FREQUENTLY AS 2 MONTHS
- READINESS—1982 TO 1989
- STRONGEST GROUPING DRIVERS—ORBIT, POINTING REFERENCE, AND ACCURACY

POINTING ACCURACY/STABILITY ERROR DEFINITIONS

Pointing accuracy and stability requirements are important since it must be determined what the basic platform can reasonably provide, higher order requirements being left to equipment carried as part of individual payloads. For purposes of this study, pointing accuracy was used as the primary control requirement since this was the parameter which was most often defined for a given payload.

POINTING ACCURACY/STABILITY ERROR DEFINITIONS



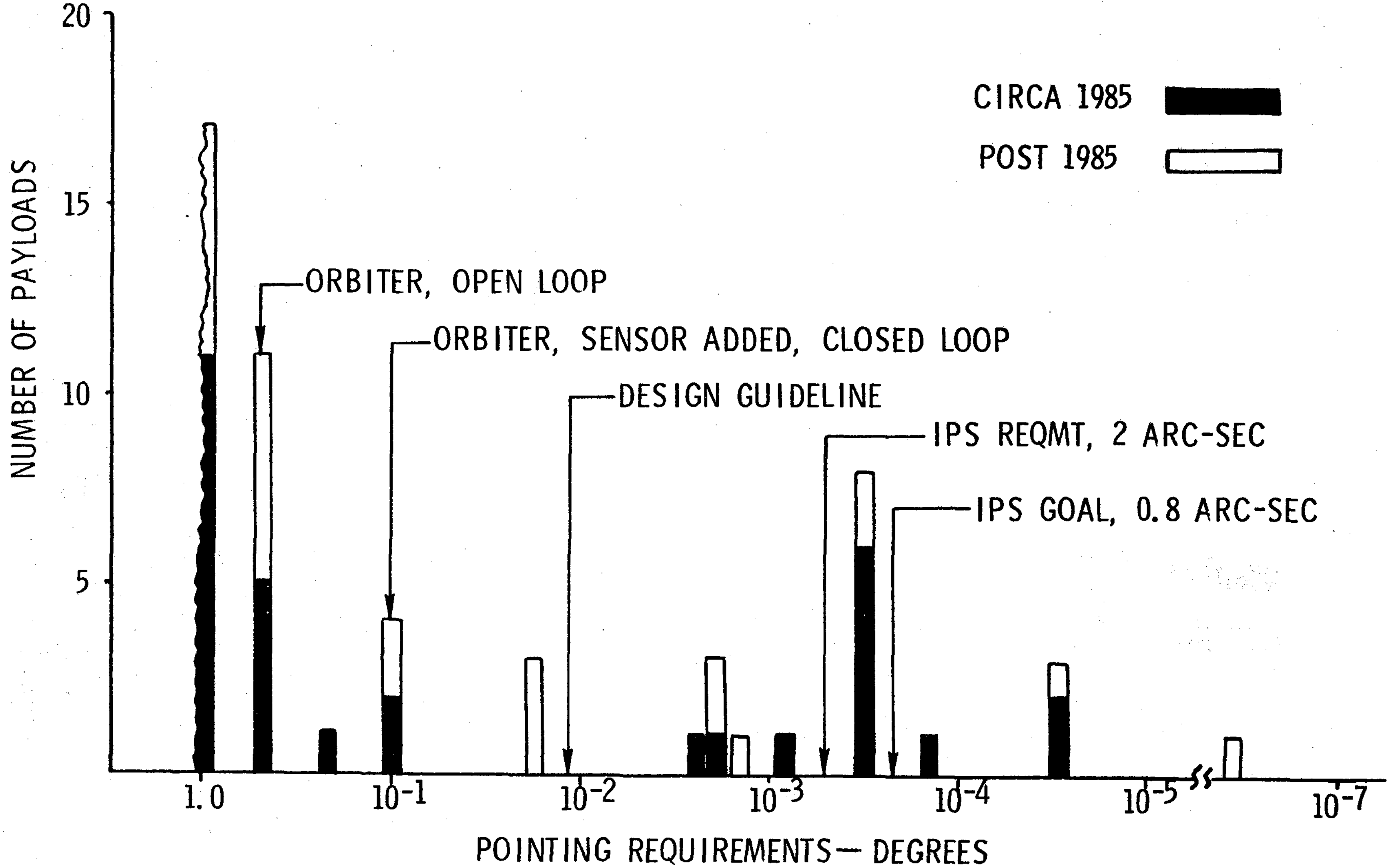
Platform Pointing/Payload Capture Capability, 55 Payloads

Current state of the art in control systems is represented by the Shuttle orbiter. The basic, open-loop system has a ±0.5-degree pointing capability. Incorporating systems of similar capability in the platforms would satisfy the requirements for about 50 percent of the payloads. Adding an accurate sensor in a closed-loop mode would capture five additional payloads. This requires no upgrading of current capability and is entirely feasible, either using a primary platform-mounted sensor or a sensor incorporated as part of one payload.

The OAST design guideline was to provide an order-of-magnitude improvement to ±0.01 degree. However, this provides no additional capture potential of payloads prior to 1985, and only three thereafter. Therefore, the design guideline requirement was revised to 0.1 degree.

The instrument pointing system (IPS) will provide pointing of ±2 arc-seconds with a goal of ±0.8 arc-seconds. Therefore, the use of the IPS should be considered for any payloads requiring accuracy greater than 0.1 degree. It was assumed that the IPS design goal was attainable within current state of the art. Five payloads require higher accuracy which would need advanced technology in the form of a "Super IPS." Only three payloads, however, are captured in the 1985 time frame by a system which exceeds the standard IPS design goal.

PLATFORM POINTING/PAYLOAD CAPTURE CAPABILITY
55 Payloads



PLATFORM REQUIREMENTS SUMMARY (CIRCA 1985)

Of the 55 payloads considered, a total of 31 would be accommodated through 1985 by the three platforms. Of these, 18 require no compromises in desired orbit inclination or altitude. A standard IPS will be required by 11 payloads, plus six with gimbals for orientation. A "super IPS" is required only on P-1 for two payloads. Cryogenics are required on P-1 and P-2 which will make these platforms subject either to servicing every 1 or 2 months or corresponding increases in weight associated with long-term cryogenic storage.

PLATFORM REQUIREMENTS SUMMARY (CIRCA 1985)

PLATFORM	P-1	P-2	P-3
ORBIT	400 KM, 28^{o}, CIRC.	575 KM, 90^{o}, CIRC.	400 KM, 57^{o}, CIRC.
STABILIZATION	INERTIAL	INERTIAL, SOLAR PTG.	EARTH LOCAL VERTICAL
PAYLOAD MANIFEST	SOLAR 3,13 ATM 15b,16a H.E. 26,31 ASTP 35a,36,38, 39,40	SOLAR 1,2,11 H.E. 27 E.O. 52,55,65	SOLAR 12, 14 ATM 15a,16b,20 ASTP 35b E.O. 56,57,59,60, 62,63,64
TOTAL WEIGHT (KG)	15,300	18,500	10,105
TOTAL AVG PWR (KW)	14.5	6.6	16.4
STD. IPS*	3,16a,35a,38	1,11	16b,35b,62,64
GIMBAL ONLY*	20 (15b WITH 16a)	52,55,65	14,20
SUPER IPS	36,39,40	NONE	NONE
PLATFORM SPACES (PALLET EQUIV.)	13	12	11
CRYOGENS	15b, 16a,40	TBD	16b,56

- ABOVE ASSUMES NECESSARY COMPROMISES CAN BE MADE TO ACCOMMODATE EXTRAORDINARY ORBIT, VIEWING, AND REPLENISHMENT REQUIREMENTS.
- BASED ON OPTIMUM GROUPING OF PAYLOADS THROUGH 1985.

*Additional IPS or gimbals may be required to provide necessary FOV.

Baseline Platforms - Capture Assessment (31 Payloads through 1985)

When arranged in terms of orbit inclination, payloads generally fall into three groups: low, high, and mid-range. Eleven low-inclination payloads can be accommodated by platform, P-1, having a 28-degree circular orbit at an altitude of 400 km. This platform would be controlled to an inertial reference. Accommodation of one payload, No. 36, requires a compromise in altitude. Several compromises in altitude and inclination are assumed for the high- and mid-inclination platforms. Insufficient information is available at this time to determine if such compromises would be acceptable.

Several of the payloads assigned to platform P-2 in the polar orbit have altitude requirements significantly above the assembly altitude of 400 km. The transfer to these higher orbits will significantly impact the platform design (propulsion stage) and exceed the orbiter's payload carrying capability into the polar orbit. The P-2 platform is proposed to be operated at 575 km which might compromise some of the solar terrestrial missions.

BASELINE PLATFORMS—CAPTURE ASSESSMENT
(31 Payloads through 1985)

PLATFORM	P-1	P-2	P-3
• "GOOD-FIT" PAYLOADS	3, 15b, 16a, 26, 31, 35a, 38, 39, 40	1	12, 14, 20, 35b, 59, 62, 63, 64
• POTENTIAL PROBLEM PAYLOADS	13 (ORBIT TBD) 36 (300 KM)	2 (SUN SYNCH TO 50% SUNLIGHT) 11 (450 KM) 27 (500 KM MAX DES.) 52 (85-87°, 600-800 KM) 55 (98.14°, 705 KM) 65 (98°, 700-1600 KM)	15a (70° DESIRED) 16b (70° DESIRED) 56 (180-280 KM) 57 (180-225 KM) 60 (300 KM MAX)

NOTE: WHERE CHOICE WAS PERMITTED, PAYLOADS WERE ASSIGNED TO LOWEST-INCLINATION ORBIT.

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3.0 UTILITIES MODULE AND PLATFORM CONFIGURATION

A brief, overall description of the Space Sciences and Applications Platform is now described. That description includes the module that provides the power, thermal control, data handling, and attitude control services provided to the payloads as well as the platform on which the payloads are mounted. Specific details of each subsystem are described in Section 4.0.

SERVICE PLATFORM
PRINCIPAL CONFIGURATION DRIVERS

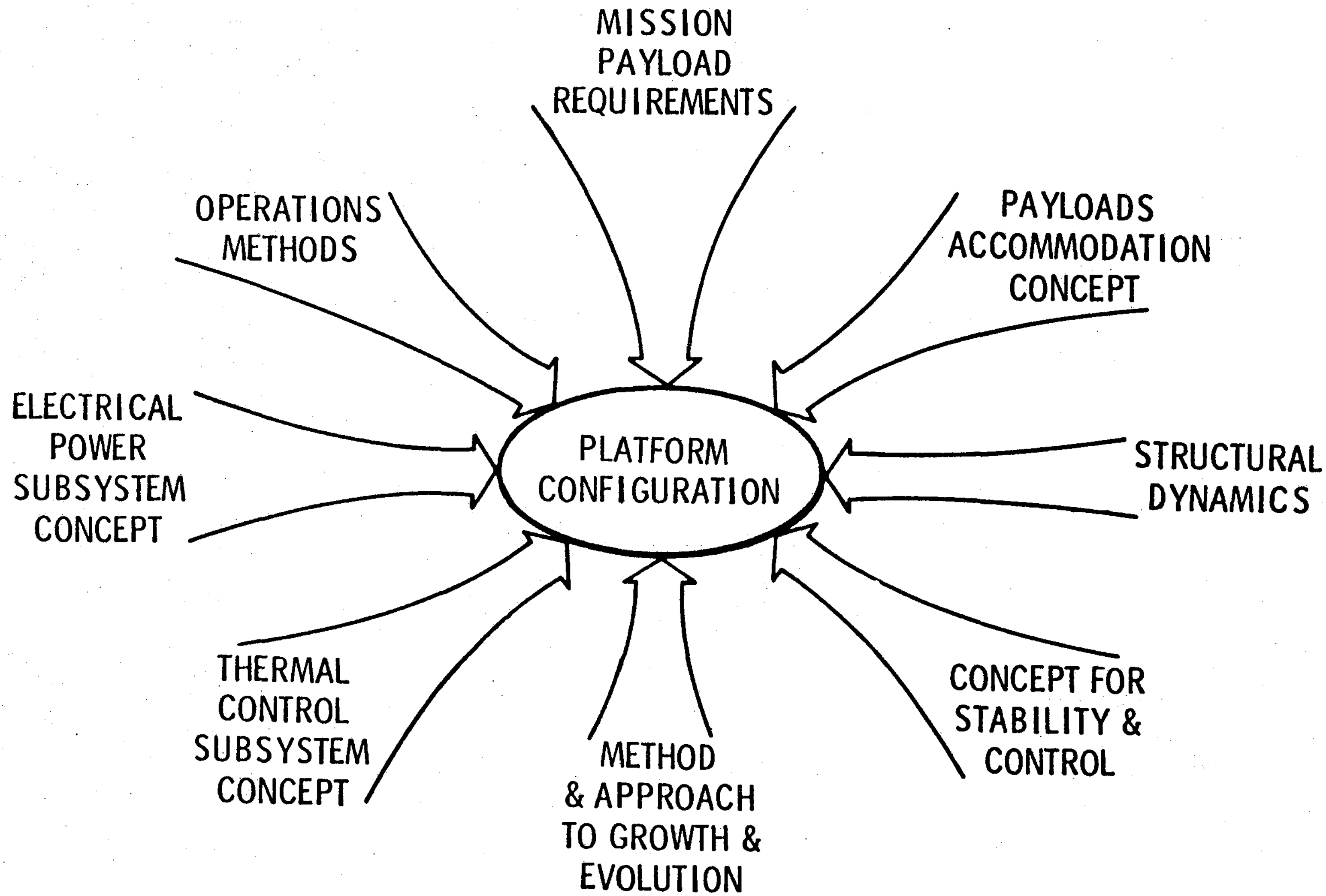
The figure opposite depicts some of the more significant factors that influence platform configuration. These, and others, are discussed in some detail in Section 4. The basic configuration concept, however, is treated here in order to establish a baseline for all the following subsystem discussions.

As an item of clarification, a distinction is made between the "services" or "utilities" module and the payload accommodations platform. The former is defined as the free-flying vehicle system which supplies power, heat rejection, stability and control, and data handling services to a group of attached payloads. The payload accommodations platform is that structure attached to the utilities/service module which functions as host to the assigned payloads and through which the utilities module services are distributed to individual payloads.

In order to most effectively use this study's resources on the "platform" aspects of the orbiting systems, the utilities module portion of the orbiting system was that developed by Rockwell under its earlier orbiting power module studies. That "utilities" module concept, then, was "baseline" to all platform and payload accommodation trade studies.

That "baseline utilities module" is briefly described next.

SERVICE PLATFORM
PRINCIPAL CONFIGURATION DRIVERS

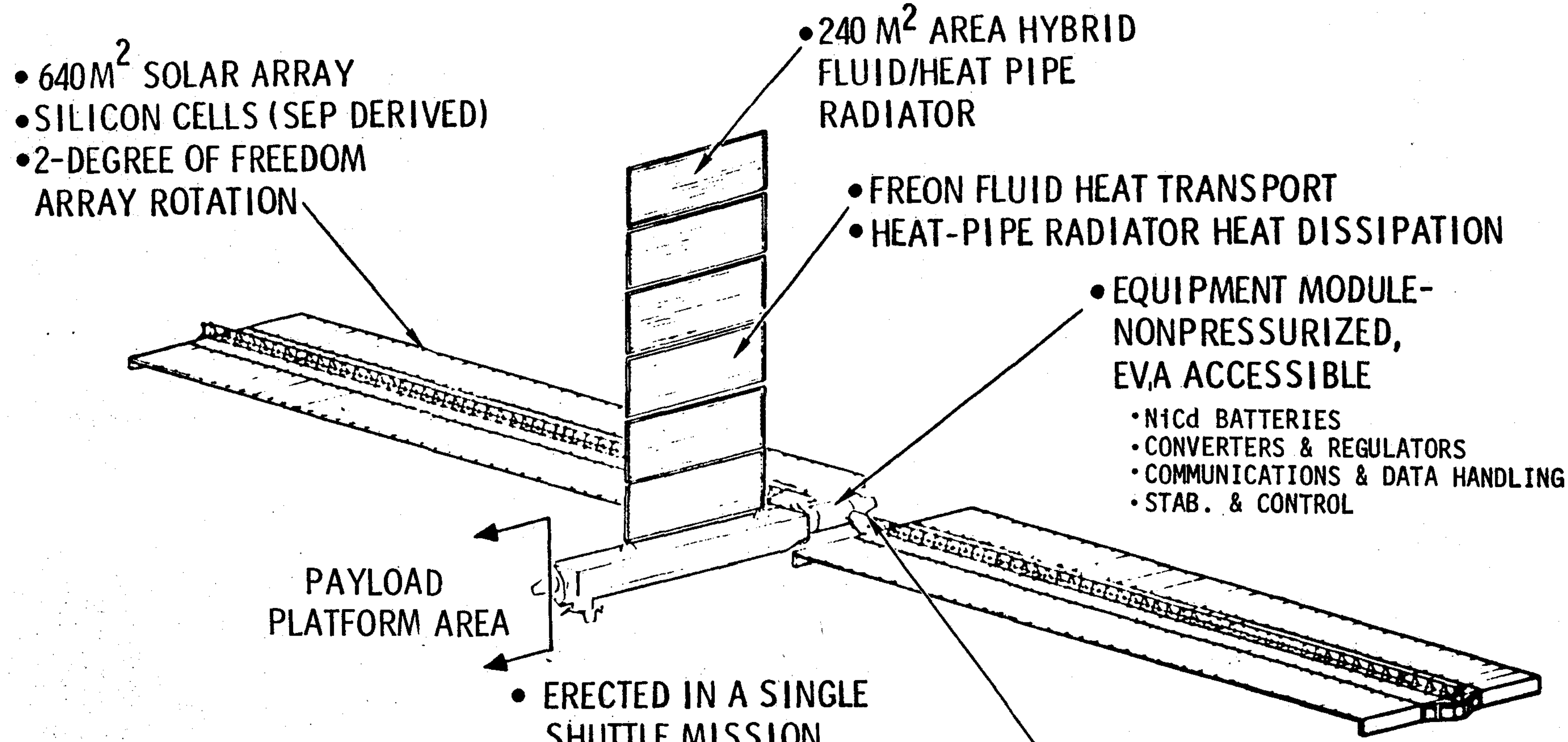


Baseline Utilities Module

Shown opposite is the basic power/utilities module used in conducting the rest of the study described herein. Its principal characteristics are as shown on the figure.

It will be seen later that modifications to this basic concept were required and were developed in response to the specific needs of the NASA defined mission model. Other changes and additions were introduced to accommodate newly understood operational requirements. All these will be discussed as appropriate in later sections of this report.

BASELINE
FREE-FLYING (25 KW - 35 KW) POWER MODULE



- ERECTED IN A SINGLE SHUTTLE MISSION
- OPERABLE WITH OR WITHOUT SHUTTLE ATTACHED
- ACTIVELY STABILIZED

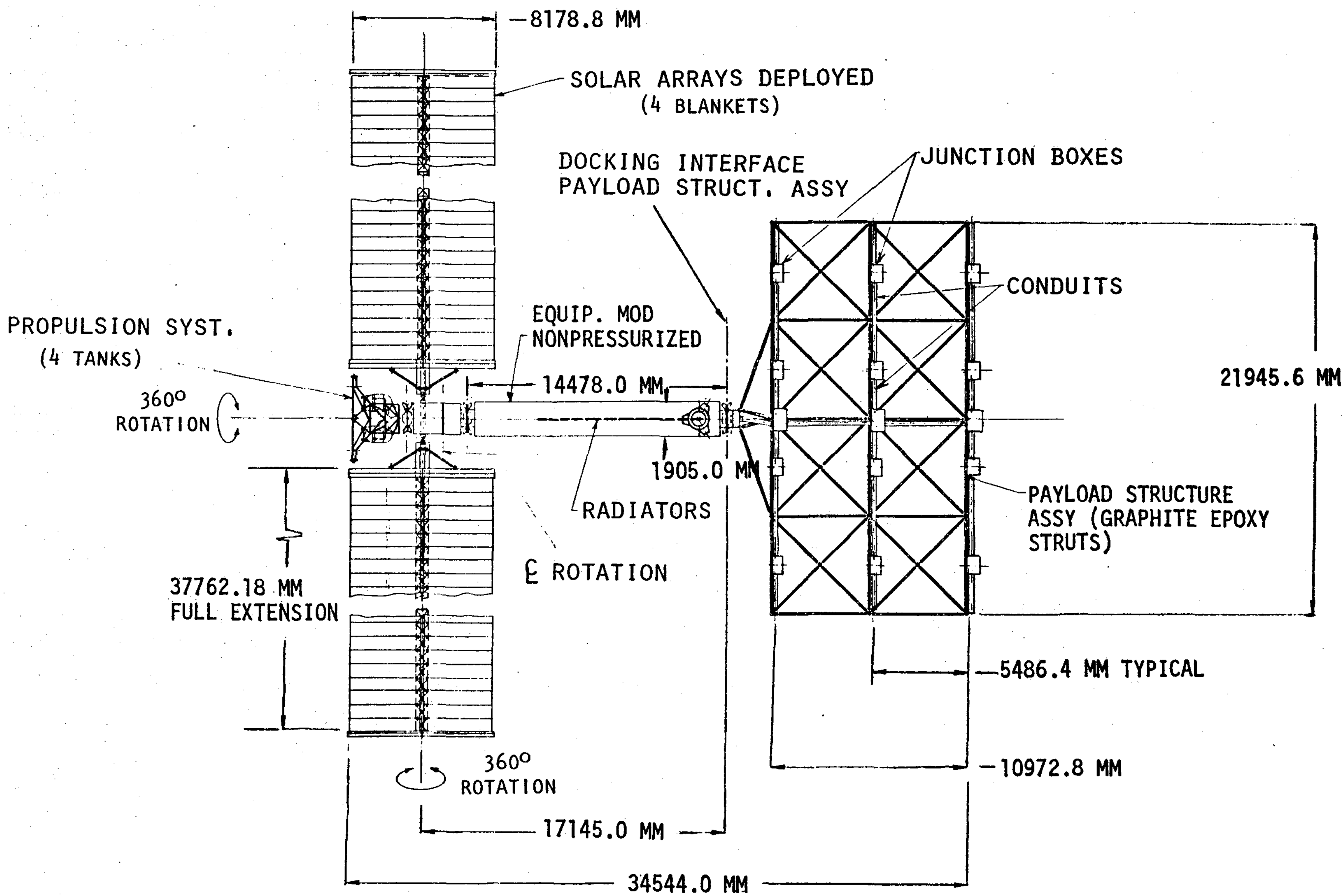
Baseline Configuration - Platform 1 (8 Cells, Pentahedral Area Nodal Mounting)

It will be recalled that, in Section 2, an analysis of mission and payload requirements resulted in the identification of three separate and distinct platforms to accommodate the mission model. The first of these, Platform No. 1 at 400-km altitude and 28.5° circular orbit, is the one which was selected as baseline for subsequent technology planning. Its principal characteristics are a 20- to 25-kW electric power to payload capacity and a payload mount capacity of up to 23 positions.

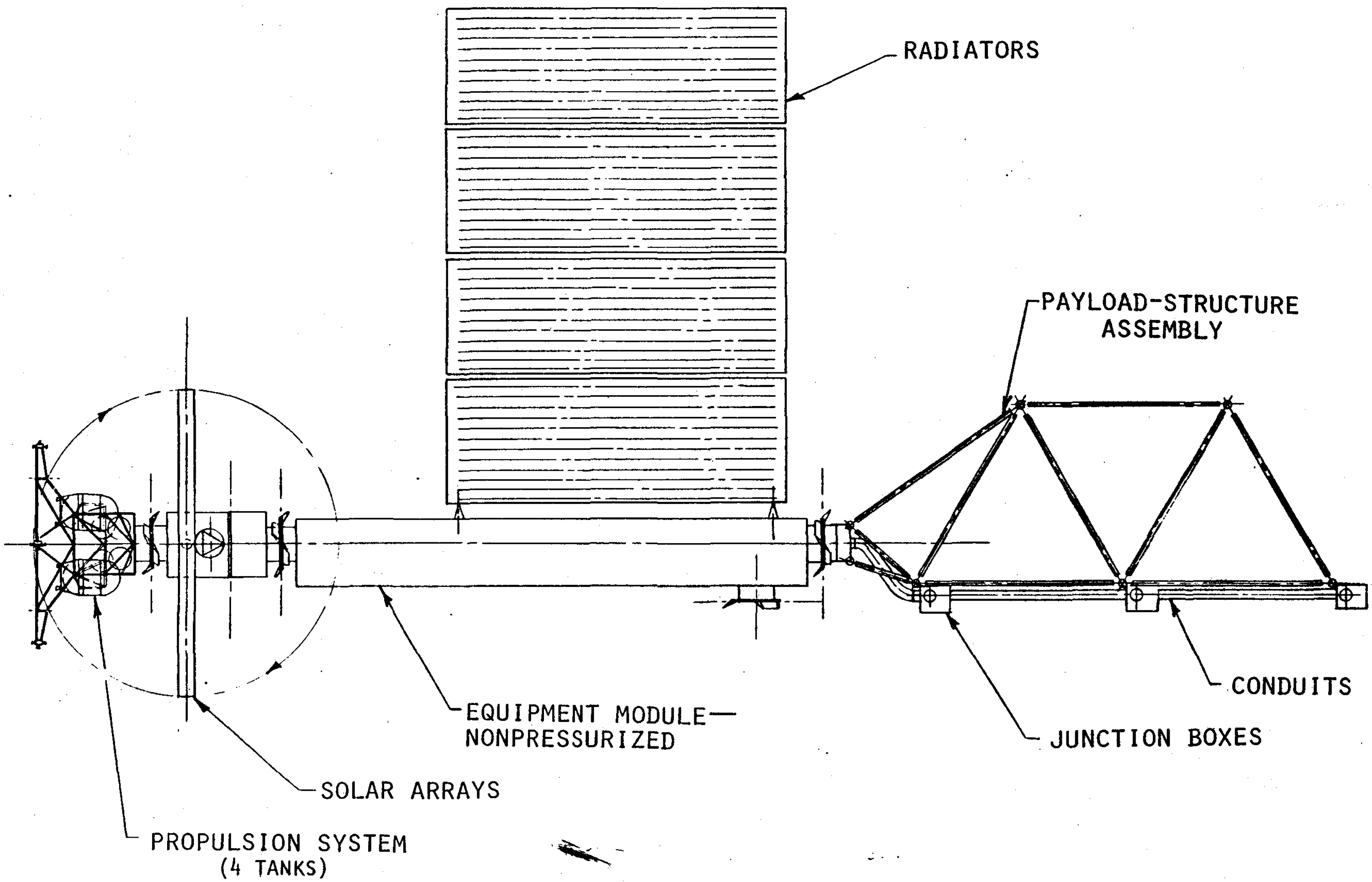
Three fundamental concepts for platform design were considered to satisfy the baseline platform requirement. The first was a "triangular strongback"; the second was a "pentahedral strongback." These two will be briefly discussed later. The third, a "pentahedral area nodal mounting" concept, was the one selected as baseline for this study by a process which is described later. This concept consists of a series of pentahedral cells, assembled into an area matrix. Payloads are mounted at nodal "hardpoints" on the structure, which also supports the distribution system for the services provided by the utilities/services module. As will be shown later, the entire orbiting complex, as shown, can be delivered to orbit and assembled from a single Shuttle orbiter flight. Mission payloads can be delivered with the second flight.

The next three figures depict the total orbiting vehicle system (utilities/services module plus experiments accommodations platform, plus propulsion module) that resulted from the requirements and optimization studies to select a single concept. It is this total configuration which formed the baseline for the technology planning discussed under the various subsections of Section 4.

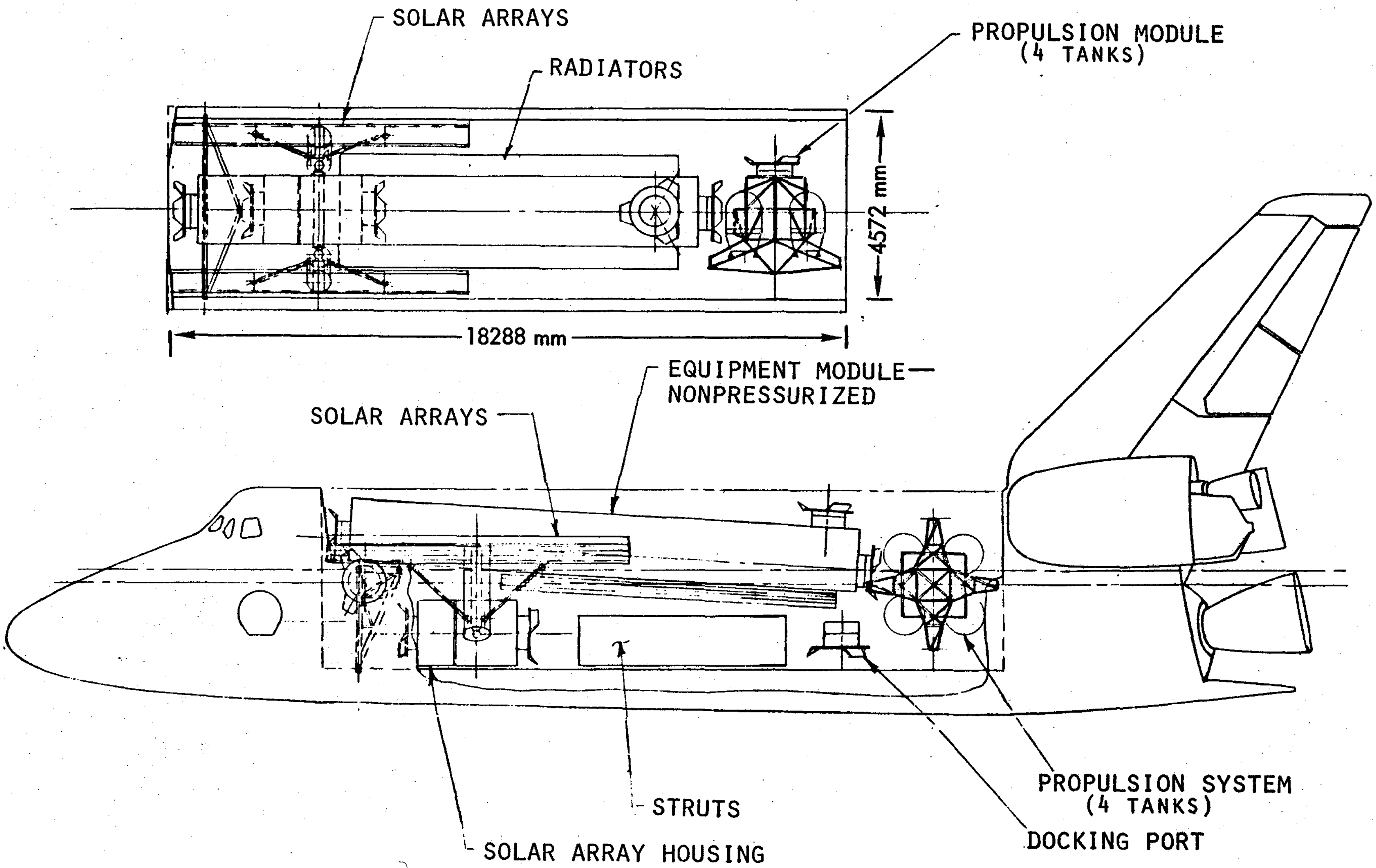
BASELINE—PLATFORM 1
PENTAHEDRAL AREA NODAL MOUNT
(8 CELLS)



PENTAHEDRAL AREA NODAL MOUNT
(8 PAYLOAD CELLS)



MODERATE SIZE PLATFORM (8 CELLS, 4 ARRAYS)
SHUTTLE PACKAGING



Alternate Pentahedral Area Nodal Mount Configurations

The following four figures depict modular variations to the baseline platform concept just discussed. The first two show a configuration version characteristic of Platform No. 2, discussed in Section 2, operating at 575 km in a 90° circular orbit. This configuration provides from 10 to 12 kW of electric power to the payloads and therefore only 2 (instead of 4) solar arrays and 2 (instead of 4) radiator panels are required. The payload platform consists of 4 pentahedral cells (instead of 8) which provide 13 nodal mounting positions.

The second pair of following figures depict a growth version of the concept which provides for additional electrical power (up to 35-40 kW) via 6 solar arrays, heat rejection via 6 radiator panels, and additional platform area for payload accommodation.

The same basic design is used in all the versions illustrated. Initial studies indicate that smaller sizes may be erected on orbit first, and then expanded to larger sizes as mission requirements dictate.

Details of the pentahedral area nodal mount concept are discussed in Section 4.

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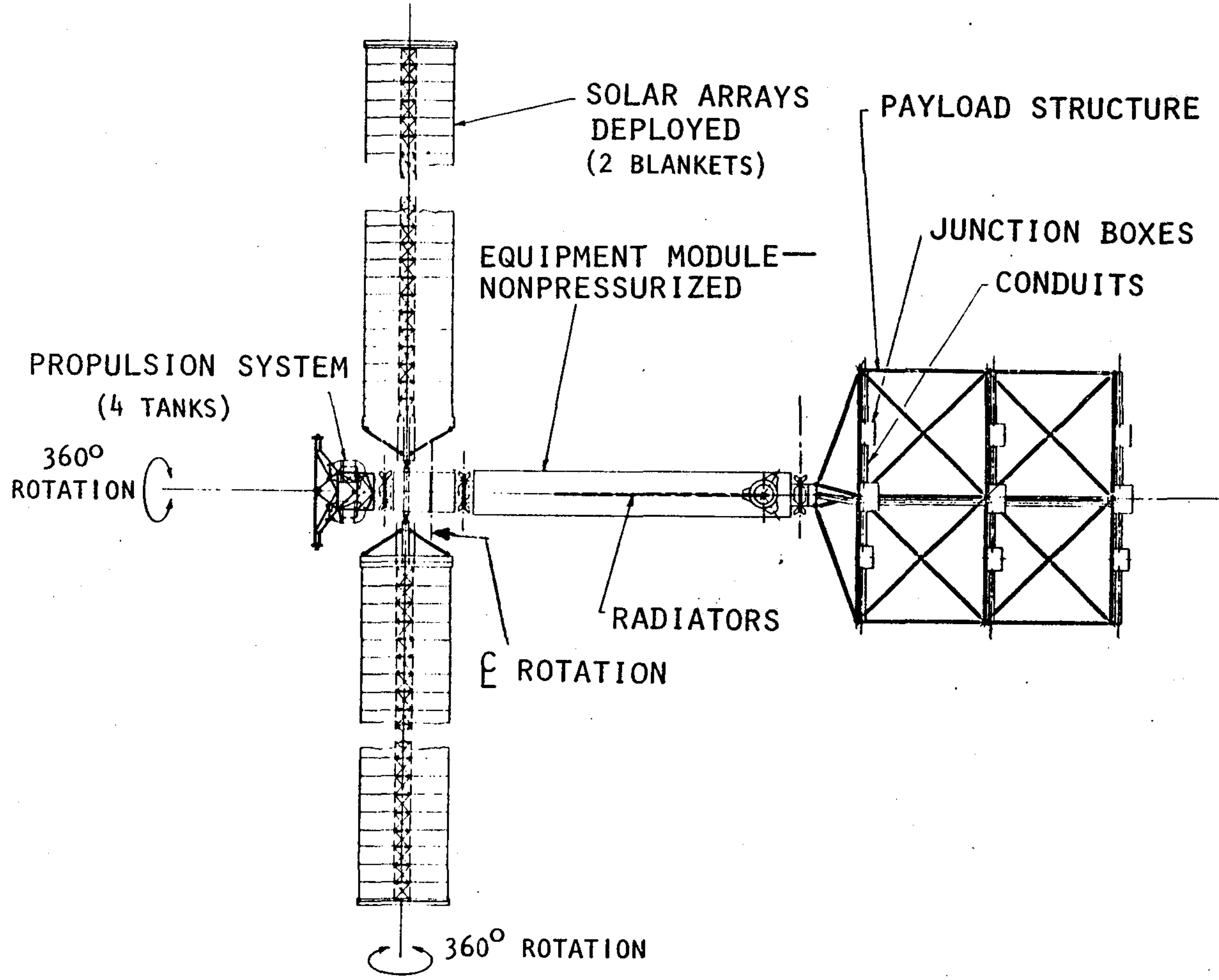
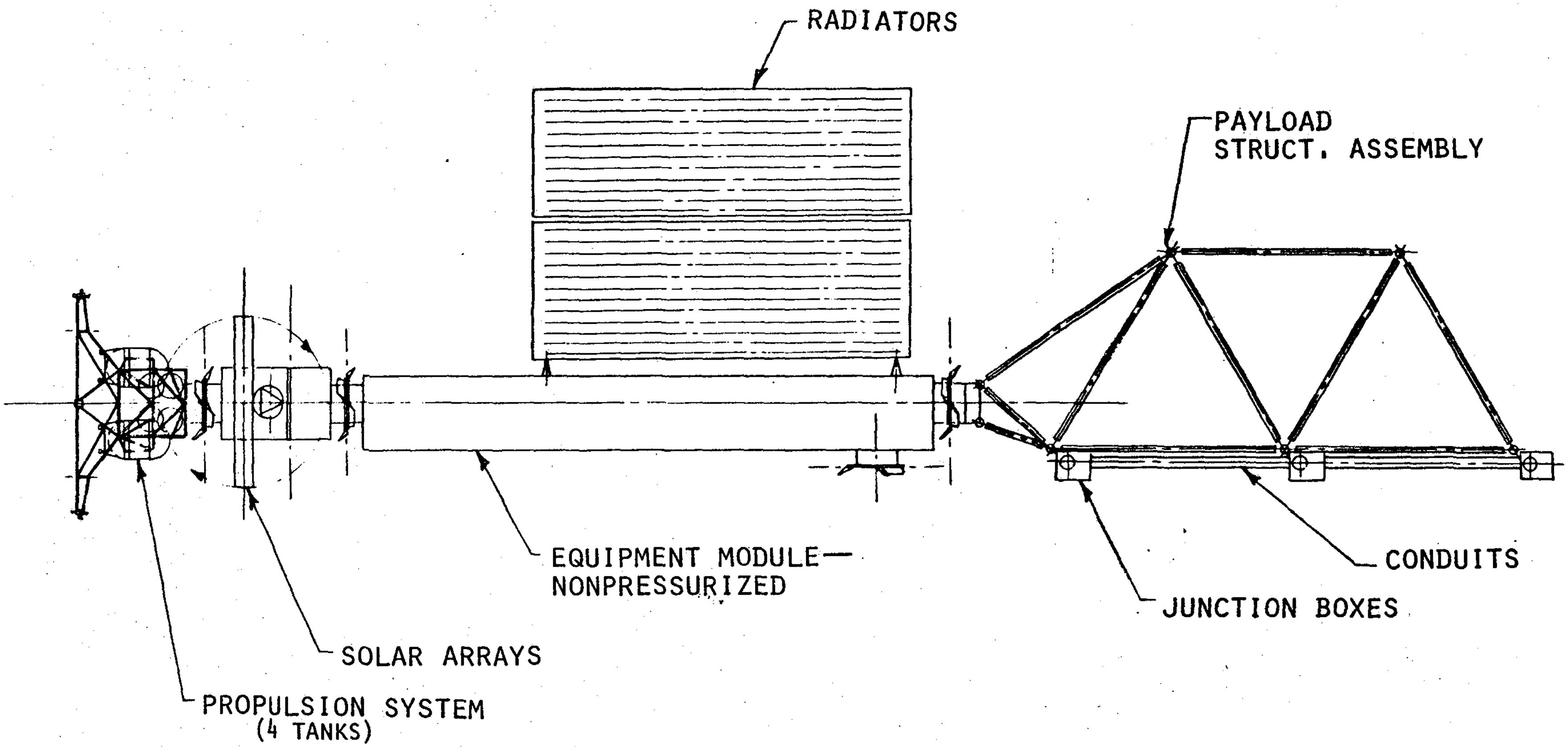
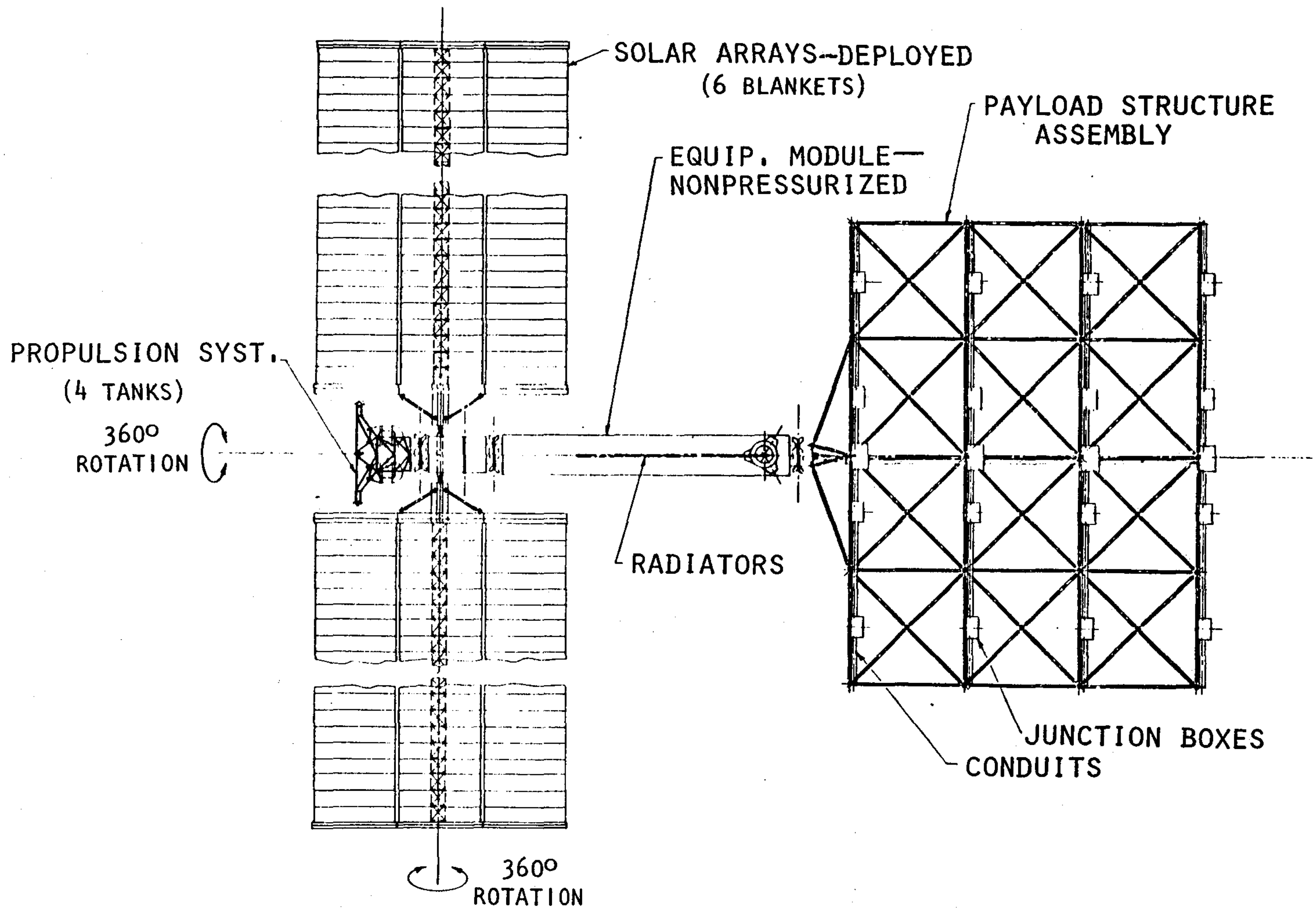
PLATFORM 2 CONFIGURATION
PENTAHEDRAL AREA NODAL MOUNT
(4 CELLS)



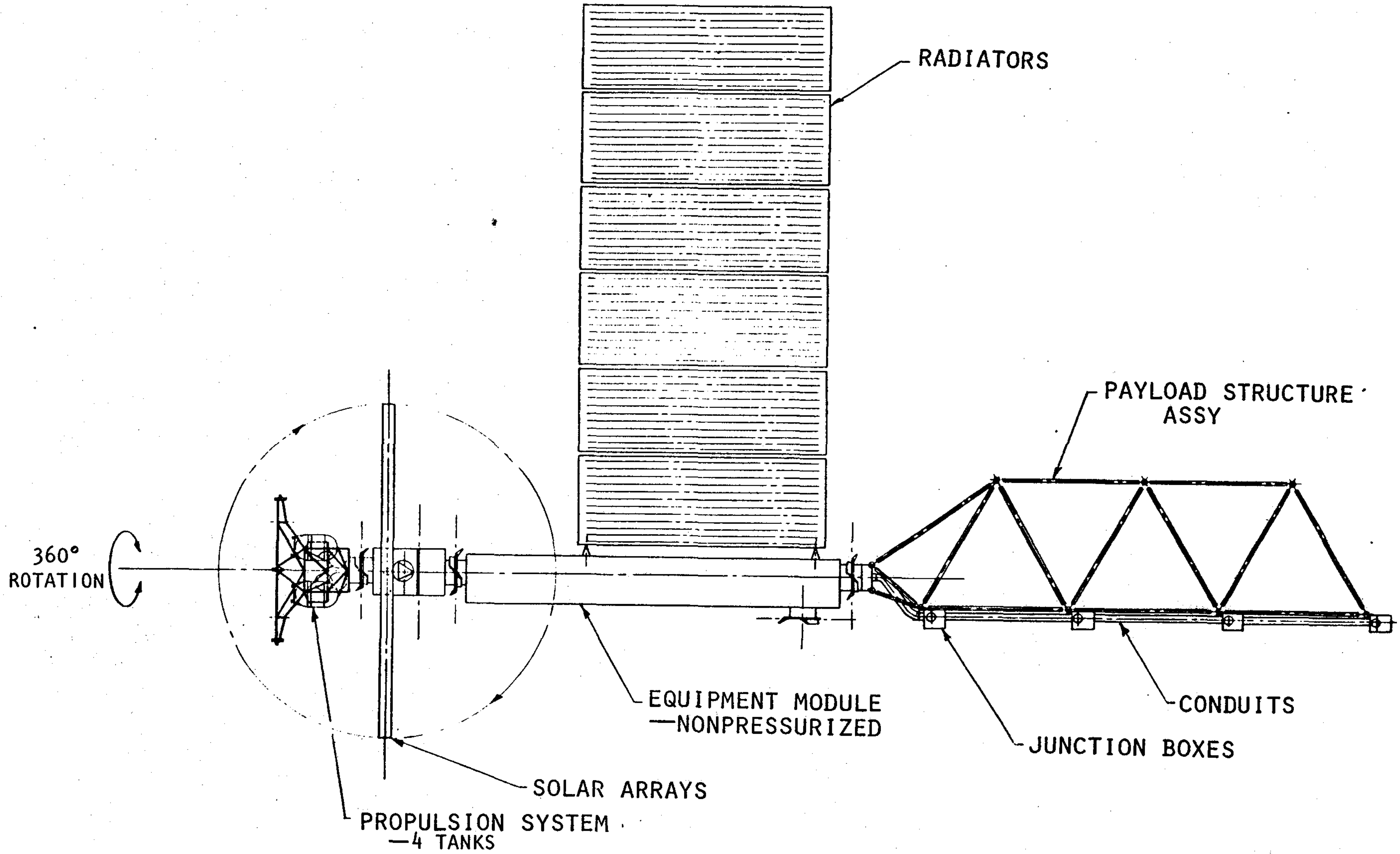
PENTAHEDRAL AREA NODAL MOUNT
(4 CELLS)



GROWTH CONFIGURATION
PENTAHEDRAL AREA NODAL MOUNT
(12 CELLS)



PENTAHEDRAL AREA NODAL MOUNT
12 CELLS



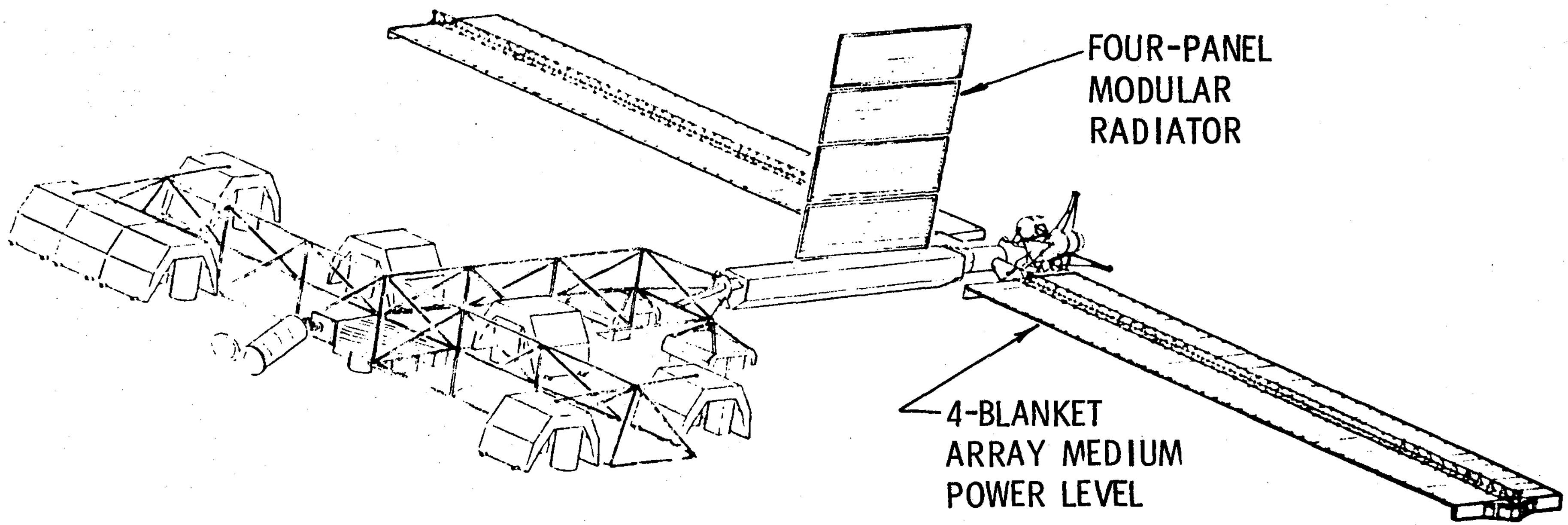
Pentahedral Strongback - Medium Platform

The alternate platform depicted here uses the pentahedral cell. In this concept, the cells, which are built up of a series of composite struts and unions, are joined end to end in order to form a spine, and the end is extended to form a pair of ribs. Pallets and payloads are then attached to the frames of the structure at hardpoints along the pentahedron's side.

The concept shown has 14 payload or pallet positions. The orientation of each payload/pallet can be rotated by varying the length of the stabilizing strut attached to the bottom of the pallet. Utilities are mounted along the platform's structure as they were in the area nodal-mount case. Details of utility service deployment are discussed in Section 4.

In all the alternate platform concepts, the utilities module remains similar to the baseline concept.

PENTAHEDRAL STRONGBACK—MEDIUM PLATFORM



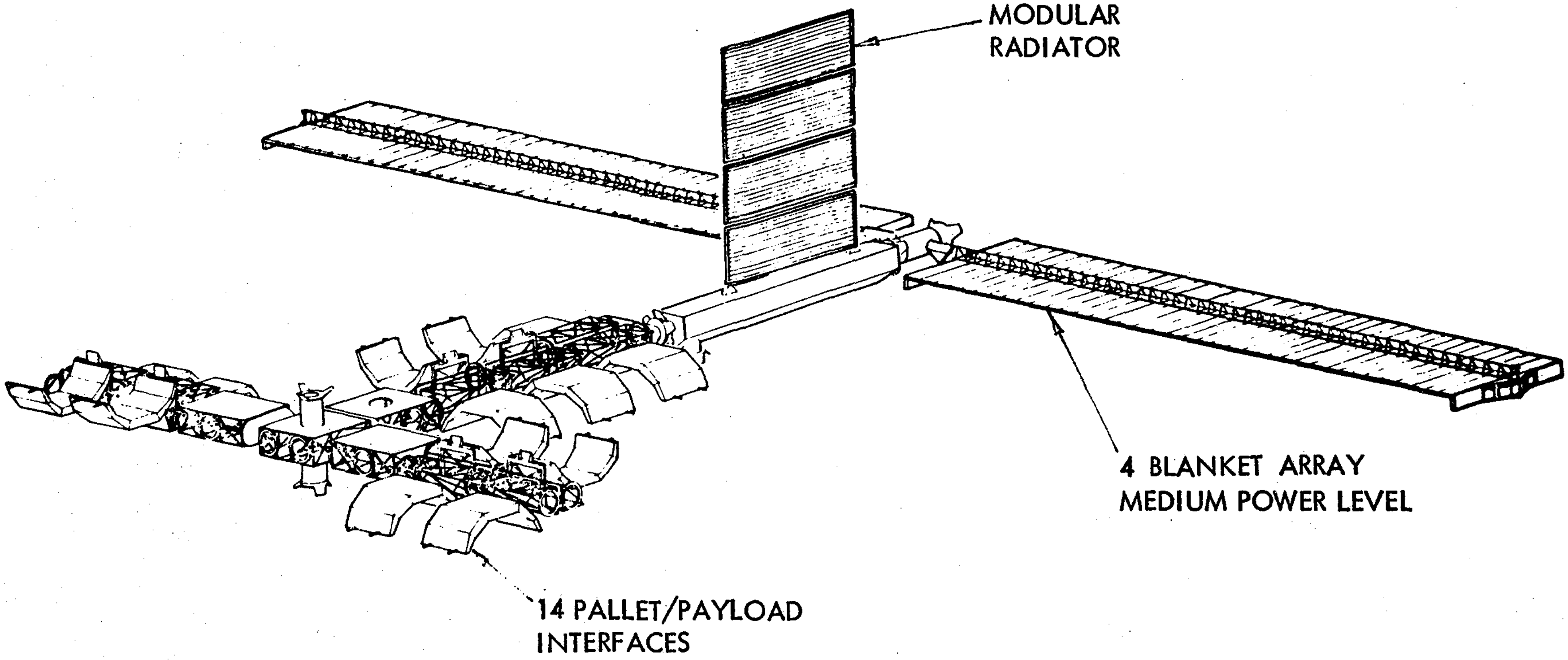
- 14-SIDE MOUNTED PAYLOAD/PALLET INTERFACES

Triangular Strongback - Medium Platform

Opposite is shown the triangular strongback concept arranged to accommodate the requirements of the baseline platform - No. 1. The arrangement will accommodate up to 14 pallets or payload positions. The concept permits the rotation of individual strongback sectons so that there is flexibility in pointing any payload pallet over the range of about 270 degrees.

Details of the triangular strongback are shown on the next chart.

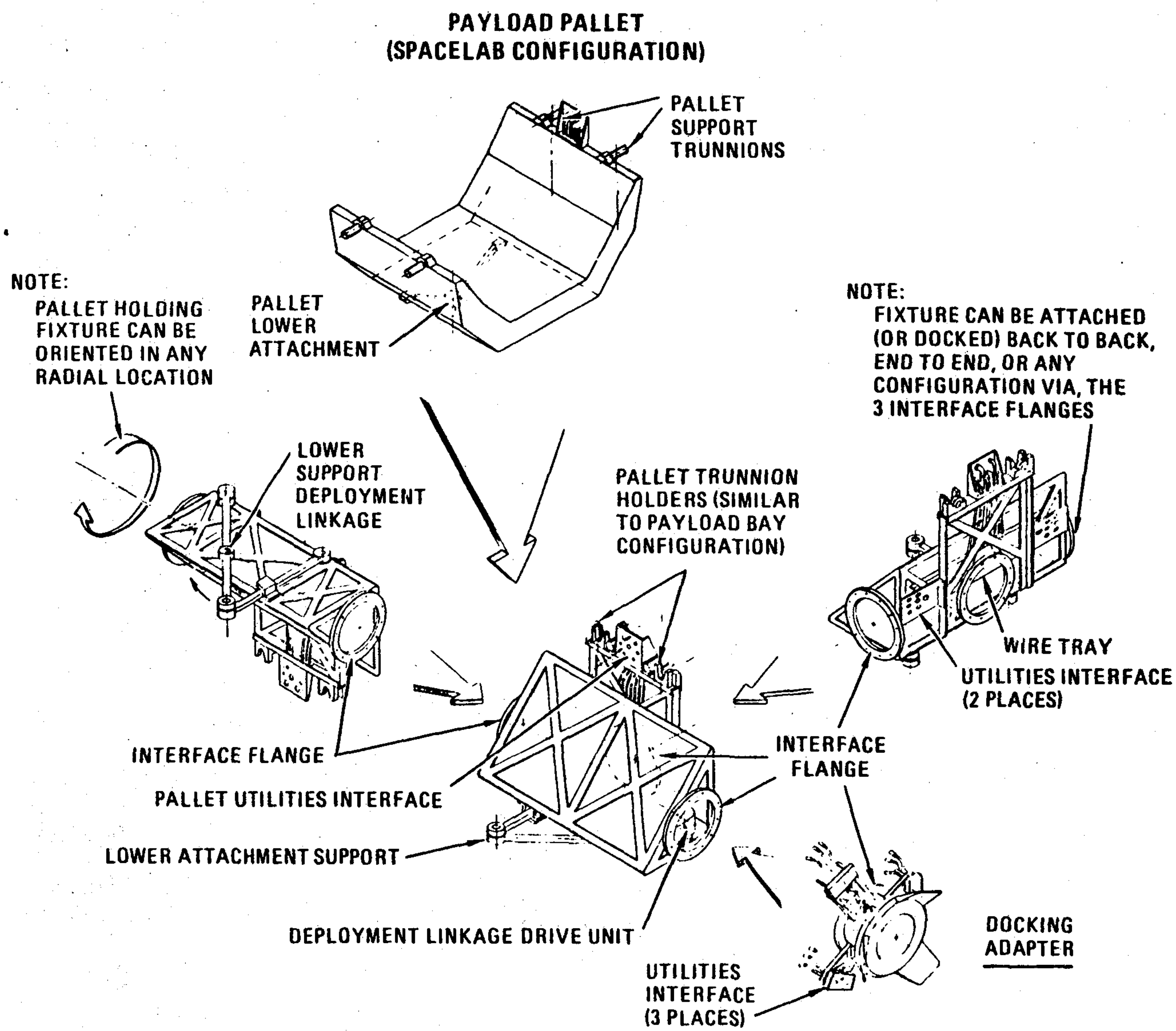
TRIANGULAR STRONGBACK-MEDIUM PLATFORM



Triangular Strongback

An alternative concept to payload accommodation is based on the use of triangular strongback sections depicted opposite, Reference 5 Free Flying Power Module Studies. The sections are joined in pairs, and then end to end to form a payload accommodations spine. At the end of a central column, other triangular sections are joined to form ribs. Utilities are carried on lines mounted in utilities trays along the length of each section. The lines are connected between sections by appropriate jumpers.

TRIANGULAR STRONGBACK



- **SUPPORT CONCEPT CONTAINS**
  - **PALLET–PHYSICAL ATTACH PROVISIONS (SAME AS ORBITER)**
  - **SUPPORT PHYSICAL ATTACH FLANGES–PROVIDES PALLET TO PALLET OR PALLET TO OSM ATTACH VIA BOLT RINGS**
  - **PALLET UTILITIES INTERFACE –POWER, DATA, AND FLUID INTERFACES ARE PROVIDED (SAME AS ORBITER)**
  - **DOCKING ADAPTER – PERMITS ATTACH TO OSM INCLUDING UTILITIES SERVICE**

Platform and Installed Payload Weight Summary

Initial weight estimates for the total complex in orbit are shown in the next four pages for the baseline Platform No. 1 at 28 degrees, as well as Platform No. 2 (a smaller version at 90°), and Platform No. 3 (essentially the same configuration as No. 1, but with a different payload set). The two summary weight statements on the next two pages also include a column identified as "Platform 2 and 3." In this arrangement, a larger platform configuration was conceived for a 90° orbit, and functions as a single host vehicle for all the payloads originally intended for Platforms 2 and 3.

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PLATFORM AND INSTALLED PAYLOAD WEIGHT SUMMARY (LB)

	PLATFORM 1 Inclination, 28°		PLATFORM 2 Inclination, 90°		PLATFORM 3 Inclination, 57°		PLATFORMS 2&3 Inclination, 90°	
PLATFORM VEHICLE WEIGHT	(18,580)		(13,269)		(18,580)		(23,725)	
• PLATFORM STRUCTURE		623		367		623		866
• UTILITY SERVICE MOD.		15,233		11,075		15,233		19,323
• UTILITY DISTRIBUTION		1,035		621		1,035		1,680
• CONTINGENCY		1,689		1,206		1,689		2,157
PROPULSION STAGE WEIGHT	(4,624)		(4,970)		(4,624)		(10,060)	
PAYLOAD MOUNTING	(22,425)		(15,165)		(20,420)		(35,580)	
• ERNO PALLETS (MOD)		9,000		7,200		12,600		19,800
• PAYLOAD TABLES		3,525		2,115		2,820		4,930
• INSTR. POINTING SYST.		9,900		3,300		3,300		6,600
• GIMBAL SYSTEM		-		2,550		1,700		4,250
PAYLOAD EQUIPMENT	(40,260)		(22,660)		(18,634)		(41,294)	
TOTAL ON-ORBIT WEIGHT	(85,889)		(56,064)		(62,258)		(110,659)	

PLATFORM AND INSTALLED PAYLOAD WEIGHT SUMMARY (KG)

	PLATFORM 1 Inclination, 28°		PLATFORM 2 Inclination, 90°		PLATFORM 3 Inclination, 57°		PLATFORMS 2&3 Inclination, 90°	
PLATFORM VEHICLE WEIGHT	(8,428)		(6,019)		(8,428)		(10,762)	
• PLATFORM STRUCTURE		283		166		283		393
• UTILITY SERVICE MOD.*		6,910		5,024		6,910		8,764
• UTILITY DISTRIBUTION		470		281		470		762
• CONTINGENCY		765		547		765		978
PROPULSION STAGE WEIGHT	(2,097)		(2,254)		(2,097)		(4,563)	
PAYLOAD MOUNTING	(10,172)		(6,879)		(9,263)		(16,139)	
• ERNO PALLETS (MOD)		4,082		3,266		5,715		8,981
• PAYLOAD TABLES		1,599		959		1,279		2,236
• INSTR. POINTING SYST.		4,491		1,497		1,497		2,994
• GIMBAL SYSTEM		-		1,157		771		1,928
PAYLOAD EQUIPMENT	(18,262)		(10,278)		(8,452)		(18,731)	
TOTAL ON-ORBIT WEIGHT	(38,959)		(25,430)		(28,240)		(50,195)	

*Breakdown of utilities module on next page.

ELECTRICAL POWER SUBSYSTEM WEIGHT

ITEM	COMPONENT WEIGHT, KG (LB)					
	6-ARRAY		4-ARRAY		2-ARRAY	
ORIENTATION ELECTRONICS	10	(22)	10	(22)	10	(22)
SOLAR ARRAY DISTRIBUTOR	44	(96)	29	(64)	15	(32)
POWER COND. DISTRIBUTOR	109	(240)	73	(160)	36	(80)
BATTERY MODULE (Ni-Cd)	2768	(6102)	1845	(4068)	923	(2034)
BATTERY CHARGER	98	(216)	65	(144)	33	(72)
H.V. DC DISTRIBUTOR	54	(120)	54	(120)	54	(120)
SUBSYSTEMS DISTRIBUTOR	65	(143)	65	(143)	65	(143)
LOW VOLTAGE (28 VDC) REG.	65	(144)	65	(144)	33	(72)
400 Hz 3 Ø INVERTER	68	(150)	68	(150)	68	(150)
WIRE HARNESS	181	(400)	181	(400)	181	(400)
TOTAL	3462	(7633)	2455	(5415)	1418	(3125)

UTILITIES SERVICE MODULE WEIGHT SUMMARY

COMPONENT	WEIGHT, KG (LB)					
	LARGE		MEDIUM		SMALL	
SOLAR ARRAY ASSEMBLY	925	(2,040)	617	(1,360)	308	(680)
ARRAY MAST	104	(230)	104	(230)	104	(230)
ARRAY ORIEN. MECH.	240	(530)	240	(530)	240	(530)
ELEC. POWER SUBSYSTEM *	3462	(7,633)	2461	(5,415)	1418	(3,125)
THERMAL CONTROL SUBSYS.	1878	(4,140)	1340	(2,948)	798	(1,760)
UTILITY MODULE STRUCT.	1338	(2,950)	1340	(2,950)	1338	(2,950)
INSUL. & MET. PROT.	227	(500)	227	(500)	227	(500)
GUIDANCE & CONTROL	456	(1,000)	456	(1,000)	456	(1,000)
DOCKING MODULE	136	(300)	136	(300)	136	(300)
TOTAL	8764	(19,323)	6910	(15,233)	5024	(11,075)

* Breakdown of electrical power subsystem weight on next page.

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4.0 SUBSYSTEM AND TECHNOLOGY PROGRAM DEFINITION

Each subsystem of the Space Sciences and Applications Platform is now described in turn. For each subsystem, the discussion will (1) describe the subsystem and/or major issues pertinent to that subsystem; (2) define the trade studies carried out in this initial definition; (3) identify technology advancement issues for that subsystem, and rate them as "Enabling", "Enhancing", or "Routine" for phase C/D development; (4) describe the technology effort in terms of its objectives, current status, approach, and expected results; (5) and finally, suggest a schedule and budget for each technology program element.

After each subsystem has been treated, one further broad assessment was made, to provide a measure of current "technical adequacy" for each technology issue.

4.1 ELECTRICAL POWER SUBSYSTEM

The electrical power subsystem consists of solar arrays for power generation, batteries for energy storage, power conditioning to provide conditioned power for battery charging and end item use, and distribution elements to distribute and control electrical power to the utility module subsystems and the payload positions on the platform.

A nominal power level of 20 kW for the payloads with an additional 2 kW for subsystem (22 kW total) was selected to baseline the power subsystem elements. The 20 kW satisfies the circa 1985 payload requirements with some margin. It is also sufficient to allow technology advancement requirements to be identified and evaluated.

In the following charts the subsystem requirements, evaluation of subsystem element alternative, the selected baseline subsystem configuration and the necessary technology development program are presented.

Electrical Power System Requirements

Payload power requirements for circa 1985 platforms 1, 2 and 3 have been estimated at 14.5 kW, 6.6 kW, and 16.4 kW respectively. A power output capability of 20 kW would satisfy these requirements with some margin. This power level plus 2 kW for housekeeping requirements (22 kW total) was used as the baseline requirement for selecting and sizing the subsystem elements. An interface with the orbiter is provided to supplement orbiter power when it is necessary for the orbiter to stay with the platform for extended periods.

Power forms required (28 V dc and 115 V, 400 Hz, 3Ø) for payloads are based on providing the same services provided by the spacelab. 115 V, 400 Hz, 3Ø power is required by the thermal control subsystem for freon pumps and flow control valves.

The subsystem is modular in design to accommodate growth by serial addition of solar array blankets, batteries, power conditioning, etc.

ELECTRICAL POWER SUBSYSTEM REQUIREMENTS

- CONTINUOUS POWER OUTPUT—EOL
    - 20 KW TO PLATFORM PAYLOADS
    - 2 KW TO SUBSYSTEMS HOUSEKEEPING
- 10-YEAR LIFE WITH MAINTENANCE
- VOLTAGE COMPATIBLE WITH ORBITER AND PAYLOAD REQUIREMENTS
    - 27.5 V DC TO 32.5 V DC (SUBSYSTEMS, PAYLOADS, & ORBITER)
    - 115 V, 400 Hz, 3 Ø (SUBSYSTEMS & PAYLOADS)
- ORBIT PARAMETERS
    - 400 KM, 28.5 DEGREES INCLINATION—PLATFORM 1*
    - 575 KM, 90 " " —PLATFORM 2
    - 400 KM, 57 " " —PLATFORM 3
- PLATFORM DISTRIBUTION
    - 5-KW CAPABILITY TO EACH PAYLOAD INTERFACE
    - DEAD-FACE SWITCHING AT EACH PAYLOAD INTERFACE
- ACCOMMODATE GROWTH

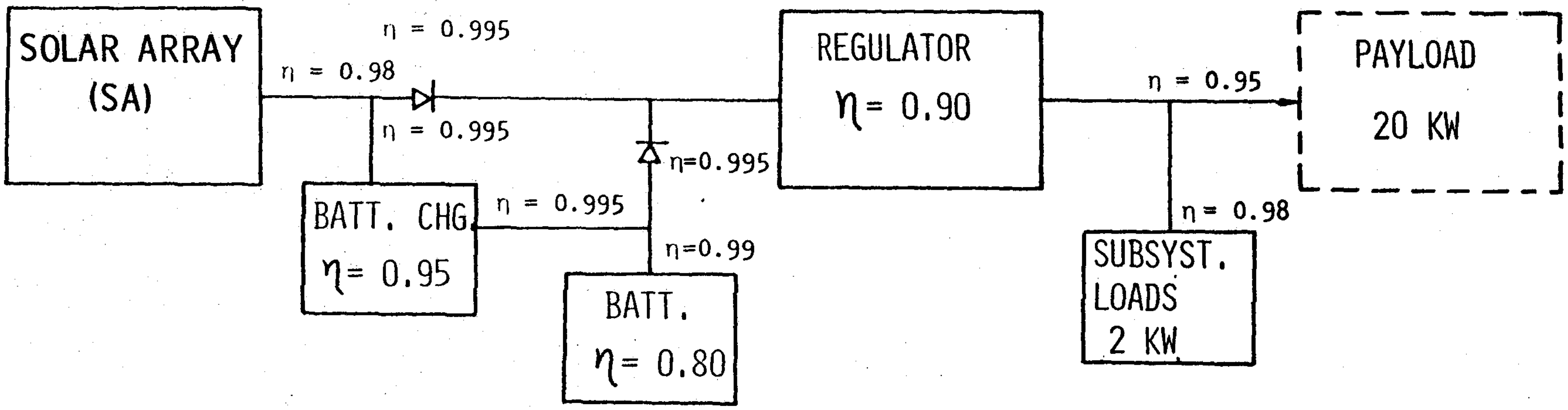
*Baseline System

Electrical Power Subsystem Efficiency Diagram

The figure represents the efficiency diagram of the electrical power subsystem. For the subsystem with these efficiencies to provide 20 kW to payloads, a solar array output of 49 kW minimum is required. Four Solar Electric Propulsion (SEP) size arrays will provide sufficient solar array area for an end of life (EOL) level and are used for the baseline.

The T-shadow/T-eight (Ts/Tl) ratio varies with Beta angle (angle between orbit plane and sun line) and is the greatest when Beta angle equals zero. The Ts/Tl ratio shown in the figure are for zero Beta angle. The Ts/Tl ratio also varies with altitude, becoming less as altitude increases. Thus the Ts/Tl ratio for platform 2 (575 km vs. 400 km for platforms 1 and 3) is lower than that for platforms 1 and 3.

ELECTRICAL POWER SUBSYSTEM EFFICIENCY DIAGRAM



$$SA = \frac{\frac{20}{.95} + \frac{2}{.98}}{(.90)(.995)(.99)(.80)(.99)(.995)(.95)(.995)(.98)}\left(\frac{T_S}{T_L}\right) + \frac{\frac{20}{.95} + \frac{2}{.98}}{(.90)(.995)(.98)}$$

$$SA = 35.68\,\frac{T_S}{T_L} + 26.31$$

	PLATFORM 1	2	3
$\frac{T_S}{T_L}$	0.6383	0.5868	0.6383
SA (KW)	49.08	47.25	49.08

Comparison of Solar Array Orientation Concepts

The angle between the sunline and the spacecraft flight path (Beta Angle) represents the sum of two cyclic conditions. One of these results from the annual movement of the earth around the sun ($\pm$ 23.44 degrees), and superimposed on it is a cycle with a magnitude that is $\pm$ the orbit inclination. As the Beta angle increases, the eclipse duration decreases. The Beta angle also defines (or partially defines) the incidence angle of the sun to the solar array for fixed or single degree of freedom arrays.

The data shown on the chart was derived for a 900 km orbit and while there will be some variations for other altitudes the comparisons are the same. The data reflects performance for four orientation concepts as follows:

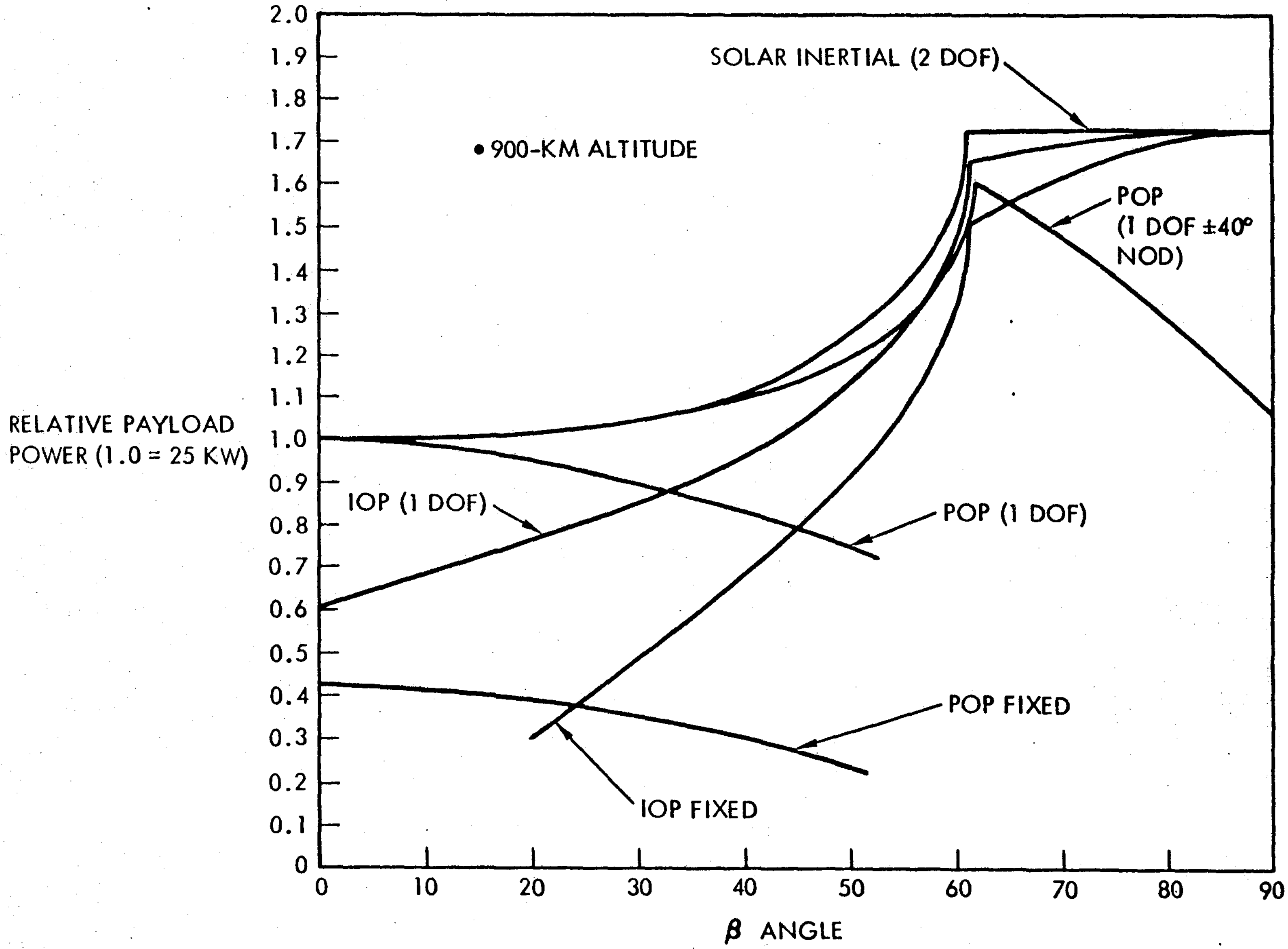
(a) Fixed - The spacecraft orientation has the arrays perpendicular to the orbit plane for low Beta angles. The spacecraft turns 90 degrees to place the array in the orbit plane for high Beta angles.

(b) Single degree of freedom - Same spacecraft orientation requirements as (a) above.

(c) Single degree of freedom with $\pm$ 40 degrees of NOD - Operates with array perpendicular to orbit plane.

(d) Two degree of freedom - Any spacecraft orientation acceptable.

COMPARISON SOLAR ARRAY ORIENTATION CONCEPTS



Solar Cell Selection

High efficiency silicon solar cells, based on current and near term projections of cost and performance data on space rated solar cells, are the more cost effective for this platform application. Lower efficiency silicon cells, while less expensive per cell, are more expensive on an array basis due to the additional area required. Gallium Arsenide (GaAs) solar arrays are more expensive because of cell costs, although for higher orbits where radiation is more intense they start to become cost effective due to their inherent greater radiation resistance.

On an area basis, GaAs arrays require (depending on radiation environment) less than 70% that of high silicon arrays. Area of the array is an important factor due to its contribution to the control dynamics problem and orbit make-up requirements. When those area contributions are quantified (not accomplished in this study) GaAs will look that more attractive.

SOLAR CELL SELECTION

- BASED ON PRESENT COST AND PERFORMANCE DATA ON SPACE-RATED SOLAR CELLS:
  - HIGH-EFFICIENCY SILICON CELLS LOWEST COST FOR MOST LEO APPLICATIONS WHERE RADIATION IS NOT A MAJOR FACTOR
    - —NOT CONSIDERING ANY PENALTY FOR ARRAY AREA
  - WHEN INTEGRATED RADIATION DOSES EXCEED 2×10^{15} ELECTRONS/CM^2 (1 MEV ELECTRON EQUIV.) GAAS STARTS TO BECOME EFFECTIVE
    - —FOR A 900-KM ORBIT AND TEN-YEAR LIFE, GAAS ARRAY COSTS ARE ABOUT THE SAME AS SILICON
- TECHNOLOGY DEVELOPMENT
  - SILICON — ACTIVITY AIMED AT MASS-PRODUCTION FOR COST REDUCTION AND LARGE DEPLOYABLE ARRAY DESIGN
  - GAAS — ACTIVITY IN CELL DEVELOPMENT, BLANKET/ARRAY DESIGN (PLANAR AND CONCENTRATED), AND MASS PRODUCTION
- CELL SELECTION
  - NEAR-TERM — SILICON CELL—COST REDUCTIONS WILL OFFSET GAAS PERFORMANCE GAIN
  - FAR-TERM — GAAS—PERFORMANCE GAINS AND COST REDUCTION SHOULD MAKE GAAS COST EFFECTIVE FOR:
    - (A) HIGH-POWER APPLICATIONS WHERE THE ARRAY AREA MUST BE CONSIDERED
    - (B) APPLICATIONS INVOLVING HIGH-RADIATION EXPOSURE

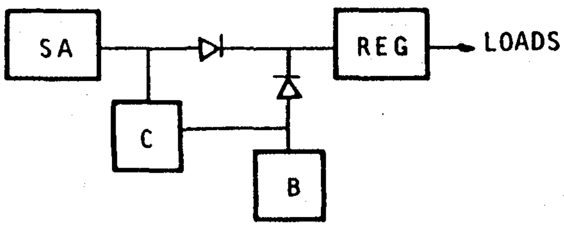
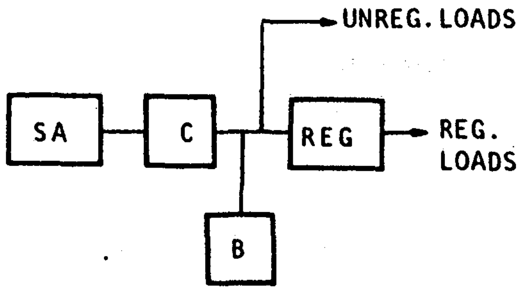
Power Conditioning Alternatives

Two alternative power conditioning concepts were evaluated for this application: (1) unregulated direct energy transfer with parallel peak power tracking battery charger, and (2) in-line (series) regulated battery charger.

The unregulated direct energy transfer concept uses a parallel peak (maximum) power (V_{pm}) tracker for battery charging. The solar array maximum power voltage point is tracked by the battery charger to extract power for battery charging that is in excess of load requirements. This system was used by the Skylab ATM along with an additional regulator to control the system output voltage to the level desired by the loads. The system offers a good energy transfer efficiency, however, with the wide voltage excursion possible (+100%, -18%) almost all loads would require some additional regulation which detracts from overall system efficiency. Another consideration, although minor, is the condition where the load (time-limited) exceeds the solar array output and the battery must share part of the load. The solar array voltage drops to the battery discharge voltage level under this condition and, when operating at this voltage level, is about 20% down from its maximum power capability.

The in-line regulator concept routes the entire solar array output through the peak power tracker battery charger. In this manner, tracking of the maximum power voltage is accomplished over the range of array temperature variations, yet the output voltage is controlled to a reasonable range. It is slightly less efficient in energy transfer due to the line regulation losses, but does offer advantages in load sharing since it does not suffer the 20% array loss of the parallel charger concept when the array and battery are sharing the loads. Another minor advantage relates to the maximum reverse bias voltage that can exist across a faulted (shadowed or failed) cell on the solar array. The maximum voltage for this condition is the open-circuit voltage for the cell string minus the array output voltage. Unlike the parallel concept that can operate below the maximum power voltage, the array for this concept will always operate at or above it; thus, the potential reverse bias voltage is less. A modified multi-mission spacecraft standard power regulator is compatible with either approach. Either of these approaches is acceptable and selection is not critical to the technology program.

POWER CONDITIONING ALTERNATIVES

	PARALLEL CHARGER	SERIES CHARGER
		
• PRIOR PROGRAM USE	SIMILAR TO ATM	SIMILAR TO MMS
• PERFORMANCE		
• V_{PM} TRACKING	CHARGER ONLY	FULL
• LOAD SHARING (SA & B)	APPROX. 20% SA LOSS	NO PENALTY
• UNREG BUS EXCURSION	+100/-18%	± 15%
• EFFICIENCY	HIGHEST IF ALL LOADS NEED REGULATED PWR	HIGHEST IF AT LEAST 25% OF LOAD CAN USE UNREG. BUS
• HARDWARE TECHNOLOGY	MODIFIED MMS	MODIFIED MMS

Energy Storage Alternatives

The next two charts show comparison data for energy storage alternatives. Practical energy storage alternatives for the application are Nickel-Cadmium (Ni-Cd) and Nickel hydrogen ($Ni-H_2$) batteries, and a regenerative fuel cell (RFC) system. The Ni-Cd battery has been the standard means of energy storage for satellites from the start. Its operational characteristics are fairly well known, as are the methods for dealing with its shortcomings.

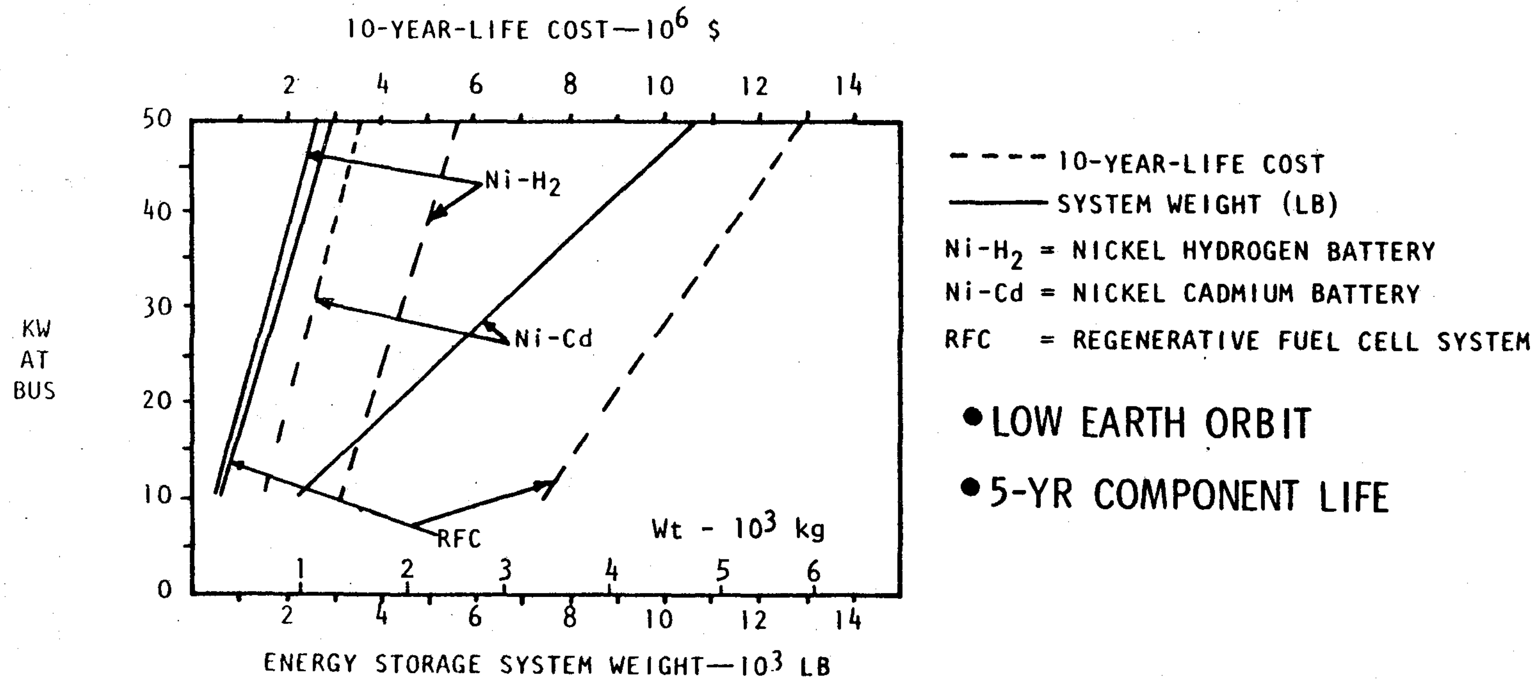
The $Hi-H_2$ battery is a relatively recent development with effort initiated for development around 1970. The system is very attractive due to its potential long life and improved useful specific energy over Ni-Cd. However, it has been used thus far on only a couple of spacecraft. The Air Force Aeropropulsion Lab (AFAPL) presently has a program in work to develop and qualify a $Ni-H_2$ battery cell specifically for low earth orbit application.

A regenerative fuel cell system consists of an electrolysis module, gaseous reactant and with storage tanks, and accessories of pumps, valve plumbing, etc. The fuel cell part of this system has proven reliable space operation, but the electrolysis part of the system has been limited to the laboratory. System efficiency is somewhat less than that for a battery system.

The $Ni-H_2$ battery offers advantages in weight over the other concepts. While the RFC is about the same weight as a $Ni-H_2$ battery system, its costs are greater due in part to the additional solar array required to compensate for its lower efficiency. A Ni-Cd battery system is much heavier than the others, but it does have the advantage of availability. The costs shown do not include transportation to orbit costs. For the Ni-Cd battery system with its much greater weight this could be significant.

Selection: Ni-Cd early; $Ni-H_2$ later

ENERGY STORAGE ALTERNATIVES



- LOW EARTH ORBIT
- 5-YR COMPONENT LIFE

- CURRENT CELL DESIGNS FOR Ni-Cd AND $Ni\text{-}H_2$
- ADVANCED FUEL CELL TECHNOLOGY
- COSTS INCLUDE DDT&E & PRODUCTION; TRANSPORTATION TO ORBIT & TRANSPORTATION OF REPLACED PARTS TO EARTH NOT INCLUDED. FOR THE Ni-Cd, WITH ITS GREATER WEIGHT, THIS IS A SIGNIFICANT COST.
- REGENERATIVE FUEL CELL VALUES INCLUDE DELTA SOLAR ARRAY WEIGHT AND COST DUE TO LOWER SYSTEM EFFICIENCY (AS COMPARED TO BATTERIES)

ENERGY STORAGE SYSTEM COMPARISON

(See text on previous chart)

ENERGY STORAGE SYSTEM COMPARISON

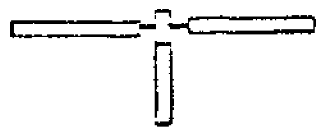
	ADVANTAGES	DISADVANTAGES
NICKEL-HYDROGEN ($Ni-H_2$)	• PERFORMANCE Overdischarge stability Overcharge tolerance Operating temperature • SYSTEM WEIGHT • POTENTIAL LIFE	• DEVELOPMENT STATUS • VOLUME
NICKEL-CADMIUM (Ni-Cd)	• AVAILABILITY • EXPERIENCE	• SYSTEM WEIGHT • LIFE
REGENERATIVE FUEL CELL SYST. (RFC)	• SYSTEM WEIGHT • POTENTIAL LIFE • INCREASED STORAGE CAPACITY WITH MINIMUM WEIGHT IMPACT	• DEVELOPMENT STATUS • DDT&E COSTS • LOWER EFFICIENCY

Utility Module Power Output

The power output data shown on this chart are those that would be obtained by using SEPS size solar arrays with high efficiency silicon solar cells. These data are for conditions of zero Beta angle. As Beta angle increases there will be increases in power output capability due to the decreased eclipse duration. For all three configurations, housekeeping loads were assumed to be 2 kW.

UTILITY MODULE POWER OUTPUT

POWER (KW) TO PLATFORM ($\beta = 0$)

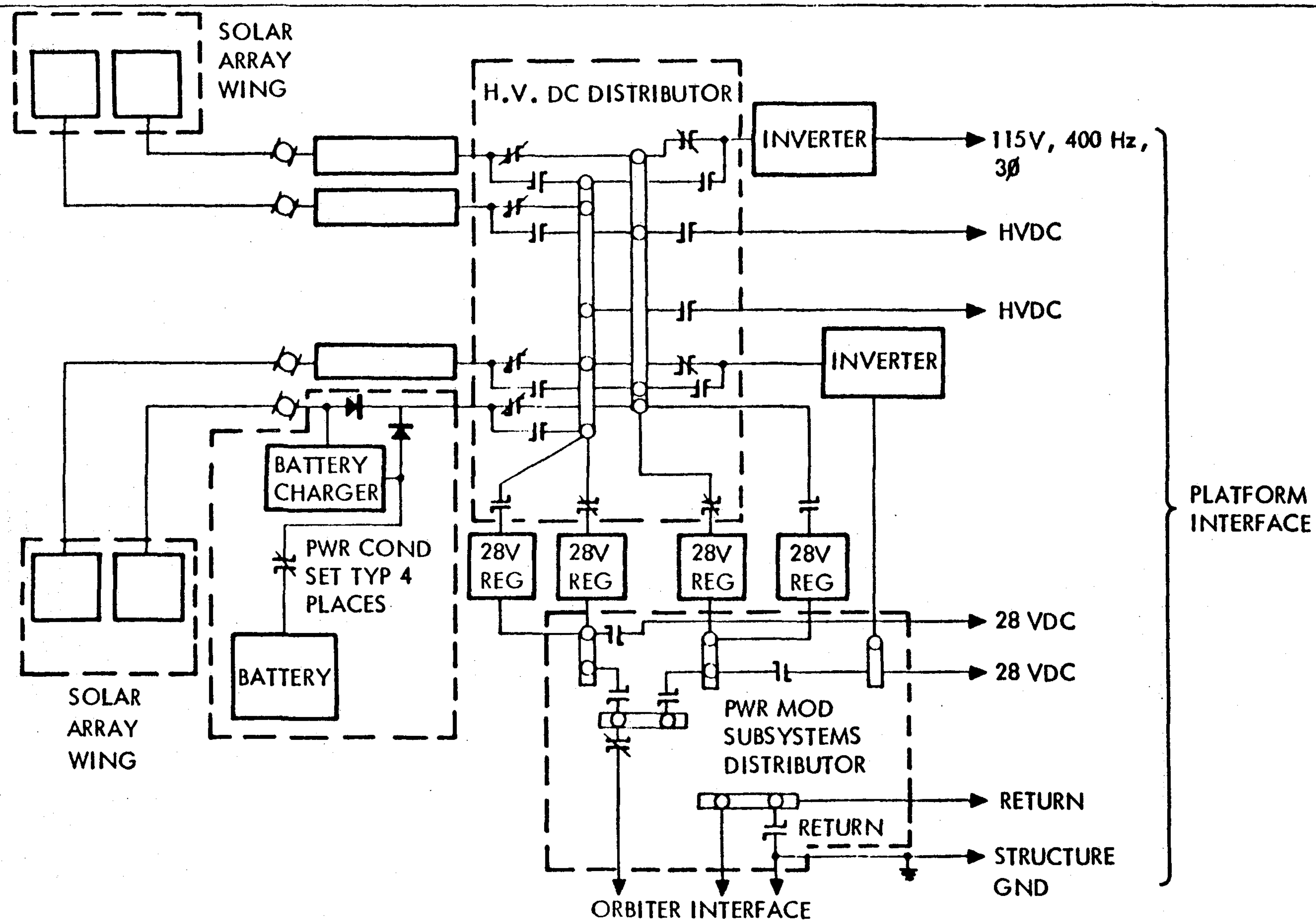


	2-ARRAY	4-ARRAY	6-ARRAY
400 KM, 28.5° INCL. (P-1)			
BOL	11.6	25.1	38.6
5 YR	9.9	21.7	33.5
10 YR	9.3	20.6	31.9
575 KM, 90° INCL. (P-2)			
BOL	12.1	26.1	40.2
5 YR	9.0	20.0	30.9
10 YR	8.4	18.7	28.9
400 KM, 57° INCL. (P-3)			
BOL	11.6	25.1	38.6
5 YR	9.7	21.2	32.7
10 YR	9.1	20.1	31.1

Power Subsystem

The power subsystem concept shown here has each of the solar arrays associated with a battery and battery charger as a set. Each of these sets is capable of feeding either of two redundant main high voltage dc (HV dc) buses, but would normally be dedicated to one. Regulators buck the HV dc down to 28 V dc for utility module subsystems and for the orbiter interface. The orbiter interface also allows the orbiter to provide utility module initialization power and control. HV dc is the primary supply to the platform where regulators buck it down to 28 V dc for payload use. Inverters provide 115 V, 400 Hz, 3Ø for payloads and utility module subsystems (freon pump, etc.).

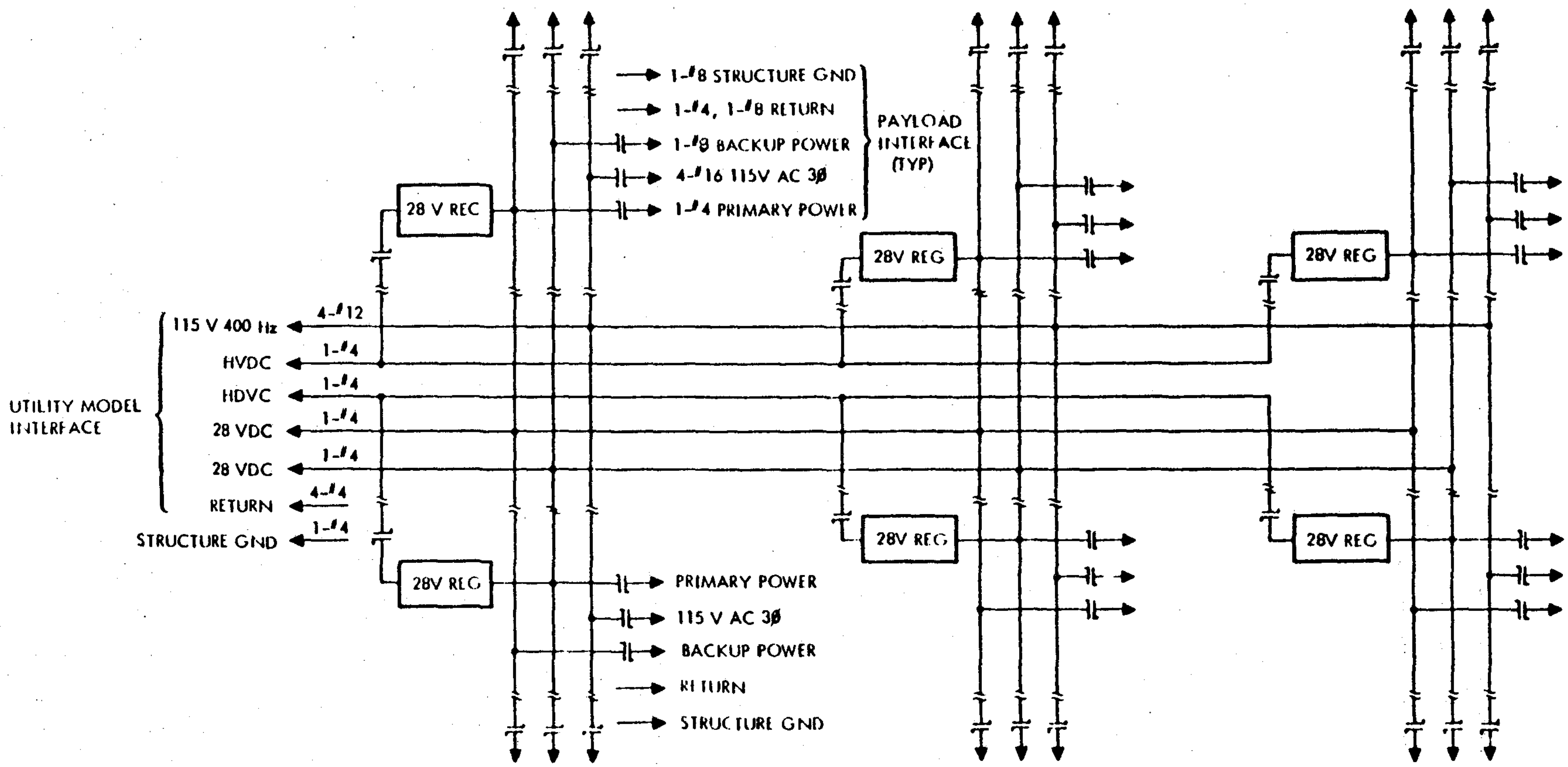
POWER SUBSYSTEM



Power Subsystem - Platform Distribution Concept

The chart shows a concept for distributing power to the payloads on the platform. In this concept each payload interface would be identical and would include primary 28 V dc at 5 kW peak capability, up to 2 kW 28 V dc backup power, and 2 kW maximum 115 V ac power. It is not expected that a payload would use the maximum capability of each power source at the same time. Regulators are located on the platform to buck HV dc to 28 V dc near the payloads. Having the regulators near the payloads reduces wire weight, improves efficiency, and reduces voltage variation at the payload interface.

POWER SUBSYSTEM - PLATFORM DISTRIBUTION CONCEPT



Electrical Power Subsystem Equipment

This chart shows the estimated size and weight of the power system equipment located within or on the utility module. It does not include power distribution items on the platform.

ELECTRICAL POWER SUBSYSTEM EQUIPMENT

ITEM	UNIT WT (KG)	UNIT SIZE (CM)	QUANTITY 6-ARRAY	QUANTITY 4-ARRAY	QUANTITY 2-ARRAY
SOLAR ARRAY BLANKET*	154	158 (4 m) × 1220 (31 m)	6	4	2
ARRAY ORIENTATION MECH.	240		1	1	1
ORIENTATION ELECTRONICS	10	25×46×15	1	1	1
SOLAR ARRAY DISTRIBUTOR	7	20×30×15	6	4	2
POWER CONITIONING DIST.	18	25×43×25	6	4	2
BATTERY MODULE (Ni-Cd)	51	30×46×20	54	36	18
BATTERY CHARGER	16	25×66×15	6	4	2
H.V. DC DIST.	54	40×91×25	1	1	1
SUBSYSTEMS DISTRIBUTOR	65	40×91×25	1	1	1
LOW VOLTAGE (28 VDG) REG.	16	25×66×15	4	4	2
400 ~ 3ϕ INVERTER	34	46×53×18	2	2	2
WIRE HARNESS	181		1	1	1

*DOES NOT INCLUDE EXTENSION MAST

Power Subsystem Trades/Selections

The next two charts summarize the trades and selections made for the components of the power subsystem and assesses the level of technology for each.

The Solar Electric Propulsion technology program is the principle effort that is on-going with respect to near term large solar array development. This technology is directed primarily toward design of the blanket and the deployment mechanism. The type of solar cell (silicon, Gallium Arsenide, etc.) used on the array is a secondary issue. Silicon cells, of course, are well developed and any technology efforts expended should be pointed toward reducing cell costs via automated manufacturing processes. GaAs solar cells are not so well developed, but since they do offer advantages in efficiency, radiation resistance and greater compatibility with concentrated arrays [concentration ratio (CR) greater than one] over silicon cells, it is a technology that should be pursued for introduction on later elements.

The SEP program array development technology is designed for deployment of a single array. On the utilities module it will be necessary to deploy multiple arrays, thus a delta technology effort is necessary.

The Ni-Cd battery is a state-of-the-art device, while the Ni-H_2 battery and the regenerative fuel cell both need technology advancement. The Hi-H_2 battery technology is being supported and is moving ahead, but is unlikely to be ready for initial platform use. However, since it does have potential for longer operating lifetimes it should be pursued for later introduction.

POWER SUBSYSTEM TRADES/SELECTIONS

COMPONENT	ALTERNATES	SELECTION	TECH. LEVEL*
POWER GENERATION			
SOLAR CELLS	SILICON	Si—CURRENT S.O.A.—LOWEST RISK/COST	2
	GaAs	GaAs—Δ FOR LATER INTRODUCTION	2
SOLAR ARRAYS	SEP DERIVED—CR = 1	SEP—CR = 1, LOWEST RISK/INITIAL COST	2
	ADVANCED—CR > 1		
DEPLOYMENT CONCEPT	ERECTABLE ARRAYS		
	DEPLOYABLE ARRAYS	DEPLOYABLE—SEP DERIVED TECHNOLOGY	1
ORIENTATION CONCEPT	1 DEGREE OF FREEDOM (DOF)		
	1 DOF + NOD		
	2 DOF	2 DOF—MISSION FLEXIBILITY	3
ENERGY STORAGE			
BATTERY	NiCd	NiCd—AVAILABILITY & LOWEST RISK	3
	NiH_2	NiH_2—Δ FOR LATER INTRODUCTION	2
	REGENERATIVE FUEL CELLS		

*TECHNOLOGY LEVELS

1 - ENABLING

2 - ENHANCING

3 - NORMAL ØB/C/D DEVELOPMENT

Electrical Power Subsystem Trades/Selections (Cont.)

High voltage dc with local conversion to 28 V dc where required is more efficient, requires considerably less cabling weight, and provides better voltage regulation for the payloads than a conventional 28 V dc approach. Distribution of HV dc may also encourage payload designers to use power at the high voltage level. The parallel battery charging concept was selected because of its higher energy transfer efficiency. The series concept is almost as good, however, and it would be better suited because of its closer voltage regulation for conditions where the payloads could use the high voltage dc directly.

Pulse width modulation buck regulation is the conventional approach to converting from high voltage dc to lower voltage. The efficiency of this approach is adequate for the ratio of input to output voltage expected for this program. For higher ratios, which may be encountered on later programs where the solar array may be operated at higher voltages, its efficiency degrades and the transformer coupled converter would be a better selection. Technology for either approach is adequate for normal Ø B/C/D development.

ELECTRICAL POWER SUBSYSTEM TRADES/SELECTIONS (CONT.)

Sheet 2 of 2

COMPONENT	ALTERNATES	SELECTION	TECH. LEVEL*
POWER DISTRIBUTION			
PRIMARY DISTRIBUTION VOLTAGE LEVEL	HI-VOLTAGE DC (WITH LOCAL CONVERSION TO 28 VDC) 28 VDC	HVDC—LOWEST CABLING WT. & POWER LOSS	3
POWER CONDITIONING CONCEPT (BATTERY CHARGING)	SERIES PARALLEL SHUNT	PARALLEL—EFFICIENCY	3
POWER REGULATION/ CONVERSION			
- LOW VOLTAGE REG.	BUCK REGULATOR TRANSFORMER-COUPLED CONV.	CURRENT S.O.A.—LOWER INITIAL INVESTMENT	3
- BATTERY CHARGE REG.	BUCK REGULATOR	NO PRACTICAL ALTERNATIVES—S.O.A.	3
- INVERTER	INTEGRATED INDIVIDUAL PHASE	INTEGRATED—LOWER COST TO USERS—S.O.A.	3

*TECHNOLOGY LEVELS

1 - ENABLING

2 - ENHANCING

3 - NORMAL ØB/C/D DEVELOPMENT

Electrical Power Subsystem Technology Candidates

The next four charts summarize the technology development program required for the power subsystem.

The Department of Energy (DOE) is sponsoring several programs aimed at decreasing the cost of silicon solar cells for terrestrial application. Material and process development, automated manufacturing, and alternative materials research are examples. The results of some of these programs will undoubtedly have application to aerospace cells. It is the application of those developments without significantly impacting cell efficiency that should be accomplished.

Gallium Arsenide (GaAs) solar cells are superior to silicon solar cells with respect to conversion efficiency and radiation resistance. It would be desirable to phase GaAs in for later platform use when GaAs cells have been more fully developed and cell costs reduced. There appears, however, to be a marginal supply of gallium available to support large solar array production due to two conditions. First there is a limited production capacity and second the present method of cell production wastes about 90% of the GaAs bole from which the cell is cut. If GaAs solar cells are to be used on future large solar arrays, it will be necessary to increase the national gallium recovery capacity and/or significantly reduce the amount wasted in the cell production process.

ELECTRICAL POWER SUBSYSTEM TECHNOLOGY CANDIDATES

SILICON SOLAR CELLS

OBJECTIVE

- DEVELOP LOW-COST AEROSPACE CELL

TECHNOLOGY ASSESSMENT

- PRESENT AEROSPACE CELL COST—$100/PEAK WATT
- AEROSPACE CELL EFFICIENCY—14% → 15%
- TERRESTRIAL CELL COSTS:

  $15/PEAK WATT → $2/PEAK WATT (DOE 1982 GOAL)

APPROACH

- AUTOMATED PRODUCTION PROCESSES
- APPLICATION OF TERRESTRIAL CELL TECHNOLOGY

EXPECTED RESULTS

- HIGH-EFFICIENCY CELLS (13-15%) AT LOWER UNIT COST

GALLIUM-ARSENIDE SOLAR CELLS

OBJECTIVE

- PROVIDE HIGH-EFFICIENCY GaAs SOLAR CELLS FOR LARGE SOLAR ARRAY PRODUCTION

TECHNOLOGY ASSESSMENT

- AFAPL PROGRAM TO DEVELOP CELL EFFICIENCY—17% → 20%
- MARGINAL GALLIUM SUPPLY TO SUPPORT LARGE SOLAR ARRAY PRODUCTION

APPROACH

- INCREASE GALLIUM RECOVERY CAPACITY
- CELL PROCESS DEVELOPMENT TO REDUCE GALLIUM WASTE IN PROCESS
- DEVELOP CELL PRODUCTION CAPABILITY

EXPECTED RESULTS

- INCREASED SUPPLY OF GALLIUM MATERIAL
- HIGH-EFFICIENCY, RADIATION-RESISTANT SOLAR CELL
- LOW-COST PRODUCTION PROCESSES

Electrical Power Subsystem Technology Candidates

The SEP solar array program is developing a lightweight large-area deployable array. While the array panel design concept should be supportive of an automated cell "laydown" process, development of this automated process is not a part of the present SEP solar array program.

The deployment mechanism for the SEP array is designed, and will be tested on orbit, to deploy a single array. For this application it will be necessary to have multiple array development. It is anticipated that such a concept can be readily developed using the SEP concept as a point of departure.

Technology for the $Ni\text{-}H_2$ battery is near at hand. The main problem inhibiting its availability for the platform is the lack of operating experience. Commitments by users, such that production capacity and a performance data base can be established, is needed before commiting the platform program to $Ni\text{-}H_2$ batteries in lieu of available Ni-Cd batteries.

ELECTRICAL POWER SUBSYSTEM TECHNOLOGY CANDIDATES

SOLAR ARRAY

OBJECTIVE

- LOW-COST SOLAR ARRAY

TECHNOLOGY ASSESSMENT

- SEPS PROGRAM TO DEVELOP LIGHT-WEIGHT, LARGE-AREA DEPLOYABLE ARRAY
- ARRAY PANEL ASSEMLY (CELL LAY-DOWN) COSTS
  - NON-AUTOMATED, SMALL SATELLITE $190/PEAK WATT
  - AUTOMATED, SEPS CONCEPT, EST. <$20/PEAK WATT

APPROACH

- AUTOMATED ARRAY POWER ASSEMBLY PROCESSES

EXPECTED RESULTS

- LOW-COST SOLAR ARRAY PANELS

SOLAR ARRAY DEPLOYMENT

OBJECTIVE

- CONCEPT PERMITTING REPETITIVE ARRAY DEPLOYMENT AND RETRACTION FOR MULTIPLE ARRAYS

TECHNOLOGY ASSESSMENT

- SEPS PROGRAM TO DEVELOP A SOLAR ARRAY DEPLOYMENT & RETRACTION MECHANISM FOR SINGLE ARRAY
- DEPLOYMENT & RETRACTION EXPMENT ON SHUTTLE FOR SEPS ARRAY CON-FIGURATION IN PLAN

APPROACH

- CONTINUE SEPS PROGRAM & CONDUCT ORBITER DEPLOYMENT EXPERIMENT
- DESIGN MULTIPLE BLANKET DEPLOY-MENT MECH. BASED ON SEPS AND QUALIFY BY SIMILARITY

EXPECTED RESULTS

- FLIGHT-QUALIFIED DEPLOYMENT MECHANISM

$Ni-H_2$ BATTERY

OBJECTIVE

- PROVIDE A LONG-LIFE LIGHTWEIGHT BATTERY

TECHNOLOGY ASSESSMENT

- AFAPL PROGRAM TO DEVELOP 50 AH $Ni-H_2$ CELL FOR LEO OPERATION
- CELL COSTS ARE HIGH (~$5000/CELL) AND PRODUCTION CAPACITY LIMITED
- PERFORMANCE & LIFE DATA ARE LIMITED
- NEEDS COMMITMENTS BY USERS TO GAIN MATURITY

APPROACH

- DEV. $Ni-H_2$ CELL FOR UTILITY MODULE USE BASED ON AFAPL PROGRAM DESIGN
- DESIGN, FAB, & TEST $Ni-H_2$ BATTERY
- INCREASE PRODUCTION CAPACITY (POSSIBLY SECOND SOURCE)

EXPECTED RESULTS

- QUALIFIED $Ni-H_2$ BATTER FOR SPACE PROGRAM USE

ELECTRICAL POWER SUBSYSTEM TECHNOLOGY PROGRAM PLANNING

(This chart is self explanatory)

ELECTRICAL POWER SUBSYSTEM TECHNOLOGY PROGRAM PLANNING

	1979	1980	1981	1982	1983	1984	1985	1986
			PHASE C/D GO-AHEAD			PLATFORM LAUNCH		
<u>SOLAR CELLS</u>								
SILICON AUTOMATED PROCESS DEVELOPMENT AUTOMATED EQUIP. DEVELOPMENT PILOT RUNS		$ 500 K	500	500				
GaAs MARKET DEVELOPMENT GALLIUM RECOVERY CELL DESIGN/DEVELOPMENT PILOT RUNS ORBITAL FLIGHT TEST		DEPENDENT ON TOTAL NATIONAL REQMTS = $ X$\bar{M}$						
<u>SOLAR ARRAY</u> PANEL DESIGN PRODUCTION EQUIP. DEVELOPMENT PILOT RUN TEST AND EVALUATION		$ 250 K	250	250				
<u>SOLAR ARRAY DEPLOYMENT</u> DEPLOYMENT MECH DESIGN & FAB TEST AND EVALUATION		$ 200 K	250	300				

ELECTRICAL POWER SUBSYSTEM TECHNOLOGY PROGRAM PLANNING
(Continued)

(This chart is self explanatory)

ELECTRICAL POWER SUBSYSTEM TECHNOLOGY PROGRAM PLANNING (CONT.)

	1979	1980	1981	1982	1983	1984	1985	1986
			PHASE C/D GO-AHEAD			PLATFORM LAUNCHED		
NiH_2 BATTERY CELL DESIGN AND DEVELOPMENT BATTERY DESIGN AND FAB. TEST AND EVALUATION PRODUCTION CAPACITY DEV.					$ 250 K	250	250	250

(AVAILABLE FOR FIRST BATTERY REPLACEMENT IN ORBIT, 1988-1989)

4.2 THERMAL CONTROL SUBSYSTEM

Thermal control of the platform/utilities module involves collecting the waste heat from the power conversion equipment, the batteries, the utility module and platform subsystems, and the payloads. The collected heat is then directed through a radiator that is mounted in an appropriate location to provide a view of space to reject its heat. The next several charts will discuss the solar array temperature (the prime item obstructing the radiator's view of space), radiator sizing, radiator arrangement alternatives, heat pipe radiator concept, payload interface alternatives, subsystem schematics, and the required technology development program.

Thermal Control Subsystem Requirements

The symmetric thermal control concept requires that the utilities module thermal control system, including the radiator, have the capability to reject all the heat generated by the utilities module and the payloads. This results in a larger radiator area on the utilities module than would be required if each payload were responsible for rejecting at least a portion of its heat load, but it does avoid the integration problem that could be involved with each payload having its own radiator. This is, however, a subject that should be investigated in any follow-on effort since a significant reduction in the centralized radiator area would be highly beneficial, particularly to the attitude control system.

A thermal control interface with the orbiter has been assumed as a requirement and is so reflected on subsequent charts. As the program and subsystem requirements become more clearly defined, however, this requirement may go away.

THERMAL CONTROL SUBSYSTEM REQUIREMENTS

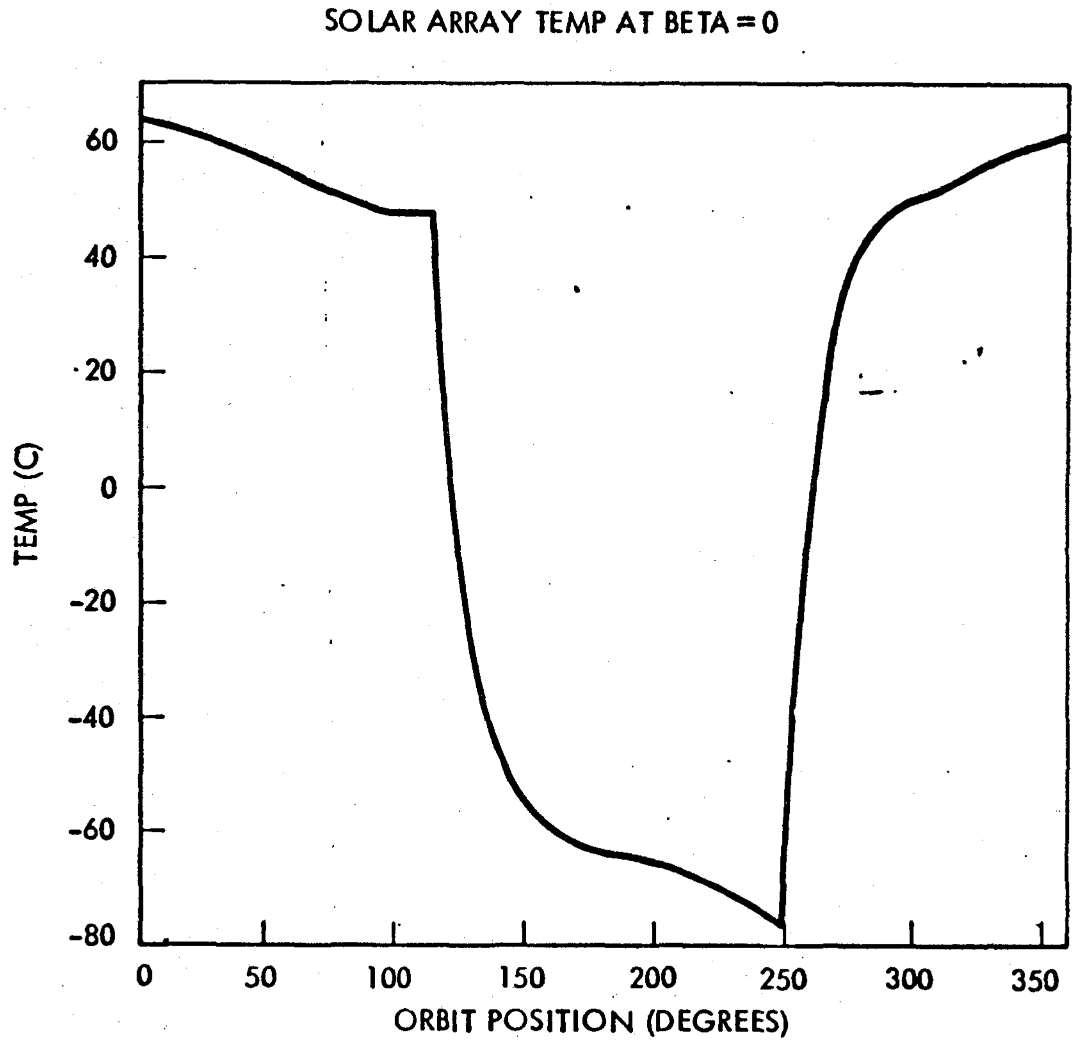
- SYMMETRIC THERMAL CONTROL CONCEPT
  (REJECT ALL POWER GENERATED BY THE ELECTRICAL POWER SUBSYSTEM)
- ORBIT PARAMETERS
  - 400 KM, 28.5° INCLINATION—PLATFORM 1*
  - 575 KM, 90° INCLINATION—PLATFORM 2
  - 400 KM, 57° INCLINATION—PLATFORM 3
- INTERFACE WITH ORBITER AND PLATFORM/PAYLOADS
- 10-YEAR LIFE WITH MAINTENANCE
- ACCOMMODATE GROWTH

*Baseline System

Solar Array Temperature Transient

The solar array temperatures were calculated for a solar array with the same thermal properties of a SEPS solar array. These calculations were made to include the effects of the solar cell efficiency and the cell packing fraction. Including the cell efficiency reduces the resultant maximum temperature. The solar array temperature at zero degree beta varied from 63.3C at the zero orbit (noon) position to -75C at the termination of the eclipse at about the 250 degree position. This represents the solar array temperature extremes. At higher beta angles the solar array presents less view area to pick up earth emission, thus will have a slightly lower peak temperature.

SOLAR ARRAY TEMPERATURE TRANSIENT



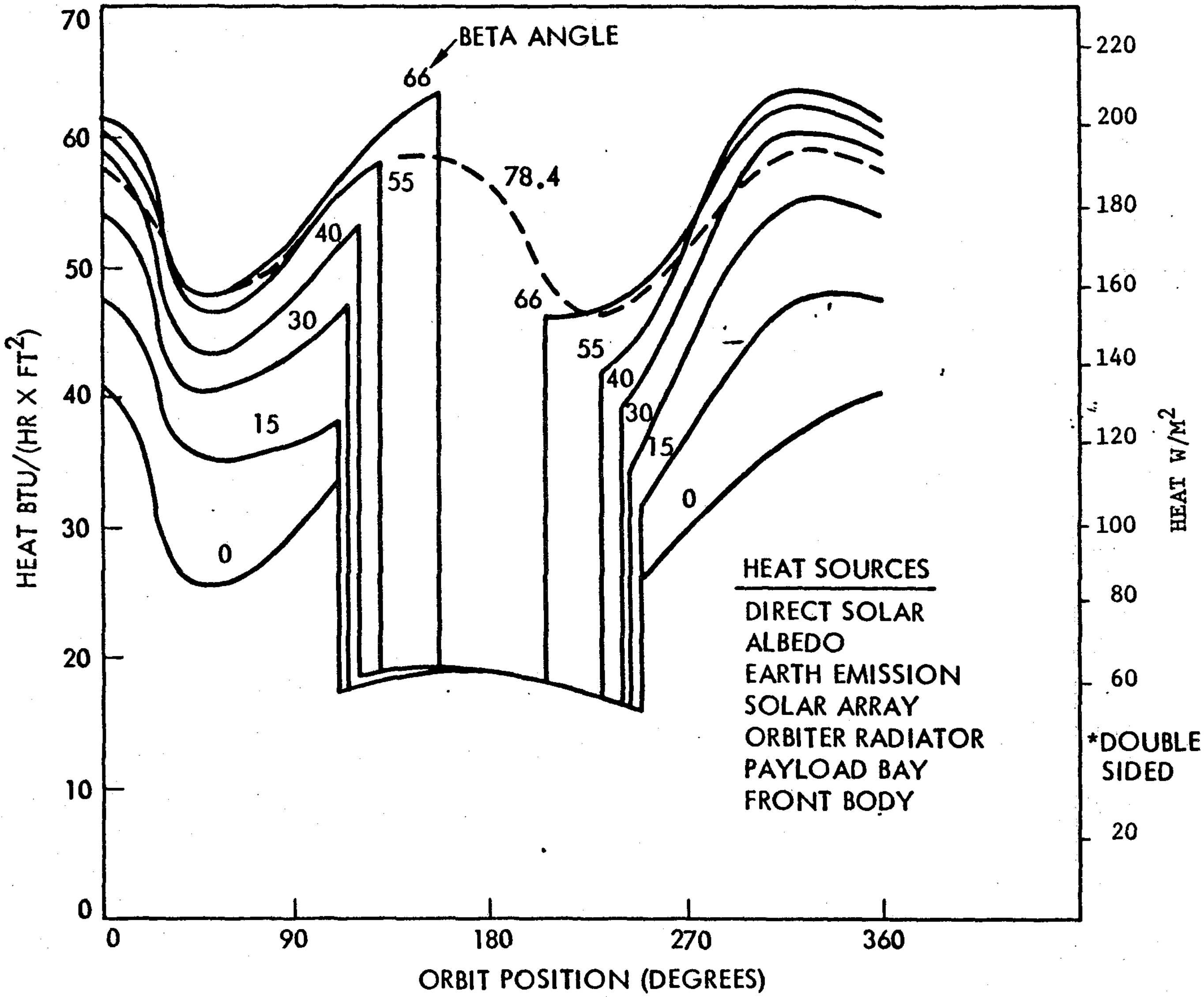
Heat Absorbed By Orbital Service Module Radiator

This chart is from an in-house power module study and shows the environmental heat absorbed by a radiator under similar conditions as would be imposed on the utilities module radiator. The environmental heat loads on the double sided radiator included direct solar, earth albedo, earth emission, the solar array, the orbiter radiator, the orbiter payload bay and the orbiter front body. These calculations were made for every 45 degrees in orbit position for beta plane angles of 0, 15, 30, 40, 55, 66, and 78.4 degrees. The maximum heat absorbed was an average of both sides of 63.4 BTU/(hr x ft^2) (199.9 W/m^2).

The data shown was calculated for a vehicle with a solar array having one full degree of freedom plus 40 degrees of NOD capability. The two degrees of freedom solar array of the utilities module should be slightly less severe. It also assumes the worst case with the orbiter attached. The use of this data will result in a slightly conservative radiator size but this should not affect the results of this study program.

HEAT ABSORBED BY ORBITAL SERVICE MODULE RADIATOR*

- (+Z) LV
- (X) VEL VECT
- DOCKED ORBITER
- $\epsilon = .76$
- $a = .11$

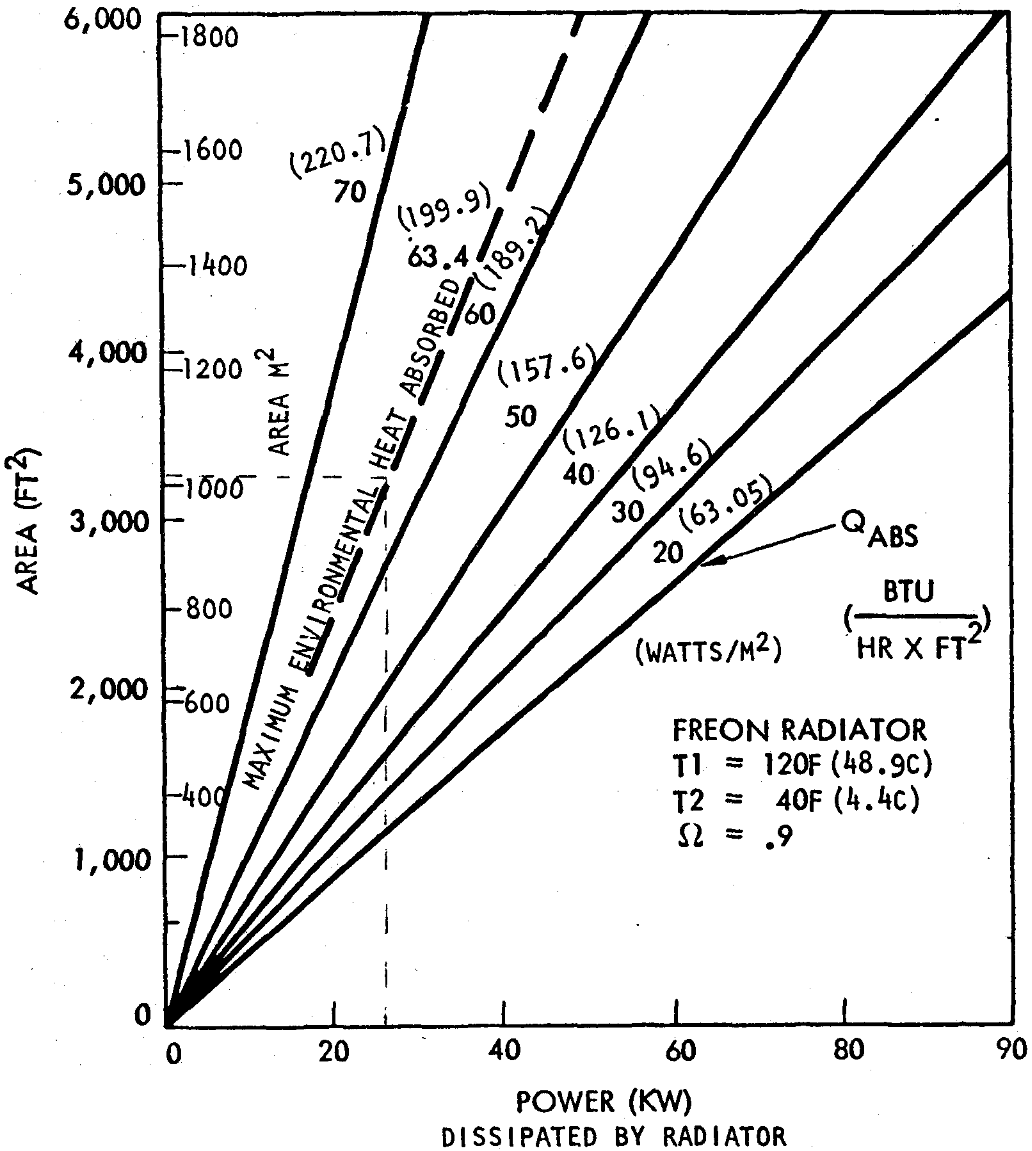


*DOUBLE SIDED

Radiator Area Required With T2 = 40F (4.4°C)

A Freon-21 fluid enters the radiator at a high temperature and drops in temperature as it rejects heat to its surroundings. The net heat loss is a function of the radiator fin temperature and the environmental heat absorbed. Conditions of power symmetry and a useful power of 20 KW available plus the power conversion loads totaling 26.8 KW to be dissipated by the radiator, resulted in a double sided radiating area of 3,246 ft^2 (302 m^2) for Freon 21 fluid operating between 48.9°C (120F) inlet to 4.4°C (40F) outlet.

RADIATOR AREA REQUIRED WITH T2 = 40 F (4.4^{0}C)



Radiator Arrangements With Freon Cooling Loop

A typical Freon 21 fluid cooling loop is shown in the figure with three different arrangements of direction of flow through the radiator. The fluid loop contains a circulating pump and reservoir for circulating the fluid through the radiator which cools the Freon and then through the batteries, interface heat exchanger and power conversion equipment. The radiator by-pass value controls the mixed radiator outlet temperature to 4°C. The simplest radiator is a double sided radiator where the heat absorbed by the radiator averages out the radiation absorbed by both sides of the radiator consisting of high and low values. This requires 302 m^2 of radiating area for the 20 KW utilities module.

A second variation of the radiator is to divide the radiator into two sides with insulation in between. This separates the radiator into one side with high radiation absorbed and a second side with low radiation absorbed. Each radiator side Freon outlet has a thermal control value which shuts off when the temperature exceeds 4°C. Each side is capable of handling the full load. Here the radiating area required is 307 m^2.

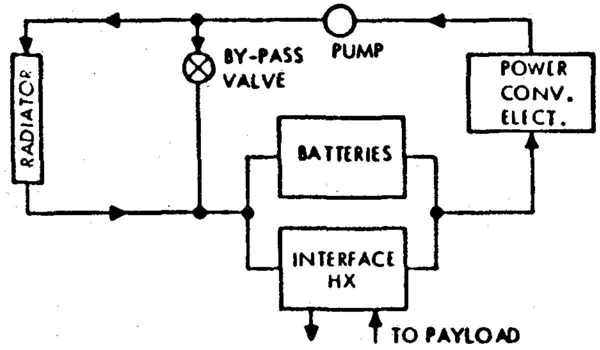
A third arrangement is the same as the second with insulation between the hot and cold sides of the radiator but using a directional control valve. This control valve has radiant flux sensors on each side of the radiator that measure the incident radiation on each surface and adjusts the control valve so the Freon flows through the hot side first and the cold side second. This takes advantage of being able to reject some heat through the hot side radiator because of the 49°C inlet temperature. A lower inlet temperature to the cold side radiator results. For this condition, a total radiator surface of only 218 m^2 is required.

Another option (not shown) is the use of a mechanical refrigeration system to extend the capability of the radiator to reject-heat in a high heat environment. The radiator size is reduced for this option (167 m^2) but the subsystem weight increases substantially as well as the electrical power required.

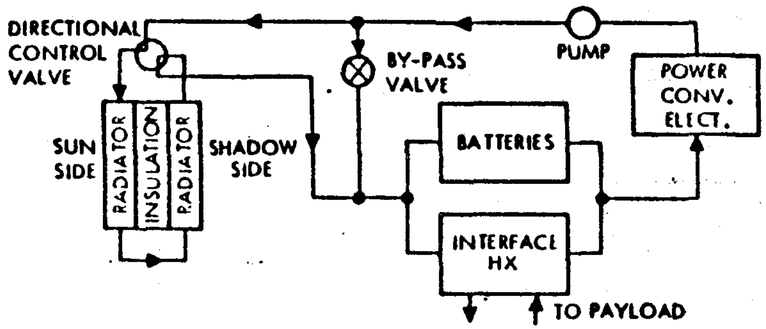
The final comparison is then between the double sided radiator with 302 m^2 and the insulated radiator with series flow with hot side first with an area of 218 m^2. The reduction in radiator is at the cost of a more complicated systems with an additional flow control valve.

RADIATOR ARRANGEMENTS WITH FREON COOLING LOOP

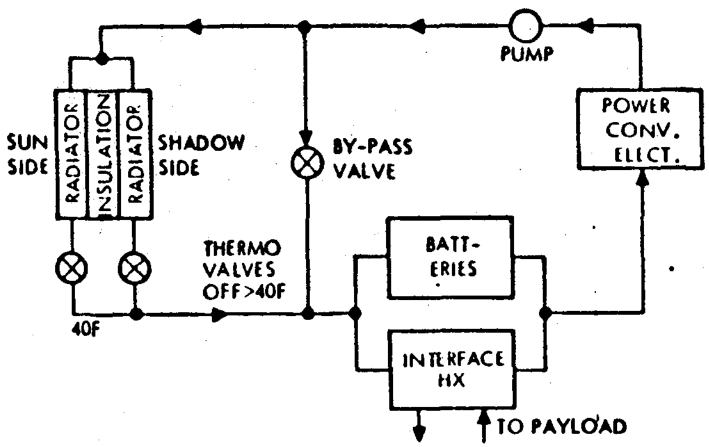
● DOUBLE SIDED RADIATOR



● INSULATED RADIATOR
SERIES FLOW HOT SIDE FIRST



● INSULATED RADIATOR
SHUT OFF HOT SIDE



● WEIGHT COMPARISONS

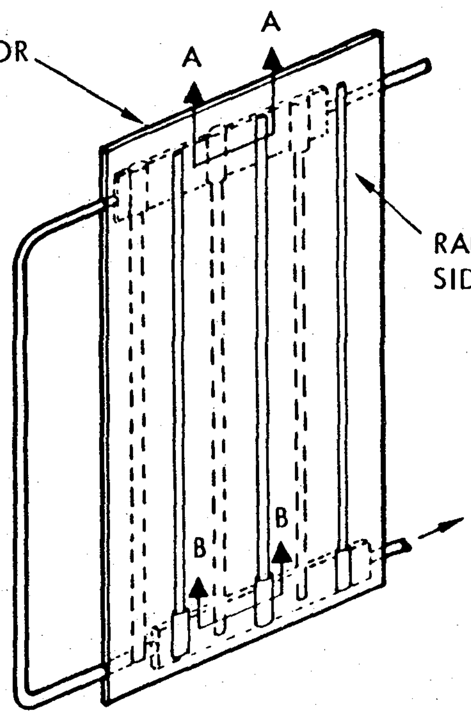
CONDITION α, ϵ η	AREA & WEIGHT	DOUBLE SIDED RADIATOR	INSULATED RADIATOR SIDES: SHUT OFF HOT SIDE	INSULATED RADIATOR SIDES: SERIES FLOW HOT SIDE FIRST	REFRIG-ERATOR & RADIATOR
NOMINAL $\alpha = .11$ $\epsilon = .76$	RADIATING AREA (m^2)	302	307	218	167
$\eta < 400$ NM	HEAT SINK WEIGHT (kg)	1068	1425	1012	1709

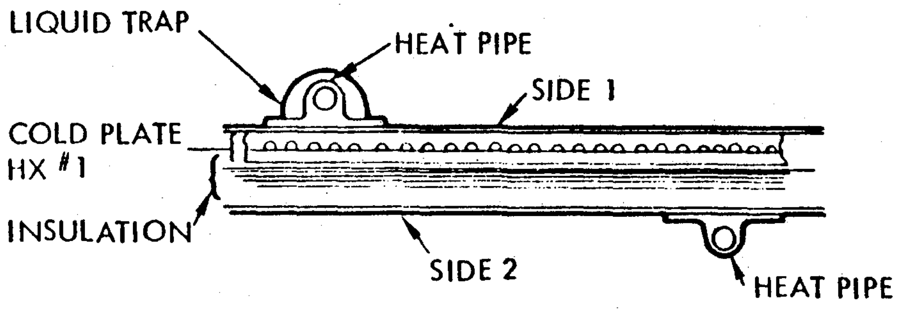
Hybrid - Heat Pipe Radiator Concept

A hybrid heat pipe radiator is an alternative to the conventional fluid tube radiator (e.g., orbiter radiator). In the hybrid heat pipe/fluid loop concept, the radiator tubes are replaced with heat pipes. Heat is transferred from fluid loop to the heat pipes via a cold plate. Each heat pipe operates independently such that a heat pipe may fail with little affect on overall radiator performance. This significantly reduces radiator sensitivity to micrometeoroid damage and allows a reduction in tube wall thickness (armor) for the same reliability from that required for an equivalent fluid tube radiator. The heat pipe radiator does however pay a slight area penalty due to the delta temperature across the cold plate. This delta temperature means that the radiator average radiating temperature must be slightly lower for the same radiator exit temperature.

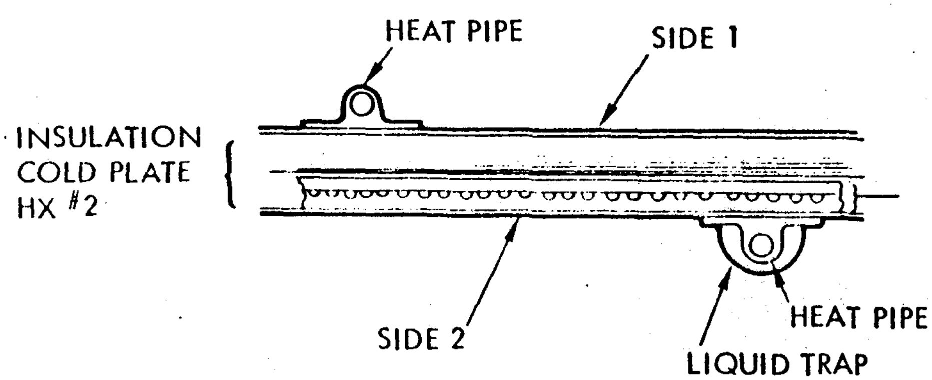
Ammonia is the choice for the heat pipe working fluid because of its high heat transport factor. It does freeze at a temperature of -78°C, but ground and flight test data indicate that ammonia heat pipes can be allowed to freeze and rethaw without damage. For 48.9°C inlet and 4.4°C outlet, with less than 20 cm heat pipe spacing (optimum spacing is less than 20 cm) and with ammonia as the working fluid, axially grooved wick (simple construction) heat pipes are adequate for panel lengths up to about 12.2 m. The maximum length considered for the utility module radiator panel is 9.14 m.

HYBRID - HEAT PIPE RADIATOR CONCEPT





A-A



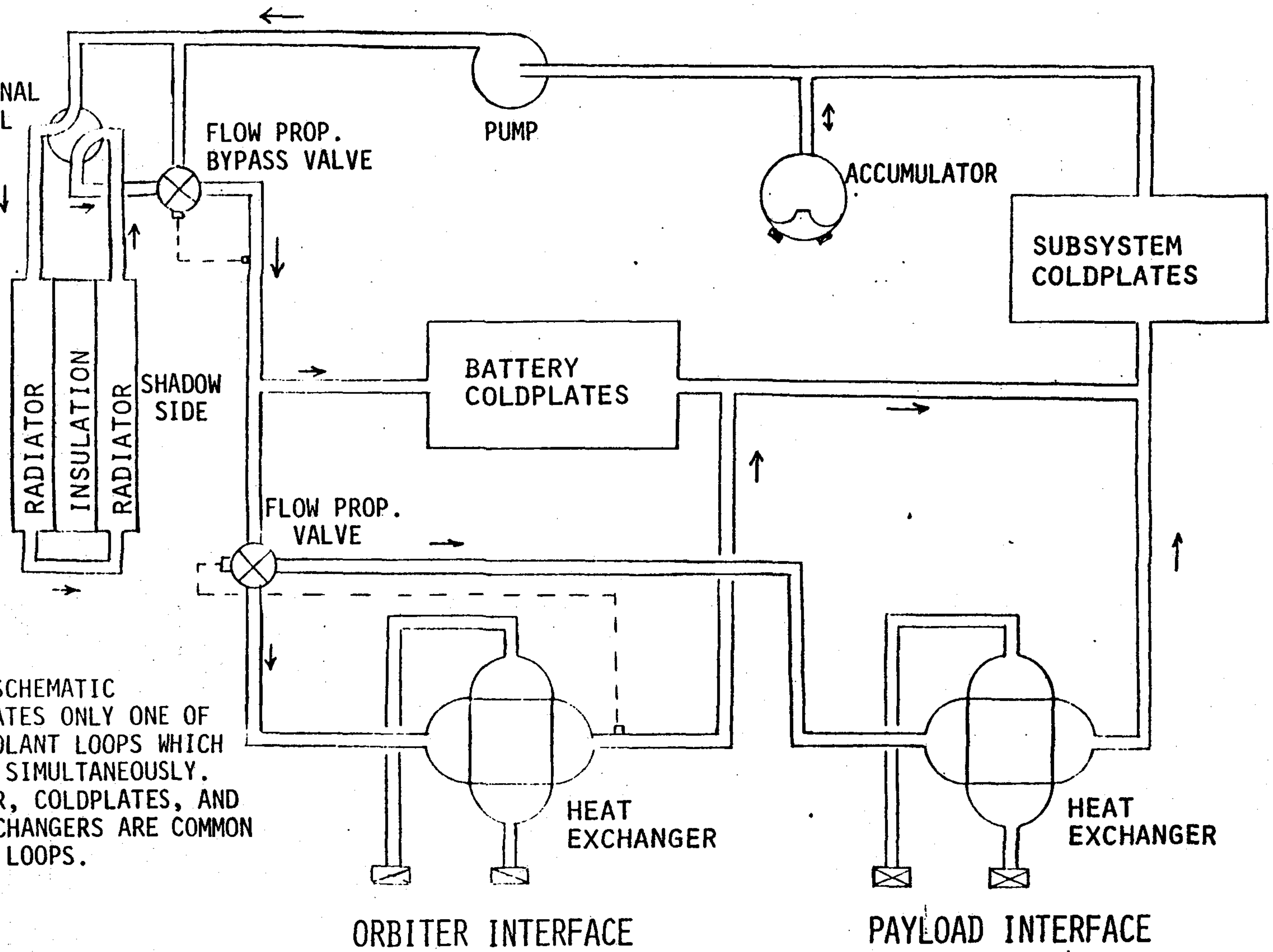
B-B

Thermal Control Subsystem

A typical Freon 21 fluid cooling loop is shown in this figure. The loop contains a circulating pump and reservoir for circulating the fluid through the radiator to cool the freon and then through the battery cold plates, interface heat exchangers and the subsystem cold plates. The radiator bypass valve controls the mixed radiator outlet temperature to 4°C.

The radiator is double sided with insulation between the sides. A directional control valve with sensors on each side of the radiator measures the incident radiation on each surface and adjusts the control valve so the freon flows through the hot side first and the cold side second. This results in about a 25% less radiator area requirement than a conventional double sided radiator.

THERMAL CONTROL SUBSYSTEM



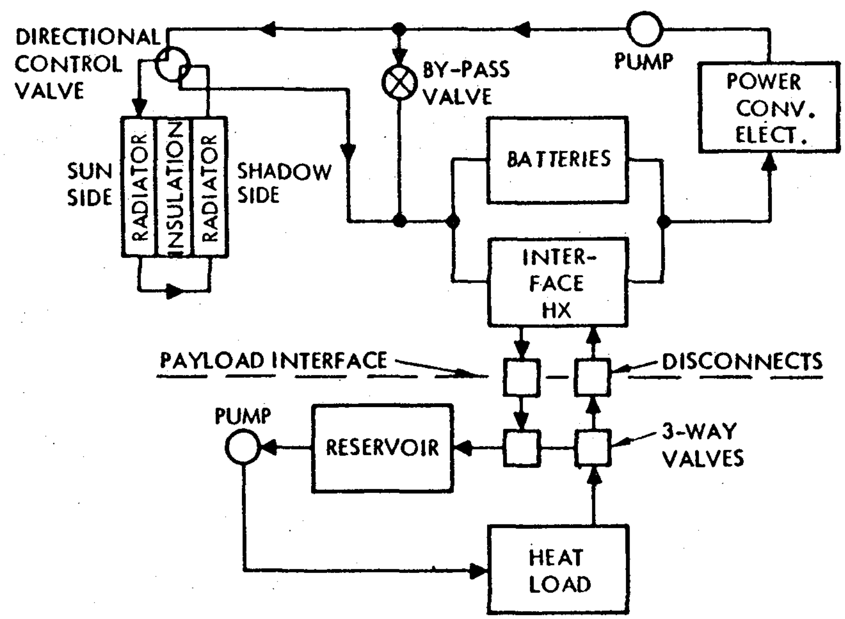
NOTE: SCHEMATIC ILLUSTRATES ONLY ONE OF DUAL COOLANT LOOPS WHICH OPERATE SIMULTANEOUSLY. RADIATOR, COLDPLATES, AND HEAT EXCHANGERS ARE COMMON TO BOTH LOOPS.

Payload Interfaces

The next two charts show three alternatives for interfacing the utilities module thermal control subsystem with the platform/payload. These charts are from in-house power module studies but most of the data is applicable for this program. Advantages and disadvantages are listed for each alternative and all three appear workable with no major problem areas. Use of an interface heat exchanger does maintain loop integrity and allows complete checkout of the utilities module loop on the ground.

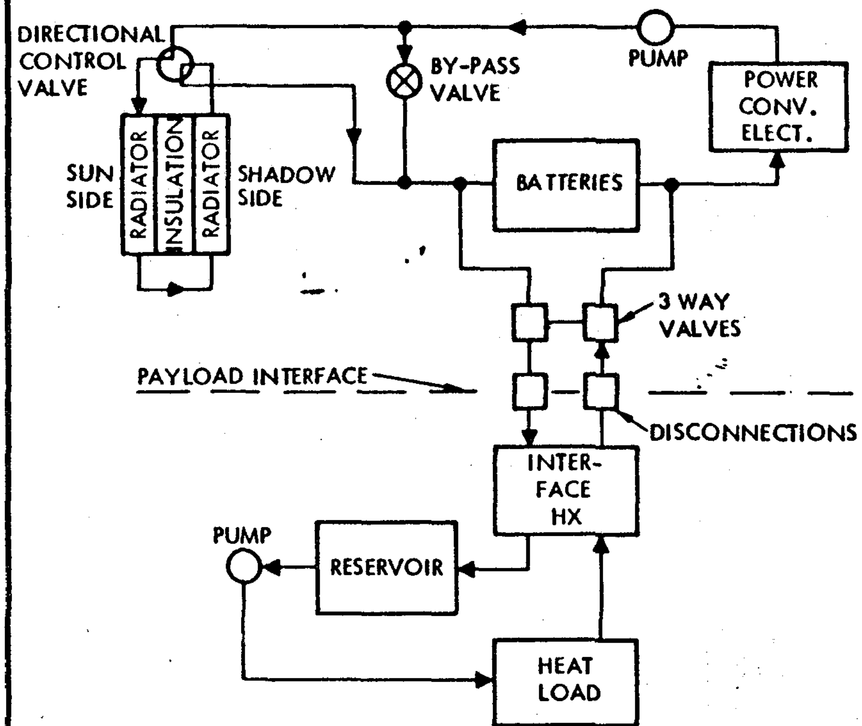
PAYLOAD INTERFACES

- INTERFACE HEAT EXCHANGER IN OSM



- PLUS
  - NO GAS ENTRAINMENT
  - TEST OSM ON GROUND
- MINUS
  - EVA OR REMOTE CONNECTION
  - ZERO LEAKAGE DISCONNECTS
  - TEST INTERFACE IN ORBIT
  - FLUID LOSS FROM PAYLOAD SYSTEM

- INTERFACE HEAT EXCHANGER IN PAYLOAD



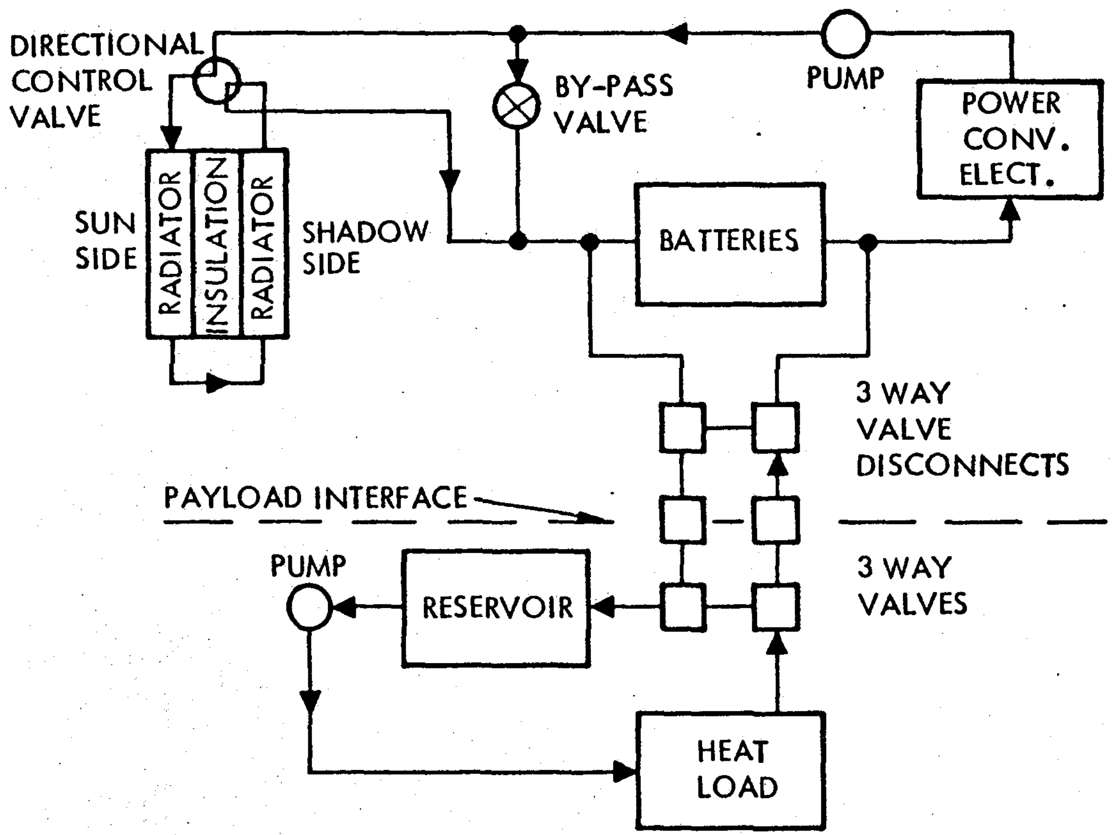
- PLUS
  - NO GAS ENTRAINMENT
  - TEST PAYLOAD ON GROUND
- MINUS
  - EVA OR REMOTE CONNECTION
  - ZERO LEAKAGE DISCONNECTS
  - TEST INTERFACE IN ORBIT
  - FLUID LOSS FROM OSM SYSTEM

PAYLOAD INTERFACE

See Previous Chart

PAYLOAD INTERFACE

● NO INTERFACE HEAT EXCHANGER



● PLUS
NO INTERFACE HEAT EXCHANGER
NO GAS ENTRAINMENT

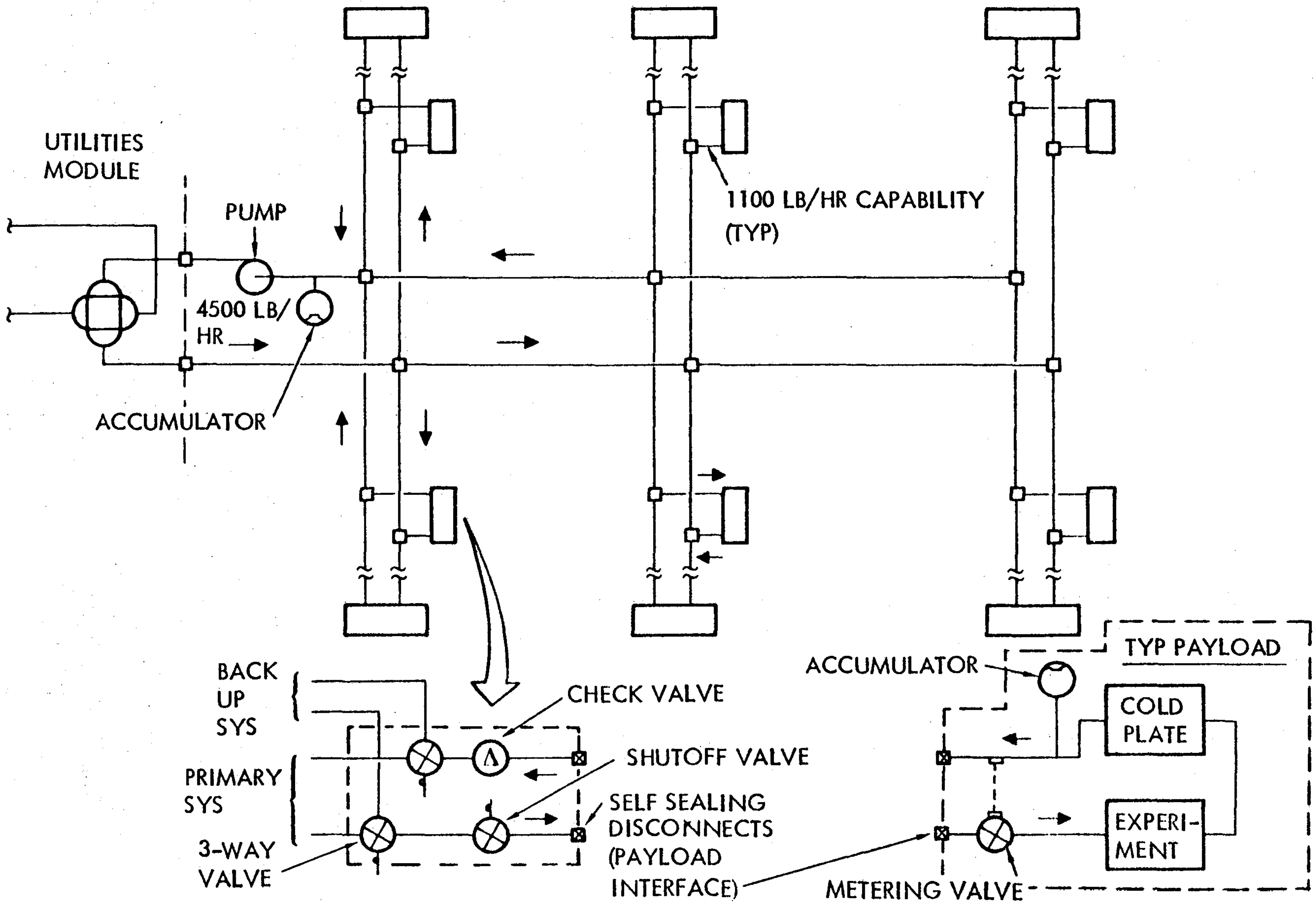
● MINUS
- EVA OR REMOTE CONNECTION
- ZERO LEAKAGE DISCONNECTS
- TEST INTERFACE IN ORBIT
- TEST COMPLETE SYSTEM IN ORBIT
- GREATER POSSIBLE FLUID LOSS FROM BOTH SYSTEMS

Thermal Control Subsystem - Platform Distribution

A concept for distribution of thermal control fluid on the platform is shown in the figure. It consists of two Freon 21 loops with non-redundant payload interfaces capable of being supplied from either loop. An accumulator is included in each loop and is pressurized to maintain loop pressure above vapor pressure under all operating conditions. The payloads are connected into the loop in parallel and each payload contains a metering valve to limit fluid flow to that required to maintain desired temperature.

(For details on fluid line installation on the payload platform and on fluid connectors see discussion under heading of Utilities Distribution Subsystem.)

THERMAL CONTROL SUBSYSTEM - PLATFORM DISTRIBUTION



Thermal Control Subsystem Components (Utilities Module)

This chart shows the estimated size and weight of the thermal control subsystem equipment located within or on the utility module. It does not include equipment located on the platform or within payload pallets.

THERMAL CONTROL SUBSYSTEM

ITEM	UNIT WT. (KG)	UNIT SIZE (CM)	QTY PER P.M.		
			6-ARRAY	4-ARRAY	2-ARRAY
RADIATOR PANEL	259	305x914	6	4	2
INTERFACE HEAT EXCHANGER	20	23x53x10	2	2	2
PUMP/ACCUMULATOR PKG	21	33x72x36	2	2	2
FLOW PROPORTIONING VALVE	3.6	18x23x10	2	2	2
COLDPLATES—BATTERY	$7.3/M^2$		$9.3\ M^2$	$6.2\ M^2$	$3.2\ M^2$
COLDPLATES—SUBSYST.	$7.3/M^2$		$13.9\ M^2$	$13.9\ M^2$	$13.9\ M^2$
PLUMBING, FLUID, DISCONNECTS, ETC.	68				

Thermal Control Subsystem Trades/Selections

Fluid loop radiatior technology is adequate for a normal Ø B/C/D development program, but would require heavy armor to survive the micrometeoroid environment for a ten year life. A heat pipe radiator has an inherently greater resistance to micrometeoroid damage since loss of a few heat pipes would have very little effect on radiator performance compared to the loss of a fluid tube. A trade study conducted for a similar application has thus concluded that a hybrid heat pipe radiator is lighter than a fluid radiator, even for a 0.9 reliability, and that substantially greater reliability can be obtained with very little weight penalty.

Silver teflon thermal coating which has quite acceptable thermal properties is used on the orbiter radiator. There are some indications, however, that these properties degrade (primarily increased absorptivity) from radiation exposure with time on orbit. This degradation, while not severe, would require a larger radiator area to compensate for degraded performance or replacing the thermal coating (or radiator) periodically (unlikely choice). An alternative is developing and selecting improved thermal coating materials.

Orbiter technology should be adequate for most of the fluid loop equipment with the freon pump being the possible exception. The orbiter freon pump has a design life of 40,000 hours (4.5 years) and would need at least one and possibly two replacements over the ten year design life of the utilities module/ platform.

THERMAL CONTROL SUBSYSTEM TRADES/SELECTIONS

COMPONENT	ALTERNATES	SELECTION/RATIONALE	TECH. LEVEL*
RADIATOR			
• TYPE	HEAT PIPE	—	-
	FLUID LOOP	MARGINAL RELIABILITY	3
	HYBRID HEAT PIPE/ FLUID LOOP	LONG LIFE/HIGH RELIABILITY	2
• CONFIGURATION	DOUBLE-SIDED	—	-
	INSULATED, SHUT-OFF HOT SIDE	—	-
	INSULATED; SERIES FLOW THRU HOT SIDE FIRST	LOWEST RADIATOR AREA	3
• THERMAL COATING	SILVER TEFLON (ORBITER DER.)	MARGINAL LIFE EXPECTANCY	3
	IMPROVED RADIATION-RESISTANT MATERIALS	LONG LIFE	2
FLUID LOOP			
• FREON PUMP	ORBITER FREON PUMP	MARGINAL LIFE EXPECTANCY	3
	IMPROVED-LIFE PUMP	REDUCED MAINTENANCE	2
• HEAT EXCHANGER	ORBITER DERIVED	—	3
• MISCELLANEOUS (VALVES, FILTERS, COLD-PLATES, ACCUM-ULATORS, ETC.)	ORBITER DERIVED	—	3

*Technology levels: 1, enabling; 2, enhancing; 3, normal ØB/C/D development

THERMAL CONTROL SUBSYSTEM TECHNOLOGY CANDIDATES

(This chart is self explanatory)

THERMAL CONTROL SUBSYSTEM TECHNOLOGY CANDIDATES

RADIATOR

OBJECTIVE

- DEVELOP A HIGH-RELIABILITY, LONG-LIFE HYBRID HEAT PIPE RADIATOR

TECHNOLOGY ASSESSMENT

- NASA-JSC PROGRAM FOR HEAT PIPE RADIATOR
- AFAPL HEAT PIPE DEVELOPMENT PROGRAM

APPROACH

- PRELIMINARY DESIGN AND HEAT PIPE TESTS
- PROTOTYPE DESIGN, FAB, & TEST
- THERMAL VACUUM TEST

EXPECTED RESULTS

- HYBRID HEAT PIPE RADIATOR DESIGN WITH IMPROVED RELIABILITY

THERMAL COATING

OBJECTIVE

- PROVIDE LONG-LIFE THERMAL COATINGS

TECHNOLOGY ASSESSMENT

- POPULAR THERMAL COATINGS TEND TO LOSE OPTIMUM THERMAL PROPERTIES WITH TIME ON ORBIT

APPROACH

- MATERIALS RESEARCH
- AGE TEST CANDIDATE MATERIALS

EXPECTED RESULTS

- SELECTION OF OPTIMUM THERMAL COATING FOR UTILITY MODULE APPLICATION
  - BETTER THERMAL PROPERTIES
  - LONGER LIFE

FREON PUMP

OBJECTIVE

- DEVELOP LONG-LIFE PUMP

TECHNOLOGY ASSESSMENT

- ORBITER PUMPS—40,000 HOURS (4.56 YR)

APPROACH

- PRELIMINARY DESIGN & LAB TEST
- PROTOTYPE DESIGN, FABRICATION
- QUAL & LIFE TESTS

EXPECTED RESULTS

- A LONG-LIFE FREON PUMP

THERMAL CONTROL SUBSYSTEM TECHNOLOGY PROGRAM PLANNING

(This chart is self explanatory)

THERMAL CONTROL SUBSYSTEM TECHNOLOGY PROGRAM PLANNING

	1979	1980	1981	1982	1983	1984	1985	1986
			PHASE C/D START			PLATFORM LAUNCH		
RADIATOR • PRELIM DESIGN AND HEAT PIPE TESTS • PROTOTYPE DESIGN, FAB, AND TEST • THERMAL VACUUM TEST		$ 200 K	250	350	150			
THERMAL COATINGS • MATERIAL RESEARCH • MATERIAL TESTING		$ 100 K	100	100	100	75		
FREON PUMP • PRELIM DESIGN & LAB TESTS • PROTOTYPE DESIGN AND FAB • TEST			$ 100 K	150	200			

4.3 COMMUNICATIONS, COMMAND, AND CONTROL SUBSYSTEM

The communications, command, and control (sometimes referred to as communication and data handling) subsystem consists of those elements that provide for collecting, processing, and transmission of data (housekeeping and payload) to ground; on board data storage; receipt and distribution of ground commands; and receipt, processing and distribution of Global positioning satellite (GPS) time and position data. Subsequent charts will discuss the subsystem requirements, subsystem arrangement, major equipment items and alternatives, and the required technology development program.

Communications, Command, Control Subsystem Requirements

Requirements for the subsystem were based on an analysis of the experiments selected for each platform. Continuous monitoring of all housekeeping and command functions are required. There data and the experiment data are to be transmitted in real time or stored, when the real time link is unavailable, for later transmittal. This established the necessary data rates and storage requirements. Uplink command rates of up to 32 K BPS was established with a downlink of variable capacity from 5 K BPS to 15 M BPS.

The experiments selected for the platform, require that the platform system have the capability to know its location in the sky to 6 meters at 3 sigma. Also, precision experiment timing is required via an atomic clock interface with stability of 10^{-14}. Data can be stored on board, transmitted in real time as well as continuous monitoring of all housekeeping and command functions.

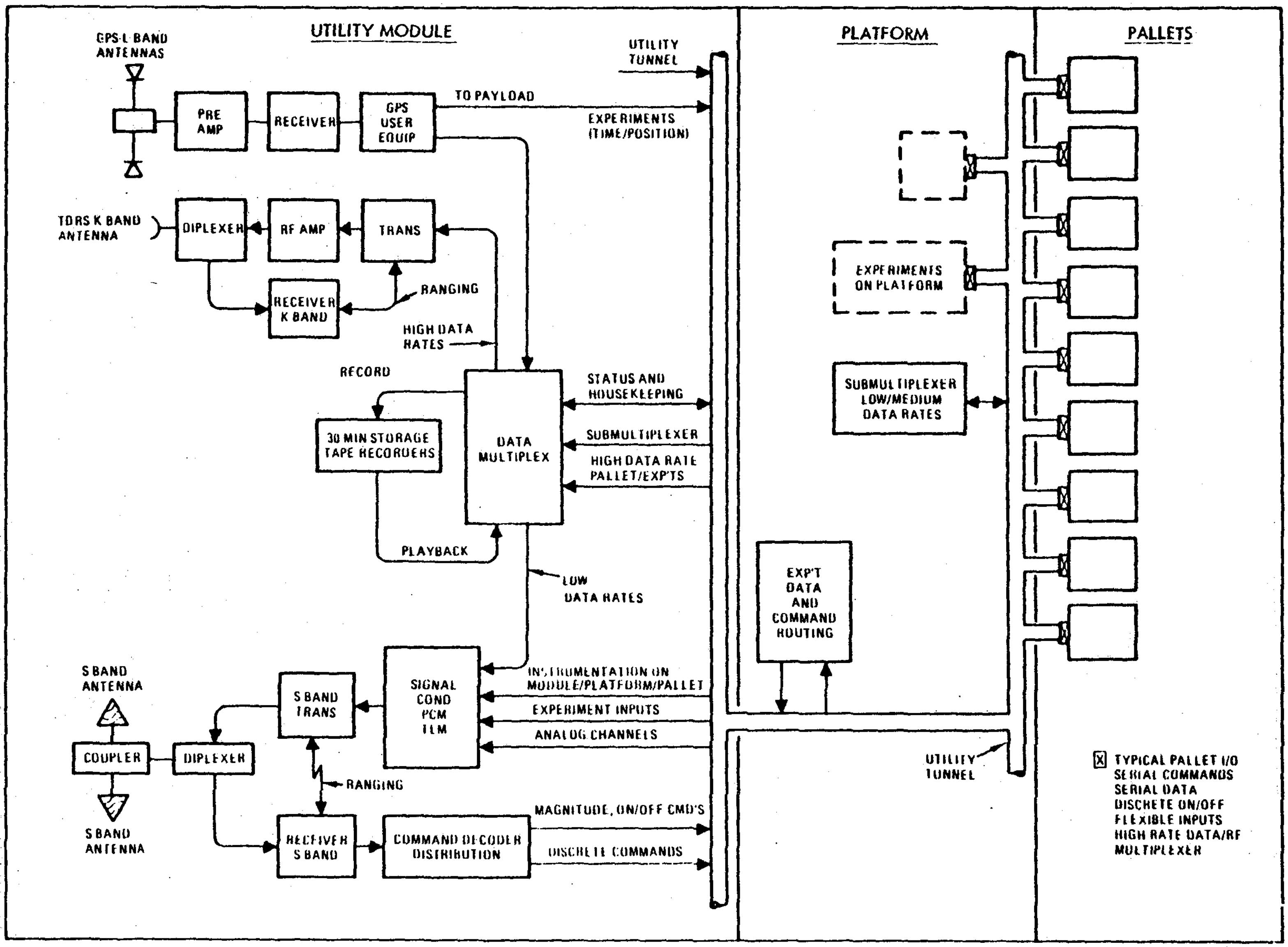
COMMUNICATIONS, COMMAND, CONTROL SUBSYSTEM REQUIREMENTS

- CONTINUOUS TRANSMISSION OF HOUSEKEEPING/STATUS AND EXPERIMENT DATA TO NASA/USER.
- PROVIDE EXPERIMENTS WITH ACCURATE LOCATION OF THE ERECTABLE PLATFORM WITH FREQUENT UPDATES TO PERMIT ACCURATE POINTING AND KNOWLEDGABLE VIEWING OF EARTH/SPACE BY THE EXPERIMENTS.
- PROVIDE ATOMIC CLOCK REFERENCE STANDARD FOR PRECISION TIMING OF EXPERIMENT EVENTS.
- OPERATE CONTINUOUSLY AND STORE DATA ONBOARD FOR LATER TRANSMISSION TO USER WHEN FIELD OF VIEW OF TDRS OR STDN IS UNAVAILABLE.
- COMMAND, CONTROL, OPERATION, STATUSING OF EXPERIMENTS VIA STANDARDIZED I/O INTERFACES.

Communications, Command, Control Subsystem Block Diagram

The block diagram shows the location (utility module or platform or pallet) of the major components of the subsystem and their interconnections. Service is furnished by the Global Positioning Satellite system for position and time accuracy. Interface is also provided to Tracking and Data Relay Satellite for high data rate transmission and via S-band for command control and low data rates such as housekeeping status, etc. Note the small multiplexer provided at each pallet/experiment interface to reduce the number of wires needed between the experiment and the transmission system on the Utility Module.

COMMUNICATIONS, COMMAND, CONTROL SUBSYSTEM



Utility Module - CC&C Subsystem

This chart summarizes the equipment, that is located on the Utility Module and its functions. The subsystem provides the capability to monitor and maintain the Utility Module via the Command link as well as the experiments on the platform. Tape recording is also provided to ensure records are available of all the data.

MAJOR EQUIPMENT ALL LOCATED ON THE "UTILITY MODULE"

- THREE ANTENNA/SUBSYSTEMS
  (1) K-BAND - TRANSMIT/RECEIVE TO TDRS SATELLITES, WITH 5 METER DIAMETER DISH ANTENNA.
  (2) S-BAND - TRANSMIT/RECEIVE TO STDN WITH MULTIPLE ANTENNAS
  (3) L-BAND - RECEIVE "USERS SET", TO GPS SATELLITES WITH SMALL OMNI ANTENNA.
- THE S AND K BAND SYSTEMS ARE CROSS-CONNECTED FOR DATA (COMMAND/ CONTROL USAGE.
- ONBOARD DATA RECORDING/PLAYBACK FOR CONTINUOUS REAL TIME EXPERIMENTS.
- COMMAND/CONTROL/DATA COLLECTION TO/FROM EACH EXPERIMENT/PALLET/ PLATFORM LOCATION.
- CC&C EQPT (INCLUDES ABOVE ITEMS) PCM/ENCODER, SIGNAL CONDITIONING, MUX/DEMUX, COMMAND/CONTROL LOGIC, COMMAND DECODER/DISTRIBUTION/ TIMER, AND DATA STORAGE.
- UTILIZES STANDARD SPACE-RATED SUBSYSTEMS COMPONENTS - MINIMUM OF DELTA QUAL NEEDED, LOW RISK DESIGN

Communications, Command, Control Subsystem Trades/Selections

A study was made of the various subsystem hardware that is space rated today and will be available shortly. All the subsystem devices with the exception of automated GPS user equipment are current state of the art and present no major technology problems for usage on the Erectable Platforms.

COMMUNICATIONS, COMMAND, CONTROL SUBSYSTEM
TRADES/SELECTIONS

COMPONENT	ALTERNATES	SELECTION	TECH LEVEL
ANTENNA SUBSYSTEMS K, S, AND L BAND	-DIFFERENT USER FREQUENCIES	CURRENT S.O.A., SPACE-QUALIFIED AND IN USE BY NASA, USAF S/C	3
PLATFORM VELOCITY/POSITION/TIME	-STDN EPHEMERIS UPDATES -GPS UPDATES	GPS/SHUTTLE ORBITER EQUIPMENT, AUTOMATED FOR SPACECRAFT USAGE. MORE ACCURATE DATA TO EXPERIMENTS.	2
EXPERIMENT OUTPUT DATA STORAGE	-BUFFER STORAGE -TAPE RECORDERS -DIRECT TRANSMISSION	TAPE RECORDERS CURRENT S.O.A. SPACE QUALIFIED.	3
SIGNAL CONDITIONING AND MULTIPLEXING	SPACE QUALIFIED HARDWARE	NO ALTERNATES, 3 SOURCES (S.O.A.) FOR HARDWARE SPACE QUALIFIED.	3
COMMAND/DECODER/ ENCODER/DISTRIBUTION	SPACE QUALIFIED HARDWARE	NO ALTERNATES, 4 SOURCES (S.O.A.) FOR SPACE QUALIFIED HARDWARE	3
TRANSMITTER/RECEIVER	SPACE QUALIFIED HARDWARE	NO ALTERNATES, 3 SOURCES (S.O.A.) FOR SPACE QUALIFIED HARDWARE CURRENTLY FLYING IN SPACE ON SEVERAL PROGRAMS.	3

Technology Candidates—Communications, Command, Control Subsystem

Man-operated GPS user equipment is being space rated for use on the Shuttle Orbiter. Also there are satellite programs for the military that are currently funded to prepare automated GPS user equipment for use on free-flying spacecraft. Hardware and technology from these programs should be available in the early 1980's as a base to develop the user equipment for this program.

GPS USER EQUIPMENT -

OBJECTIVE:	PROVIDE ACCURATE POSITION, VELOCITY, AND TIMING REFERENCE FOR THE ERECTABLE SPACE PLATFORM EXPERIMENTS.
TECHNOLOGY ASSESSMENT:	NOT AVAILABLE FOR SPACECRAFT UTILIZATION TODAY. NOT SPACE-QUALIFIED, REQUIRES MAN IN THE LOOP.
APPROACH:	FURTHER SPACE QUALIFY AND AUTOMATE THE LRU DESIGN OF THE SYSTEM SCHEDULED FOR THE SHUTTLE ORBITER FOR REMOTE AUTOMATED OUTPUTS.
EXPECTED RESULTS:	PROVIDE PAYLOAD EXPERIMENTS WITH 6-10 M EPHEMERIS LOCATION, LESS THAN A FOOT/SECOND VELOCITY ACCURACY, AND AN ATOMIC CLOCK FOR BETTER THAN 10^{-9} TIMING STABILITY REFERENCE.

Technology Candidates—Communications, Command, Control Subsystem

The current NASA TDRS System will need to be expanded so that several high data rate spacecraft can be serviced at one time. Also, an analysis of the usage priorities appears to be necessary in order that continuous data may be transmitted from the orbiting platform as well as the Shuttle Orbiter when it is spaceborne.

IMPROVED TDRS USAGE -

OBJECTIVE:	PROVIDE CONTINUOUS DATA TRANSMISSION LINK FROM ERECTABLE SPACE PLATFORM HIGH DATA RATE EXPERIMENTS TO NASA/USER.
TECHNOLOGY ASSESSMENT:	THE SINGLE HIGH DATA RATE ANTENNA ON TDRS IS REQUIRED TO TRACK SHUTTLE/ORBITER WHEN IT IS IN SPACE. NO HIGH DATA RATE CAPABILITY EXISTS FOR OTHER USERS.
APPROACH:	PERFORM ANALYSIS OF CURRENT/FUTURE TDRS SINGLE ACCESS AND MULTI ACCESS OPERATIONAL PROCEDURES FOR HIGH DATA RATE USERS. REVISE TDRS PROCEDURES.
EXPECTED RESULTS:	EXPANDED HIGH DATA RATE COMMUNICATION LINK USAGE OF TDRS BY THE ERECTABLE SPACE PLATFORM MOUNTED EXPERIMENTS AND OTHER NASA FREE FLYERS WITH REDUCTION IN SHUTTLE/ ORBITER PRIORITIES.

Communications, Command, and Control Subsystem Technology Program Planning

The budget planning chart shows in 1979 dollars the anticipated costs for further reducing the technical risks of using GPS and TDRS satellite subsystems for the Erectable Platform as part of the proposed subsystem.

COMMUNICATIONS, COMMAND, AND CONTROL SUBSYSTEM TECHNOLOGY PROGRAM PLANNING

	1979	1980	1981	1982	1983	1984	1985	1936
			PHASE C/D GO-AHEAD			PLATFORM LAUNCH		
GPS USER EQUIPMENT		50K	150K	200K	150K			
DESIGN FOR REMOTE/AUTOMATED OPERATION								
FABRICATE AND VERIFY								
PERFORM "SPACE" QUAL TESTS								
ORBITAL FLIGHT TESTS								
IMPROVED TDRS USAGE		50K	50K					
PERFORM OPERATIONS ANALYSIS								
REVISE PROCEDURES								

4.4 STRUCTURES SUBSYSTEM

The erectable space platform has three distinct structural elements: (1) utilities module, (2) solar array and mast, and (3) the platform's erectable structure. The utilities module has been discussed in depth by several NASA contracts and studies (References 6, 7, and 8) and Rockwell International independent research study (Reference 5). Information from these references was considered in the design formulation of the basic structural core of the utilities module. The structural concepts are state-of-the-art technologies and will not influence the selection of the structural elements of the platform.

Electrical power for the utilities module will be supplied by highly flexible arrays employing SEPS-type technology. A comprehensive analysis contained in Reference 9 was used as the background data for the erectable platform's solar array. Reference 9 indicated a first bending frequency for the array's deployable mast of approximately 0.04 Hz and the array blanket first frequency of 0.02 Hz. The remainder of the structural system (utilities core module and payload platform) are at least an order of magnitude stiffer and should not interact significantly (first order) with the dynamics of the array and mast.

The principal concern for this study was the selection of structural elements and spatial arrangements of the platform structure. In-house information relating to strut and union design from References 3 and 5 was used extensively in developing concepts for the platform structure.

The basic functions of the platform structure are as follows. (1) Provide a structural framework with (or to) which the mission/payload elements can be accurately installed and oriented. (2) The platform shall be capable of withstanding the worst combination of gravity-gradient and aerodynamic loads while being assembled in low-earth orbit. (3) The partially assembled platform shall withstand assembly/erection loads. (4) Provide a framework for attitude and stabilization control reference. (5) React forces and moments associated with attitude and control actuation, as well as propulsive forces for orbit transfer and stationkeeping. (6) Minimize the distortions arising from adverse thermal gradients during orbital operations.

Platform Structure Configuration Evaluation Criteria

The evaluation of each configuration was based on the performance and design environment requirements previously defined and the criteria listed on the facing chart.

The most important criterion was ease of assembly which was measured in terms of the number of primary structural joints required. Although not measurable, the implied access to joints by an orbiter-based remote manipulator system (RMS) and the avoidance of excessive maneuvering were also significant.

The next two criteria were also of major importance. Suitability of the configurations to a wide range of missions and payload requirements was obviously a desired feature as well as application to wide variations in installing, removing, and positioning of the payload/pallets. The configuration should also require the least number and types of structural elements and operations.

The next two criteria (minimum mass and high stiffness) were not too significant with respect to selecting the structural members. Since the mass of the platform struts are less than one percent of the platform/utility module/payload system, the strut design should be based on ease of manufacture, assembly, etc., with weight as a minor criterion. The total number of struts required for the platform does not impose concerns on their packageability within the orbiter's cargo bay. All of the platforms considered have stifness considerably higher than the flexible solar array mast and should not present dynamic interaction problems.

The last two criteria (thermal and control) were important as they impact the platform's pointing accuracy. Effects of these criteria are discussed under material selection in this section and in Section 4.6, Attitude Stabilization and Control Subsystem.

PLATFORM STRUCTURE CONFIGURATION EVALUATION CRITERIA

- EASE OF ASSEMBLY
  - Number of primary structural joints
  - Accessibility of joints and attachments
- COMPATIBILITY WITH POTENTIAL MISSION APPLICATIONS
  - Adaptable to sizing up/down
  - Accommodation of various types of mission equipment
  - Accommodation of attaching subsystem structure (e.g., utility distribution)
  - Potential for various attitude orientations
- REQUIRES MINIMUM VARIETY OF STANDARD STRUCTURAL ELEMENTS AND ASSEMBLY OPERATIONS
- MINIMUM MASS FEATURES
- HIGH STIFFNESS FEATURES
- MINIMIZES THERMAL PROBLEMS
- MINIMIZES CONTROL PROBLEMS

Platform Structure and Payload/Pallet Attachment

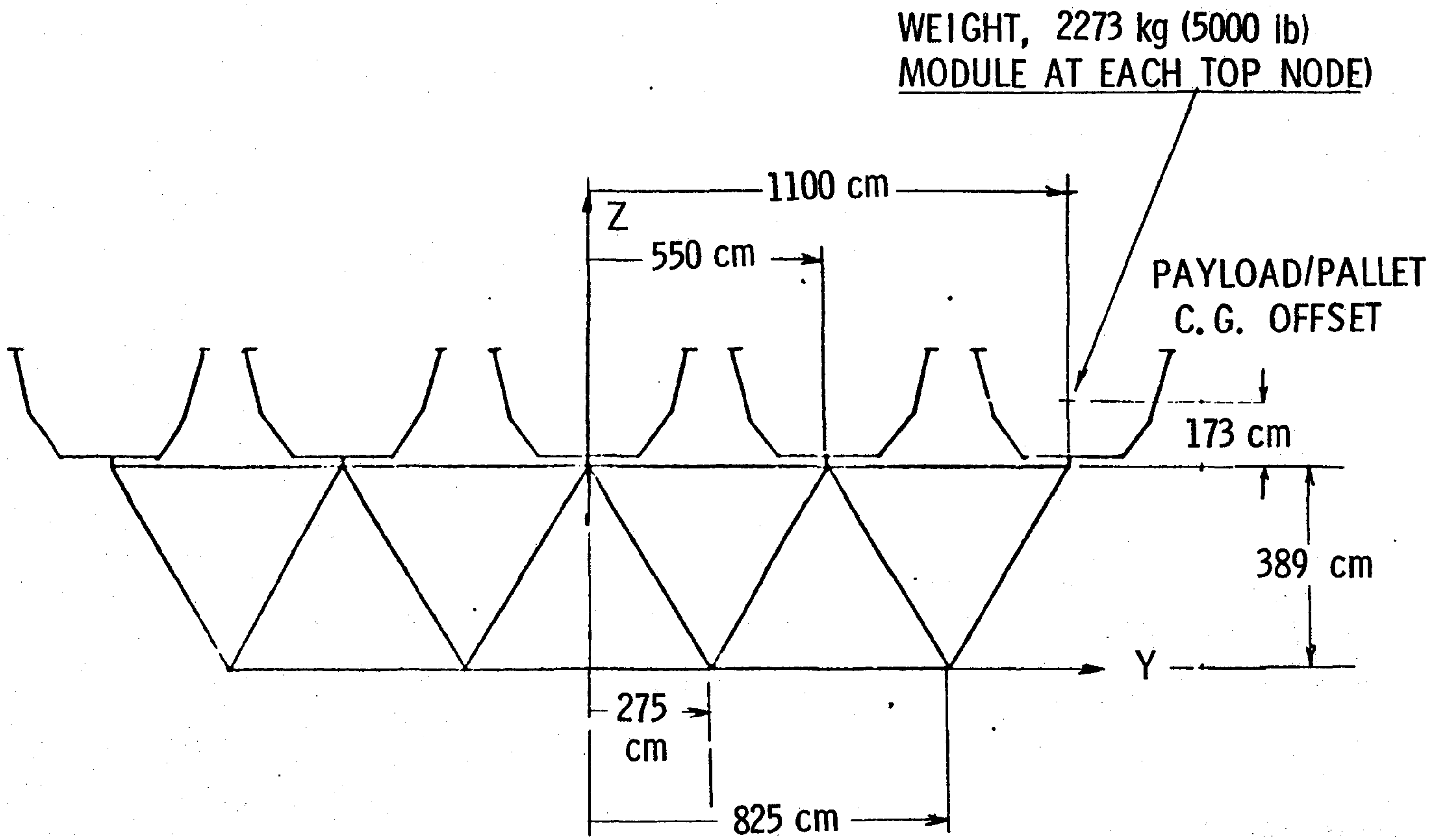
The payload/pallet arrangement in the platform suggested that the strut lengths between centerline of node attachment points should be 550 cm (216 in.). This distance will provide sufficient clearance between adjacent pallets to easily accomplish pallet installation and removal, and utilities connect and disconnect operations either remotely with the RMS or using EVA.

The most significant internal loads in the structural platform result from assembly operations, transfer of the assembled structure and payload from low earth orbit to higher orbits, and thermal gradients. It is recognized that the handling and packaging of the individual struts before assembly must be given consideration to ensure that these operations do not impose significant loads that will dictate the design of the structural elements. The movement of payload elements, attitude control reactions, etc., are forces that are localized in nature and will influence, perhaps, the design of secondary structure of the platforms and are peculiar to the mission equipment. Structural loads from other sources such as gravity gradient, solar pressure, and aerodynamic drag are considered negligible.

The structural design criteria for the struts were for a fully stressed design where the local buckling stresses and column buckling stress are equal to the applied comprehensive loading. A minimum tube wall thickness of 0.050 cm (0.020 in.) was selected due to handling and manufacturing considerations. The aluminum tube 10.2-cm (4-in.) diameter with a 0.050 cm (0.20 in.) wall thickness provided sufficient factor of safety to withstand the preliminary design loads and account for strut eccentricities due to manufacturing tolerances, thermal gradients, and off-sets of load application and center joints.

A finite model was built and analyzed which represented the platform structure composed of either aluminum or graphite epoxy struts and union. The strut/union joints were considered to be fully fixed. This is a reasonable assumption considering the low flexual rigidity of the long struts. At each node point there was a 2273-kg (5000-lb) payload and pallet to represent a fully loaded platform (worst case).

PLATFORM STRUCTURE AND PAYLOAD/PALLET ATTACHMENT



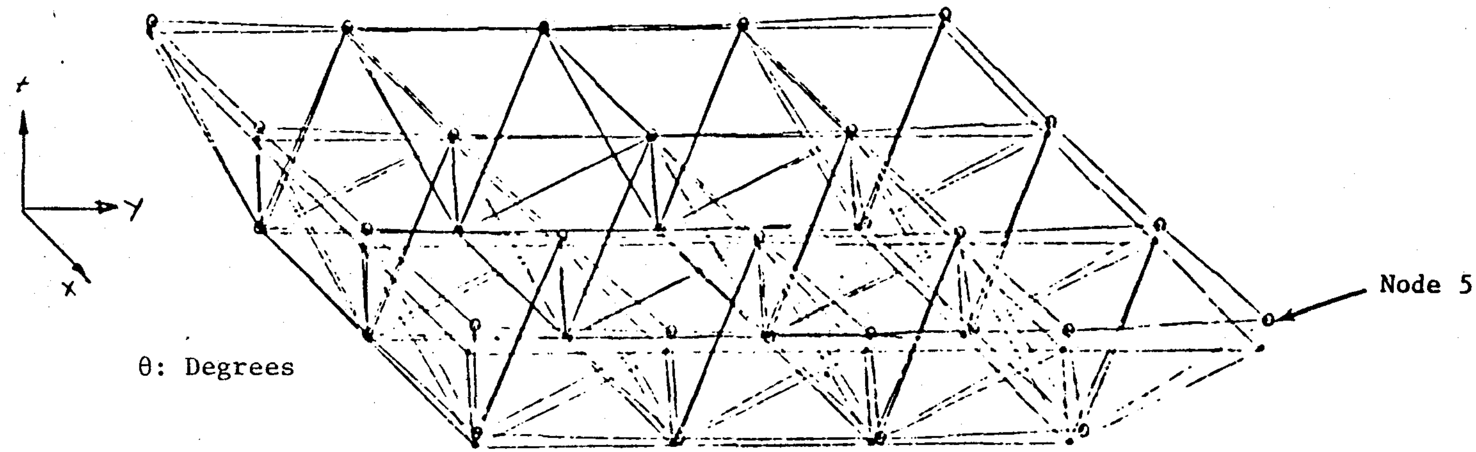
Deflection Shape of Area Platform

The finite element model was subjected to anticipated orbital thermal gradients and 0.01 g acceleration loads. Platform distortions must be minimal during continuous orbital operations. The thermal cycles will occur each orbit during payload operations, while the acceleration loads occur periodically during orbit transfer, etc., when the payloads are inoperative. Of interest are the maximum distortions of any node point/payload interface with respect to a reference system within the utilities core module.

The translation and rotation of the corner node (No. 5) with respect to the attachment points on the opposite side of the platform are shown in the accompanying chart. The rotational deflections for the aluminum structure due to thermal loading are greater than 0.1 degree which, by themselves, exceed the pointing accuracy requirements of 0.01 degree. These deflections can be reduced by nearly an order of magnitude when graphite epoxy struts are used for the platform.

Although the structure is stable with the fixed joints at the nodes, a second concept was considered using diagonal bracing struts. There was no significant change in the thermal or acceleration induced deflections. The addition of the diagonal struts significantly increased the platform's first natural frequency to about 1 Hz, thus removing the platform frequency response away from the flexible solar array mast at 0.04 Hz.

DEFLECTED SHAPE OF AREA PLATFORM



CASE NO.	SUBCASE DESCRIPTION			DEFLECTION—ROTATION			
				NODE NO.	θ_x	θ_y	θ_z
1	THERMAL ONLY	ALUM.	DIAGONALS	5	0.0890	-.1136	0.0000
2			NO DIAGONALS	5	0.0896	-.1142	0.0000
3		GR/EPOXY	DIAGONALS	5	0.0109	-.0139	0.0000
4			NO DIAGONALS	5	0.0110	-.0140	0.0000
1	THERMAL PLUS 0.01 g ACCELERATION	ALUM.	DIAGONALS	5	-.0179	-1.667	-.1123
2			NO DIAGONALS	5	-.0053	-1.666	-.0995
3		GR/EPOXY	DIAGONALS	5	-.0414	-0.420	-.0511
4			NO DIAGONALS	5	-.0348	-0.418	-.0455

Structural Material Selection

The selection of the material for the structural elements is influenced by the operating environment and mission performance criteria of the structural platform.

Aluminum, steel, titanium, and nonmetals, such as graphite polyimide and graphite epoxy, were considered as candidate materials. Graphite composites are expensive and not as versatile as aluminum, steel, and titanium from the standpoint of assembly and fabrication. Thermal stresses are high in steel, intermediate in titanium and aluminum, and low in graphite composites. Thermal distortions are high for aluminum, intermediate for steel and titanium, and low for graphite composites.

The relatively few struts required for the platform will not present a packaging and stowing problem within the cargo bay. Their length and diameter suggest they can be an efficient one-piece monolithic closed cross-section concept. Weight trades conducted in Reference 3 indicated potential weight reductions by using the NASA-Langley Research Center's folded nestable columns (dixie cups). While the dixie cups hold significant weight and packaging advantages for large platform structures (approximately 100 m), they present additional operational, deployment, and manufacturing complexities on this study's erectable platform which outweight their weight and packaging benefits.

For the selected strut concept 10.2 cm (4 in.) circular tubes, the manufacturing complexity is minimized both for metalic tubes and lay-up and wrapping of composite materials. The thermal distortion constraints are the predominant drivers toward the use of composite material for the platform structure.

STRUCTURAL MATERIAL SELECTION

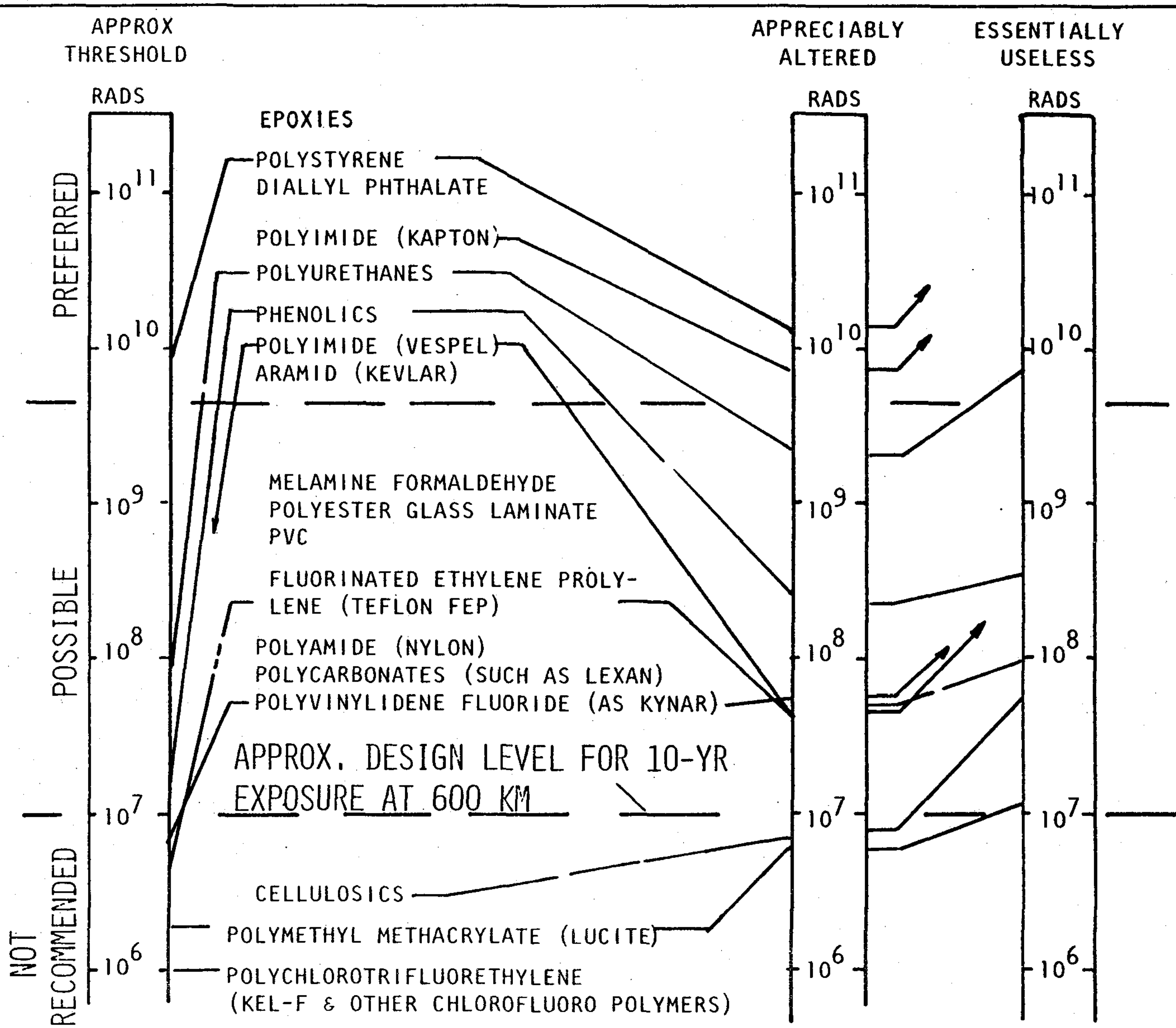
PROPERTY	GRAPHITE-EPOXY	ALUMINUM 6061-T6	STEEL 1020	TITANIUM 6A1-4V
STIFFNESS/MASS N·M/KG (LBF-IN/LBM)	67×10^6 (268×10^6)	25×10^6 (100×10^6	25×10^6 (100×10^6	25×10^6 (100×10^6)
THERMAL EXPANSION CM/CM/°K (IN./IN./°F)	1.8×10^{-6} (1×10^{-6})	23.4×10^{-6} (13×10^{-6})	11.7×10^{-6} (6.5×10^{-6})	9×10^{-6} (5×10^{-6})
HI-TEMP RESISTANCE % E AT 450°K (350°F)	0.93	0.92	0.95	0.91
RADIATION RESISTANCE	TBD	EXCELLENT	EXCELLENT	EXCELLENT
DIMENSIONAL ACCURACY AND STABILITY	EXCELLENT(?)	EXCELLENT	EXCELLENT	EXCELLENT
MANUFACTURING COMPLEXITY (1 = BEST)	4	2	1	3
FATIGUE RESISTANCE ENDURANCE LIMIT N/M^2 (PSI)	1.38×10^8 (20,000)	1.38×10^8 (20,000)	2.76×10^8 (40,000)	2.76×10^8 (40,000)

Relative Radiation Susceptibility of Plastics

A major problem in the application of composites is the lack of information on the long-term aging in the space environment. Polymeric materials, due to their covalent bond linkage are susceptible to degradation due to the photon (UV and X-ray) and particulate (electron, proton, and alpha) radiation in the outer space environment. This radiation is energenic enough to cause breakage of the polymer bonds resulting in embrittlement due to crosslinking of the polymeric chains and/or strength loss due to chain scission. At present, it is difficult to predict the long-term aging behavior of polymeric because of the complexity of the degradation mechanisms that occur under radiation exposure.

Actual test data on the individual materials are limited and generally of shorter duration than the ten-year exposure used in this study. A radiation level of 10^7 rad is chosen as representative for a ten-year exposure at a 600-km altitude. The chart shows this level compared to the radiation susceptibility of generic classes of polymers. As can be seen from the chart, the more cyclic polymers such as polyimides and epoxies appear to have the best potential as the matrix material for the composites used in this study. Some design allowances have to be made due to degradation of these materials, but any values must be verified with actual test data.

RELATIVE RADIATION SUSCEPTIBILITY OF PLASTICS



Current Strut and Joint Concepts

The efficient joining of structural elements and assemblies in space is a most critical operational requirement that will affect the time and energy needed for assembly, the structural mass, and rigidity of the platform. In developing a joining concept, it is essential, then, to emphasize operational simplicity coupled with positive engagement and minimum force to effect the joint. In addition, the joint must be capable of being effected without complex tools, and must be forgiving in terms of the angularity of the strut when introduced into the union. The joint could also include length adjustment features to compensate tolerance buildups in the construction. It is also desirable that verification of joint engagement be provided by visual or other positive means.

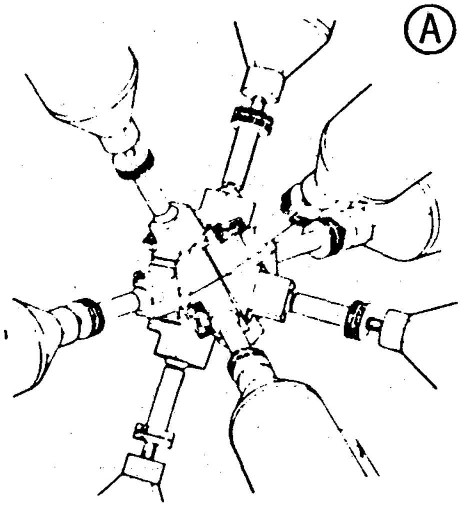
Joints can fall under one of two classifications—rigid joints and pinned joints. Rigid joints are the type that connect two relatively stationary objects and can be either of the multi-point attachment (e.g., a multi-bolted lap joint) or the single-point attachment (e.g., a bayonet connection). Pinned joints, on the other hand, allow relative angular motion between the connected elements. In their design criteria study for NASA/LaRC, Reference 10, Boeing showed that fixed joints could lead to undesirable thermal stresses and distortions in large metallic space structures. Larger thermal distortions with rigid joints could present adjustment problems during the assembly operations. Highly redundant structures with pinned joints, such as the diagonally braced platform, have thermal stresses and assembly mismatch problems.

Reference 3 assessed several strut/union concepts including the LaRC-Rockwell ball-socket swivel joint. Subsequent RMS simulation tests have indicated that this concept is a viable candidate (Ref. 11). Other experimental hardware concepts have been built; some are shown on the accompanying chart.

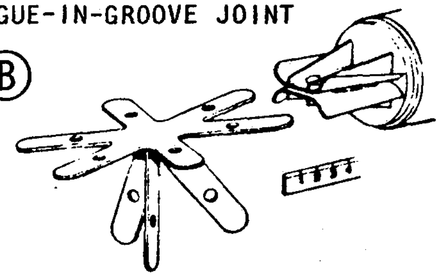
The prime criteria for any joint concept must be the ease of assembly, tolerances to directional and positional misalignment, and the resulting assembly has no noticeable backlash to cause non-linearities (deadbands) in the platform's behavior.

CURRENT JOINT CONCEPTS

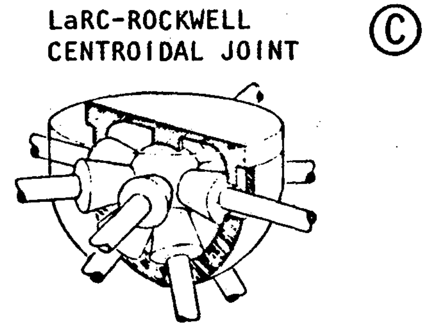
LaRC-ROCKWELL
BALL-SOCKET SWIVEL JOINT



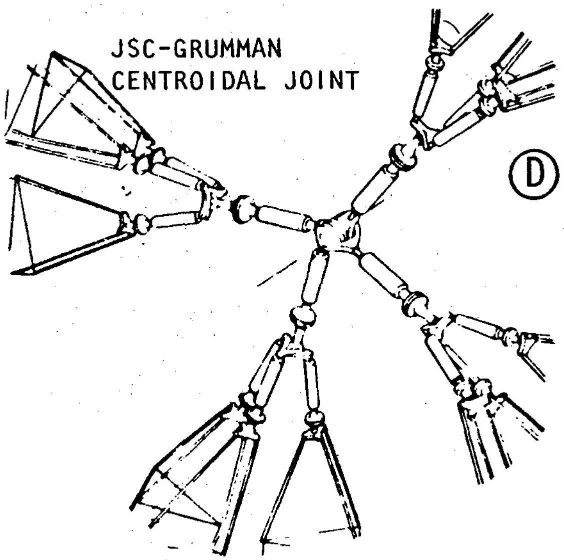
LaRC
TONGUE-IN-GROOVE JOINT



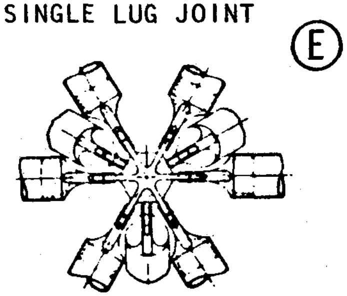
LaRC-ROCKWELL
CENTROIDAL JOINT



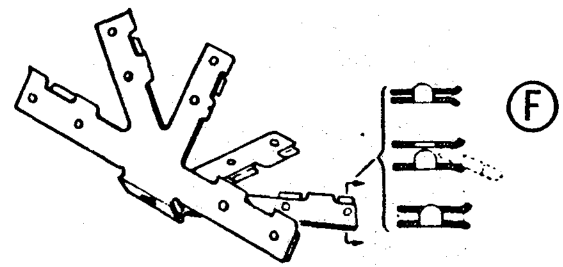
JSC-GRUMMAN
CENTROIDAL JOINT



JSC-BOEING
SINGLE LUG JOINT



GENERAL DYNAMICS
TONGUE-IN-GROOVE JOINT



Ball-Socket Swivel Joint Concept

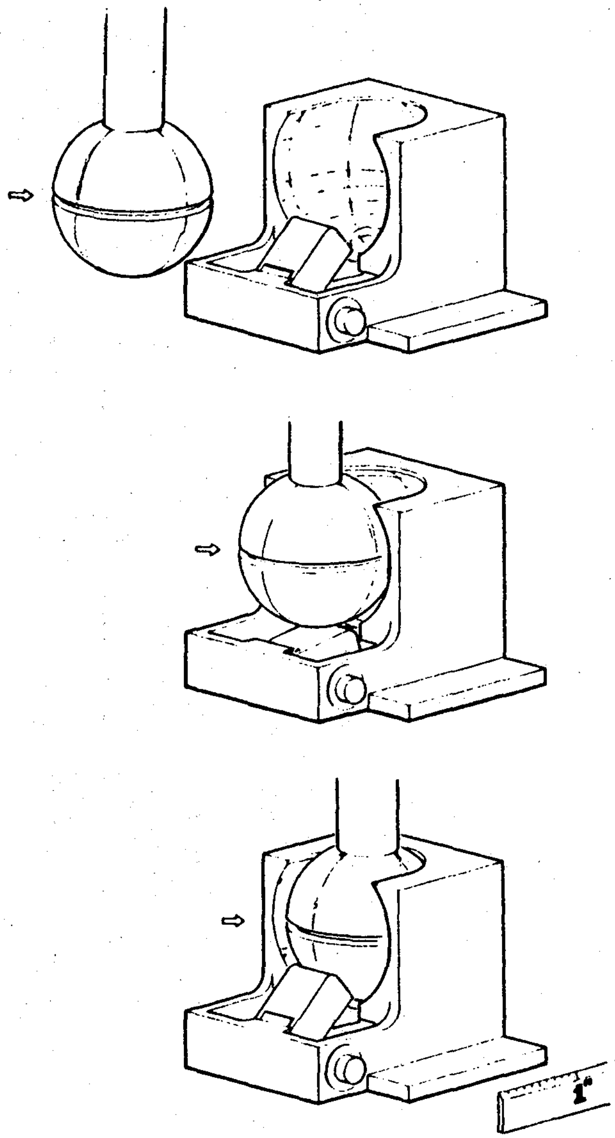
The ball and socket swivel joint originated at Satellite Systems Division, Rockwell International, under NASA-Langley Contract NAS1-14116. The ball and fitting is shown attached to the end of a dixie-cup strut. The union shown is a nine-faceted female fitting for a tetrahedral cell construction. The pentahedral nodal platform will need 10-sided unions for the diagonal braced surface and 8-sided unions for the other surface.

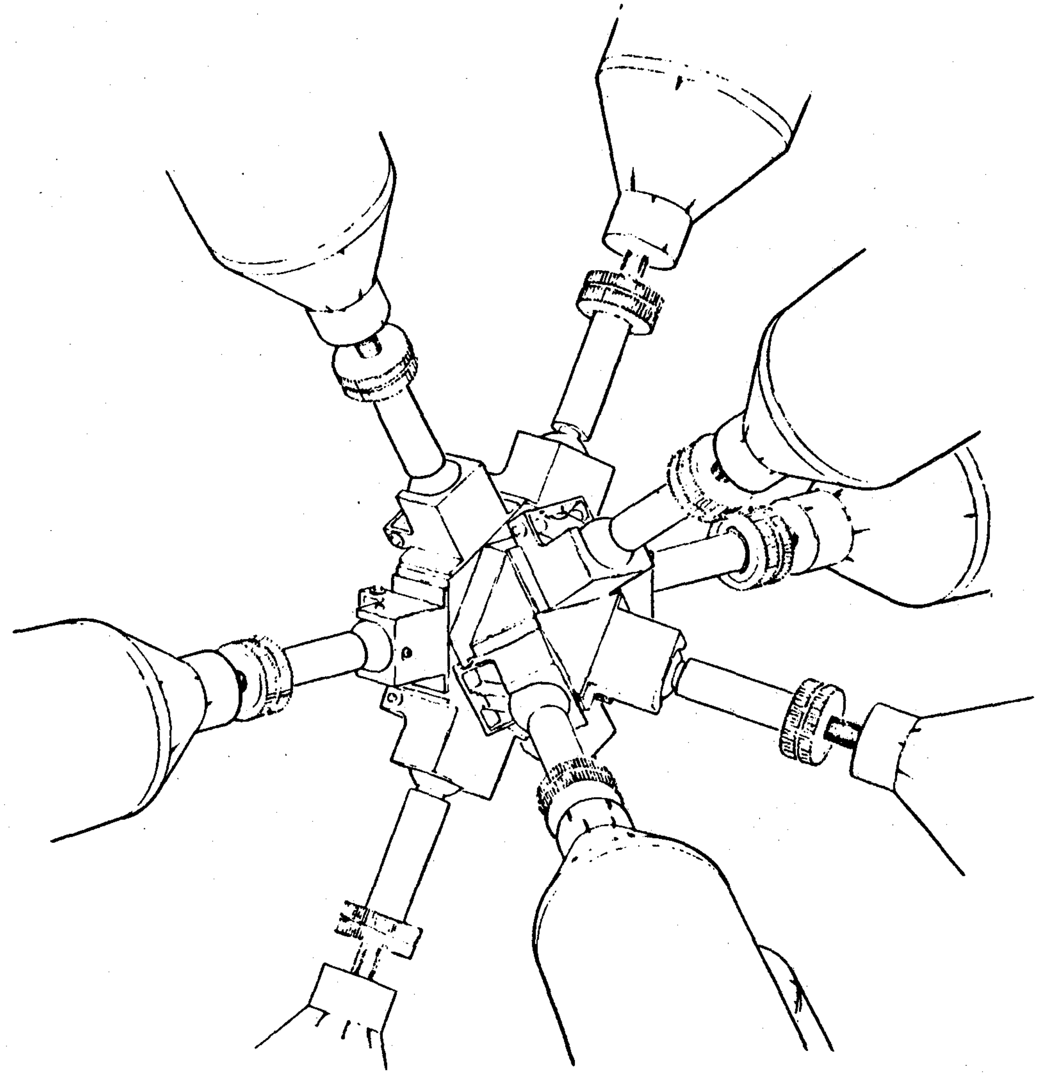
The chart clearly shows the engagement procedure of the strut fitting with the socket. The ball and spherical opening provides allowances for angular and positional mismatch during the engagement procedure. A trip latch stops the ball from disengagement after completion of the joining operation. This concept provides a pinned joint.

Around the ball there is a circular groove which engages with spring-loaded pins deployed with the socket. This will create a fully fixed joint concept.

Removal of the strut from the union can be accomplished by depressing the trip latch either mechanically or electrically.

BALL-SOCKET SWIVEL JOINT CONCEPT





REF. CONTRACT NAS1-14116

Structural Subsystem Technology Candidates

Due to their high strength-weight ratio and low coefficient of thermal expansion, certain filamentary composites are being seriously considered as candidates for large space structures.

The major weakness in application is the lack of information on long-term aging effects, caused by the service environment, on changes in these properties. To compensate for this lack of information, most of the design allowables on these materials are degraded in some arbitrary proportion to the time and severity of the anticipated service. This applies both in the materials for primary members and for joining terminal hardware to these members. In addition to the basic materials, continuing effort is needed to develop integrated structural concepts and hardware taking such factors as thermal distortion, tolerances, and mechanical/electrical interfaces into account.

STRUCTURAL SUBSYSTEM TECHNOLOGY CANDIDATES

STRUCTURAL MATERIALS AND PROCESSING

OBJECTIVE

- DETERMINE ABILITY OF GRAPHITE/EPOXY TO WITHSTAND RADIATION EXPOSURE.

TECHNOLOGY ASSESSMENT

- LACK OF LONG-TERM AGING DATA PRECLUDES ESTABLISHMENT OF DESIGN ALLOWABLES

APPROACH

- ACCELERATED TESTING IN SIMULATED SPACE ENVIRONMENT
- CHARACTERIZE AGING EFFECTS ON CHEMICAL STRUCTURE TO ENABLE FAILURE ANALYSIS
- REAL-TIME AGING USING LDEF

EXPECTED RESULTS

- DEVELOPMENT OF DESIGN ALLOWABLES TO HIGH LEVEL OF CONFIDENCE

JOINTS AND JOINING

OBJECTIVE

- DEVELOP AND VERIFY JOINTS FOR STRUCT. ASSEMBLY

TECHNOLOGY ASSESSMENT

- THERMAL EFFECTS, TOLERANCES, CONSTRUCTION OPERATIONS, INTERFACE DESIGN MUST BE INVESTIGATED RELATIVE TO SPECIFIC JOINING AND JOINT CONCEPTS

APPROACH

- DETAILED ANALYSES OF CANDIDATES TO SELECT JOINT DESIGNS BASED ON ABOVE CONSIDERATIONS
- DEVELOPMENT OF JOINT/STRUT INTERFACE DESIGNS, MATERIALS (BONDING AGENTS) AND PROCESSING
- LONG-TERM AGING EFFECTS ON INTERFACES

EXPECTED RESULTS

- VERIFICATION OF MANUFACTURING, OPERATIONAL, AND PERFORMANCE FACTORS RELATIVE TO SPECIFIC JOINT DESIGNS

Structures and Materials Technology Program

A structural and materials program is needed, leading to component selection for final prototype and test article fabrication. The component program would require abour $1.3 million, expended over three to four years, followed by qualification tests of a full-scale prototype.

STRUCTURES AND MATERIALS TECHNOLOGY PROGRAM

	1979	1980	1981	1982	1983	1984	1985	1936
			PHASE C/D GO-AHEAD			PLATFORM LAUNCH		
<u>STRUCTURAL MATERIALS RESEARCH</u>		150	250	350	100			
MATERIALS SELECTION								
CHAMBER PREPARATION								
GROUND TESTING								
FLIGHT TESTING								
QUALIFICATION								
<u>STRUCT. & JOINT TECHNOLOGY</u>		150	250	100				
CONCEPTS DESIGN								
MATERIALS SELECTION/TEST								
FABRICATION								
PROTOTYPE HARDWARE TEST								

4.5 PROPULSION SUBSYSTEM

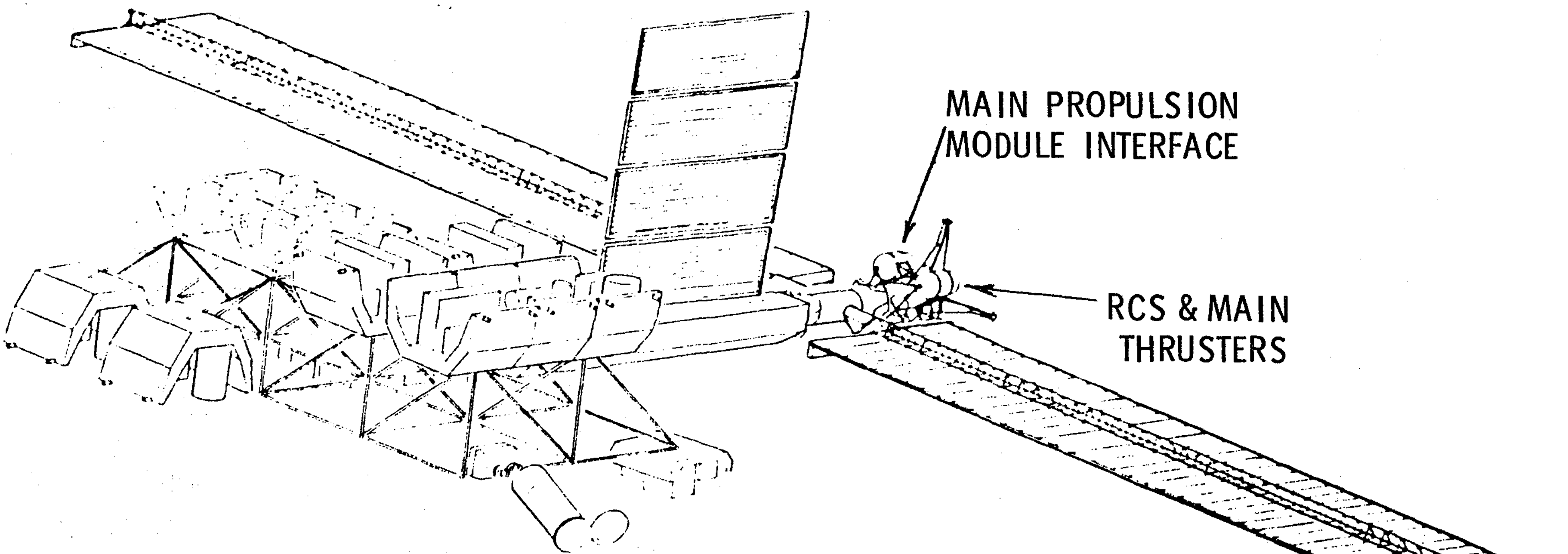
The propulsion subsystem has two primary functions. A main propulsion function for orbit make-up and orbit-to-orbit transfer, and reaction control function for stationkeeping and attitude control momentum management. The propulsion subsystem is configured as a separate self-contained module which can be removed, replaced and for return to earth for refurbishing and/or re-fueling. Subsequent charts in this section discuss the orbit make-up delta V and annual propellant requirements, delta V and propellant requirements for orbit transfer, the propulsion subsystem configuration, and the required technology program.

Propulsion Module Installation

The propulsion module is required as the main propulsion system for orbit make-up and orbit-to-orbit transfer, and as a reaction control system for attitude control. The module has to be installed during platform assembly operations and either refilled in orbit or replaced by another self-contained module on orbit.

A possible alternate position for the RCS module is at the extreme ends of the solar array. This position will provide greater torque control authority and hence reduce the propellant requirements. The thrust level at this long lever arm would be 1-5 pound (4.4-22.4 N) force and will impose loads on the solar arrays equivalent to the main propulsion thrust-to-weight loads.

PROPULSION MODULE INSTALLATION



Propulsion Subsystem Requirements

Shown is a summary of the requirements for the propulsion subsystem of the erectable platform. The main propulsion will be used for orbit transfer, if required, and for orbit make-up velocity to counteract the aerodynamic drag effects. A reaction control subsystem using common tankage with the main propulsion is used for periodic desaturation of the momentum storage devices.

The thrust-to-weight ratio of the propulsion subsystem should be less than 0.01g so as not to impose significant design loads on the flexible structures. In order to alleviate the need for on-orbit refueling technology, the propulsion module is modular in concept, with easy disconnect features to allow return of the module to Earth for refilling and refurbishment.

The potential problem of contamination of the space platform payloads and solar arrays has to be considered when placing the propulsion module to ensure zero or absolute minimum plume impingement on particular payloads.

PROPULSION SUBSYSTEM REQUIREMENTS

FUNCTIONAL REQUIREMENTS

MAIN PROPULSION

- PROVIDE TRANSFER CAPABILITY FOR PLATFORMS ASSEMBLED AND SERVICED IN LOW EARTH ORBIT (400 KM) WHICH ARE TRANSFERRED TO HIGHER ALTITUDES (575 KM) FOR THEIR MISSION OPERATIONS
- USED FOR ORBIT MAKE-UP ΔV

REACTION CONTROL SYSTEM

- STATIONKEEPING MANEUVERS
- DESATURATION OF MOMENTUM STORAGE DEVICES EACH ORBIT

DESIGN REQUIREMENTS FOR TOTAL SUBSYSTEM

- LONG-TERM SPACE STORAGE PRIOR TO REACTIVATION
- REPETITIVE CYCLES AND LONG LIFE
- MODULAR CONCEPT FOR
    - ON-ORBIT INSTALLATION
    - QUICK-DISCONNECTS FOR RESUPPLY
    - PROPELLANT REFILLING AND REFURBISHMENT
- MINIMAL-TO-ZERO CONTAMINATION OF SPACE PLATFORM PAYLOADS AND SOLAR ARRAYS
- LOW THRUST-TO-WEIGHT RATIO < 0.01 g

Requirements for Orbit Make-Up ΔV

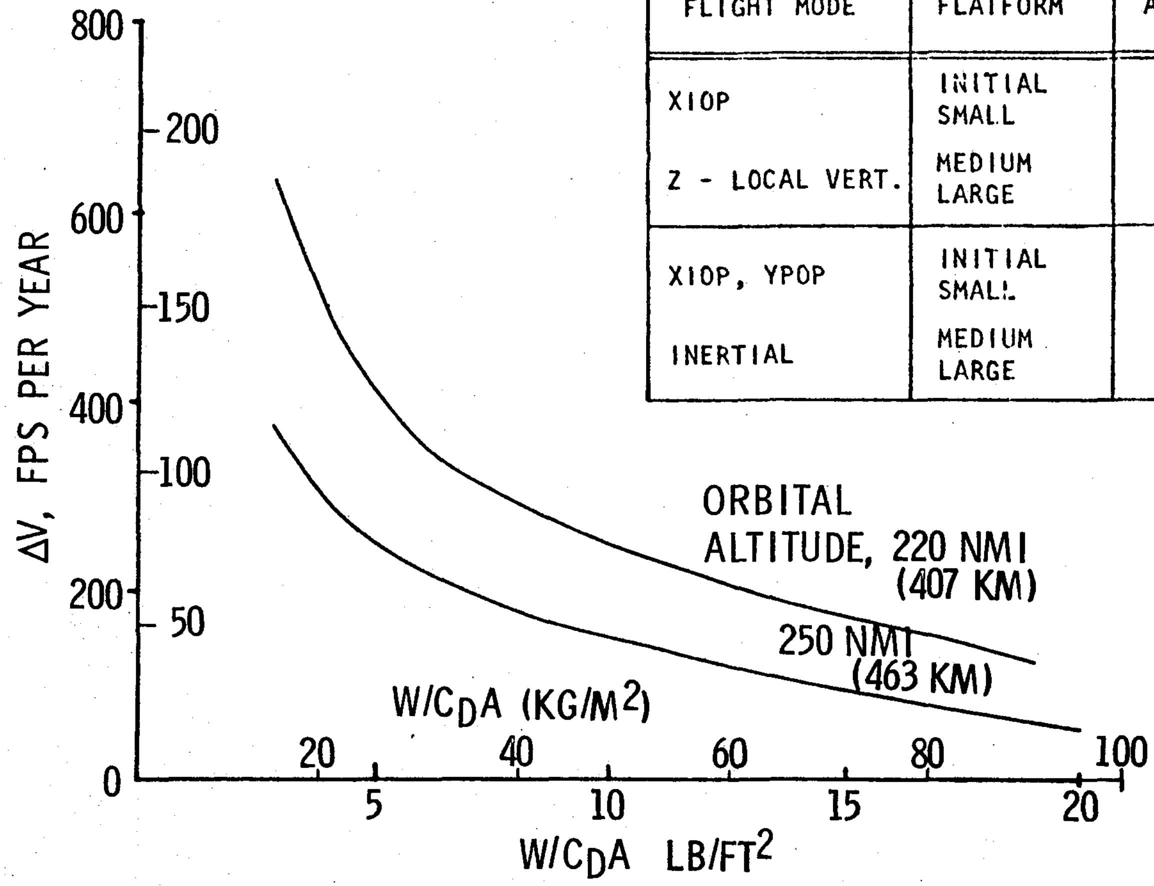
The platform assembly and operation is in low earth orbit and will be subject to orbit decay due to aerodynamic drag. An atmospheric density model was used for the decay velocity based on the nominal solar maximum.

Ballistic coefficients (W/C_DA) were evaluated for the platform during its initial assembly through the fully loaded operational phases for the platform flying X-axis in the orbit plane (XIOP) and for two flight modes (Z local-vertical and inertial hold). This attitude (XIOP) will present the least frontal area (i.e., radiator surface in the orbit plane), the highest W/C_DA and hence the least velocity make-up requirements to counteract the orbit decay.

The table shows the time taken to cause a ten-percent altitude decay for the various configurations. The initial assembly phases with the solar arrays deployed but no payloads installed on the platform can result in decay times as short as 42 days. As additional payloads are installed the W/C_DA increases and the ΔV requirements are reduced drastically.

ORBIT MAKEUP ΔV

FLIGHT MODE	FLATFORM	DAYS FOR DECAY 10% h	
		ALT, 220 NMI (407 KM)	ALT, 250 NMI (463 KM)
XIOP	INITIAL	50	92
	SMALL	116	248
Z - LOCAL VERT.	MEDIUM	105	241
	LARGE	108	246
XIOP, YPOP	INITIAL	42	70
	SMALL	88	176
INERTIAL	MEDIUM	85	170
	LARGE	84	168



- MAKEUP INTERVAL FOR Δh = 10%
- NOMINAL SOLAR MAXIMUM
- BASED ON AVERAGE W/C_DA THROUGHOUT EACH ORBIT

Annual Orbit Make-Up Propellant Requirements

The annual propellant requirements were defined for a hydrazine system with 230 secs (2256 N-sec/kg) specific impulse and a bi-propellant with a higher specific impulse of 260 seconds (2550 N-sec/kg). The three platforms identified in the payload/mission section were considered to determine the propellant needs of each platform. Platform #1 at 28° inclination, which is the heaviest vehicle, was considered to be in an inertial hold flight mode. For commonality, it is recommended that identical propulsion stages be employed for all the platforms. Thus, the propulsion module on platform #3 will be replaced less often than the module on platform #1.

Platform #2 will spend some time at the lower altitudes both during assembly, maintenance and replacement of payloads. The majority of its operation is at 575 km where the propellant requirements for orbit make-up are insignificant compared to the orbit transfer requirements.

ANNUAL ORBIT MAKE-UP PROPELLANT REQUIREMENTS

	HYDRAZINE, I_{sp} = 230 SEC (2256 N-SEC/KG)		BI-PROPELLANT, I_{sp} = 260 SEC (2550 N-SEC/KG)	
ALTITUDE (nmi)→	220 (407 KM)	250 (463 KM)	220 (407 KM)	250 (463 KM)
PLATFORM 1 XIOP/YPOP INERTIAL HOLD	*3930 (1783 KG)	2459 (1115 KG)	3227 (1464 KG)	2017 ((915 KG)
PLATFORM 2 ** XIOP/YPOP INERTIAL HOLD	2380 (1080 KG)	1473 (668 KG)	1954 (886 KG)	1209 (548 KG)
PLATFORM 3 XIOP Z LOCAL VERTICAL	2185 (991 KG)	1429 (648 KG)	1794 (814 KG)	1173 (532 KG)

*Identical propulsion stages have been adopted for the Baseline Platforms 1 and 3.

**Platform 2 operational altitude is at 575 km, where the propellant for orbit make-up ΔV is significantly less.

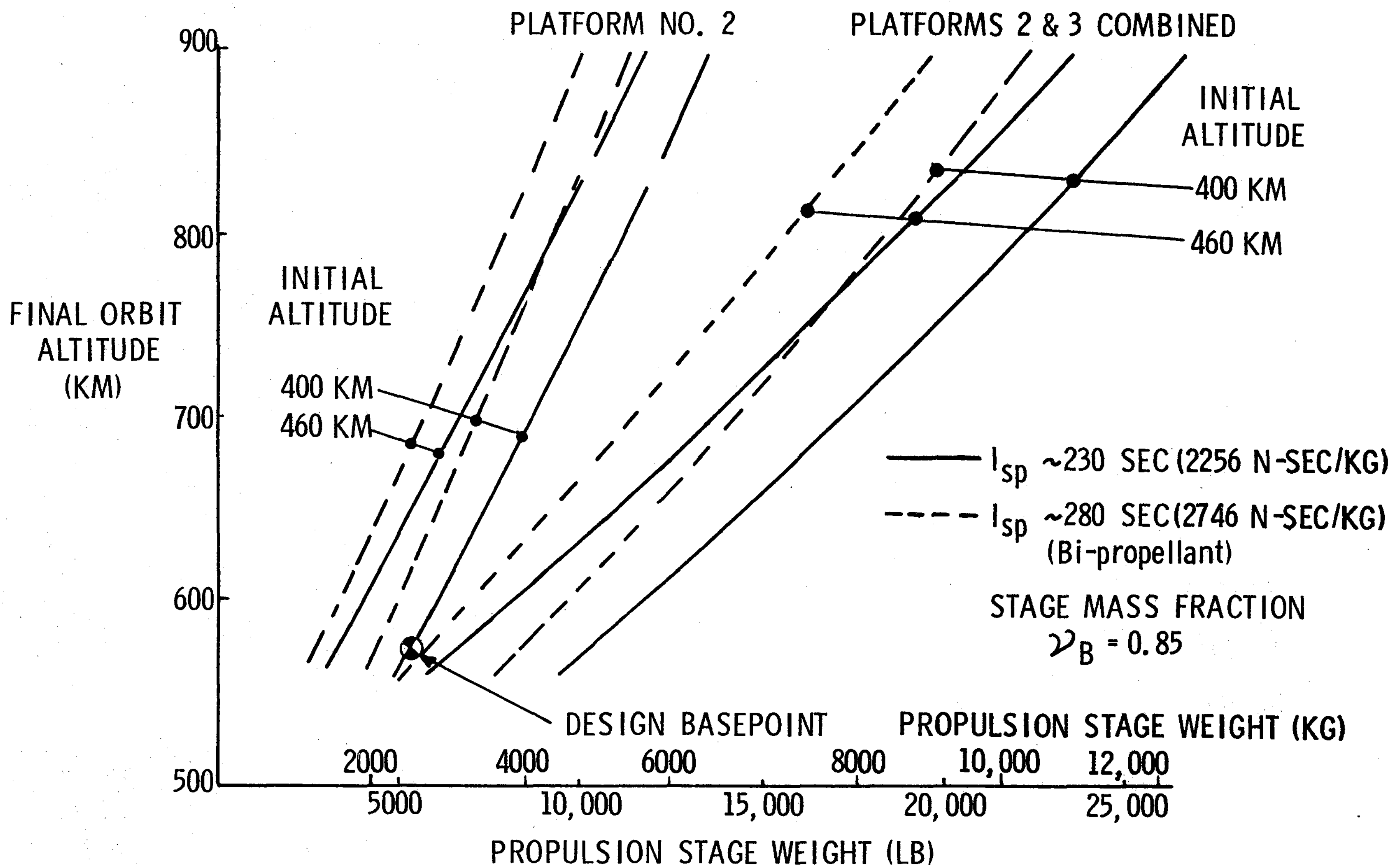
Propulsion Stage for Orbit Transfer

Initial design guidelines suggested that the platform be transferred to a 900 km orbit. This chart shows the significant weight impact on the propulsion system requirements for the round trip to the 900 km orbit. For the platform #2, which is a near polar orbit, the propulsion stage weight is about 13,500 lbs (6123 kg) if the assembly orbit is 400 km, and the propellant is hydrazine. The Shuttle orbiter payload capability at polar orbit is nearly 23,000 lbs (10433 kg) which allows for 9,500 lbs (4309 kg) of payload other than the propulsion stage (i.e., payload experiments, pallets and cradle to carry the propulsion stage).

The mission analysis section suggested a preferred altitude of 575 km for platform #2 which results in a basepoint weight for the propulsion stage of slightly over 5,000 lbs (2268 kg). Therefore, the orbiter has sufficient payload margin to polar orbit to accommodate several useful experiment payloads.

If only two platforms are considered by combining platforms #2 and #3 with the resulting platform in polar orbit, the propulsion stage weight significantly increases and thus detracts from the orbiters ability to carry experimental payloads.

PROPULSION STAGE FOR ORBIT TRANSFER



Platform No. 2 Transportation to 90° Inclination Orbit

The most demanding mission on the orbiter's capability is its ability to assemble and supply platform #2 in its near polar orbit. It is required to assembly the utilities module structural platform and propulsion stage with the first flight. This will allow the platform to be a stable, controllable vehicle for subsequent visits of the orbiter for payload installation. The payload mission nodal shows that the requirements are for a small size platform with a two blind array. The packaging can be accomplished inside the orbiter's cargo bay and an initial weight estimate shows that the 22,750 lbs (10319 kg) weight budget is within the orbiter's delivery capability.

The payloads (circa 1985) to be installed on this platform can be organized into two separate flights that do not exceed the orbiter's polar orbit payload capability. Included are weights for the payload pallets, platform tables and pointing systems.

PLATFORM NO. 2 TRANSPORTATION TO 90° INCLINATION ORBIT

LAUNCH NO. 1
- (22,744 LB) (10316 KG)
- UTILITIES SERVICE MODULE
- STRUCTURAL PLATFORM (SMALL)
- PROPULSION STAGE
- SUPPORT STRUCTURE & ASSY FIXTURES

LAUNCH NO. 2
- (22,030 LB) (993 KG)
- PAYLOADS 2, 11, 27, & 65
- 2 ERNO PALLETS
- 2 PLATFORM TABLES
- 1 INSTRUMENT POINTING SYSTEM
- 1 PAYLOAD GIMBAL MOUNT

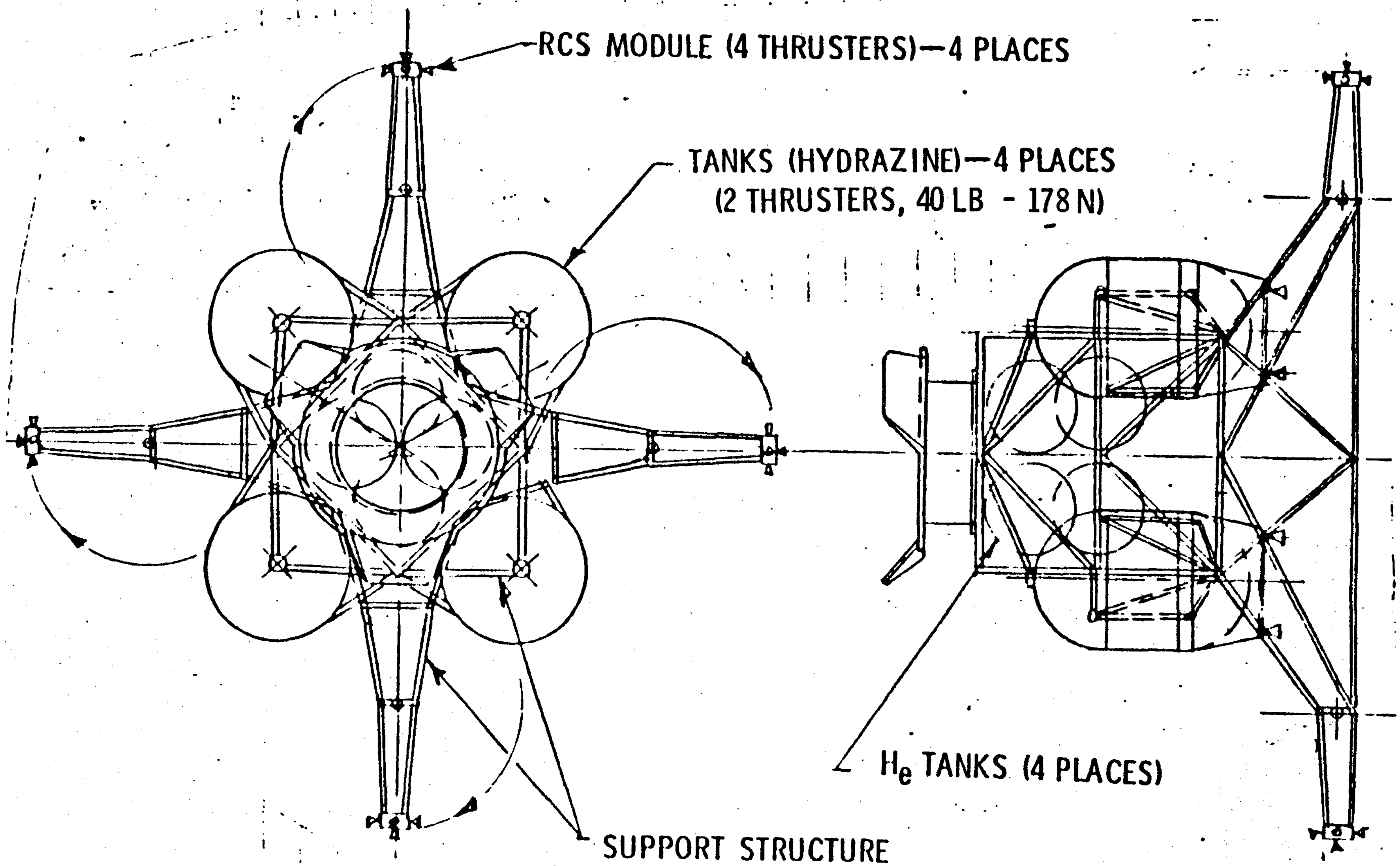
LAUNCH NO. 3
- (21,055 LB) (9550 KG)
- PAYLOADS 1, 52, & 55
- 2 PALLETS
- 1 PLATFORM TABLE
- 1 INSTRUMENT POINTING SYSTEM
- 2 PAYLOAD GIMBAL MOUNTS

Propulsion Subsystem - Four-Tank Version

Based on the propellant requirements for main propulsion and attitude control, the propulsion tankage, as shown, is characteristic of the teleoperator. The Teleoperator Retrieval System (TRS) development has been deleted as part of the NASA Shuttle program class vehicle. The propulsion subsystem module is a non-intelligent module, with all commands and power being supplied by the utilities module of the space platform. The teleoperator tankage, propulsion kits, attitude thrust kits, and remote manipulator system (RMS) grappling fixture are directly applicable to the needs of the erectable platform.

Thruster modules should be on extendable arms in order to increase the control authority. This will necessitate foldable structural arms for packaging and flexible fuel feed lines from the common tankage to the thruster nozzles.

PROPULSION SUBSYSTEM—FOUR-TANK VERSION

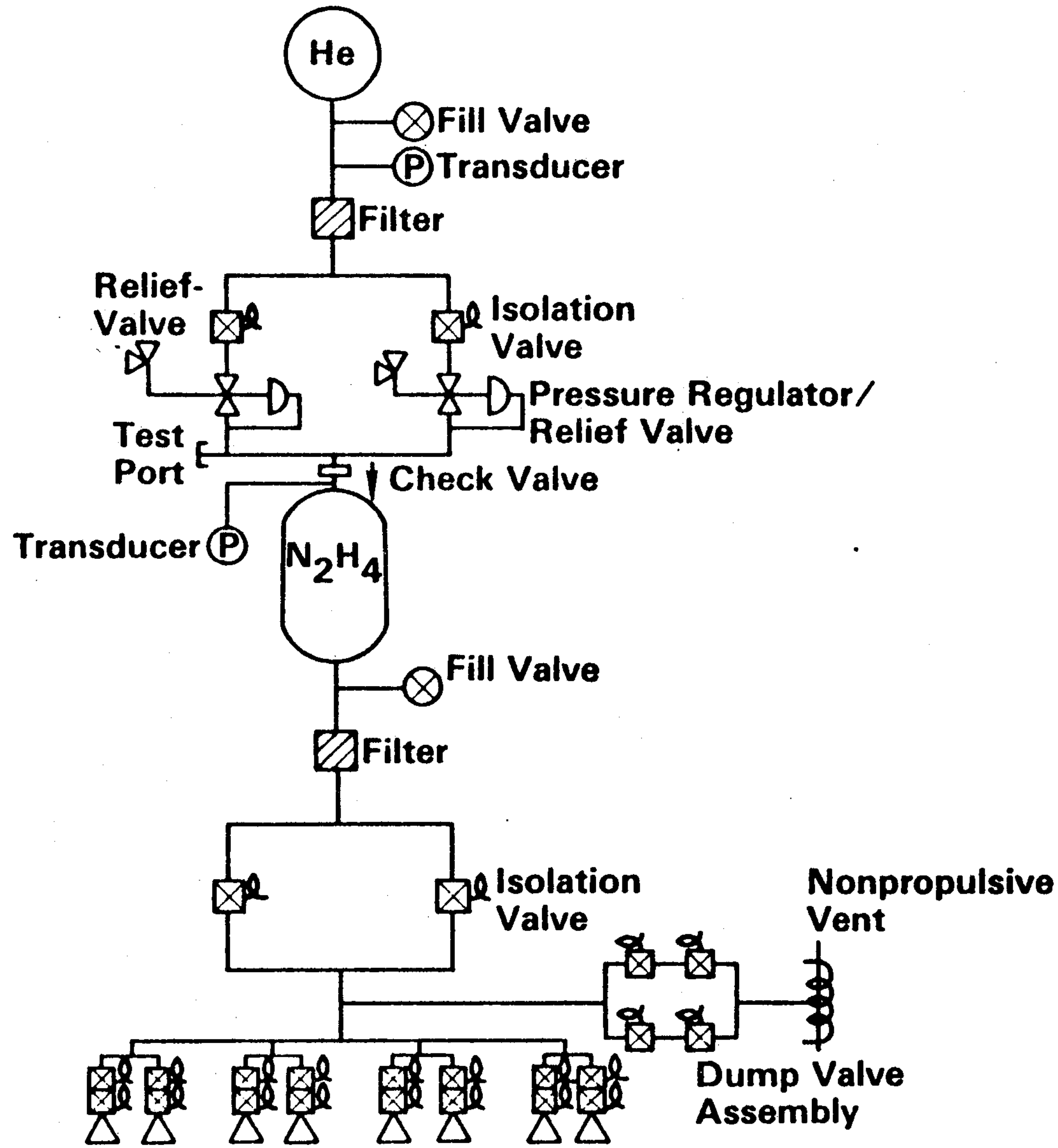


Kit Hydrazine Propulsion Subsystem

Shown is a pressure regulated subsystem (teleoperator concept) which is directly amenable to the erectable platform requirements. An alternate approach would be a blow-down system which would not require the separate helium storage tanks.

One requirement of the propulsion subsystem would be the capability of multiple re-use. The refill and refurbishment of the tankage is currently conceived to be accomplished by returning the propulsion module to the ground. In-flight refueling will require new technology development.

KIT HYDRAZINE PROPULSION SUBSYSTEM



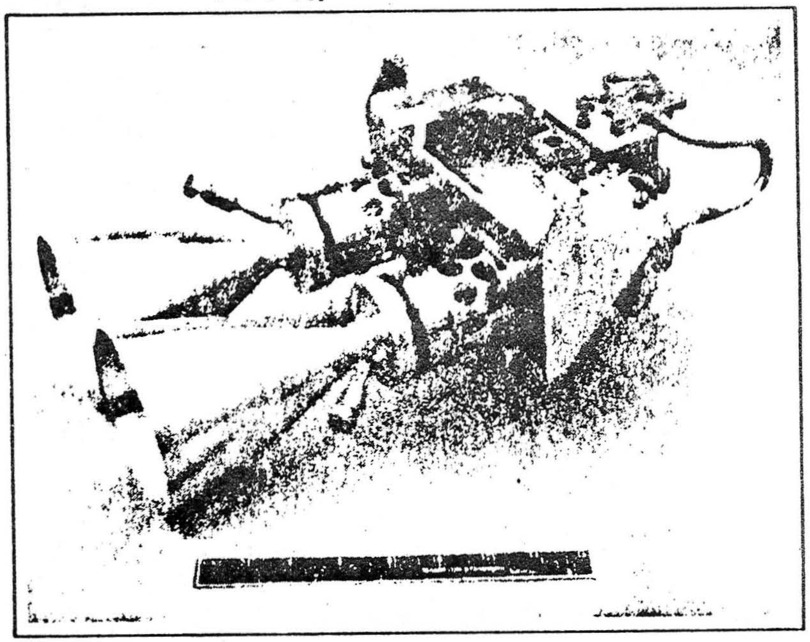
Hydrazine Thruster Units

In order to restrain the thrust-to-weight to levels acceptable to the lightweight flexible structure of the erectable platform, the thruster units are 40 lbs (178 N) or less. These units need to have the capability of long life and in the case of the RCS thrusters, operate in the pulse mode for thousands of cycles.

Shown are two typical thruster units that have been built and appear to meet the design life requirements of the erectable platform propulsion subsystem.

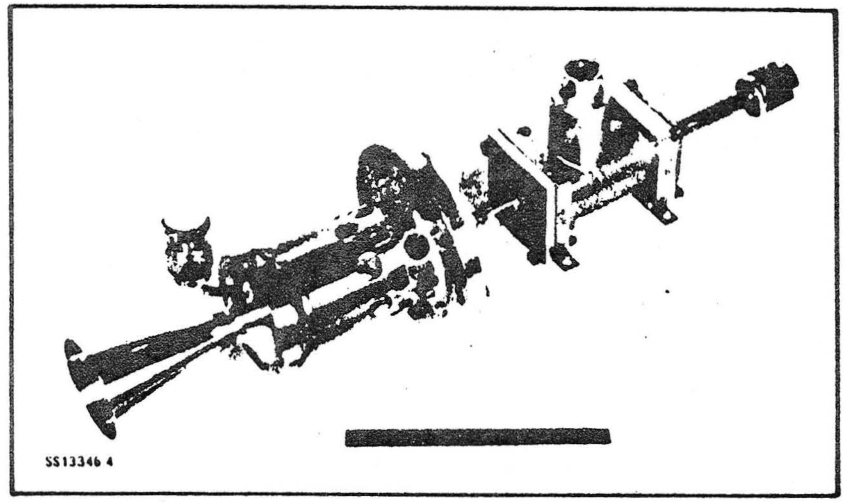
HYDRAZINE THRUSTER UNITS

40-LB_f MAIN THRUSTER (178-N)



- BUILT FOR TELEOPERATOR
- GROUND TESTED
- POSSIBLE FURTHER TEST WILL BE CONDUCTED

5-LB_f RCS THRUSTER (22-N)



- OVER 260,000 LB_f-SEC (1,156,532 N-SEC) TOTAL IMPULSE
- 1150 LB_m (522 KG) N_2H_4 THRU-PUT
- SIX 2-HR BURNS, 19 HR TOTAL STEADY STATE
- 22,500 PULSES

Propulsion Subsystem Trades/Selection

The various design alternates considered during this study are shown on this chart together with our selections and their associated technology level of criticality.

The propellant recommended is hydrazine, it being state-of-the-art and low risk, but with reservation concerning potential contamination problems. The contamination problem must be solved before the propellant or any other propellant is used, thus the assignment of #1 (enabling) technology level. Cold gas is too inefficient, and still presents contamination (ice crystals) to payloads having "super cooled" sensors.

Most of the other selections require only normal or exhancing technology development. The only critical item is the modular construction which allows for installation and removal of the module when in orbit.

PROPULSION SUBSYSTEM TRADES/SELECTION

COMPONENT	ALTERNATES	SELECTION	TECH. LEVEL
PROPELLANT MODULE	COLD GAS	TOO INEFFICIENT	-
	HYDRAZINE	S.O.A.—LOW-RISK/COST, POTENTIAL CONTAMINATION PROBLEMS	1
	BI-PROPELLANT	—	
PROPELLANT MANAGEMENT SYSTEM	DIAPHRAGM	S.O.A. SUITED FOR EXTENDED MISSION	3
	SURFACE TENSION	MAIN PROPULSION SETTLING USING RCS	3
MODULAR CONSTRUCTION	ON-ORBIT REFUELING	—	-
	REPLACE MODULE	NEEDS ELECTRICAL SIGNAL & STRUCTURAL INTERFACES	1
RCS THRUSTERS	PIVOTED FROM MAIN PROPELLANT TANKS	FLEXIBLE PRESSURIZED FUEL LINES	2
	ATTACHED TO ENDS OF SOLAR ARRAYS	MODULE AT END OF ARRAYS SHOULD BE CONSIDERED—LONG LEVER ARM	-
MAIN THRUSTERS	10-40 LB_f THRUSTER (44-178 N)	S.O.A.—NEED SEVERAL THRUSTERS FOR EACH TANK MODULE	3
	100-500 LB_f THRUSTER (445-2224 N)	HIGHER THRUST LEVEL, LONGER LIFE, HIGH RELIABILITY	2

Propulsion Subsystem Technology Candidates

The first area of technology investigation should be to determine the compatibility of the hydrazine propellant and the currently suggested non-contamination requirements. The relative size and positioning of the payloads, arrays, etc., and the propulsion module require contamination studies to be conducted not in the near field but up to 100 foot (30.5 m) distance from the nozzle exit plane and at oblique view factors to the plume center line. Some studies have shown that low levels of contamination impinging on the near surface of solar arrays and optical glasses do not seriously degrade their performance.

The technology of on-orbit installation and removal of a self contained propulsion module throughout the platform 10 year design life will require extensive development to determine the interface requirements (power, signal) and the operational procedures for installation using the RMS.

PROPULSION SUBSYSTEM TECHNOLOGY CANDIDATES

HYDRAZINE PROPELLANT

OBJECTIVE

- DETERMINE THE RELATIVE CONTAMINATION OF DIFFERENT PAYLOAD SURFACES AT DISTANCES FROM 50 TO 100 FT (15.2 TO 30.5 M) FROM NOZZLE EXIT

TECHNOLOGY ASSESSMENT

- CONTAMINATION STUDIES IN NEAR FIELD CONDUCTED ON SOLAR ARRAY AND OPTICAL GLASSES

APPROACH

- DETERMINE A REALISTIC LEVEL OF CONTAMINATION ACCEPTABLE TO PAYLOAD EXPERIMENTS
- CONDUCT GROUND TESTS TO FIND ACTUAL CONTAMINATION ON SUPER-COOLED SURFACES
- MEASURE PARTICULATE EMISSION FROM CATALYST

EXPECTED RESULTS

- ACCEPTABLE PROPULSION SYSTEM AND ITS PLATFORM POSITIONING WHICH SATISFIES MOST END USERS

MODULE ON-ORBIT INSTALLATION

OBJECTIVE

- SELF-CONTAINED MODULE CAPABLE OF BEING INSTALLED ON PLATFORM IN ORBIT AND MODULE REMOVED AND RETURNED TO EARTH FOR REFILLING

TECHNOLOGY ASSESSMENT

- ON-ORBIT INSTALLATION & REMOVAL OF PROPULSION MODULE NOT ACCOMPLISHED YET
- ADVANCE ASSEMBLY TECHNIQUES REQUIRED
- MMS PLANNED ON ORBIT REMOVAL

APPROACH

- MINIMIZED TOTAL AMOUNT OF INTERFACE REQUIREMENTS
- USE RMS SIMULATION TO INSTALL "MOCK-UP" CONCEPT
- DETERMINE THE ACCEPTABLE CONNECTIONS TO BE MADE IN ORBIT

EXPECTED RESULTS

- SELF-CONTAINED PROPULSION MODULE WITH DISCRETE STRUCTURAL ATTACHMENT POINTS AND MINIMUM OF POWER & SIGNAL INTERFACES

Propulsion Subsystem Technology Candidates (Cont.)

In order to increase the efficiency and control authority the recommendation is a concept of RCS thrusters on pivotted arms sharing a common propellant tankage. This will involve the technology associated with flexible low pressure fuel lines that can be extended and restowed during the propulsion module installation and removal on orbit. These features should be developed and ground tested to demonstrate their feasibility.

An alternative to the multiple small thruster for the main propulsion system would be the development of highly reliable larger thrusters. These thrusters would provide a legacy for other large space structure systems of the future. There are currently thrusters in this thrust class; the need would be to extend their mission life and reliability.

PROPULSION SUBSYSTEM TECHNOLOGY CANDIDATES (CONT.)

RCS THRUSTER PIVOT ARMS

OBJECTIVE

- DEVELOP FLEXIBLE PRESSURIZED FUEL LINES FOR HYDRAZINE THRUSTERS AND INCREASE CONTROL AUTHORITY ABOUT ROLL AXIS OF PLATFORM VEHICLE.

TECHNOLOGY ASSESSMENT

- GAS DUCTED ON EXTENDED NOZZLES, BUT PRECISION PULSES DIFFICULT
- BELLOW LINES FEASIBLE FOR LOW PRESSURE

APPROACH

- DEVELOP AND TEST FLUID LINES THAT ARE FLEXIBLE
- GROUND TEST AND QUALIFY FLEXIBLE LINES

EXPECTED RESULTS

- PIVOT RCS THRUSTER FOR GREATER CONTROL AUTHORITY AND SIGNIFICANTLY LESS PROPELLANT REQUIREMENTS

100-500 LB_f THRUSTERS

OBJECTIVE

- DEVELOP 100-500 LB_f (445-2224 N) HYDRAZINE THRUSTERS CAPABLE OF REPETITIVE LONG-DURATION FIRING TIME

TECHNOLOGY ASSESSMENT

- SMALLER THRUSTER USED FOR RCS SYSTEM —REPETITIVE FIRINGS
- 40-LB_f THRUSTERS WERE UNDER DEVELOPMENT FOR TELEOPERATOR BEFORE CANCELLATION

APPROACH

- IDENTIFY THE TECHNOLOGY REQUIRED FOR LONG-LIFE, LONG DURATION FIRINGS FOR THE LARGER, HIGHLY RELIABLE THRUST SYSTEMS
- DEVELOP PROTOTYPE THRUSTER AND CONDUCT EXTENDED GROUND TESTS

EXPECTED RESULTS

- 100-500 LB_f (445-2224 N) HIGHLY RELIABLE HYDRAZINE THRUSTERS SUITABLE FOR MAIN PROPULSION SYSTEM KITS FOR FLEXIBLE SPACE STRUCTURES REQUIRE LOW THRUST-TO-WEIGHT

Propulsion Technology Program Planning

Shown are estimates for the funding and scheduling of the previously identified technology candidates.

PROPULSION TECHNOLOGY PROGRAM PLANNING

	1979	1980	1981	1982	1983	1984	1985	1936
			PHASE C/D GO-AHEAD			PLATFORM LAUNCH		
HYDRAZINE PROPELLANT		150K						
CONTAMINATION REQUIREMENTS								
GROUND TEST CONTAMINATION LEVELS								
MODULE ON-ORBIT INSTALLATION		200K	300K	300K				
INTERFACE REQUIREMENTS								
GROUND SIMULATION								
RCS THRUSTERS			100K	200K				
DEVELOPMENT FLEX LINES								
GROUND TEST RELIABILITY								
100-500 LBF (445-2224 N) THRUSTERS		200K	200K	200K				
DESIGN & DEVELOPMENT PROTOTYPE								
GROUND TEST FIRINGS								

4.6 ATTITUDE STABILIZATION AND CONTROL SUBSYSTEM

If the platform is required to operate at any prescribed attitude and flight mode as dictated by potential payload users, then significant demands are imposed on the pointing accuracy and long term control requirements. There could be stringent contamination requirements from payloads with super-cooled sensors that would rule out all chemical thrusters as a control subsystem. This section indicates some of the potential problems of a control system that has to meet all the mission/payload requirements. Serious consideration should be given to relaxing some of the flight operational modes in order to reduce the momentum requirements.

Attitudes which are gravity gradient stable and alignment of principal axis with earth local vertical would assist in reducing the momentum build up and influence the selection of momentum desaturation concepts.

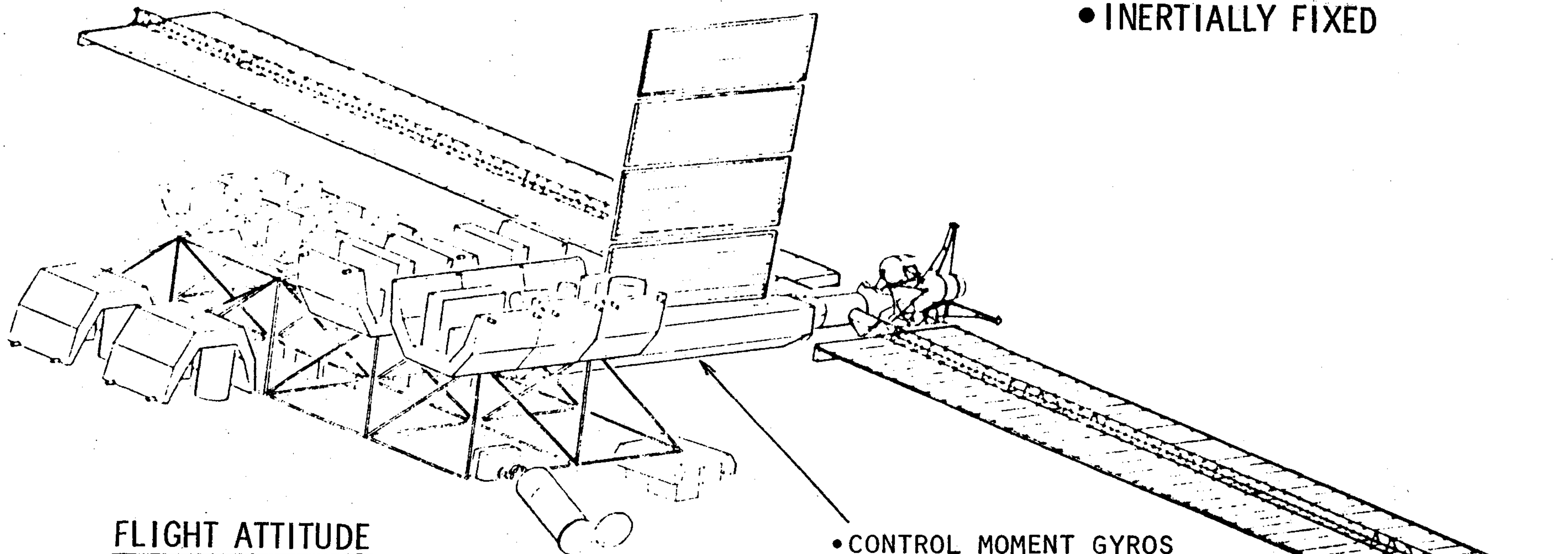
The plant (platform/utilities module payload) build-up and uncertainties during the mission life time will effect the control laws and structural dynamics of the overall system. The large and flexible structural elements will present significant technology issues with respect to structural interaction, control formulation, and control responsiveness.

Attitude Stabilization and Control Subsystem

The basic features of the attitude stabilization control subsystem are divided into three functions: a pointing and stability function provided by six CMG's in a cone cluster configuration, an attitude determination system containing a star tracker, gyros, and computers to provide the necessary attitude and attitude rate information, and a momentum management system to control CMG saturation.

The attitude stabilization and control subsystem requirements were determined by considering XIOP and XPOP flight attitudes in local vertical and inertial stable flight modes at various sun angles.

ATTITUDE STABILIZATION AND CONTROL SUBSYSTEM



PLATFORM POINTING

- EARTH LOCAL VERTICAL
- INERTIALLY FIXED

FLIGHT ATTITUDE

- XIOP: -X AXIS IN ORBIT PLANE
- XPOP: -X AXIS PERPENDICULAR TO ORBIT PLANE

- CONTROL MOMENT GYROS
- ATTITUDE REFERENCE SYSTEM—STAR TRACKER
- MOMENTUM MGMT. SYSTEM

Control Issues

The chart presents various issues and options that were considered in the design of the attitude control systems. Some of the factors driving the design are the variable geometry of the spacecraft during buildup and operational orbit, environmental disturbance torques, disturbance loads during construction and docking, structural flexibility, pointing requirements, and orbit lifetime.

CONTROL ISSUES

ISSUE	OPTIONS	CONSIDERATIONS AFFECTING DESIGN
✓ ATTITUDE CONTROL	• GRAVITY GRADIENT • MOMENTUM BIAS • SCISSORED HOOPS • GEOMAGNETIC • JETS • IPACS • CMG'S	• CONFIGURATION • ORBIT • ORIENTATION • POINTING ACCURACY • P/L VIEW ANGLES AND • GIMBAL COMMANDS • MANEUVERS (ATTITUDE & DELTA-V) • OPERATIONAL CONSTRAINTS • STRUCTURAL STIFFNESS • CONSTRUCTION METHOD • DOCKING • LOADS • LIFE
✓ PAYLOAD POINTING	• POINT PLATFORM • REACTIONLESS GIMBALS • GIMBALS	
✓ ATTITUDE REFERENCE	• CENTRAL REF + P/L ALIGNMENT • AUTONOMOUS REF FOR EACH P/L	
✓ DELTA-V CONTROL	• PULSED • CONTINUOUS	
✓ CONTROL SYSTEMS &/OR STRUCTURE INTERACTIONS	• STIFF-VS-SOFT STRUCTURE • INDEPENDENT-VS-INTEGRATED CONTROL SYSTEMS	
✓ CONTROL DURING ASSEMBLY	• SHUTTLE CONTROL • GRAVITY GRADIENT • PLATFORM OPERATIONAL ACS • FREE DRIFT • AUX. STAB. SYSTEM	

Attitude Control Analysis—Initial Observations

Initial analysis shown that large gravity gradient and aerodynamics disturbance torques are present on the spacecraft. These torques result from inertia unbalance, assymetry, and CP-CG offset. As a result of the torques, there is a significant momentum buildup and large cyclical momentum requirements. For a pointing requirement of .01 degrees, there is a control system/structural dynamics interaction problem. The first bending frequency of the solar blanket is within the control bandwidth (i.e., no separation) for pointing errors of .01 degrees or less.

ATTITUDE CONTROL ANALYSIS—INITIAL OBSERVATIONS

- CONFIGURATION ASYMMETRIES CAN PRODUCE LARGE DISTURBANCE TORQUES (GRAVITY GRADIENT & SOLAR ARRAY AERODYNAMICS)

  → MINIMIZE INERTIA DIFFERENCE & CP TO C.G. OFFSETS

- WORST CASE INERTIAL ORIENTATIONS IMPOSE EVEN MORE SEVERE CONTROL REQUIREMENTS THAN LOCAL LEVEL MODES

- PRELIMINARY ANALYSIS INDICATES THAT CMG's IN SKYLAB SIZE RANGE CAN MEET WORST CASE REQUIREMENTS

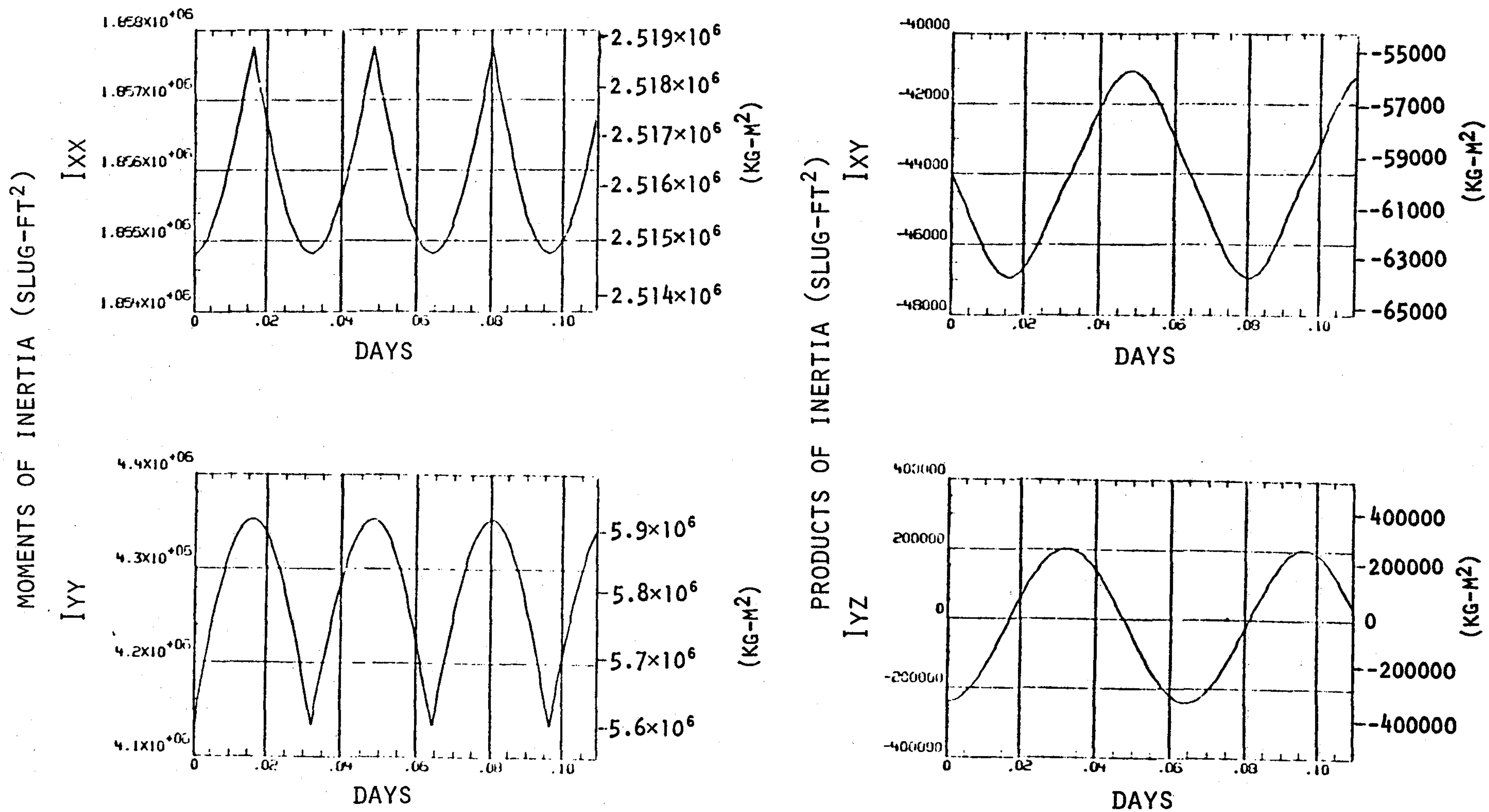
- 0.01-DEG POINTING ACCURACY REQUIREMENT AT PAYLOAD INTERFACE APPEARS DIFFICULT, REQUIRES FURTHER ANALYSIS AND EVALUATION

Inertia Time History

The chart shows the time varying properties of some terms in the inertia matrix for the XIOP attitude. The solar arrays which are tracking the sun for maximum power via their two axes rotation gimbals result in the time varying principal and product of inertias. The sinusodially varying Iyz term provides a major contribution to the momentum buildup along the X and Z body axes. The Ixz term, not shown, is a sinusoid and it is a significant factor in the momentum buildup along the Y body axis. A major contribution to the secular momentum is the assymetrical property of the spacecraft. A reduction in momentum buildup could be achieved by providing better inertial balancing if possible without imposing severe design problems on the other subsystems.

INERTIA TIME HISTORY

FLIGHT MODE XIOP/ZLV, SUN @ 45°

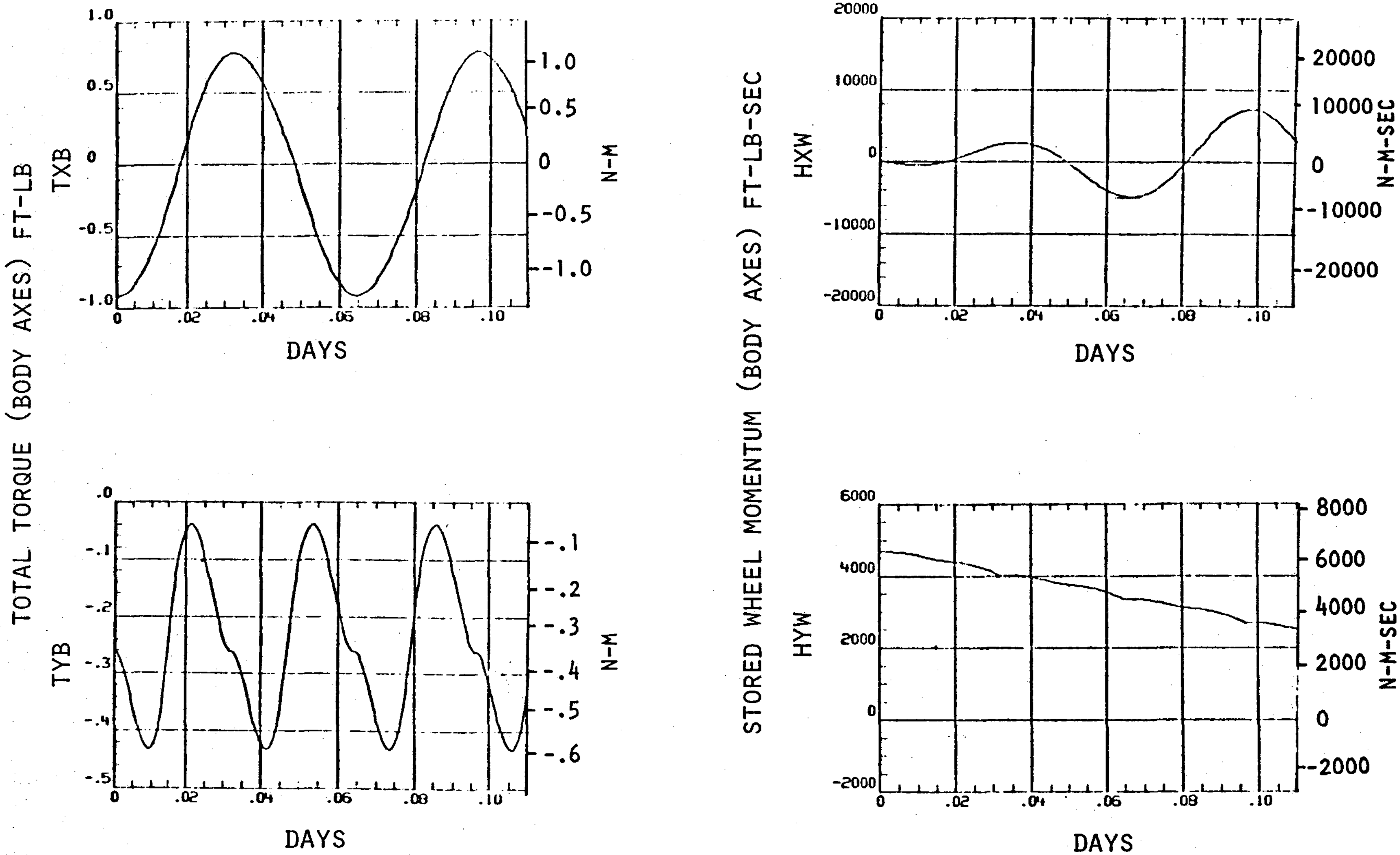


Torque and Momentum Time History

Torque history along with the corresponding angular momentum history for the X and Y axes are presented for the XIOP local vertical mode at a sun angle of 45 degrees. In both cases there is a significant secular momentum buildup. The X axis momentum exhibits a profile of a divergent sinusoid. The Z axis, not shown, exhibits a momentum profile similar in shape and magnitude to the X axis. This divergent shape is due to a gravity torque produced by a sinusoidal varying Iyz shown on the next chart and the aerodynamic torque varying sinusoidally about the Z-body axis. This case clearly demonstrates the problem of momentum buildup that must be countered to control the CMG saturation.

TORQUE AND MOMENTUM TIME HISTORY

FLIGHT MODE XIOP/ZLV, SUN @ 45°



Principal Disturbance Torques and Momentum Requirements

The following four charts present a summary of the principal disturbance torques and the corresponding momentum values. The data is given for a XIOP and XPOP flight attitude at three values of sun angle (0°, 45°, 90°).

Large disturbance torques are experienced in the inertial mode for the XIOP attitude. These large torques are expected in this orientation. However, the torques are cyclical and give rise to large cyclical momentum requirements and significantly less secular momentum buildup than in the local vertical case. In the local vertical orientation, the smaller torques are predominantly secular resulting in a large momentum buildup per orbit. As previously mentioned, the secular buildup is due to the time varying assymetrical cross product terms and the sinusodial aerodynamic torques.

For the XPOP flight attitude, the gravity gradient torques are large about the X axis in the inertial mode. The aerodynamic torques are the same order of magnitude for the local vertical and inertial mode. These torques are essentially cyclical resulting in large cyclical momentum requirements and smaller momentum buildup relative to XIOP attitude. However, in XPOP the secular momentum is significant for all three sun angle conditions in the local vertical mode. The momentum requirements for the XPOP attitude are greater than for the XIOP and on this basis the XIOP is the preferred orientation.

PRINCIPAL DISTURBANCE TORQUES

XIOP ATTITUDE	TORQUE (N·m)					
	GRAVITY GRADIENT			AERODYNAMIC		
	X	Y	Z	X	Y	Z
LOCAL VERTICAL						
$\beta = 0$	-0.104	-0.43	0.	0.04	2.06	0.104
$\beta = 45$	-0.125	-0.42	0.	-0.37	0.37	1.69
$\beta = 90$	-0.096	-0.42	0.	0.07	0.07	0.01
INERTIAL						
$\beta = 0$	-0.18	9.18	0.24	0.12	5.15	-0.01
$\beta = 45$	-1.26	8.61	0.75	0.19	2.13	1.65
$\beta = 90$	-0.18	8.04	0.24	0.04	0.47	-0.01

PRINCIPAL DISTURBANCE TORQUES

XPOP	TORQUE (N·m)					
	GRAVITY GRADIENT			AERODYNAMIC		
	X	Y	Z	X	Y	Z
LOCAL VERTICAL						
$\beta = 0$	-1.25	-0.42	0.	-0.84	1.80	6.35
$\beta = 45$	-1.23	0.72	0.	-0.61	0.72	3.36
$\beta = 90$	-0.09	-0.42	0.	-0.49	0.	1.67
INERTIAL						
$\beta = 0$	3.82	-0.45	-0.35	0.49	5.15	1.67
$\beta = 45$	3.82	0.46	-0.35	0.49	2.25	1.67
$\beta = 90$	3.82	-0.45	-0.35	0.49	0.47	1.67

ANGULAR MOMENTUM

XIOP	ANGULAR MOMENTUM (N∘m-sec)					
	CYCLIC			SECULAR (PER ORBIT)		
	X	Y	Z	X	Y	Z
LOCAL VERTICAL						
$\beta = 0$	240.	678.	271.	0.	1152.	0.
$\beta = 45$	*	0.	*	*	1798.	*
$\beta = 90$	169.	0.	203.	0.	1974.	0.
INERTIAL						
$\beta = 0$	102.	7118.	678.	285.	0.	630.
$\beta = 45$	678.	6440.	1220.	3475.	0.	644.
$\beta = 90$	102.	6915.	34.	271.	0.	636.

*Divergent sinusoid—H per orbit = 4750

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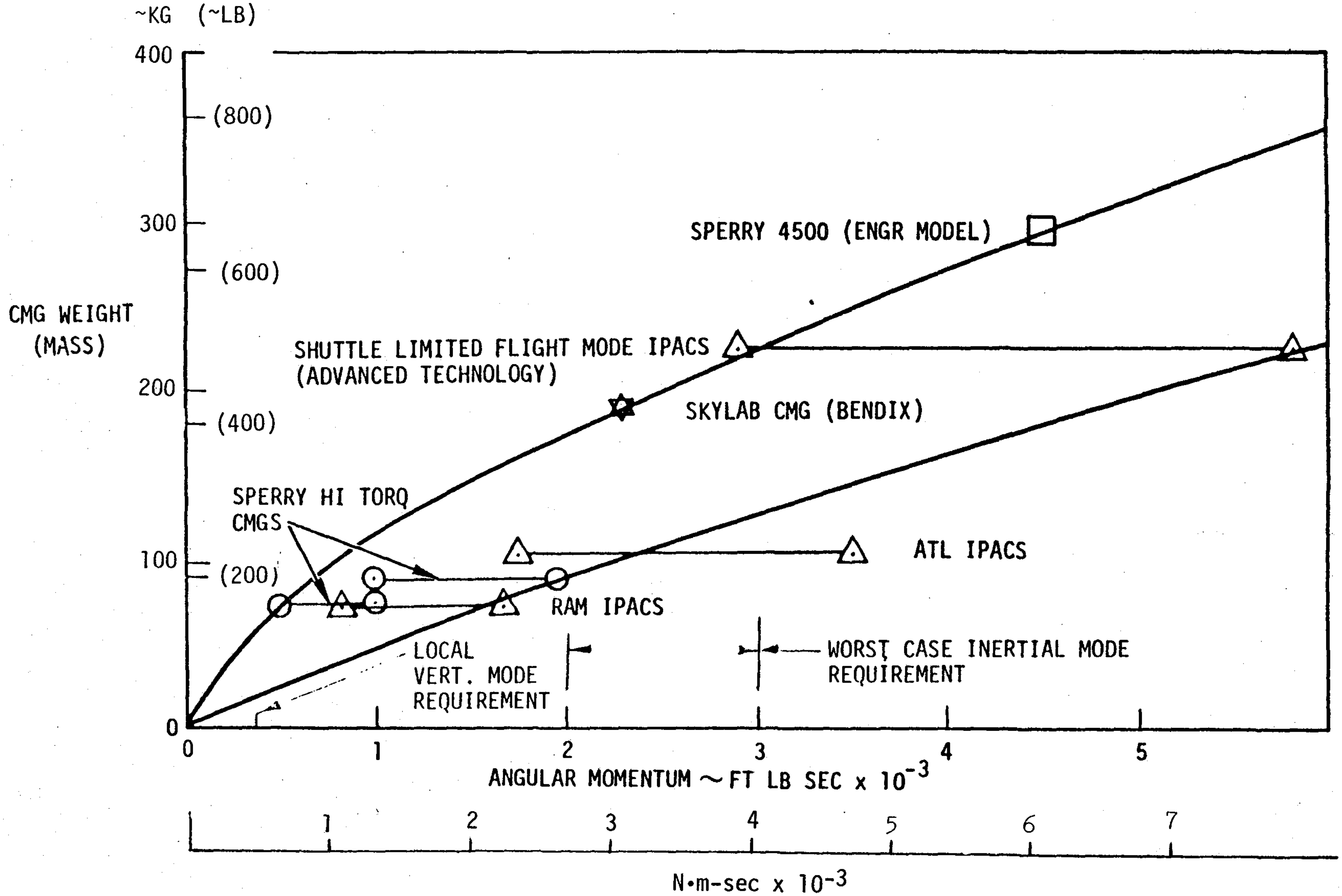
ANGULAR MOMENTUM

XPOP	ANGULAR MOMENTUM (N·m-sec)					
	CYCLIC			SECULAR (PER ORBIT)		
	X	Y	Z	X	Y	Z
LOCAL VERTICAL						
$\beta = 0$	1356.	9491.	6440.	4067.	0.	0.
$\beta = 45$	610.	5288.	3525.	3559.	0.	136.
$\beta = 90$	0.	2963.	2034.	3220.	0.	0.
INERTIAL						
$\beta = 0$	3796.	3874.	1055.	0.	1152.	662.
$\beta = 45$	3796.	1478.	949.	0.	1220.	678.
$\beta = 90$	3796.	339.	1055.	0.	1152.	644.

Double-Gimbaled CMG Mass

The cyclical momentum requirements for XIOP and XPOP attitudes require gyro capability that are within the state of the art. The chart shows that the Skylab size CMG is feasible for this application. A technology issue is the longer life requirement needed to meet the ten-year mission requirement for the erectable platform.

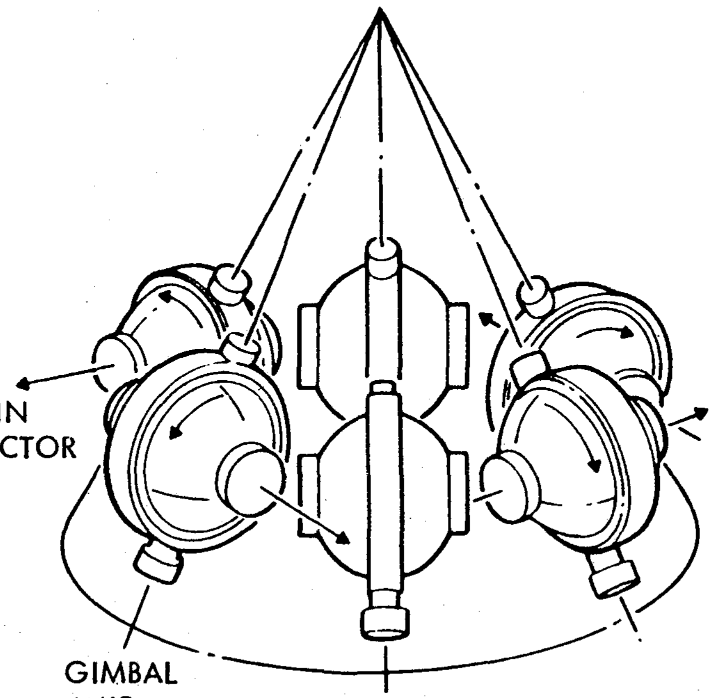
DOUBLE-GIMBALED CMG MASS



Control Moment Gyro Characteristics & Configuration (Representative)

Six SDOF Sperry CMG's in a cone cluster configuration were developed as a representative system to provide the momentum exchange for attitude control for the spacecraft in XIOP attitude. The system was sized to deliver full momentum with five gyros operating and allowing one gyro for redundancy. Assuming a five gyro cluster and a 30^{o} cone angle, the momentum delivered along the X axis which is directed toward the cone apex is (6779 N•m-sec; and the momentum in YZ plane is 11,741 N•m-sec. Without the redundant gyro, the configuration has 7% margin along the X axis and 21% margin in YZ plane. With the utilization of six gyros the momentum margins increase substantially.

CONTROL MOMENT GYRO CHARACTERISTICS & CONFIGURATION (REPRESENTATIVE)



SPERRY CMG CHARACTERISTICS

• CMG WEIGHT (BASE GIMBAL INCLUDED)	88.5 KG
• ANGULAR MOMENTUM	2712 N·m
• POWER REQUIREMENTS	
PEAK GYRO INPUT POWER	600 W
PEAK DURING RUN-UP	200 W
STEADY STATE RUNNING	40 W @ 5300 RPM
• WHEEL OPERATING SPEED	8700 RPM
• SPIN BEARINGS	
BALL BEARINGS	204H DF PAIR
• TORQUER OUTPUT	203 N·m
• CMG SIZE	101.9 x 68.6 x 77.5 CM
• TOTAL WEIGHT (INCL. ELECTRONICS)	93.4 KG

Control Accuracy Budgets

There are several sources of pointing and positional errors that can accumulate and exceed the initial pointing and stability guidelines of 0.01 degree. The budgets shown on this chart are the statistical variances that can be anticipated during the orbital and control operations if there is a centralized pointing reference sensor in the utility services module. The majority of these error budgets can be reduced if the reference sensors are placed in close proximity to the individual payloads.

The actual error budgets are indicated for the payload position on the node nearest the reference sensor on the service module and the worst case where the payloads are furthest away from the reference sensor. Subsequent charts show the derivation of each budget component.

For the erectable platform concepts it is anticipated that the pointing accuracy will range from 0.024 degree of the nearest payload to 0.201 degree for the worst-case payload. Platform distortion has been significantly reduced by using a graphite epoxy platform structure with low thermal distortion. It is interesting to note that the major pointing error arises from the thermal "hot dogging" of a side mounted aluminum pallet. If all pallets are node mounted or the pallet table is a graphite epoxy, this error is reduced to 0.029 degree.

CONTROL ACCURACY BUDGETS

ERROR CONTRIBUTIONS	DESIGN/MISSION SOLUTIONS	ACCURACY BUDGET (DEG) NEAREST PAYLOAD	ACCURACY BUDGET (DEG) WORST-CASE PAYLOAD*
• ATTITUDE REFERENCE DETERMINATION	POSITIONING OF SENSORS SINGLE OR MULTIPLE	0.0015	0.0015
• THERMAL DEFORMATION • SHORT-TERM TRANSIENTS • STEADY STATE	MATERIAL SELECTION THERMAL COATING THERMAL CONTROL	0.005	0.016 (0.20)
• MANUFACTURING & ASSY	TOLERANCE BUILDUP ON-ORBIT MEAS. OF BIAS FIGURE CONTROL	0.012	0.013 (0.010)
• CONTROL ERRORS • BANDWIDTH • FLEXIBLE DYNAMICS INTERACTION	DESIGN STIFFNESS CONTROL COMPLEXITY	0.020	0.020
	R.S.S. TOTAL ERROR	0.024	0.029 (0.201)$^{\Phi}$

*Payload position furthest away from centralized attitude reference position.
( )Side-mounted aluminum pallet/payloads
ΦTotal error reduced to 0.028° using graphite epoxy pallets or tables side-mounted.

High-Precision Attitude Determination Systems and Sensors

The charts present a summary of various attitude determination systems that have potential application. The system selected for this spacecraft is the TRW Precision Attitude Determination System (PADS) with complete redundancy. There will be four star trackers, eight gyros (4 two-pack) and two CCC 469 R^2 computers. The system will provide the attitude and attitude rate information to be used in generating the control signals with an accuracy of 5 arc-sec or 0.0015 degree.

HIGH-PRECISION ATTITUDE DETERMINATION SYSTEMS AND SENSORS

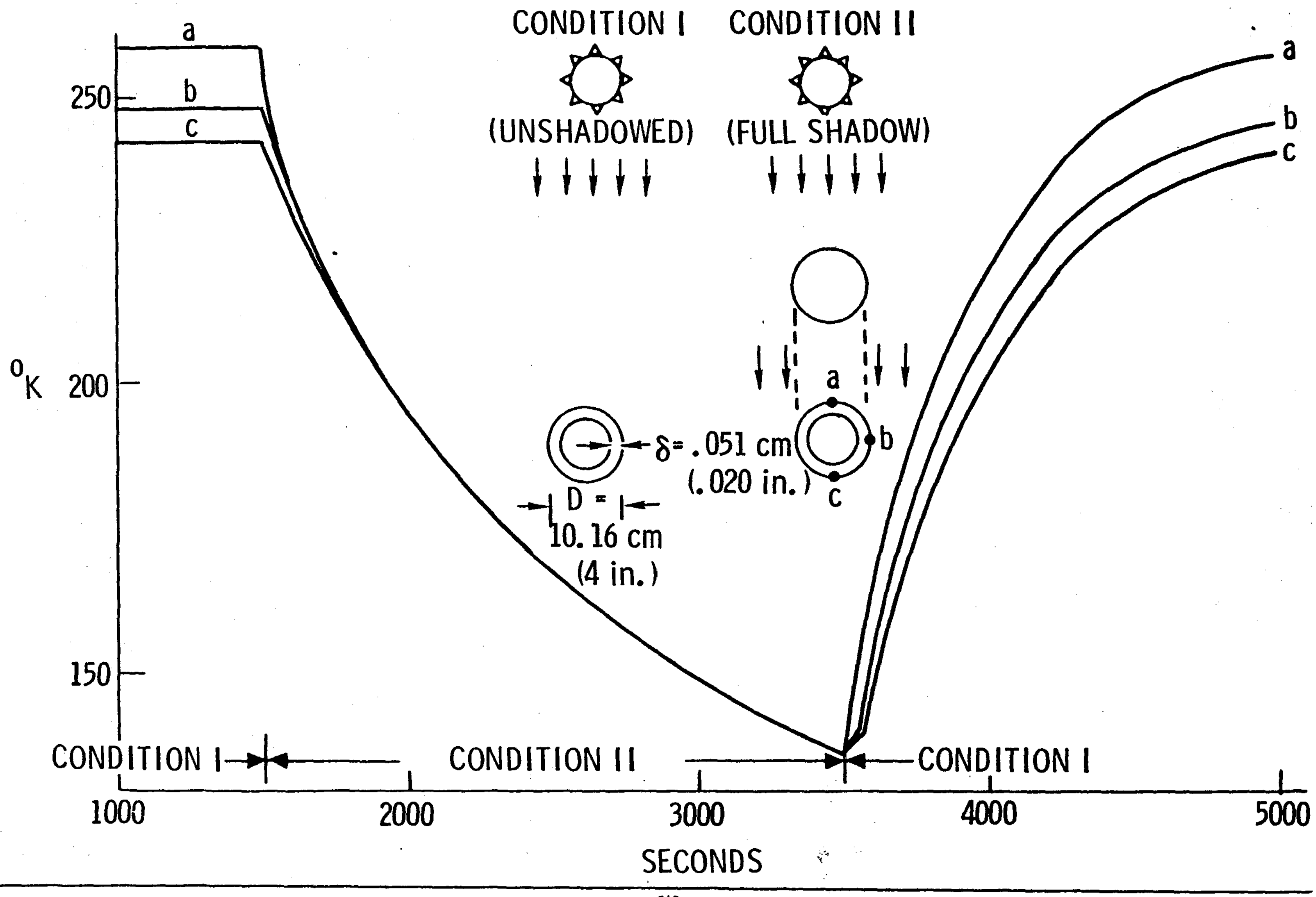
CONFIGURATION	3-AXIS RSS ACC. (SEC)	WEIGHT (KG)	POWER (W)	COMMENTS
BBRC SST	(10.0)*	8.0*	18.0*	NASA STANDARD
TRW HEAO-B	(3.0)*	45.8	93.7	HONEYWELL PHOTON ST (3), BENDIX 64-PM-RIG (4), CDC $469R^2$
HONEYWELL PADS	18.0	24.8	73.8	HONEYWELL IMU, CS, DIEU; CDC $469R^2$, P80-1, DMSP, TIROS N
BENDIX SSU	(10.0)*	4.5*	6.6*	STAR SCANNER, LOW-COST VERSION OF EXISTING HARDWARE
MM SPACE SEXTANT	0.4	25.4	120.0	MM ST (2); KEARFOTT ARU; LITTON LC-4516Δ, P89-2 EXPMT; NAVIGATION ALSO
TRW PADS	(5.0)*	32.7	81.0	TRW ST (2); BENDIX 64-PM-RIG (4), CDC $469R^2$, NASA/LaRC
JPL STELLAR	(1.0)*	3.2*	5.0*	CCD CHIP SENSOR, ENGR MODEL—NASA/AMES
ROCKWELL HAARS	2.0	6.4*	19.5*	CCD MOSAIC SENSOR, PROCESSOR IN DESIGN—USAF
IBM PRAIS	10.0	39.0	75.0	IBM PRAIS, COMPUTER, IRU, ILT USEA ATS-F
BBRC CT-401	(10.0)*	6.0*	4.0*	STAR TRACKER; SAS-C, HEAO-A & -C

*SENSOR ONLY; ( )SENSOR MEASUREMENT ACCURACY

Transient Temperature Response of Thin-Walled Aluminum Tubes

The structural platform is composed of uniform length cylindrical struts connected by standarized unions to form the pentahedral area platform. It is possible that at certain sun orientations for the struts on one surface to shadow corresponding struts on the other surface. A preliminary transient temperature response was derived for thin walled aluminum tubes. Temperature changes from 250^{o}K in full sunlight to about 140^{o}K in full shadow will be experienced by the struts. Thermal gradient around the tubes will amount to about 20^{o}K.

TRANSIENT RESPONSE OF THIN-WALLED ALUMINUM TUBES



Temperature Distortions

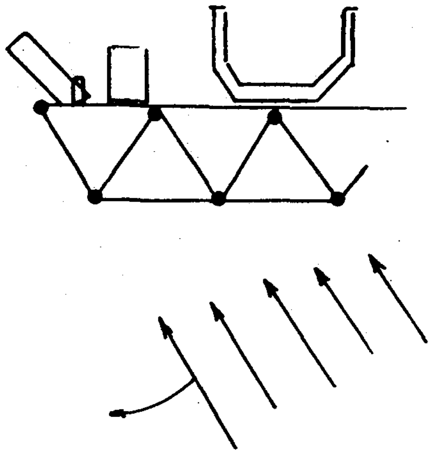
For the adverse condition of the lower platform surface shadowing the upper surface, the shadow time should be less than 0.4 minute due to the rotation rate in the low earth orbit of 463 km. With this short shadow time, the transient temperature response from the previous chart indicates that it is possible to get a thermal difference between top and bottom surfaces of about 10°C.

Due to the high thermal expansion of the aluminum struts and their long length, the maximum angular rotation of the bay furthest from the service module will be 0.157 degree. This magnitude of distortion is too large for the platform pointing accuracy requirements. A recommended alternate apporach is to use graphite epoxy material for the struts; this material has two orders of magnitude lower thermal expansion coefficient. It is recognized that there will be a high ΔT across the surfaces. A maximum pointing distortion of 0.0026 degree has been allowed for the graphite platform structure.

The ERNO aluminum pallets are subjected to large thermal gradients throughout their depth when one side looks at deep space and the other has full sunlight. This thermal gradient will induce a "hot dog-going" effect of opening out of the pallet's opposite top members. Pallets that are supported at one edge by the two trunnion attachments will have the bottom surface of the pallet, where the payloads are mounted, rotated about 0.19 degree relative to the pallet support attachment plane. This rotation is significantly reduced for pallet or platform tables made of graphite epoxy or when the pallet is attached at the keel fitting center line to one of the platform node points.

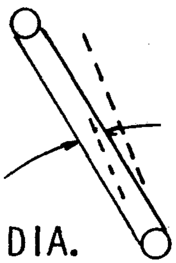
The pallets on top of the node nearest to the utilities module will experience the thermal distortion of 0.005 degree due to the struts attaching platform to the core module, core module itself, and negligible distortion of the center supported pallet. The worst condition will be for the pallets side-mounted from the corner farthest from the core module. This 0.20-degree deflection is the contribution from the platform struts, struts attaching platform to core module, core module, and the pallet "hot dogging."

TEMPERATURE DISTORTIONS



SUN RAY
ROTATION
$\approx 4°$/MIN.

10.2 cm (4-in.)



$\Delta\beta$

10.2 cm (4-in.) DIA.

SHADOW TIME $<$ 0.4 MINUTES

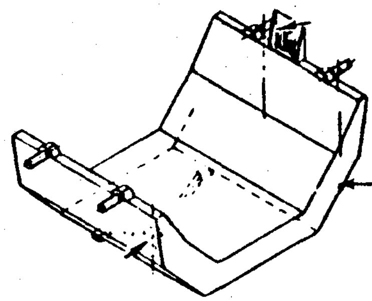
$\Delta T \approx 18°R$ (10^0C)

9 BAYS

$\Delta\beta_{AL.AL.} \sim 0.157°$

$\Delta\beta_{GR/EPOXY} \sim 0.0026°$

PAYLOAD PALLET
(SPACELAB CONFIG.)



AL. AL.

STEADY-STATE
TEMPERATURE

$T_1 \approx 540°R$ (282^0K)

RADIATION ONLY

$T_2 \approx 405°R$ (207^0K)

$\Delta T = 135°R$ (75^0C)

ALLOW FOR CONDUCTION, ETC.

$\Delta T \sim 50°R$

$\Delta\beta_{AL.AL.} \sim 0.19°$

$\Delta\beta_{GR/EPOXY} \sim 0.0031°$

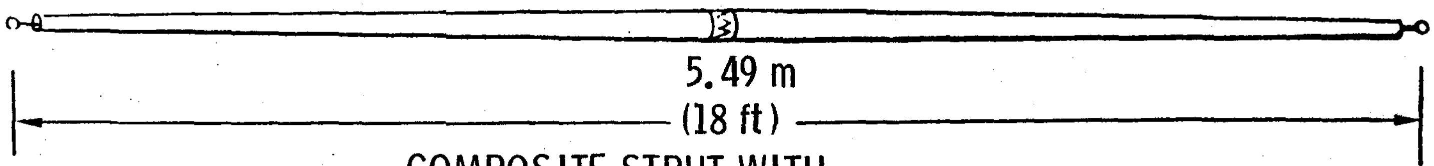
Manufacturing and Assembly Tolerances

Other sources of errors are due to manufacturing and assembly tolerances. The long graphite epoxy struts have to be trimmed and the end fittings installed. Under carefully controlled conditions, with the struts and end fittings installed in master jigs, it could be possible to adjust the length tolerance to ±0.005 cm for the struts for the platform structure.

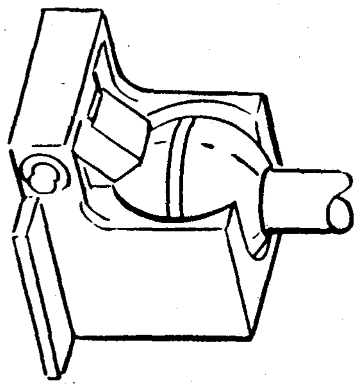
Each joint consists of a male and female connector that has to be easily assembled in space, while at the same time being tolerant to misalignment. As an optimistic assessment we will assume that these mated surfaces can be controlled to the tolerances associated with those of a "running fit" which is ±0.005 cm. Considering the root sum squared (RSS) errors of the top or bottom surfaces of the platform, the maximum error for the 5 bay configuration of the area nodal platform is 0.0057 degrees. The depth of the platform (3.861 m) helps reduce the angular displacement, since the strut length error is assumed to be independent of strut length.

There are additional misalignment errors between the platform, adapter structure and the utility service module. These misalignment tolerances are combined with the payload assembly tolerance discussed next and result in 0.012 degrees for the top node mounted pallets along the platform edge furthest away from the core module.

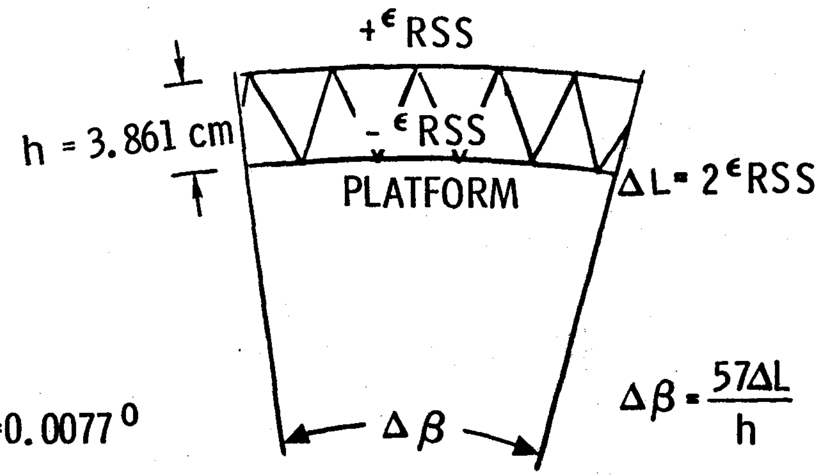
MANUFACTURING & ASSEMBLY TOLERANCES



COMPOSITE STRUT WITH
ADJUSTABLE END FITTINGS
CONTROLLED TO JIG LENGTH
ADJUSTMENT TOLERANCE ± 0.005 cm



BALL & SOCKET JOINT
EASILY ASSEMBLED ON
ORBIT. RUNNING FIT
TOLERANCES $\pm$ 0.005 cm



PENTAHEDRAL
STRONGBACK
9 BAYS

$\epsilon_{RSS} = \sqrt{9x(.005)^2x3} = 0.026$ cm; $\Delta\beta = 0.0077^0$

PENTAHEDRAL
AREA/NODAL
5 BAYS

$\epsilon_{RSS} = \sqrt{5x(.005)^2x3} \approx 0.0194$ cm $\Delta\beta = 0.0057^0$

Payload Assembly Tolerances

The attachment fitting and assembly operations of the payload and/or pallet to the platform are another source of potential pointing errors if the payload pointing reference system is centralized in the service module. Depending upon the attachment method and position, these errors can vary by at least an order of magnitude.

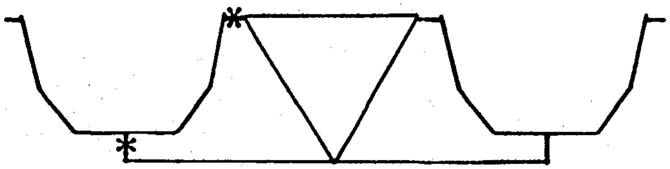
Side mounted payloads can be attached by the two upper bridge type fittings of the pallet and a single steadying strut to support the underneath of the pallet/payload. The bridge trunnions will abut against the platform structural interface. An optimistic tolerance allowance for this abutting during on-orbit assembly would be ±0.013 cm, which considering the short distance between the two trunnions will result in a pointing error of approximately 0.007 degrees. The longer base of the bottom support strut can be considered as a strut with two end fittings with tolerances of 0.005 cm and a resulting error of 0.0009 degrees.

For pallets/payloads that are mounted at the nodes with the probe and drogue fitting into the standard union joint, the tapered fitting was assumed to be equivalent to a running fit tolerance of ±0.002 inches. The maximum amount of clearance below the pallet in the orbiter is 38 cm to allow for the probe fitting. Using this as a base length, the pallet/payload could have a point error as large as 0.0106 degrees. It is interesting to note that the node mounted pallet has the largest percentage errors due to manufacturing and assembly operations, but the lowest due to thermal distorting.

The payload assembly tolerances are combined with the platform tolerances as discussed on the previous page.

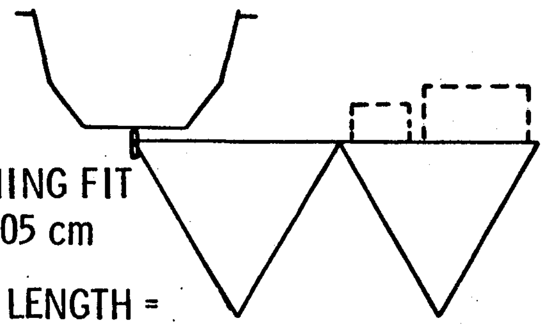
PAYLOAD ASSEMBLY TOLERANCES

RUNNING FIT
ϵ = .005 cm



STRUT ϵ = .005 cm

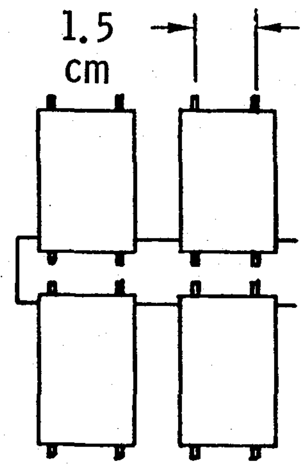
ϵ RSS = .0086 cm
$\Delta\beta$ = 0.0009^{0}



RUNNING FIT
ϵ =.005 cm

BASE LENGTH =
38 cm

ϵ RSS = 0.0071 cm
$\Delta\beta$ = 0.0106^{0}



PALLET
BRIDGE FITTINGS

POSITION TOLERANCE
0.013 cm

ϵ RSS = 0.018 cm
$\Delta\beta$ = 0.0069

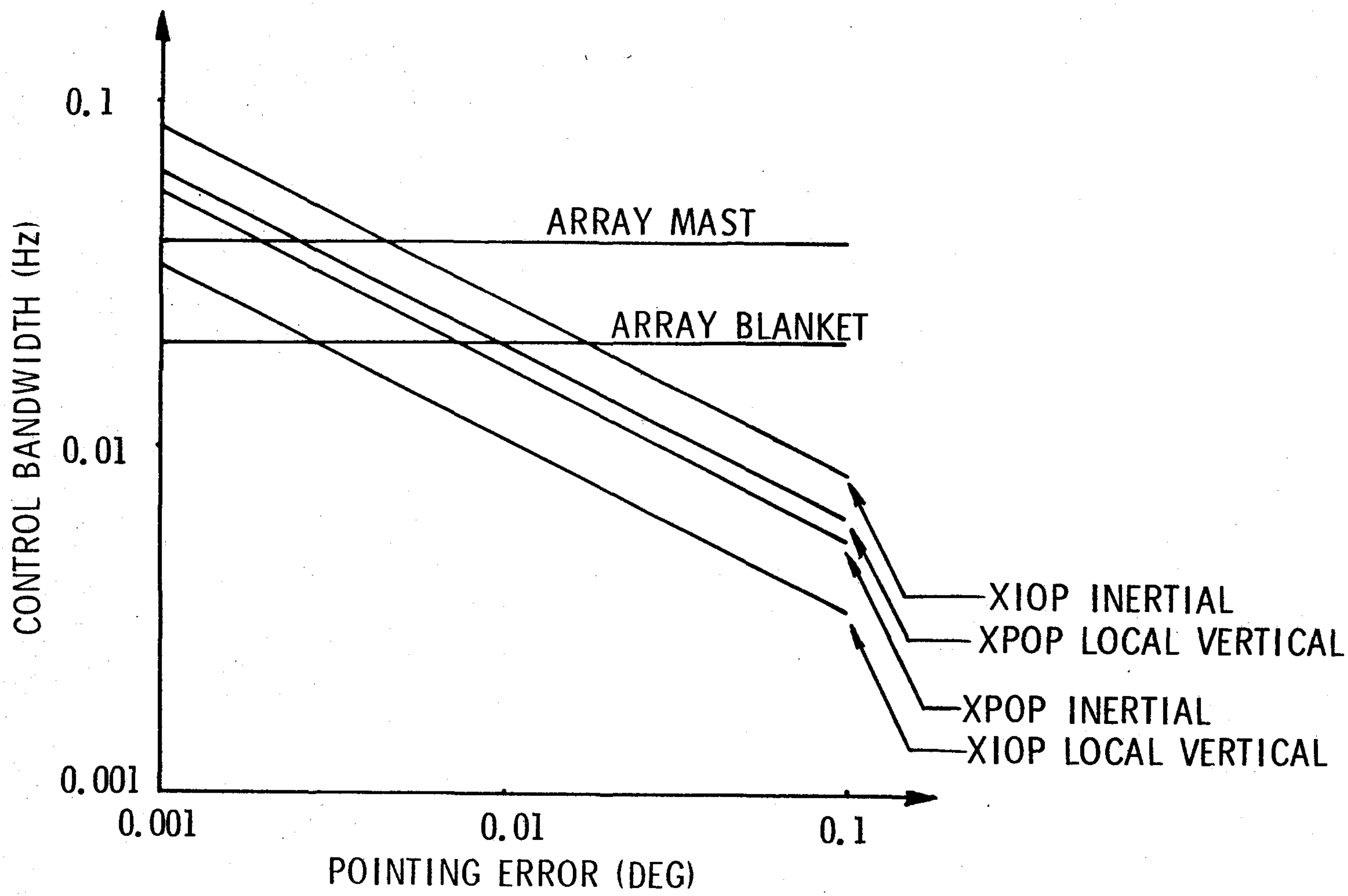
Control System/Solar Array Interaction

A pointing error requirement of .01 degrees shows that there is a control system/ structural frequency interaction problem. The most compatible attitude is XIOP local vertical. Unfortunately the platform is required to operate in inertial flight mode and then the bandwidth separation is greatly reduced. In this attitude a pointing error of .01 degrees results in a frequency separation of 2 (i.e., the bending frequency of array blanket is two times the control bandwidth). To maintain a frequency separation of two for all the flight attitudes, the pointing error must be greater than .07 degrees. However, a frequency separation of 10 would be preferable.

Due to the flexibility and multiple joints of the mast and solar array blanket there could be significant uncertainties in the platform; the bending and torsional frequencies could be significantly lower causing severe interaction problems for the control system.

Based on preliminary understanding of the anticipated structural frequencies and a slight over lap of control bandwidth and structural frequencies it appears optimistic that control errors can be held to about 0.02 degrees. This area is suspect due to the uncertainties of predicting the lower natural frequencies of these highly flexible, multi-joint structural components.

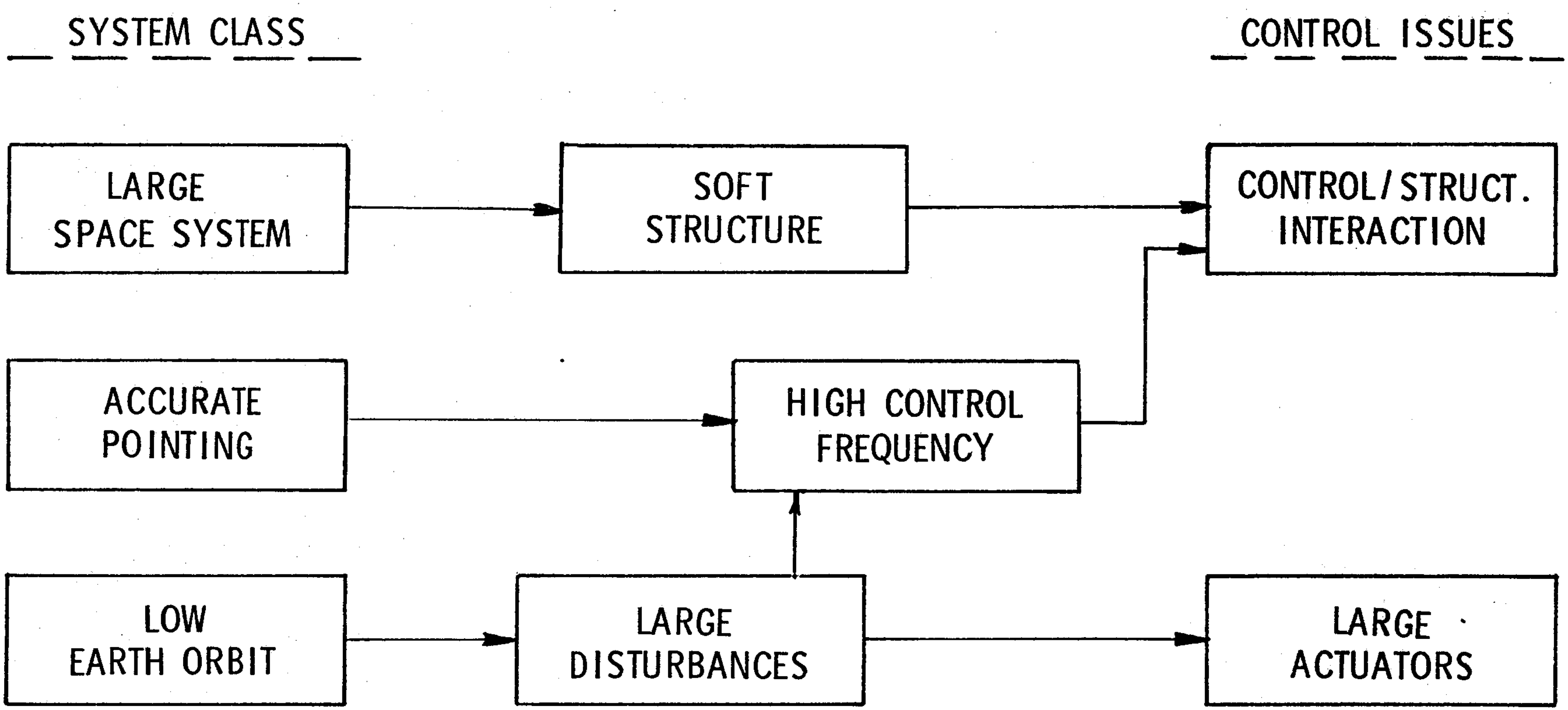
CONTROL SYSTEM/SOLAR ARRAY INTERACTION



Control Technology Issues

Large space structures pose new challenges to control system design because of their inherent low structural stiffness. The control problems are compounded by requirements for accurate pointing and the figure control. These requirements increase the control/structural frequency interaction. In LEO large structures experience large environmental torques (gravity gradient and aerodynamic) which gives rise to larger actuator forces to maintain spacecraft attitude.

CONTROL TECHNOLOGY ISSUES



CHALLENGE

√ FIND SOLUTIONS WHICH MINIMIZE SPACE CONSTRUCTION COMPLEXITY

Attitude and Control System Requirements

The chart summarizes the basic requirements for the attitude control subsystem as they pertain to the baseline erectable platform.

ATTITUDE AND CONTROL SYSTEM REQUIREMENTS

- POINTING REFERENCE AND TRACKING SENSORS WITH 5 ARC-SEC ACCURACY
- MOMENTUM STORAGE DEVICES WITH 2712 N·m-sec CAPACITY (TOTAL 6 CMG's)
- CONCEPT FOR DESATURATING CMG's EACH ORBIT—SYSTEM SHOULD NOT PRODUCE UNDESIRABLE CONTAMINANTS (DEPLOYABLE BOOMS—RCS)
- CONTROL LAWS THAT ARE CAPABLE OF HANDLING (PLATFORM, ARRAY) BUILDUP WITH CONSIDERABLE PLANT (PLATFORM, ARRAY) UNCERTAINTIES

Attitude Control Subsystem Block Diagram

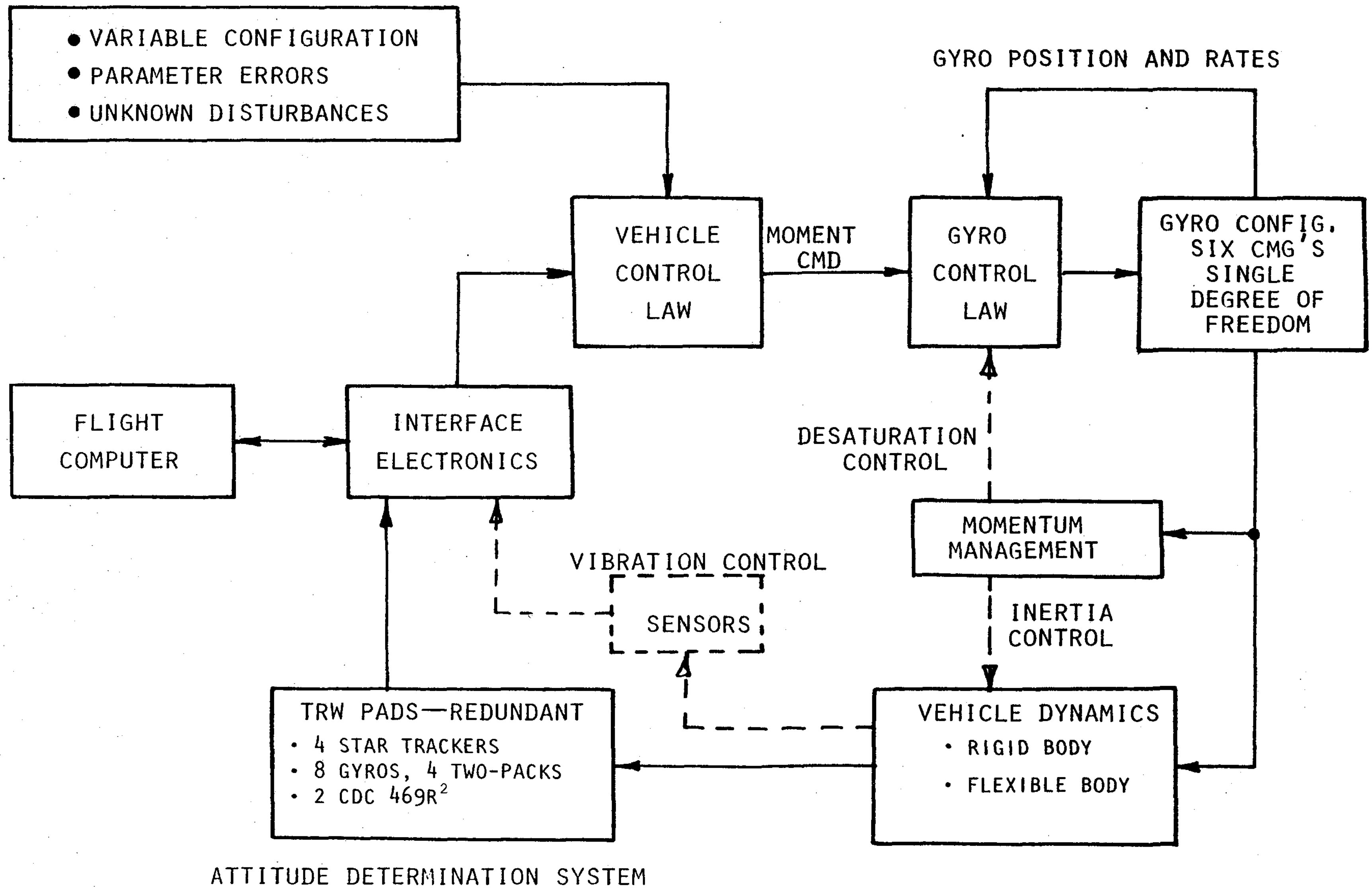
A functional block diagram for the attitude control subsystem (ACS) is shown on the chart. The ACS consists of three functional elements:

- o Pointing and Stability
- o Attitude Determination
- o Momentum Management

The pointing and control subsystems utilize SDOF CMG momentum exchange for vehicle control and the information from the attitude determination system is used to generate the torque commands. The control laws are further complicated by the variable geometry of the spacecraft, parameter uncertainties, and unknown disturbances. If vibration control becomes necessary because of the accuracy requirements imposed by the experiments, the control system increases in complexity with an increase in actuators and sensors and algorithms for modal control.

The attitude determination system (ADS) is a completely redundant system employing star tracker, gyros and CDC computer. ADS will provide rate derived information with the attitude reference. The momentum management issues has not been fully resolved. If the g loads and contamination requirements of experiments are such that RCS is ruled out for saturation control, then the use of booms for inertia balancing, the use of magnetic torques, or a combination of these techniques may be attractive alternatives.

ATTITUDE CONTROL SUBSYSTEM BLOCK DIAGRAM



Control Subsystem Trades/Selection

The next seven charts present the control subsystem trades and technology candidates indicating the various issues involved and expected results from the development programs outlined.

As space structures become larger and more flexible, the structures can no longer be reliably tested in a 1g environment. Thus, there is uncertainty concerning the dynamics of the system. The effects of these uncertainties must be better understood because of the increasingly stringent requirements for precision attitude control and vibration of the flexible modes.

There is an increase in the complexity for the active control of flexible vehicles. The burden of the control system is to be tolerant of these uncertainties which arise from the uncertainties in mode shapes and damping, mode truncation, and environmental disturbance uncertainties. The control system must also be sensitive to the varying configuration of the spacecraft. Various control system techniques must be investigated using classical theory to modern control theory which potentially provides more sophisticated controllers.

Large spacecrafts operating in LEO are subject to substantial gravity gradient and aerodynamic torques. These torques give rise to large momentum storage and desaturation requirements. Coupled with the size of CMG, the operating time must be extended to meet orbital life time requirement. Because of the contamination criteria, using gravity gradient booms as a momentum desaturation technique looks attractive. However, the RCS system must be explored to determine if such systems are efficient and satisfy the contamination criteria.

The strongest requirements for attitude vibration control lead to the need of exploring techniques that include vibration isolators and/or dampers, low frequency accelerometers, and laser optical systems. Vibration isolators are an attractive approach to uncouple the dynamic response between segments of the spacecraft. Studies on special spacecraft show that structural dampers can stabilize vibration response. It will likely be necessary to learn the actual behavior of the structure in flight before giving full control authority to the designed controller. From the identification of structural modes, the final adjustments can be made to the controller. The behavior of accelerometers with the capability of measuring low frequency vibrations must be assessed. If the platform/payload interface require very accurate pointing, the development of a more accurate measurement system must be developed using laser technology.

CONTROL SUBSYSTEM TRADES/SELECTION

COMPONENT/ TECHNOLOGY	ALTERNATES	SELECTION	TECHNOLOGY LEVEL
STRUCTURAL SYSTEM IDENTIFICATION	MODEL IDENTIFICATION AND ANALYSIS	LINEAR CURRENT S.O.A.— DESCRIPTION MIGHT NOT REPRESENT TRUE BEHAVIOR; NON-LINEAR MODELING IS PREFERRED APPROACH, BUT MORE COMPLEX MODEL	1
	MODEL UNCERTAINTIES AND BUILDUP	PERFORMANCE & MISSION FLEXIBILITY	1
CONTROL SYSTEM MODELING	CLASSICAL THEORY	SIMPLEST APPROACH—BUT MIGHT BE IMPRACTICAL	1
	MODERN THEORY	MORE COMPLEX—BUT MIGHT BE DICTATED BY FLEX. STRUCTURE	1
MOMENTUM DEVICES • STORAGE	CMG's	EXTENDED S.O.A.	2
	IPAC's	—	-
• DESATURATION	GRAVITY-GRADIENT BOOMS	ATTRACTIVE—NO CONTAMINATION	2
	RCS	COLD GAS—VERY HEAVY—POSSIBLE CONTAMINATION	3
	MOVABLE AERODYN. SURFACES	—	-
	CONSTRAINED FLT. REORIEN.	—	-
CONTROL IMPLEMENTATION	FORCE ACTUATORS	LOW RISK	3
	VIBRATION ISOLATORS	CONTROLS FOR FLEX. ARRAYS	2
	STAR TRACKER	NASA STANDARD-TYPE SYSTEM	3
MEASUREMENT SENSORS	ACCELEROMETERS	LOW-ACCELERATIONS—	1
	LASER-OPTICAL REFLECTORS	SPACE-RATED SYSTEM NEEDED—PROVIDES MEASURE OF STRUCTURAL DISTORTION	2

STRUCTURAL SYSTEM IDENTIFICATION

OBJECTIVE

- DEVELOP AND DETERMINE SPACECRAFT MODELS THAT REALISTICALLY DESCRIBE THE STRUCTURAL DYNAMICS FOR LARGE SPACE PLATFORMS

TECHNOLOGY ASSESSMENT

- MODELING OF MULTI-JOINT/HIGHLY FLEXIBLE STRUCTURE NOT FULLY UNDERSTOOD
- COMPLEX MODELS FOR STRUCTURAL ANALYSIS, BUT TECHNIQUES NOT ACCEPTABLE FOR CONTROL MODELS
- SIMPLE LINEAR AND NONLINEAR MODELS USED FOR CONTROL SYSTEM ANALYSIS
- SCALE MODEL TEST OF JOINT OR ELASTIC NONLINEARITY IS A CONCERN

APPROACH

- DETERMINE IF JOINT OR ELASTIC NONLINEARITY IS A PROBLEM AND DETERMINE EFFECTS OF THESE NONLINEARITIES ON MODEL BEHAVIOR
- INVESTIGATE INFLUENCE OF INHERENT JOINT FRICTION ON STRUCTURAL DAMPING
- DEVELOP LINEAR & NONLINEAR MODELS FOR CONTROL SYSTEM EVALUATION AND DESIGN
- COMPARE MATHEMATICAL MODEL WITH ACTUAL TEST RESULTS

EXPECTED RESULTS

- BETTER UNDERSTANDING OF VERY FLEXIBLE STRUCTURE BEHAVIOR
- NEW INSIGHTS IN THE CONTROL OF FLEXIBLE STRUCTURE, THEORY LIMITATIONS, MODELING PROBLEMS, ETC.

CONTROL SUBSYSTEM TECHNOLOGY CANDIDATES

MODEL UNCERTAINTIES

OBJECTIVES

- INVESTIGATE AREAS OF MODEL UNCERTAINTIES ARISING FROM MODEL SIMPLIFICATION
- ISOLATE MODEL CHANGES ARISING FROM STRUCTURES/ MATERIAL AGING—PAYLOAD BUILDUP

TECHNOLOGY ASSESSMENT

- STRUCTURE WITH SEVERAL JOINTS—50% MODES UNCERTAIN
- MULTI-JOINT/HIGHLY FLEXIBLE STRUCTURE WITH SLACK DEADBANDS; ANALYSES NEED DEVELOPMENT

APPROACH

- IDENTIFY THE SOURCES, MAGNITUDES, AND EFFECTS OF STRUCTURAL MODEL UNCERTAINTIES ON CONTROL RESPONSE

EXPECTED RESULTS

- RANGES OF MODEL BEHAVIOR WITH MISSION WHICH REQUIRE CONTROL MODELING

CONTROL SYSTEM MODELING

OBJECTIVES

- TO IDENTIFY STRUCTURAL FREQUENCY WITHIN CONTROL BANDWIDTH
- DEVELOP CONTROL DESIGN MODEL (MODAL REDUCTION)

TECHNOLOGY ASSESSMENT

- CURRENT CONTROL BANDWIDTH—1 ORDER SEPARATION FROM STRUCTURAL FREQUENCY
- FILTERING OUT OF 1 OR 2 LOWER STRUCTURAL FREQUENCIES
- ADVANCED TECHNIQUES NEEDED FOR MANY FREQUENCIES WITHIN CONTROL BANDWIDTH

APPROACH

- PERFORM ESTIMATE OF MODEL STRUCTURAL FREQUENCIES

  ESTIMATE CONTROL BANDWIDTH NEEDED TO MEET ACCURACY REQUIREMENT

EXPECTED RESULTS

- DEGREE OF OVERLAP BETWEEN CONTROL BANDWIDTH AND STRUCTURAL FREQUENCIES
- INSIGHT INTO ACCEPTABLE CONTROL THEORY

CONTROL SUBSYSTEM TECHNOLOGY CANDIDATES

CLASSICAL THEORY—NOTCH FILTERS

OBJECTIVE

- TO DETERMINE WHETHER CLASSICAL THEORY CAN BE ADAPTED TO THE HIGHLY FLEXIBLE PLATFORM VEHICLE

TECHNOLOGY ASSESSMENT

- CURRENT TECHNIQUES LIKE SEPARATION CONTROL BANDWIDTH & STRUCTURAL FREQUENCY
- FILTER OUT 1 OR 2 FREQUENCIES
- PLANT UNCERTAINTIES LIMITED

APPROACH

- DEVELOP CONTROL LAWS AND SUFFICIENT NOTCH FILTERS TO BYPASS LOWER STRUCTURAL FREQ.
- IMPLEMENT COMPONENT CONTROL SIMULATION WITH GROUND TESTS AND FLIGHT VERIFICATION

EXPECTED RESULTS

- SIMPLIFIED CONTROL APPROACH TO PROVIDE ADEQUATE POINT ACCURACY AND STABILIZATION
- DETERMINATION OF THE PRACTICAL ACCEPTABILITY OF CLASSICAL CONTROL THEORY

MODERN CONTROL THEORY

OBJECTIVE

- TO DETERMINE MODAL SHAPE DESCRIPTION REQUIRED FOR ADEQUATE MODEL FOR CONTROL THEORY

TECHNOLOGY ASSESSMENT

- MANY MODES CAN BE CONTROLLED BY A FEW ACTUATORS AND SENSORS
- POSSIBLE TO HANDLE SEVERAL SYSTEM CONFIGURATIONS

APPROACH

- DEFINE MODEL REDUCTION ACCEPTABLE TO ON-BOARD CONTROL & DESCRIPTIVE OF ACTUAL STRUCT. RESPONSE
- EST. MEAS. ACCURACY & SENSOR POSITIONING REQ'D TO DESCRIBE MODAL SHAPES UNDER MISSION CONDITIONS

EXPECTED RESULTS

- MODAL MODEL & SENSOR REQUIREMENTS FOR MODERN CONTROL THEORY
- TYPES, MAGNITUDE, AND FREQUENCY OF POSITION MEASUREMENTS REQUIRED

CONTROL SUBSYSTEM TECHNOLOGY CANDIDATES

CMG STORAGE DEVICE

OBJECTIVE

- LARGE LONG-LIFE ADVANCED CMG CLUSTERS

TECHNOLOGY ASSESSMENT

- CURRENT CLASS OF SKYLAB CMG
- CAPACITY, APPROX. 2000 FT-LB-SEC
- S.O.A. WITH SLIGHT ENHANCEMENT

APPROACH

- APPLY TECHNOLOGY IMPROVEMENTS FOR LONGER OPERATIONAL LIFE FOR BEARINGS, ETC., FOR LARGER CAPACITY

EXPECTED RESULTS

- LARGE LONG-LIFE SPACE-RATED CMG FOR PLATFORM APPLICATIONS

GRAVITY-GRADIENT BOOM

OBJECTIVE

- DETERMINE THE CAPABILITY OF FLEXIBLE STRUCTURAL BOOMS FOR DESATURATING CMG's

TECHNOLOGY ASSESSMENT

- SMALLER BOOMS, APPROX. 10 FT
- SMALLER MASSES, APPROX. 10 LB
- USED WITH "RIGID" STRUCTURES

APPROACH

- EXTEND G.G. BOOM CAPABILITY TO EFFECTIVELY ALTER THE MASS PROPERTIES OF EXTRA-LARGE SPACE PAYLOADS
- DEVELOP OPERATIONAL SIMULATION OF G.G. BOOM TO DESATURATE CMG's AND NOT CONTINUALLY OBSTRUCT PLATFORM FIELD OF VIEW

EXPECTED RESULTS

- CONTROL DUMPING SYSTEM THAT DOES NOT CONTAMINATE SPACE OR EXPERIMENTS

REACTION CONTROL SYSTEM

OBJECTIVE

- IDENTIFY RCS SYSTEMS THAT ARE EFFICIENT & ACCEPTABLE TO CONTAMINATION CRITERIA OF PAYLOADS

TECHNOLOGY ASSESSMENT

- COLD GAS—NITROGEN—LOW I_{SP}
  ICE PARTICLES ON SUPER-COOLED DETECTORS
- HYDRAZINE—MINUTE PARTICULATES
  CATALYST SPUTTERING—INFREQUENT OCCURRENCE
  —IMPACT NEGLIGIBLE

APPROACH

- DEFINE PRACTICAL CONTAMINATION ENVIRONMENT REQUIREMENTS FOR POTENTIAL SPACE PLATFORM EXPERIMENTS
- CONSIDER DEGREE OF CONTAMINATION FROM RCS CANDIDATES & THEIR SPACIAL POSITIONING & DESIGN TO MINIMIZE EFFLUENTS TO ACCEPTABLE LEVELS

EXPECTED RESULTS

- PRACTICAL RCS THAT WILL BE ACCEPTABLE TO THE MAJORITY OF SPACE PLATFORM EXPERIMENTS

VIBRATION ISOLATORS

OBJECTIVE

- DETERMINING WHETHER ISOLATORS CAN BE USED TO CONTROL FLEXURAL DAMPING OF LARGE ROTATING SOLAR ARRAYS

TECHNOLOGY ASSESSMENT

APPROACH

- PERFORM GROUND TESTS TO VERIFY DAMPING SYSTEMS APPLIED TO FLEXIBLE MASTS AND ARRAYS

EXPECTED RESULTS

- SYSTEM TO ISOLATE FLEXIBLE SOLAR ARRAY STRUCTURE FROM PLATFORM DYNAMICS

CONTROL SUBSYSTEM TECHNOLOGY CANDIDATES

MEASUREMENT SENSORS—ACCELEROMETERS

OBJECTIVE

- DETERMINE CAPABILITY OF LOW-FREQUENCY ACCELEROMETERS FOR DEFLECTION, VELOCITY AND ACCELERATION MEASUREMENTS

TECHNOLOGY ASSESSMENT

- S.O.A. ACCELEROMETERS MEASURE DOWN TO 2-10 Hz—STARAIN GAUGE ACCELEROMETERS FOR LOWER FREQUENCIES

APPROACH

- DETERMINE ACCELERATION LEVELS EXPERIENCED BY THE LOW FREQ. STRUCTURES
- DEVELOP APPROACHES FOR INSTALLING & RECORDING IN-FLIGHT STRUCT. DISTORTIONS

EXPECTED RESULTS

- ACCELEROMETERS TO MEASURE IN-FLIGHT STRUCTURAL BEHAVIOR

LASER-OPTICAL SYSTEM

OBJECTIVE

- DEVELOP TECHNOLOGY FOR A SPACE-RATED DISPLACEMENT MEASURING SYSTEM

TECHNOLOGY ASSESSMENT

- GROUND-BASED COMPACT THEODOLITE SYSTEM CURRENTLY AVAILABLE
- SPACE RATED SYSTEM REQUIRED WITH HIGHER ACCURACY

APPROACH

- DETERMINE SYST. REQMTS & COMPONENTS FOR ACCURATE MEASURING SYSTEM
- GROUND TEST SYSTEM TO DETERMINE MEAS. ACCURACY
- FLIGHT TEST SPACE-RATED SYSTEM

EXPECTED RESULTS

- SPACE-RATED LASER-OPTICAL SYSTEM FOR PRECISE DISPLACEMENT MEASUREMENTS

Control Subsystem Technology Program Planning

The highlights of the technology program schedule are shown on the chart. The schedule is straight forward and it is viewed that the time is adequate to explore and develop the various control technology issues.

CONTROL SUBSYSTEM TECHNOLOGY PROGRAM PLANNING

	1979	1980	1981	1982	1983	1984	1985	1986
			PHASE C/D GO-AHEAD			PLATFORM LAUNCH		
<u>STRUCTURAL SYST. IDENTIF.</u> NONLINEAR MODELING MODEL UNCERTAINTIES		115K						
<u>CONTROL SYSTEM MODELING</u> MODEL REDUCTION CLASSICAL CONTROL MODERN CONTROL		120K						
<u>MOMENTUM DEVICES</u> CMG'S AMCD GRAVITY-GRADIENT BOOMS		140K						
<u>CONTROL IMPLEMENTATION</u> VIBRATION ISOLATORS AND DAMPERS			70K					
<u>MEASUREMENT SENSORS</u> ACCELEROMETERS LASER-OPTICAL REFLECTORS			35K					

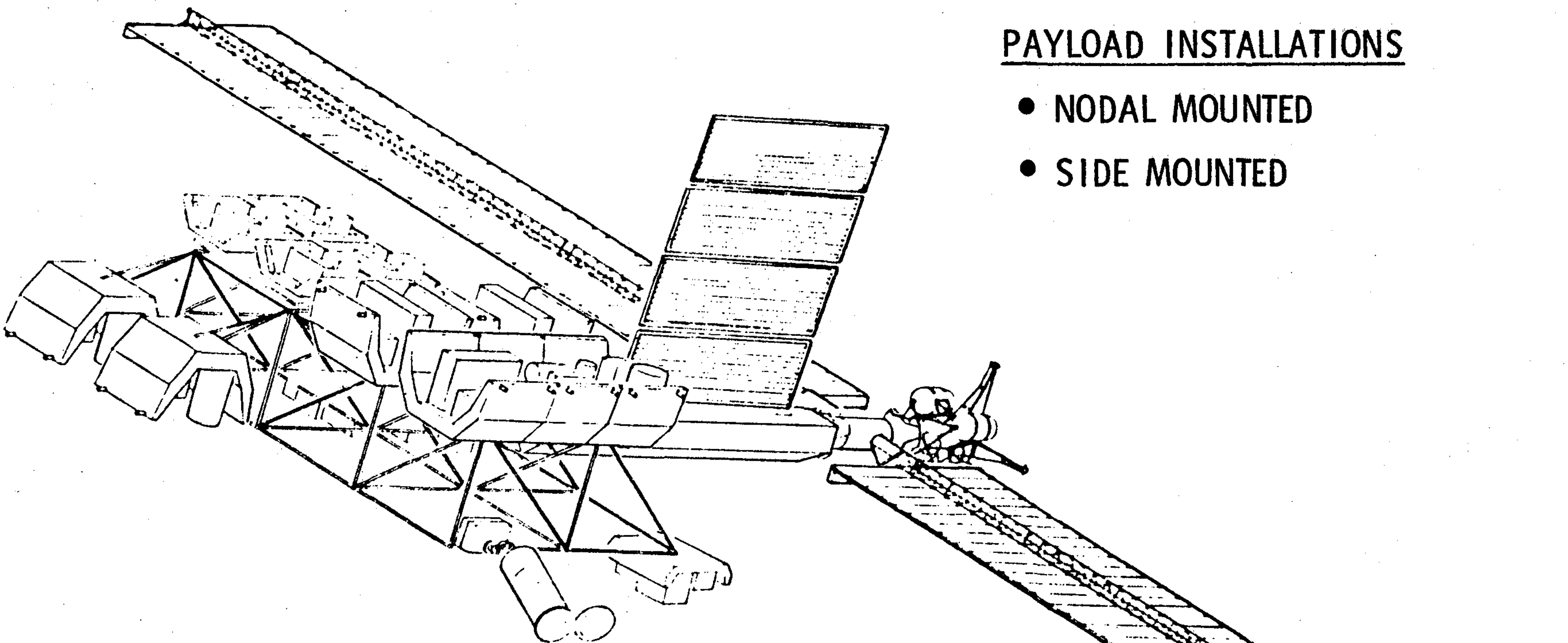
4.7 PAYLOAD ACCOMMODATIONS

The function of the erectable platform is to provide mounting provisions and utilities interface to the payloads. The payloads may be mounted directly on the platform or on a pallet (or equivalent). The pallet is then mounted on the platform. Additionally, payloads may be mounted on an instrument pointing system (IPS) which, in turn, is mounted on the platform. In the case of the pallet-mounted payloads, the pallet also serves as the payload carrier during transportation in the orbiter. The payloads (directly or mounted on pallets or IPS) may be installed on the platform, either nodal mounted (discrete nodal hard points), side mounted to nodal joings, or on a surface area table. All of these options are presented and discussed in this section. Payload accommodations technology needs and the recommended technology development program are also presented.

Payload Accommodations/Pallet Requirements

The erectable platform has the capability of mounting payloads on the top and bottom surfaces at the structural nodes and of mounting payloads at hard points around the sides of the platform with a stabilizing strut attachment to another nodal point. The payloads that are transported in the orbiter's cargo bay have a series of accommodation options on the platform as shown. These will be discussed in the subsequent charts.

PAYLOAD ACCOMMODATIONS/PALLET REQUIREMENTS



PAYLOAD INSTALLATIONS

- NODAL MOUNTED
- SIDE MOUNTED

PAYLOAD ACCOMMODATIONS

- PAYLOAD INSTALLED ONTO PALLET & PALLETS ATTACHED TO PLATFORM
- PAYLOADS DIRECTLY ATTACHED TO PLATFORM
- PAYLOADS MOUNTED ON TABLES ATTACHED TO PLATFORM

Various Payload Accommodations on Platform

A multipurpose payload accommodations platform design concept was developed that had the following salient features:

(a) Basic platform structure was sized to accommodate payloads/pallets mounted to discrete nodal hard points, e.g., union fitting in single, dual, or three-pallet arrangements. Pallets may be node-mounted in a general orientation arrangement as shown. The platform backside structure design will accommodate the same payload mounting versatility as the platform frontside. For the baseline design shown there are 15 nodal mounting points on the frontside and 8 on the backside.

(b) Single or multiple pallet/payload arrangements may be side-mounted to nodal points and may require an underside stabilizing strut.

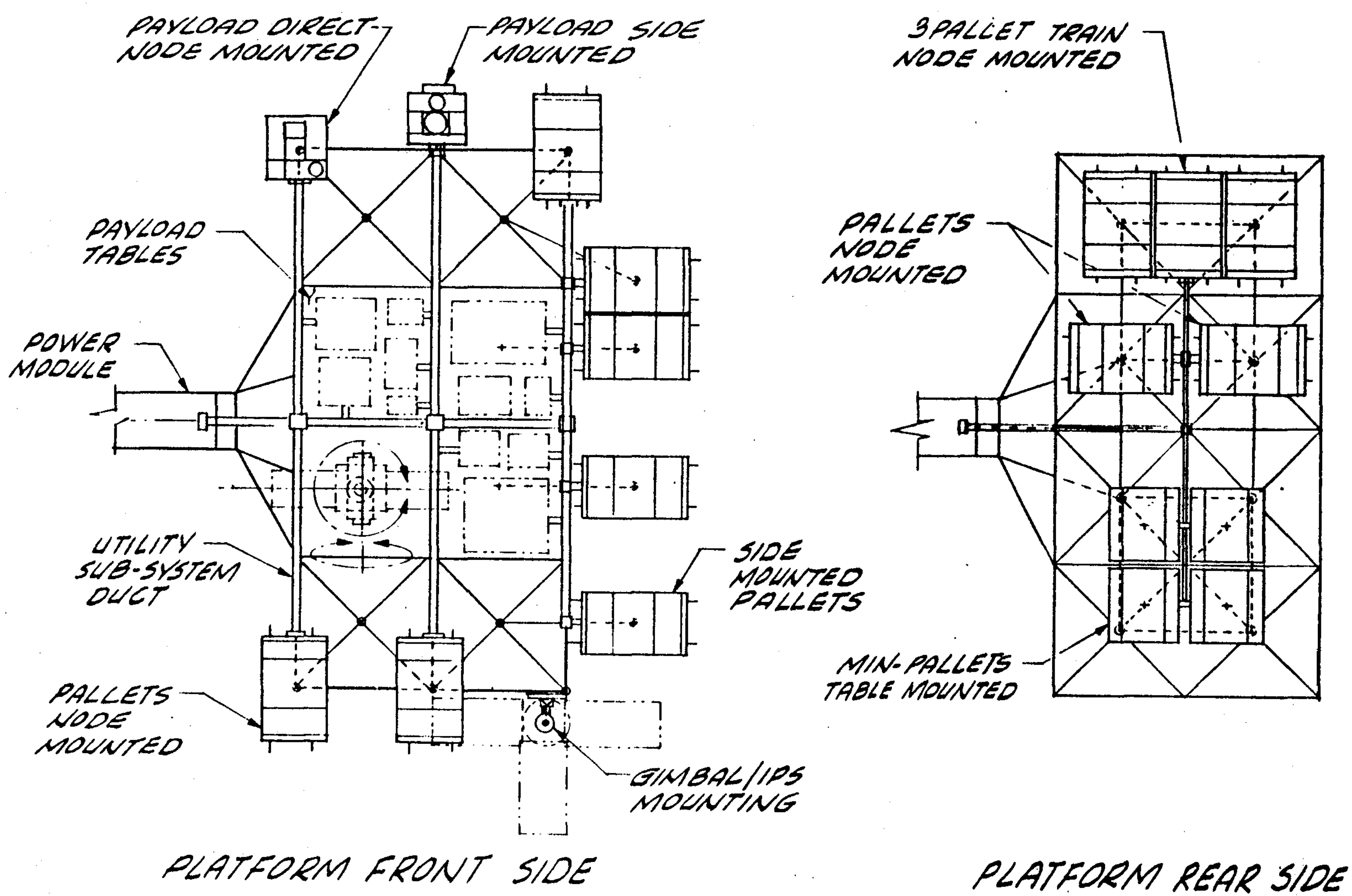
(c) In addition to payload/pallet mounting options at node points, the platform design incorporates a surface area table structure where experiments can be directly attached to the platform hard points which match the transport pallet hard points using quick-connect/disconnect coupling systems. The platform table area concept will permit efficient utilization of area for small or large experiments while enhancing mission effectiveness by providing increased viewing angle for payload.

(d) Experiments/payloads of larger sizes may be directly mounted to any of the nodal hard points on front, side, or back of platform structure as required to provide desired viewing angles; three examples are shown.

(e) Payloads can be mounted on an instrument pointing system (IPS) or gimbal system when the basic pointing accuracy or orientation of the platform is not satisfactory.

The various payload accommodations are illustrated on the next ten charts.

VARIOUS PAYLOAD ACCOMMODATIONS ON PLATFORM



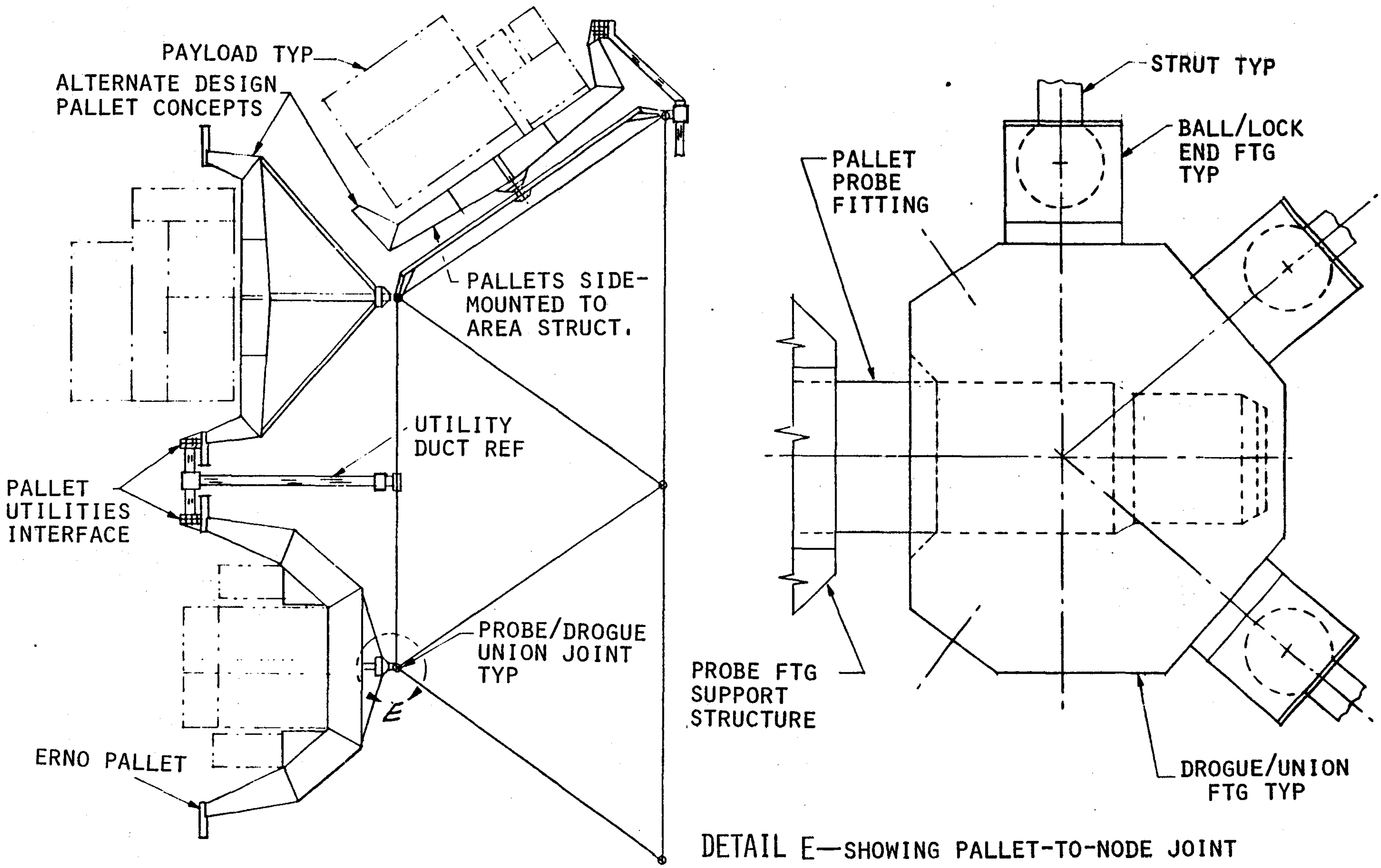
ERNO and Alternate Pallets Node-Mounted

Payload accommodation mounting options are shown for the platform backside and angle side of the platform. Nodal-mounted pallet/payloads are shown utilizing the baseline design ERNO pallet and an alternate design pallet concept.

The method of providing the structural interface between the pallet and nodal hard point union fitting incorporates a self-aligning probe/drogue attachment with quick-release features. The probe and nodal union fitting will be designed to ensure structural integrity under all on-orbit loading conditions.

Under conditions where the payload viewing angles are other than normal to the platform surfaces, a canted drogue fitting which incorporates the desired viewing angle may be used as necessary at any of the nodal hard points.

ERNO AND ALTERNATE PALLETS NODE MOUNTED

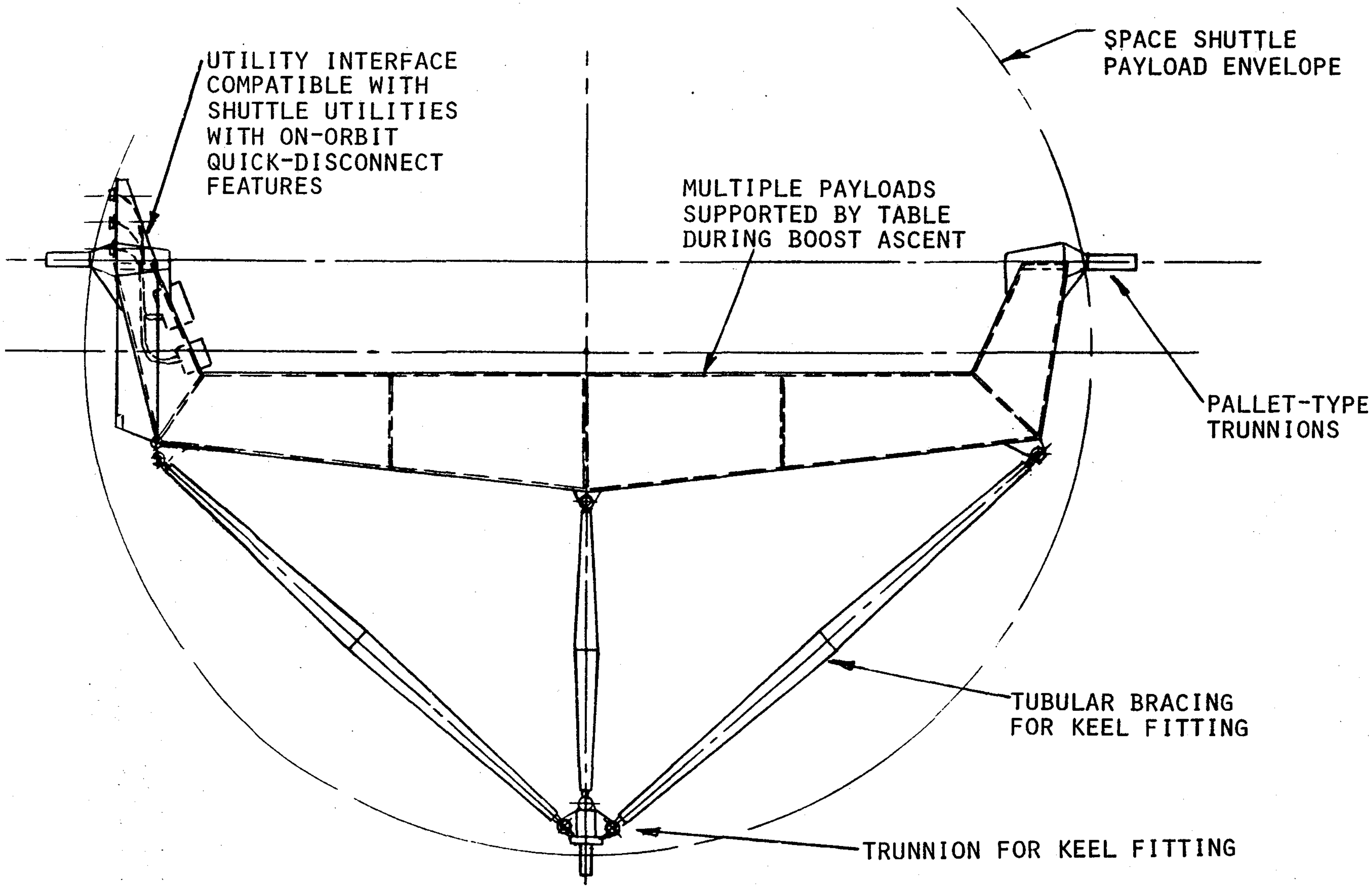


DETAIL E—SHOWING PALLET-TO-NODE JOINT

Payload Table Suspended In Space Shuttle

The alternate pallet/table concept shown incorporates the Erno/Orbiter pallet type trunnion and keel fittings. Utility interfaces are compatible with Shuttle orbiter utilities interfaces and with the additional features of on-orbit quick disconnect. With this type of pallet design, additional surface area is provided for payloads and payload viewing angles are increased due to the relatively short upstanding side structure.

PAYLOAD TABLE SUSPENDED IN SPACE SHUTTLE

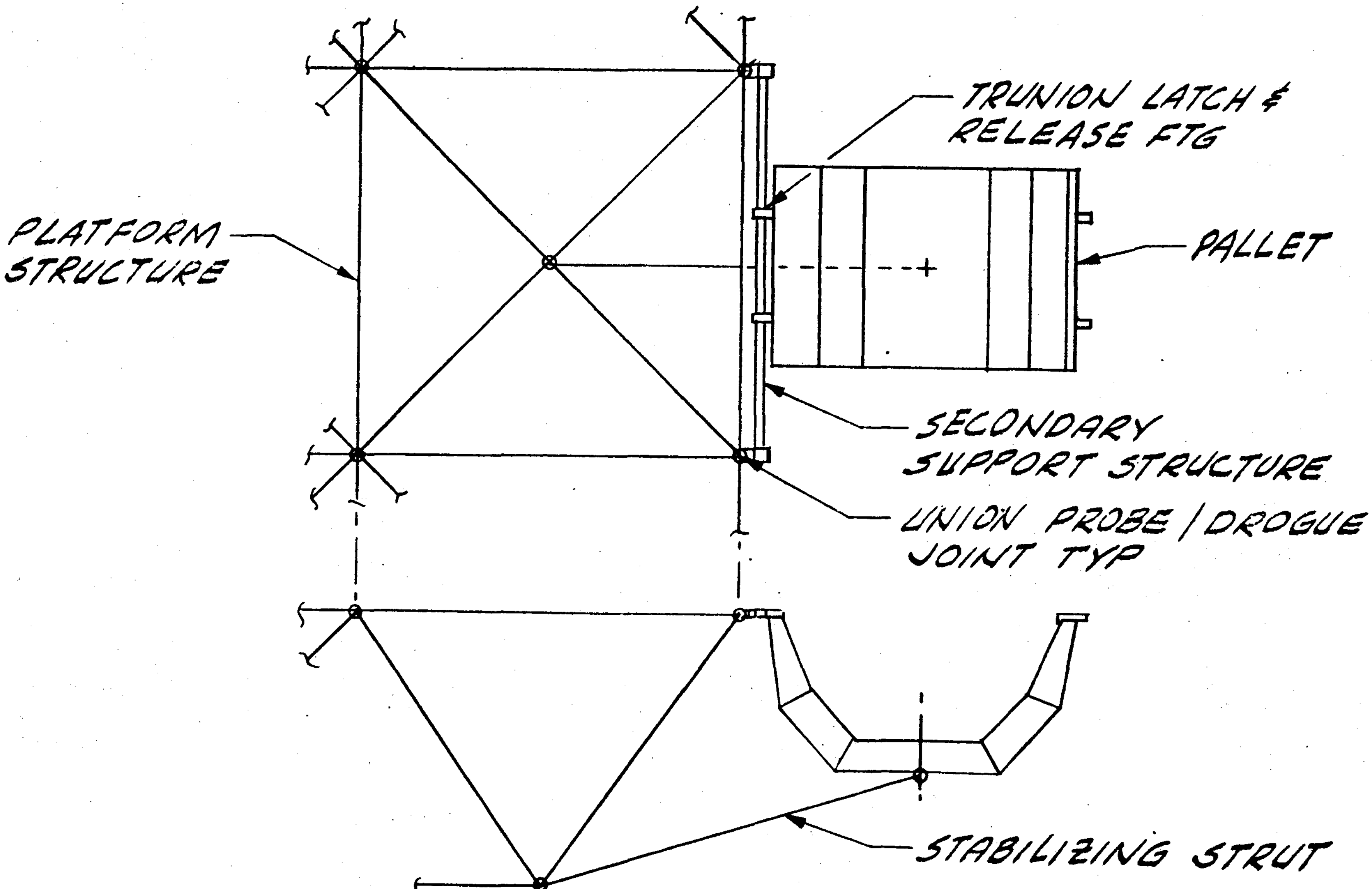


Side Mounted Pallets

A single pallet side mounted to the platform structure requires a secondary tubular support structure to provide the necessary structural stability between pallet and platform structure. The pallet trunnion fittings are attached to the secondary structure at two places with quick connect/disconnect latch type fittings. The secondary support structure attaches to the nodal hard points by means of probe/drogue fitting discussed earlier where necessary, interface position adjustments will be provided for in-orbit easy fit.

When payload pallets are side mounted as shown, a single stabilizing (bracing) strut is required at the underside. The strut will interface with the platform structure at nodal hard point with probe/drogue type attachments and will interface with the pallet underside fitting by means of ball/lock socket connections.

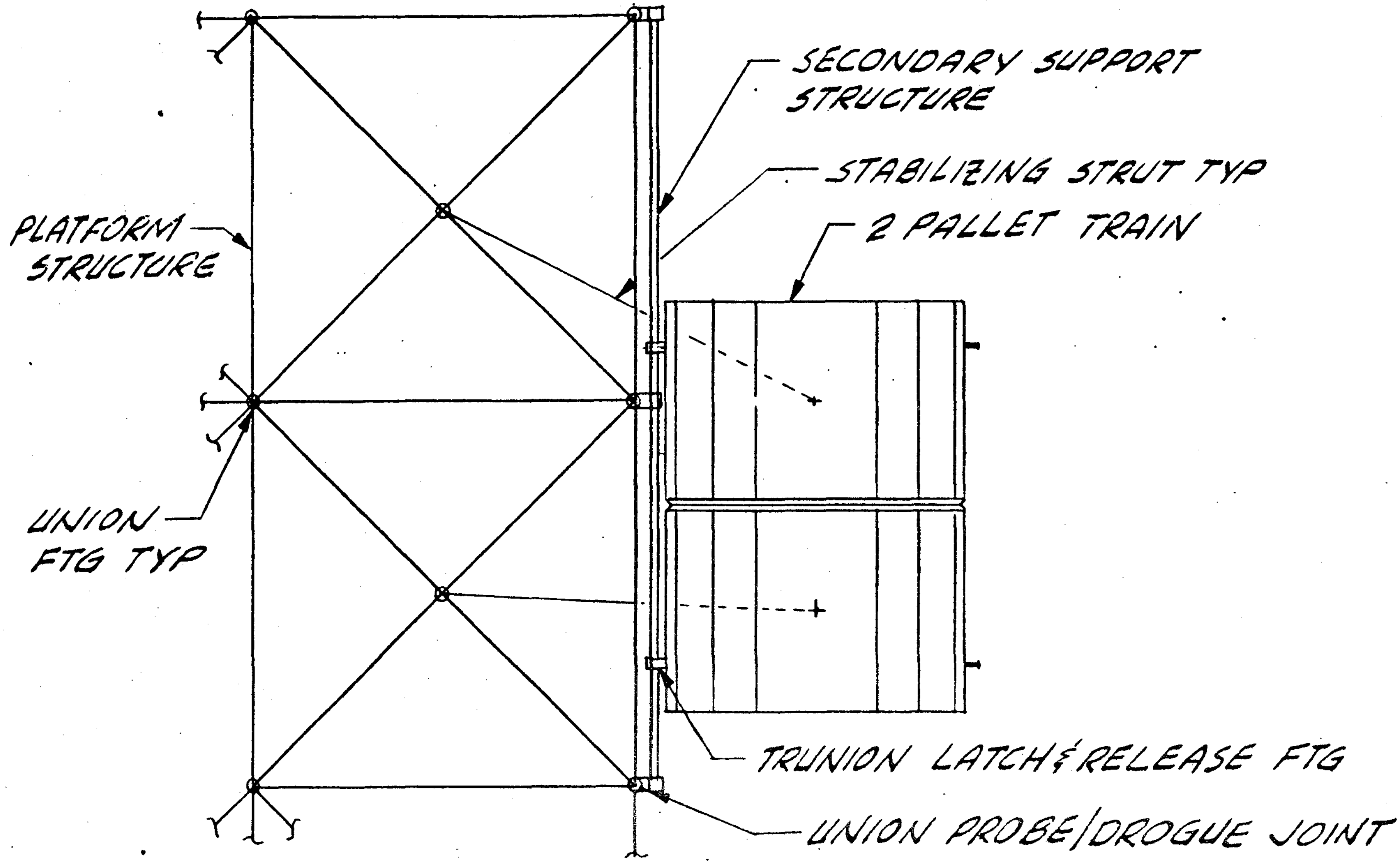
SIDE-MOUNTED PALLETS



Multi-Pallet Arrangement Side-Mounted

The arrangement shows how a multi-pallet train can be accommodated on the erectable platform. Attachments of the pallet are via the support trunnions to the secondary support structure with quick release fittings. An alternate would be to cantilever the second pallet off of the first pallet with the latter attached to the platform as shown in the previous chart.

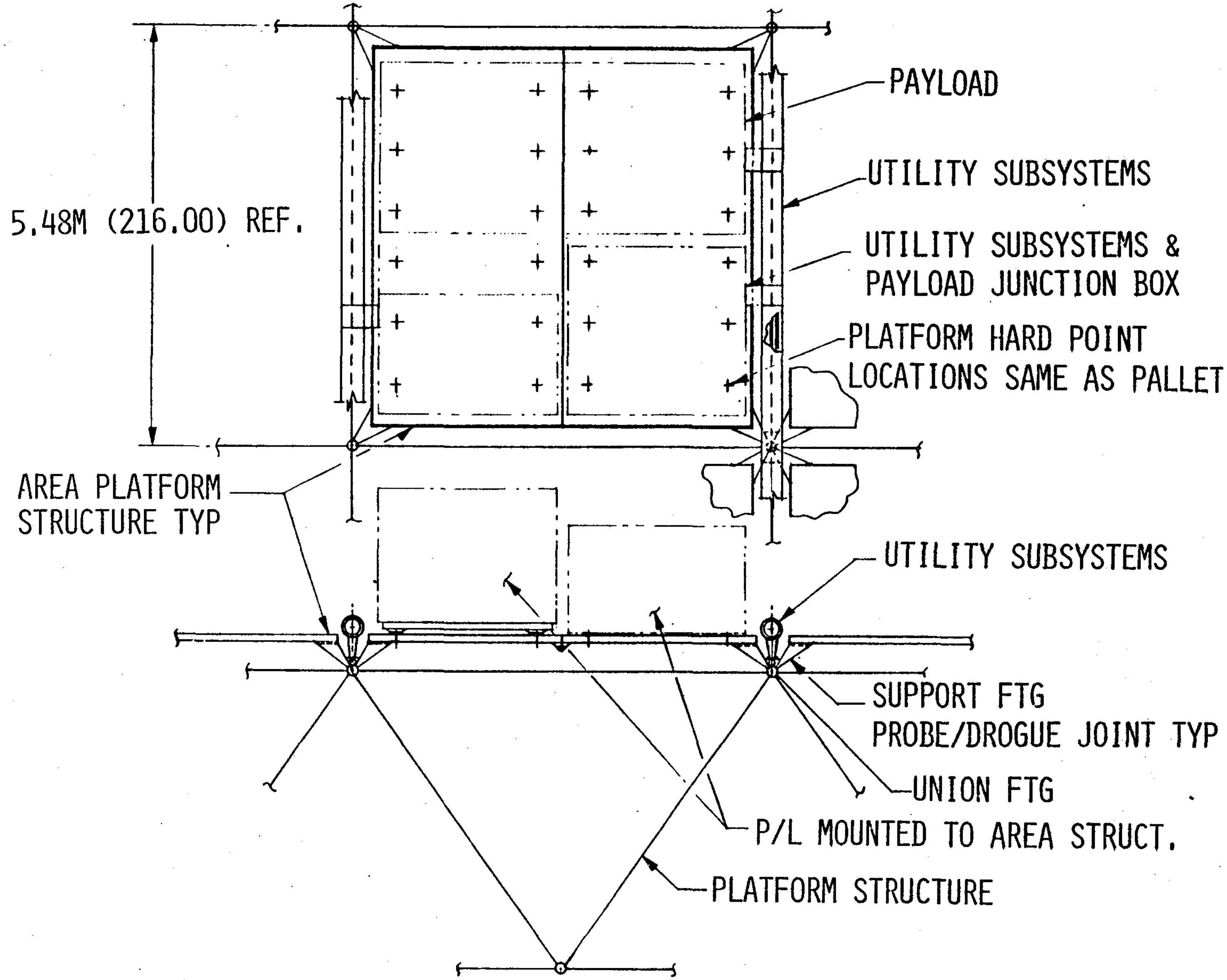
MULTI-PALLET ARRANGEMENT SIDE-MOUNTED



Payload Tables Mounted on Pentahedral Area Platform

Payload area tables are shown nodal mounted on the pentahedral area platform. The area platform structure attaches to the nodal union fitting by means of adjustable probe/drogue fittings located 3 or 4 places at the corner. Provisions for connecting the utility ducts at these same nodal junctions are provided. The payload table construction is invisioned to be an isogrid patterned sandwich panel hinged at the center for easy stowage within the shuttle cargo bay. The table has no payload installed during boost ascent and for installation onto the platform. Payloads are installed onto the table after the table is positioned on the platform; table will remain with the platform. Quick connect/disconnect hard points located within the sandwich structure will match those of the payload and transport pallet. Payloads may be positioned on the table structure to provide optimum viewing angles. Utility connections to payloads are provided. Although payloads shown in this scheme fit within a single cell platform area (approx 5.5x5.5 m) large experiments can be supported off adjacent area platforms.

PAYLOAD TABLES MOUNTED ON PENTAHEDRAL AREA PLATFORM



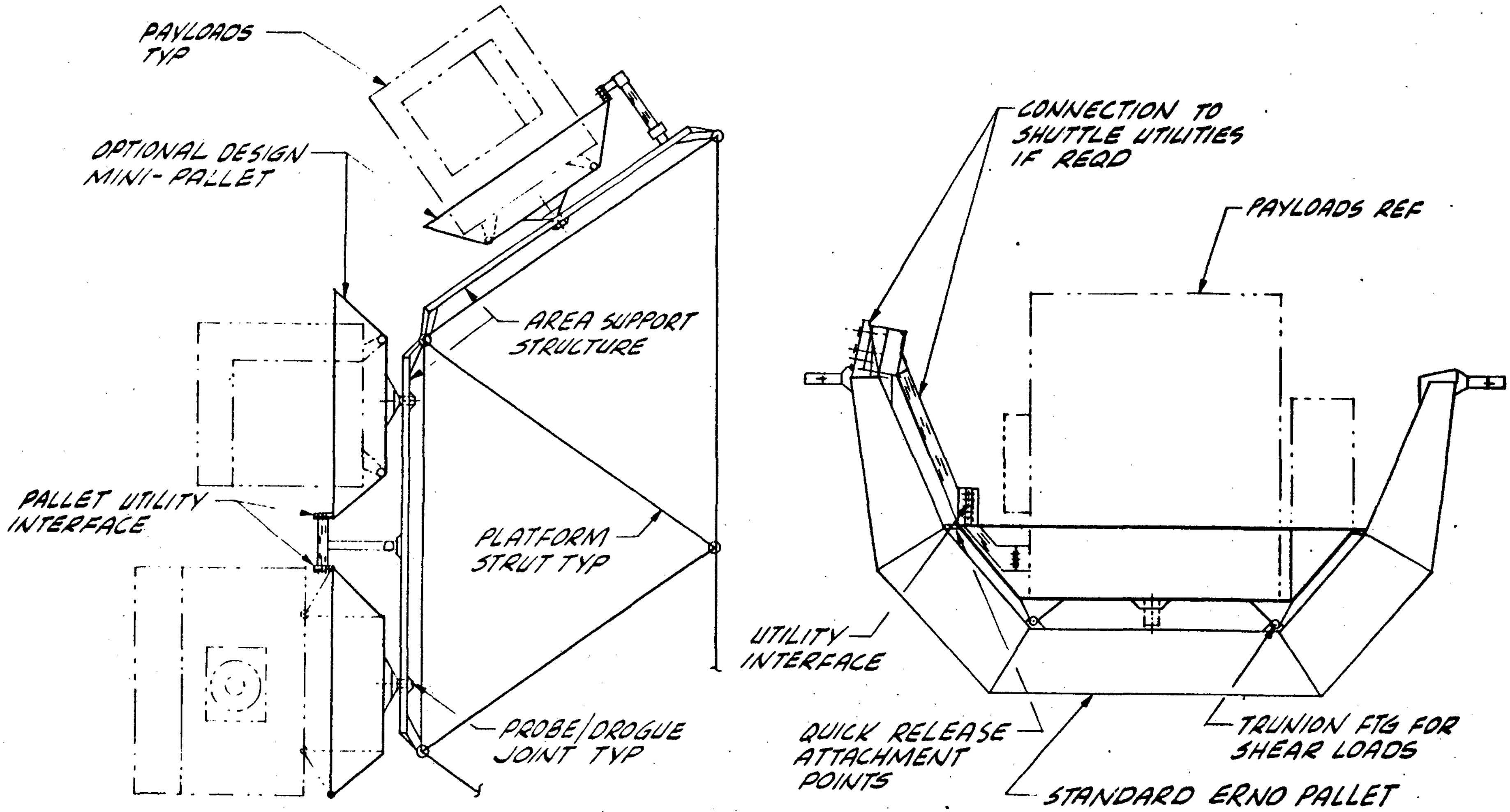
Mini-Pallets Table-Mounted Payloads

Mini-pallets shown are mounted to the area table structure located at the platform backside and angled side by means of probe/drogue fittings. The mini-pallet consists of a light-weight sandwich structure which attach/releases at the standard Erno pallet hard points. Payload launch/abort loads are directly reacted at the Erno pallet hard points. Mini-pallets can be direct mounted to the platform node points.

Utility interfaces between payload and shuttle utilities are provided for ground checkout, during boost ascent to orbit and checkout before hand-over to the platform. The mini-pallet concept provides improved on-orbit viewing angles for the payloads and permits efficient use of platform space. Also Erno pallet utility is enhanced when pallets are returned with expended payloads.

These mini-pallets can be interfaced with the Erno pallet utilities if required.

MINI-PALLETS TABLE-MOUNTED PAYLOADS

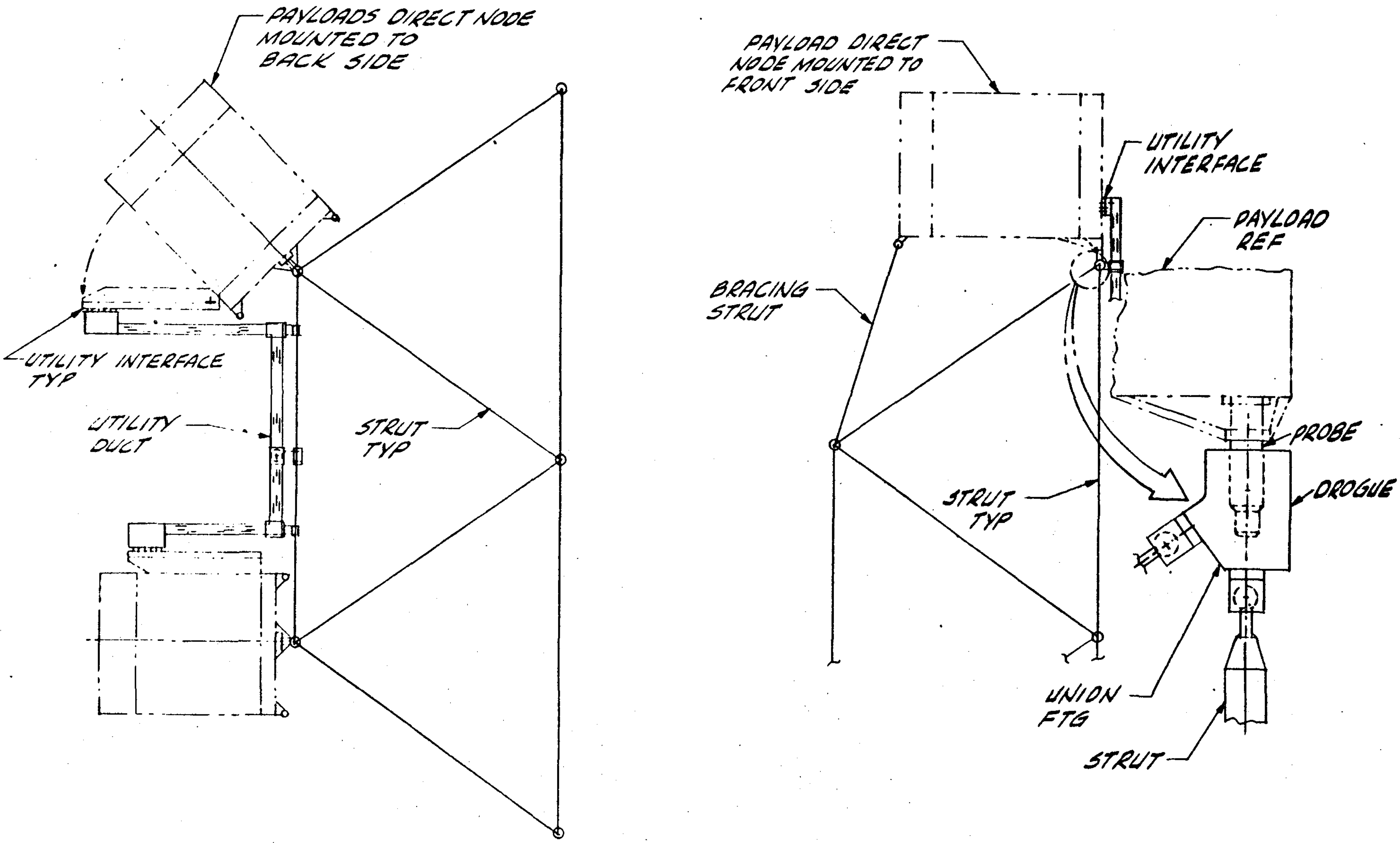


Payload Direct Mounted

Payloads are shown mounted directly to the platform nodal points by means of probe/drogue fittings discussed earlier. Payload mounting options include node mounting of payloads normal to platform backside, node mounting of payloads angled off platform, and node mounting of payloads on sides of platform. In the latter case, a stabilizing strut is required.

The secondary support structure contains the attachment fittings and utility interfaces for integration with the pallet during orbiter boost ascent/descent and with the platform for on-orbit operations. The secondary structure can be designed integrally with the payload equipment.

PAYLOAD DIRECT MOUNTED

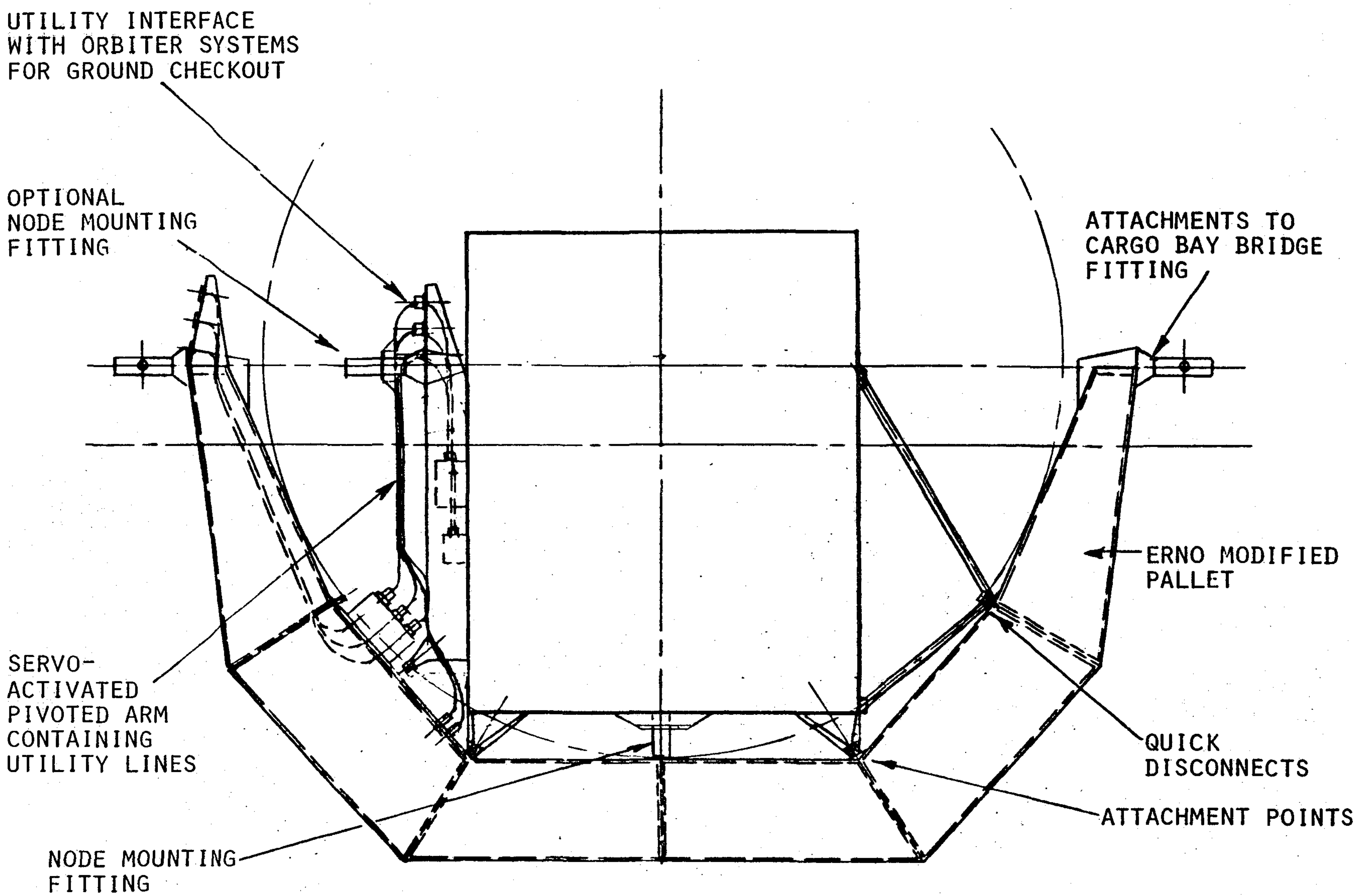


Typical Payload Installed in ERNO Pallet

A typical payload installed in an Erno pallet which is modified to provide quick connect/disconnect features is shown. The node mounting probe fitting on the payload is aligned with the payload neutral axis. An optional node mounting fitting is provided for mounting position versatility. A servo-activated pivot arm containing utility sub-systems is provided for on-orbit quick disconnects.

It is possible that the quick disconnect fittings and utility interface fittings would be standardized platform furnished equipment to the various payload developers. This would allow standardization of installation and interface operational procedures for on-orbit installation and operation.

TYPICAL PAYLOAD INSTALLED IN ERNO PALLET



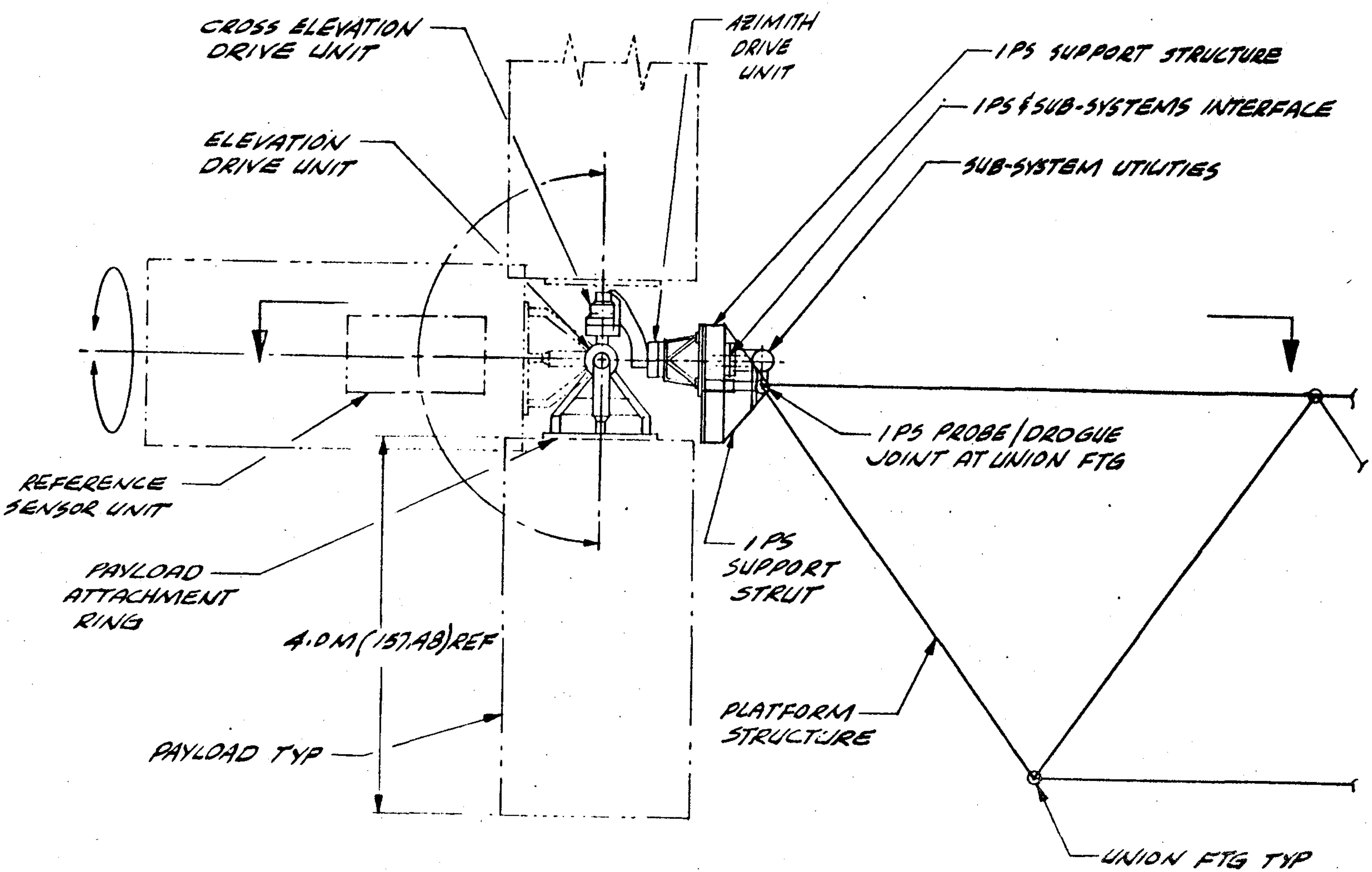
IPS or Gimbal-Mounted Payload

IPS or gimbal-mounted payloads are shown side mounted to the platform nodal union fitting by means of probe/drogue type connection. In this scheme, the baseline design IPS support structure is modified to incorporate an extendable probe fitting and stability struts. Large spherical field viewing angles are achieved with this mounting arrangement.

The IPS payload may be transported by the standard Erno pallet which must be modified to incorporate quick release latches at the payload attach hard points.

For payloads that require scan drive or field-of-view variation, a gimbal-mounting structure is provided and mounted to the platform in a similar fashion. The gimbal-mounting should not need all the sensors and precision control provided by the IPS.

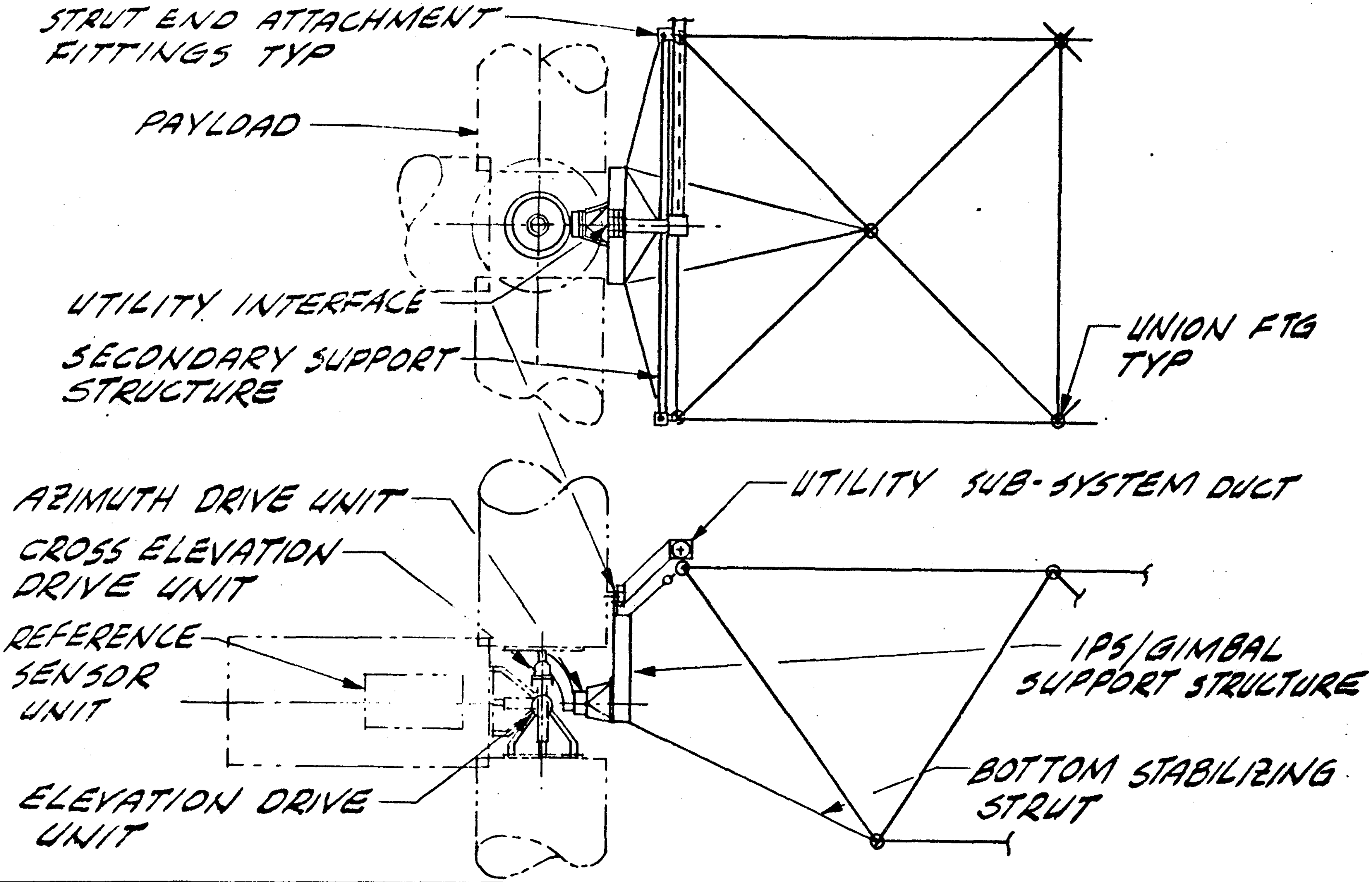
IPS OR GIMBAL-MOUNTED PAYLOAD



IPS/Gimbal Alternate Side Mounting

An IPS/gimbal alternate side mounting arrangement is shown with the payload centered between the platform node points and positioned below the platform front face. A secondary support structure provides the interface between the IPS/gimbal support structure and incorporates probe/drogue connections at the nodal points. Two bottom struts are required to stabilize the IPS platform.

IPS/GIMBAL ALTERNATE SIDE MOUNTING



Payload Accommodations/Pallet Requirements

In order to help control the complexity of on-orbit operations, there should be a series of "standardized" interfaces between the pallet/payloads, orbiter, and the erectable platform. For example, the structural fitting must have quick connect and disconnect features that can be operated remotely by the RMS end effectors.

Due to the currently projected number of ERNO pallets required, there will be insufficient available to service numerous payloads among several platforms if the pallets remain with the platform for extended periods of time. An alternative approach is to use simplified mini-pallets and platform tables designed for space operation aboard the platform.

Other types of standardized utility interfaces (power, signal, and fluid for thermal control) should be GFE to the payload users. This would assure commonality procedures for the on-orbit integration and mating of interface junction boxes with their counterparts on the erectable platform.

PAYLOAD ACCOMMODATIONS/PALLETS REQUIREMENT

- PLATFORM "STANDARDIZED" STRUCTURAL ATTACHMENT CONCEPT(S)
- QUICK CONNECT AND DISCONNECT STRUCTURAL FITTINGS FOR INSTALLATION AND REMOVAL OF PAYLOAD/PALLETS IN THE ORBITER CARGO BAY AND ON THE ERECTABLE PLATFORM
- SERIES OF MODIFIED/MINI-PALLETS AND PLATFORM TABLES AS PLATFORM-FURNISHED EQUIPMENT TO PLATFORM USERS
- UTILITY INTERFACE JUNCTION BOXES—UTILITIES TO INCLUDE POWER, SIGNAL, AND THERMAL CONTROL
- "STANDARDIZED" INTERFACE COMPONENTS FOR GROUND INSTALLATION ON PAYLOAD/PALLET—SIGNAL MULTIPLEXERS, COLDPLATES, HEAT EXCHANGERS, ETC.
- INSTRUMENT POINTING SYSTEM AND SUPPORT STRUCTURE
    - PRECISION POINTING AND REFERENCE SENSORS
    - COARSE GIMBAL TRACKING AND FIELD-OF-VIEW SCANNING
- POSITION AND POINTING ALLOCATION ENVELOPES OF PLATFORM AVAILABLE POSITIONS TO MEET PAYLOAD MISSION REQUIREMENTS

Payload Accommodation Trades/Selection

Although various types of pallets and tables have been discussed throughout the study, the alternate concepts to the ERNO pallets have been delegated to an enhancing technology level. They are better suited than ERNO pallets for the platform because of their recurring costs, their considerably lighter design, and their better viewing and other accommodations for the experiments.

The structural and utility interface attachments and connections are key items in the operational philosophy of the space platform. They represent an enabling technology that will influence the design of the platform elements and the remote manipulator system (RMS) end effectors.

Based on the payload pointing stability requirements there are several payloads that demand accuracy considerably better than can be supplied by the current instrument pointing system (IPS). These requirements relate to the "Super IPS" technology, which is nearer to the pointing systems for the large space telescope.

PAYLOAD ACCOMMODATION TRADES/SELECTION

COMPONENT	ALTERNATES	SELECTION	TECH. LEVEL
PAYLOAD ACCOMMODA-TIONS	ERNO PALLETS	S.O.A. SLIGHT DESIGN MODIFICATIONS	3
	MINI PALLETS	LOWER COSTS & WEIGHT SAVINGS	2
	PLATFORM TABLES	SUITED TO SMALLER PAYLOADS	2
	DIRECT MOUNTING	NOT DEPENDENT ON ERNO PALLET LIMITA-TIONS AND COSTS	2
STRUCTURAL ATTACHMENT CONCEPTS	NODE MOUNTING	PROBE & DROGUE MOUNTING OF PAYLOADS	1
		TRUNNION AND STRUT ATTACHMENT FOR SIDE MOUNTING OF PAYLOADS	1
	LOAD-CARRYING QUICK-DISCONNECT FITTING	REMOTE ACTIVATED FOR REMOVAL OF PAYLOADS FROM ORBITER	1
UTILITY INTERFACE CONNECTORS	AUTOMATIC ASSEMBLY	"STANDARDIZED" CONNECTORS TO BE USED BY MAJORITY USERS	1
	EVA ASSIST	COMPLEX INTERFACE REQUIREMENTS	1
POINTING & TRACKING SYSTEM	ACCURATE INSTRUMENT & POINTING SYSTEM	CURRENT IPS FOR ACCURACY TO 2 ARC-SEC	3
	SUPER ACCURATE IPS	PAYLOADS REQUIRING HIGHER ACCURACY (0.001 ARC-SEC)	2
	COARSE GIMBAL SYSTEM	PAYLOADS REQUIRING COARSE SLEWING AND TRACKING	

Payload Accommodation Technology Candidates

These two technology candidates are alternates to the modified ERNO pallets. Their potential benefits would tend to be lower production costs, light weight, and low thermal distortion.

PAYLOAD ACCOMMODATION TECHNOLOGY CANDIDATES

MINI-PALLETS

OBJECTIVE

- PRODUCE LOW-COST MINI-PALLETS WHICH ARE LIGHTER THAN ERNO PALLETS AND HAVE LOW THERMAL DISTORTION

TECHNOLOGY ASSESSMENT

- ERNO PALLETS ARE DESIGNED FOR USE IN THE SHUTTLE BOOST ENVIRONMENT—HEAVY AND HIGH COST

APPROACH

- DEVELOP MINI-PALLET CONCEPTS FOR EITHER SELF-SUPPORTING OR ERNO PALLET-SUPPORTED DURING ORBITER BOOST
- PROVIDE MULTIPLE PAYLOAD ATTACHMENT HARD POINTS AND UTILITY INTERFACES

EXPECTED RESULTS

- LOW-COST/LIGHTWEIGHT MINI-PALLETS DESIGNED TO CARRY PAYLOADS IN THE ORBITER AND SUBSEQUENT INSTALLATION ON THE PLATFORM

PLATFORM TABLES

OBJECTIVE

- PRODUCE STRUCTURAL MOUNTING TABLE FOR ERECTABLE PLATFORM—LIGHT WEIGHT AND LOW COST

TECHNOLOGY ASSESSMENT

- ERNO PAYLOAD PALLETS DESIGNED PRINCIPALLY FOR ORBITER USE

APPROACH

- DESIGN LIGHTWEIGHT TABLE FOR PLATFORM INSTALLATION—TABLE TO BE FOLDED FOR PACKAGING DURING TRANSPORTATION IN ORBITER
- DEVELOP AND VERIFY CONCEPTS FOR MULTIPLE PAYLOAD INSTALLATIONS AND REMOVALS ON ORBIT

EXPECTED RESULTS

- LOW-COST/LIGHTWEIGHT PLATFORM TABLES DESIGNED TO SUPPORT PAYLOADS ON THE ERECTABLE PLATFORM

Payload Accommodations Technology Candidates

The structural mounting attachments are a major technology development. It is necessary to develop and test various concepts in the RMS simulator to both understand their operational requirements and their impact on design of an end effector for the remote manipulator system (RMS). The degree of automation and amount of EVA assist, if required, has to be defined by these simulation tests.

Various structural disconnect systems are under consideration for payload cradles for the Multimission Spacecraft (MMS), Inertial Upper Stage (IUS), etc. Development is required to be able to disconnect the pallets via their support trunnions and multiple utility lines. It should be recognized that these disconnects are remotely activated and compatible with the RMS operation

PAYLOAD ACCOMMODATIONS TECHNOLOGY CANDIDATES

STRUCTURAL MOUNTING

OBJECTIVE

- DEVELOP MOUNTING CONNECTOR FOR IN-ORBIT INSTALLATION OF PAYLOADS ONTO THE ERECTABLE PLATFORM

TECHNOLOGY ASSESSMENT

- EXPERIMENTAL CONCEPTS BEING CONSIDERED
- PROBE & DROGUE FOR NODE-MOUNTED MODULES

APPROACH

- DESIGN & TEST SEVERAL CONCEPTS FOR NODE AND SIDE-MOUNTING OF PAYLOADS
- CONDUCT RMS SIMULATOR GROUND TEST TO DETERMINE REMOTE REQUIREMENTS FOR REMOTE AND/OR EVA-ASSIST ASSEMBLY

EXPECTED RESULTS

- MOUNTING FIXTURE TO MATE WITH PLATFORMS, NODE POINTS

LOAD-CARRYING DISCONNECT FITTING

OBJECTIVE

- DEVELOP STRUCTURAL FITTING CAPABLE OF BEING CONNECTED OR DISCONNECTED IN ORBIT REMOTELY, RMS, OR EVA ASSIST

TECHNOLOGY ASSESSMENT

- ERNO PALLET SUPPORT TRUNNIONS
- SHUTTLE PAYLOAD CRADLES FOR MMS, IUS, ETC., ATTACHMENT & RELEASE MECHANISM

APPROACH

- DESIGN & DEVELOP SIMPLIFIED FITTING FOR STANDARD APPLICATION BY PAYLOAD USERS
- PERFORM RMS SIMULATOR GROUND TEST TO DEMONSTRATE & VERIFY ACCEPTABLE DESIGNS

EXPECTED RESULTS

- STANDARDIZED STRUCTURAL FITTINGS FOR PAYLOADS AND PALLETS WITH EASY DISCONNECT FEATURES

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TECHNOLOGY PROGRAM PLANNING

	1979	1980	1981	1982	1983	1984	1985	1986
			PHASE C/D GO-AHEAD			PLATFORM LAUNCH		
MINI-PALLETS DESIGN CONCEPTS GROUND SIMULATOR			CONDITIONAL PROGRAM	CONDITIONAL PROGRAM	CONDITIONAL PROGRAM			
PLATFORM TABLES CONCEPT SELECTION GROUND ASSY SIMULATION				CONDITIONAL PROGRAM	CONDITIONAL PROGRAM			
STRUCTURAL MOUNTING CONCEPT SELECTION GROUND TESTING		200K	300K					
LOAD-CARRYING DISCONNECT DESIGN & DEVELOPMENT GROUND SIMULATION		200K	200K					

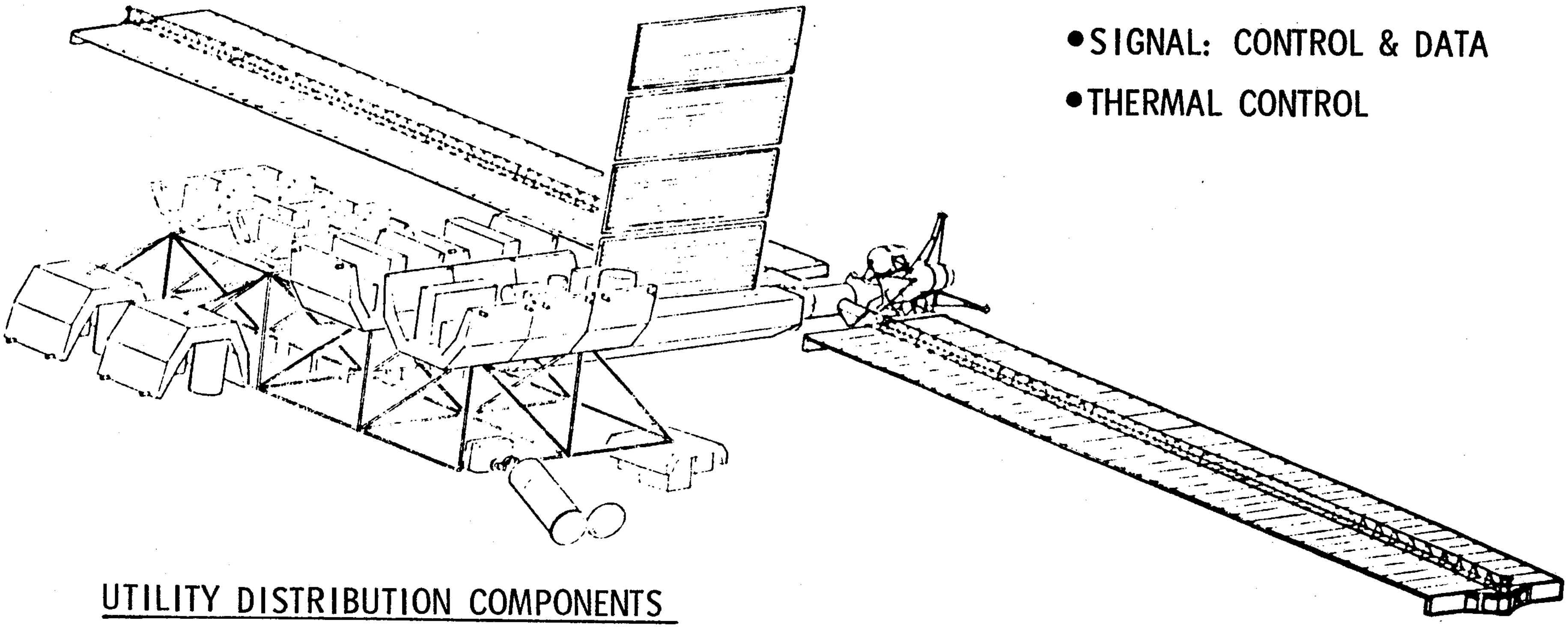
4.8 UTILITIES DISTRIBUTION SUBSYSTEM

Previous subsystem sections have discussed the type, form, amount, etc., of utilities supplied to the payloads. In this section the network for distributing these utilities from the utilities module to the payloads will be presented. The utilities include electrical power (up to 5 kW to each payload position), communication and data handling, and heat rejection. Both electrical and fluid connectors are required for the interface with the utilities module and the payloads. Design and development of these connectors for remote connect/disconnect operations are the critical technologies identified for this subsystem.

Utility Distribution System

The utility services supplied by the central service module are Power (AC & DC), signal (Control & Data) and thermal control. These services have to be supplied to every potential payload attachment point on the erectable platform. Due to the size of the distribution network, the lines have to be transported as separate modules, and once on orbit the line segments are connected together and installed onto the platform structure.

UTILITY DISTRIBUTION SYSTEM



UTILITY SERVICES

- POWER: DC & AC
- SIGNAL: CONTROL & DATA
- THERMAL CONTROL

UTILITY DISTRIBUTION COMPONENTS

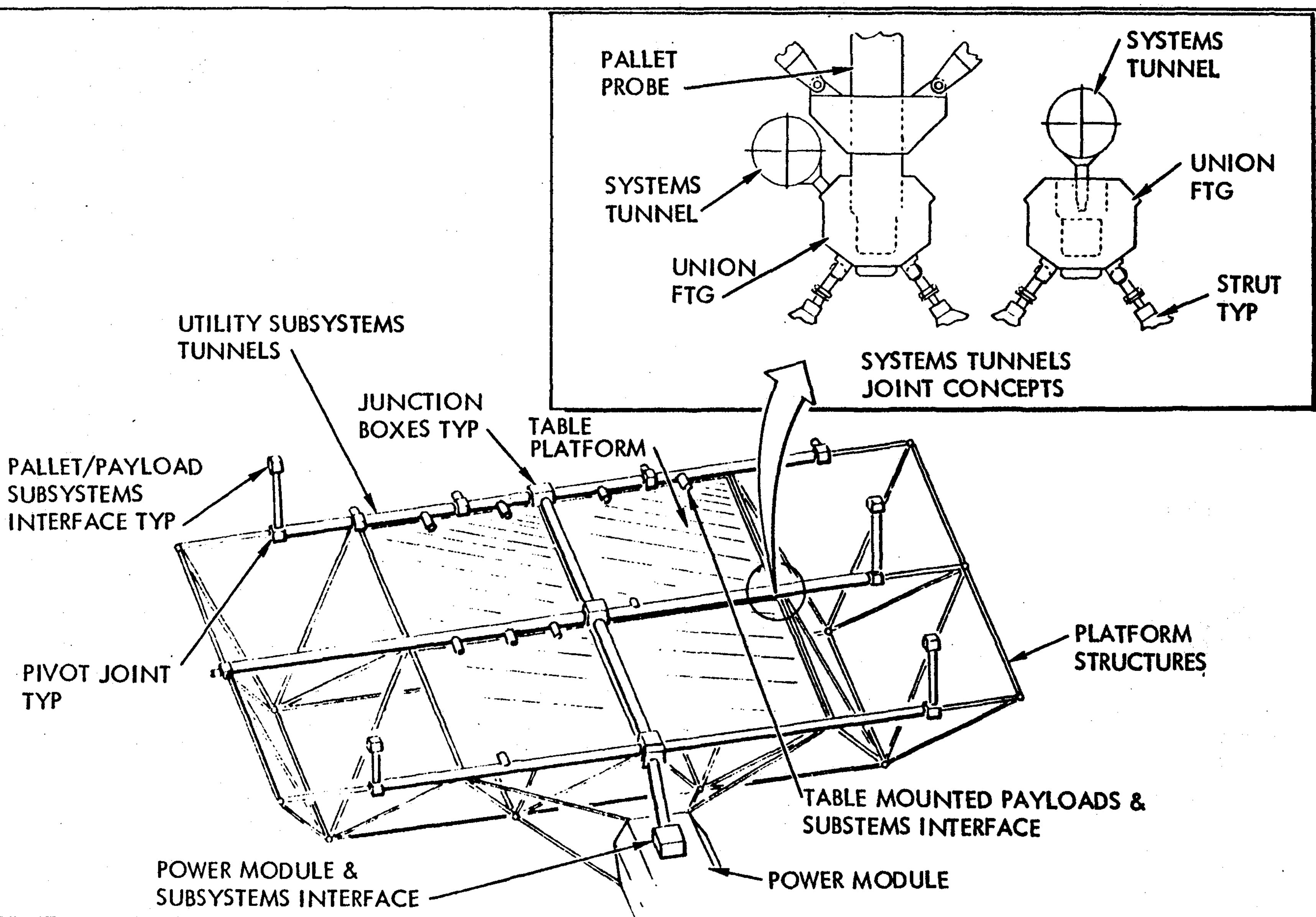
- PLATFORM DISTRIBUTION LINES & STRUCTURAL DUCT MODULES
- DISTRIBUTION LINE CONNECTORS
- UTILITY INTERFACES WITH PAYLOAD/PALLETS

Utility Subsystem Distribution for Front Side

There are numerous utility lines that have to be distributed to each payload attachment position. A systems tunnel carrying these utility lines is in long segments which are connected to the basic structure of the erectable platform. The connections are at the platform union fitting and could be achieved by a ball and socket joint similar to the basic struts.

Junction boxes are provided to interconnect the duct modules and provide the interfaces with the power module and payloads. Individual junction boxes are required for each payload. The junction box design will incorporate on-orbit remote type connects/disconnects for all subsystems. Also, the junction boxes will be designed to accommodate dimensional changes that will be caused by thermal gradients.

UTILITY SUBSYSTEM DISTRIBUTION
FOR FRONT SIDE

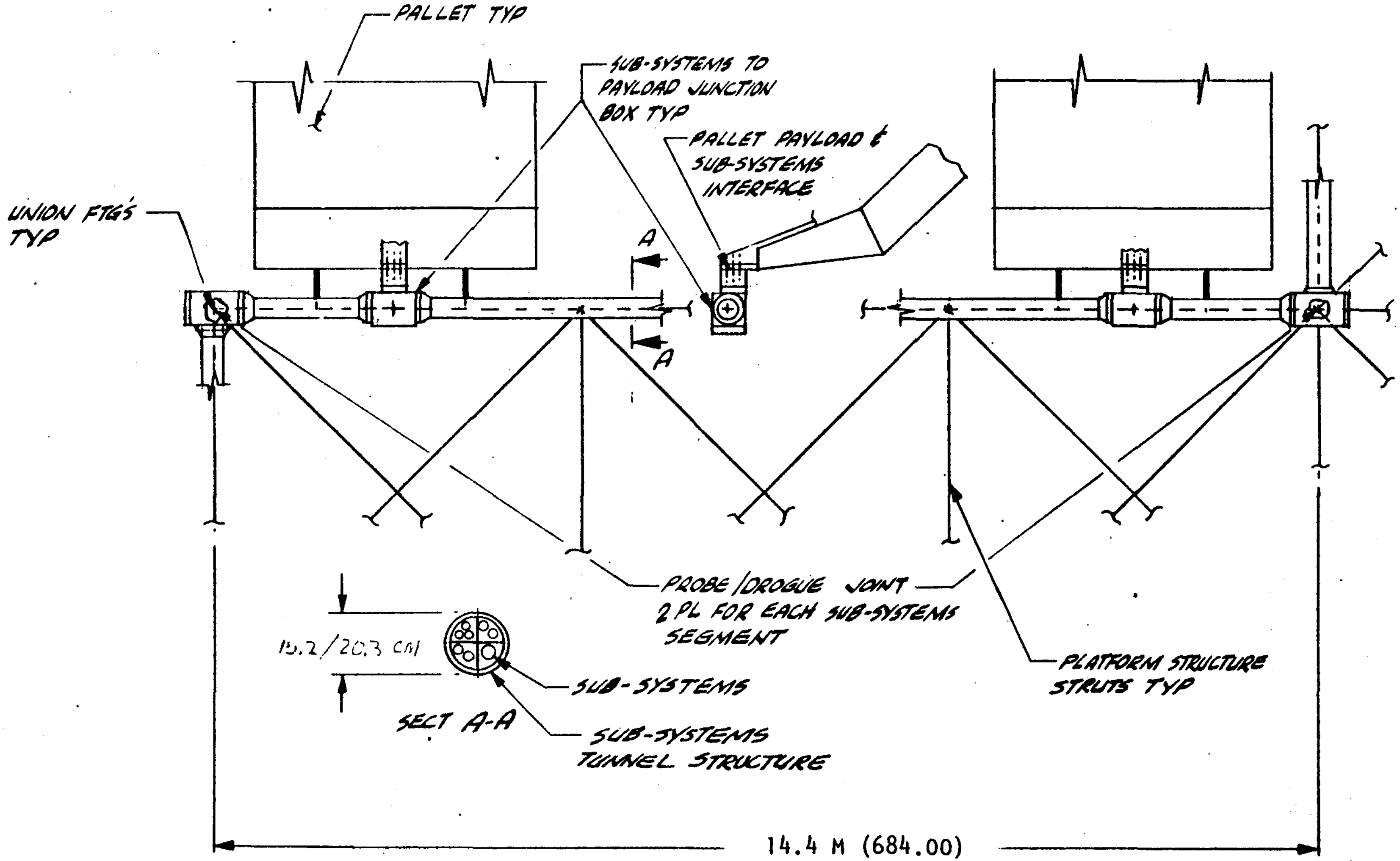


Utility Subsystem Distribution

The utility lines are enclosed within a cylindrical shape tube (duct) support structure of approximately 15 cm diameter. The duct is designed with a structural cruciform shape center section. The cylindrical outer housing is designed with access door positioned along the length to allow easy installation and servicing of subsystems. The utility lines are supported off the cruciform center section and where necessary, lines may be routed separately. The length of the utilities subsystems duct module for the baseline platform design shown range from 5.5 to 11 m.

A utilities subsystems distribution system including junction boxes are shown for a side mounted pallet arrangement. A cross section of the subsystems duct is shown. All interface connections at payload umbilicals are remote controlled connect/disconnects.

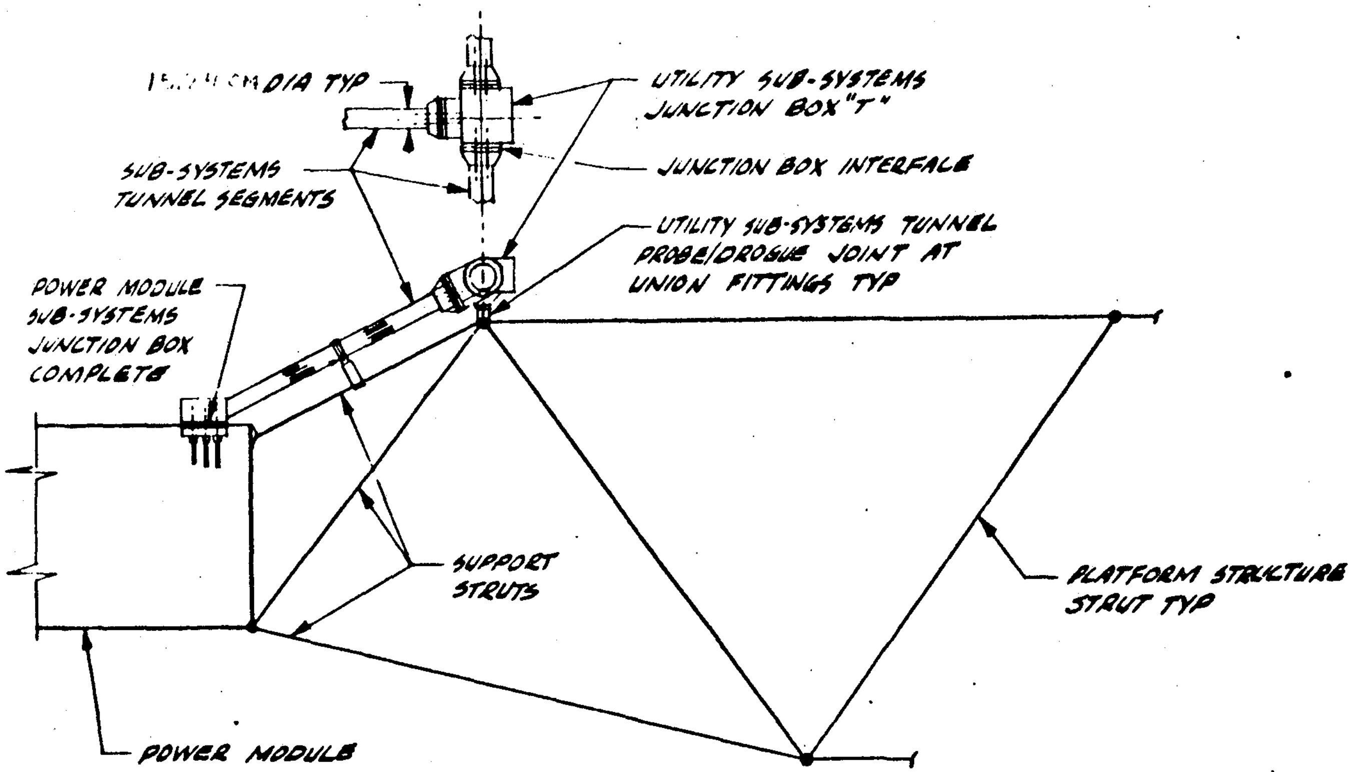
UTILITY SUBSYSTEM DISTRIBUTION



Utility Connection to Power Module

The systems distribution from the power module to the first nodal connection is shown. This segment of the utility duct could be a integral part of the power module.

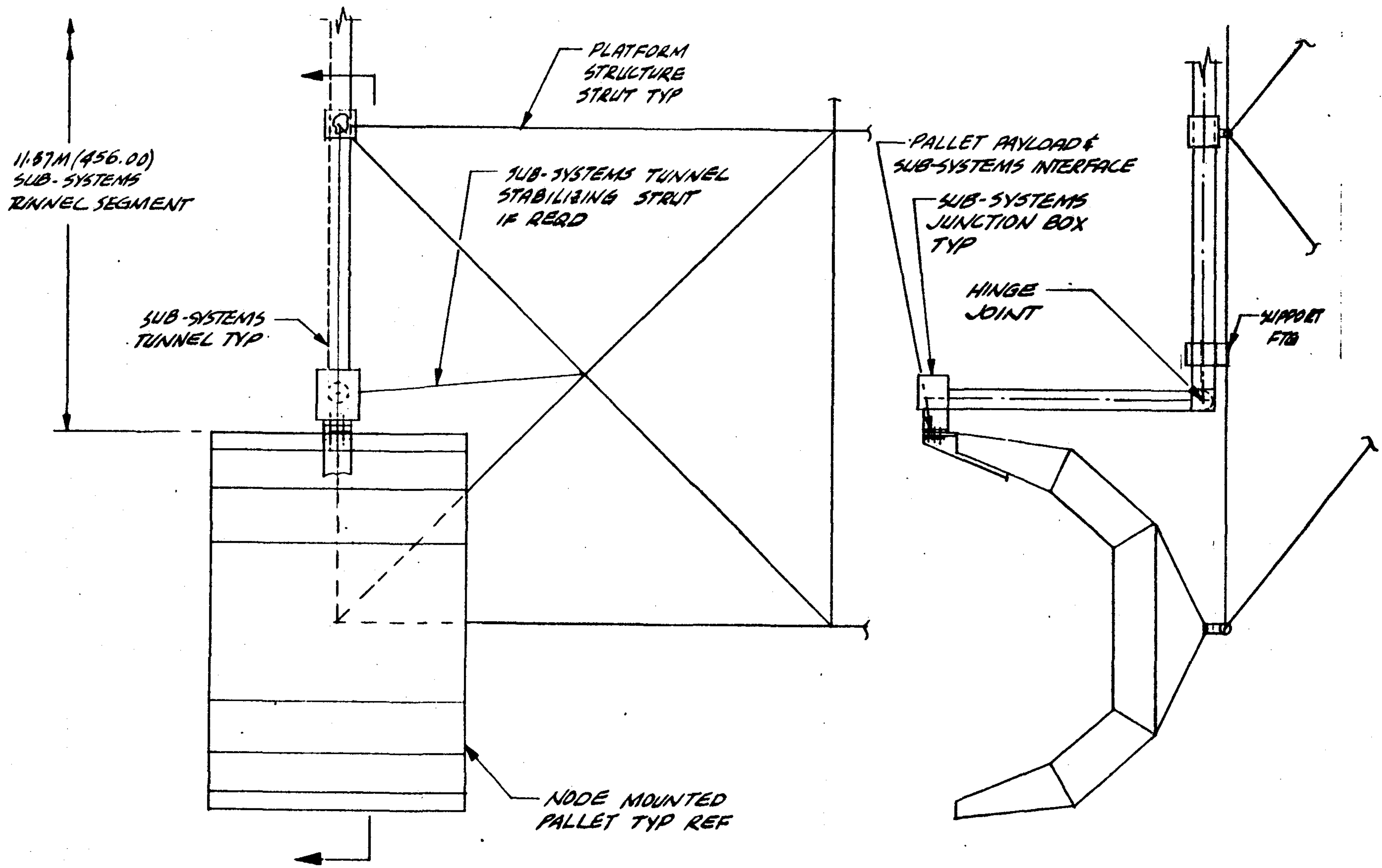
UTILITY CONNECTION TO POWER MODULE



Utility Tunnels for Node-Mounted Payloads

A subsystem distribution system for up-standing nodal mounted payload pallet is shown together with junction boxes. A hinge or pivot joint is necessary with this concept so that packaging density of the utilities distribution module is enhanced within the shuttle cargo bay.

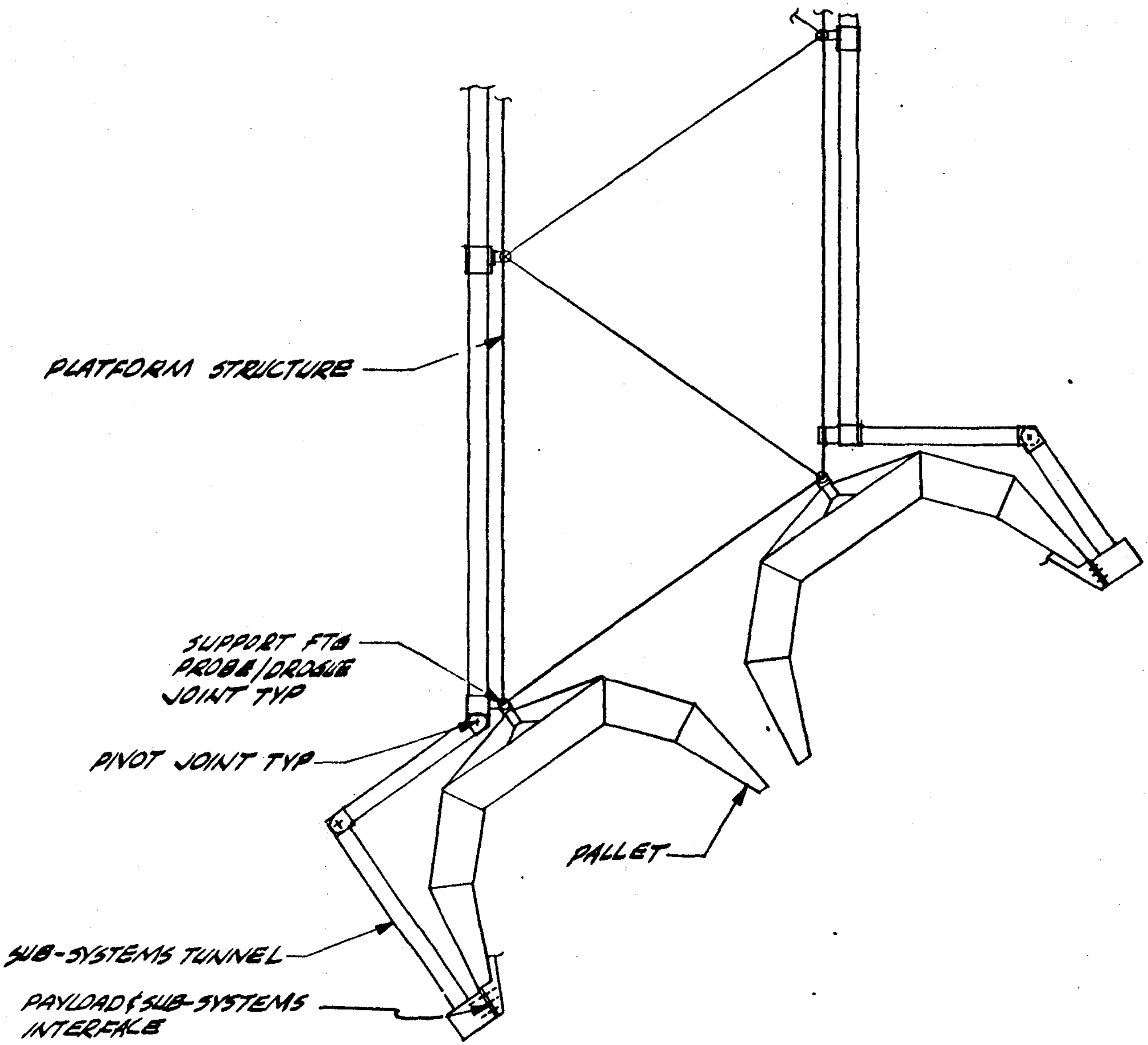
UTILITY TUNNELS FOR NODE-MOUNTED PAYLOADS



Utility Tunnels for Payload Orientation Options

Payload/pallets that are mounted at other than normal orientations can be serviced by a double-jointed utilities tunnel as shown.

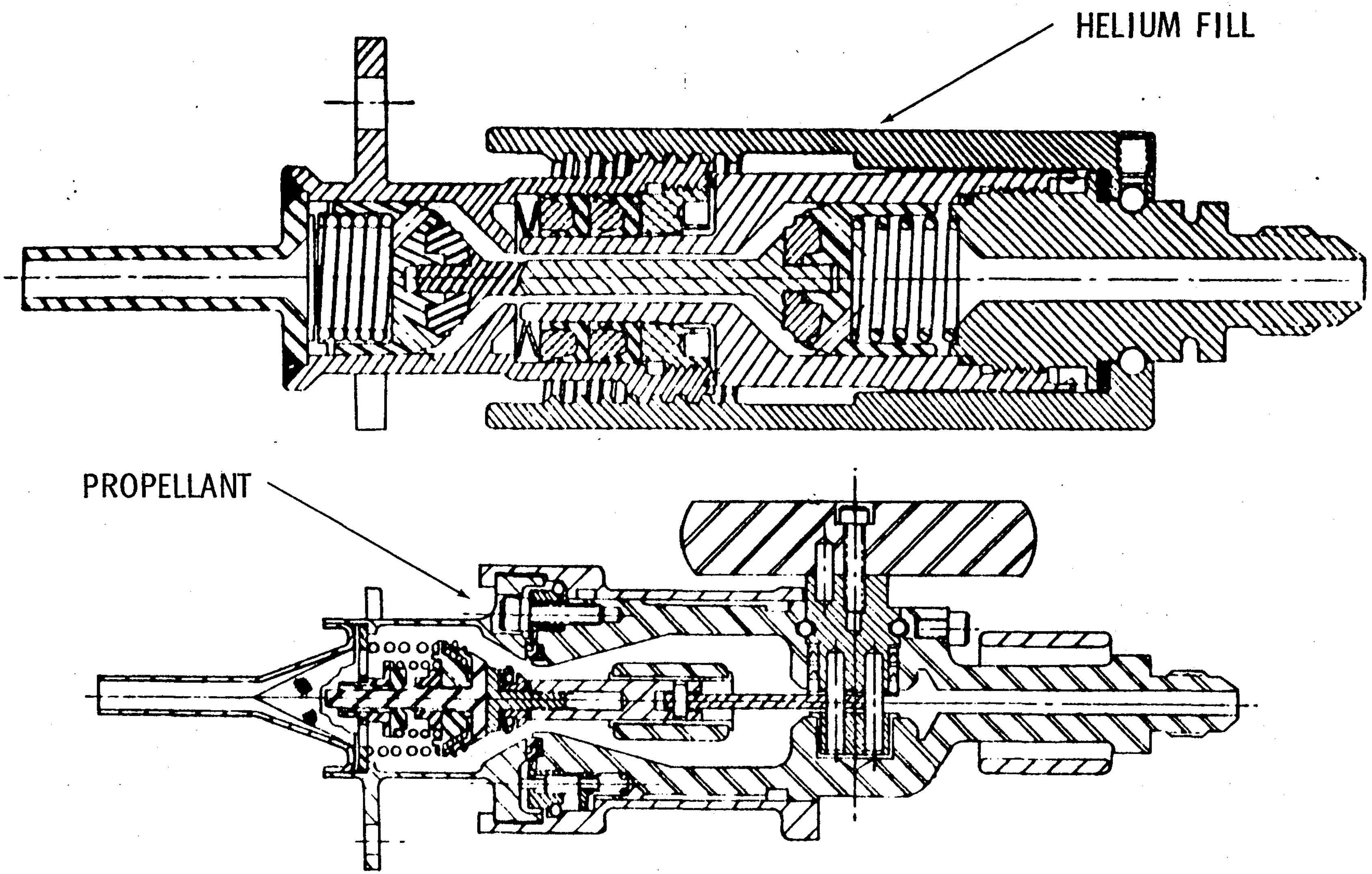
UTILITY TUNNELS FOR PAYLOAD ORIENTATION OPTIONS



State-Of-The-Art Disconnects—Apollo Program

State-of-art fluid disconnects primarily address the problems of reliability and safety in handling various hazardous fluids. Those shown here address these problems successfully, the one for propellant being a "zero-leak" connector. Neither of these is a quick-disconnect coupling. Such devices have, of course, existed for a long time. However, there is no connector, to our knowledge, which combines quick-disconnect, zero-leak, and reliability features suitable for space application.

STATE-OF-ART FLUID DISCONNECTS—APOLLO PROGRAM

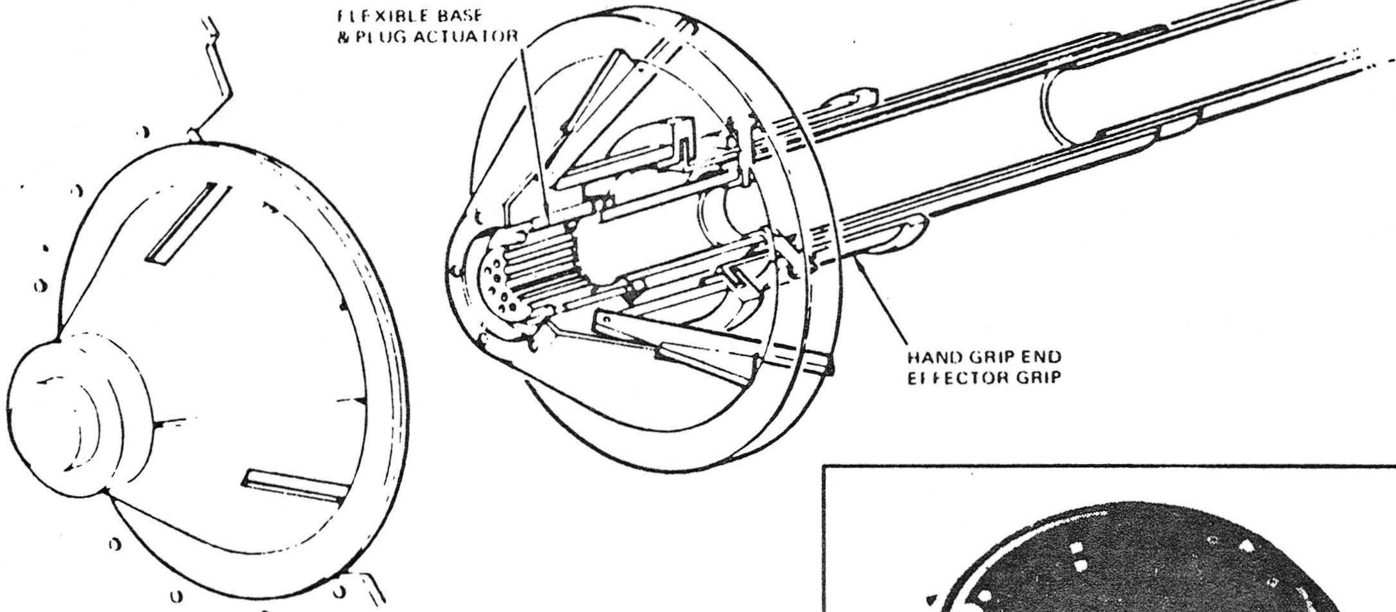


Electrical Connector

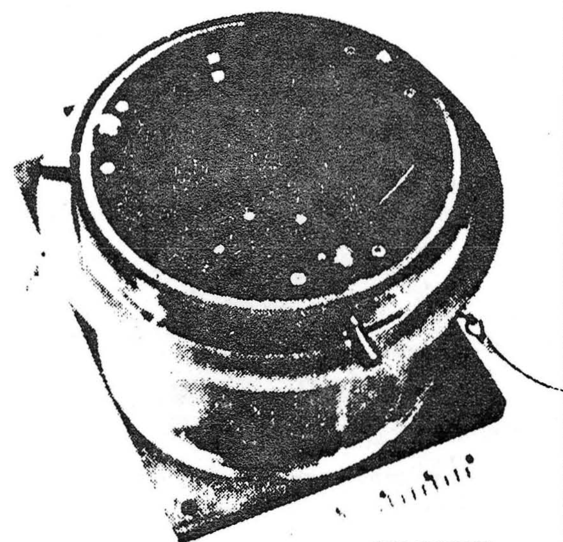
Various concepts for utility connectors have been investigated at Rockwell. Shown is a probe and drogue electrical connector that was developed at Rockwell under IR&D funding. This concept allows considerable misalignment tolerances on the approach of the two parts and still achieve a successful connection. There are guide keys which will ensure correct mating of corresponding electrical circuits and a release mechanism for disconnecting the connector.

This connector concept has been tested successfully at the NASA remote manipulator facility at JSC, Houston. Repeated connections have been completed using the RMS simulator.

ELECTRICAL CONNECTOR



PROBE

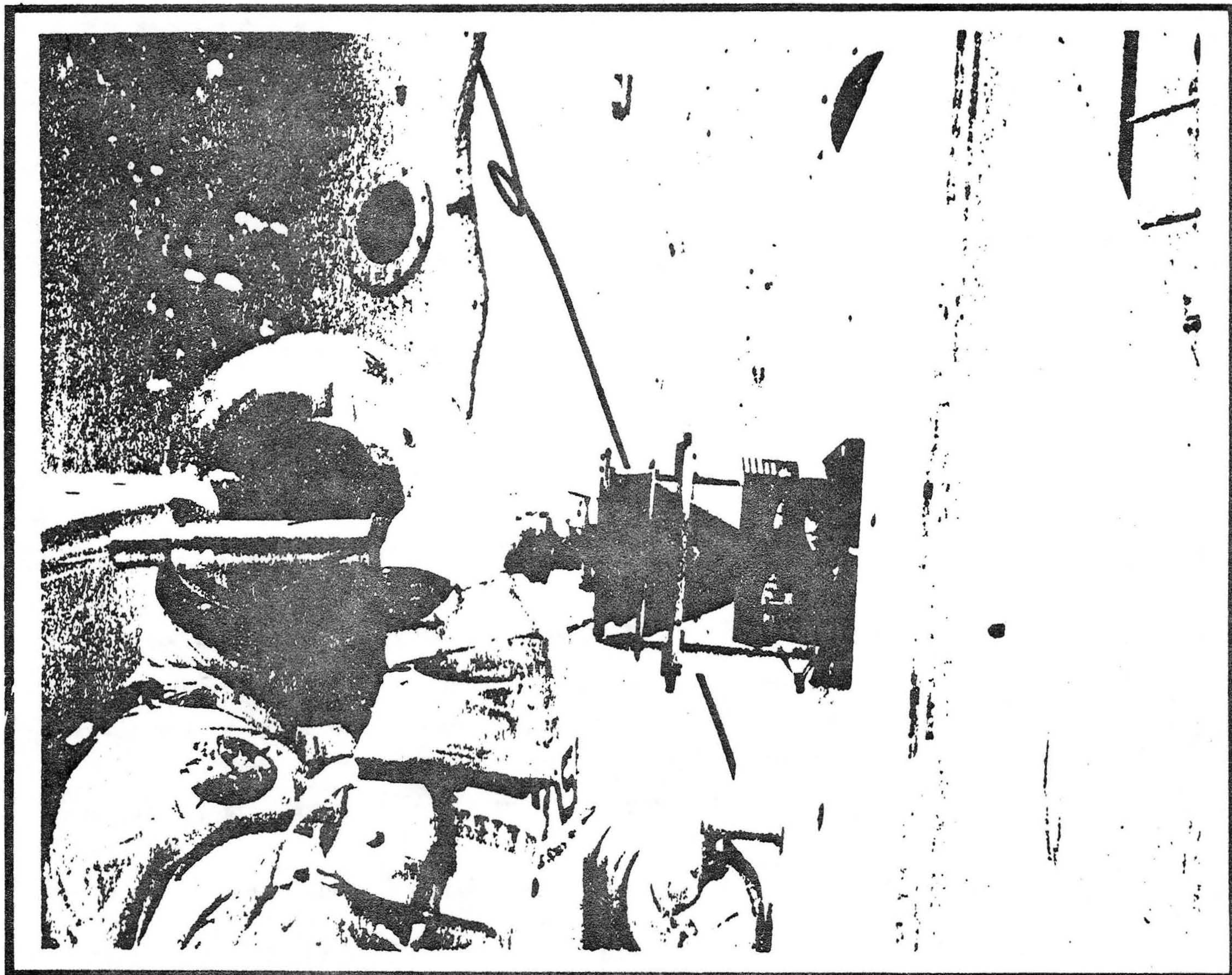


DROGUE

Electrical Connector Emplacement

The electrical connector shown on the previous page has been tested (Reference #12) under simulated EVA conditions. These tests were conducted in the NASA neutral buoyancy tank at MSFC, Huntsville. The demonstration indicated the misalignment tolerance capability of the probe and drogue concept and the difficulty of exerting forces for the required mating of the multi-pin connector.

ELECTRICAL CONNECTOR EMPLACEMENT

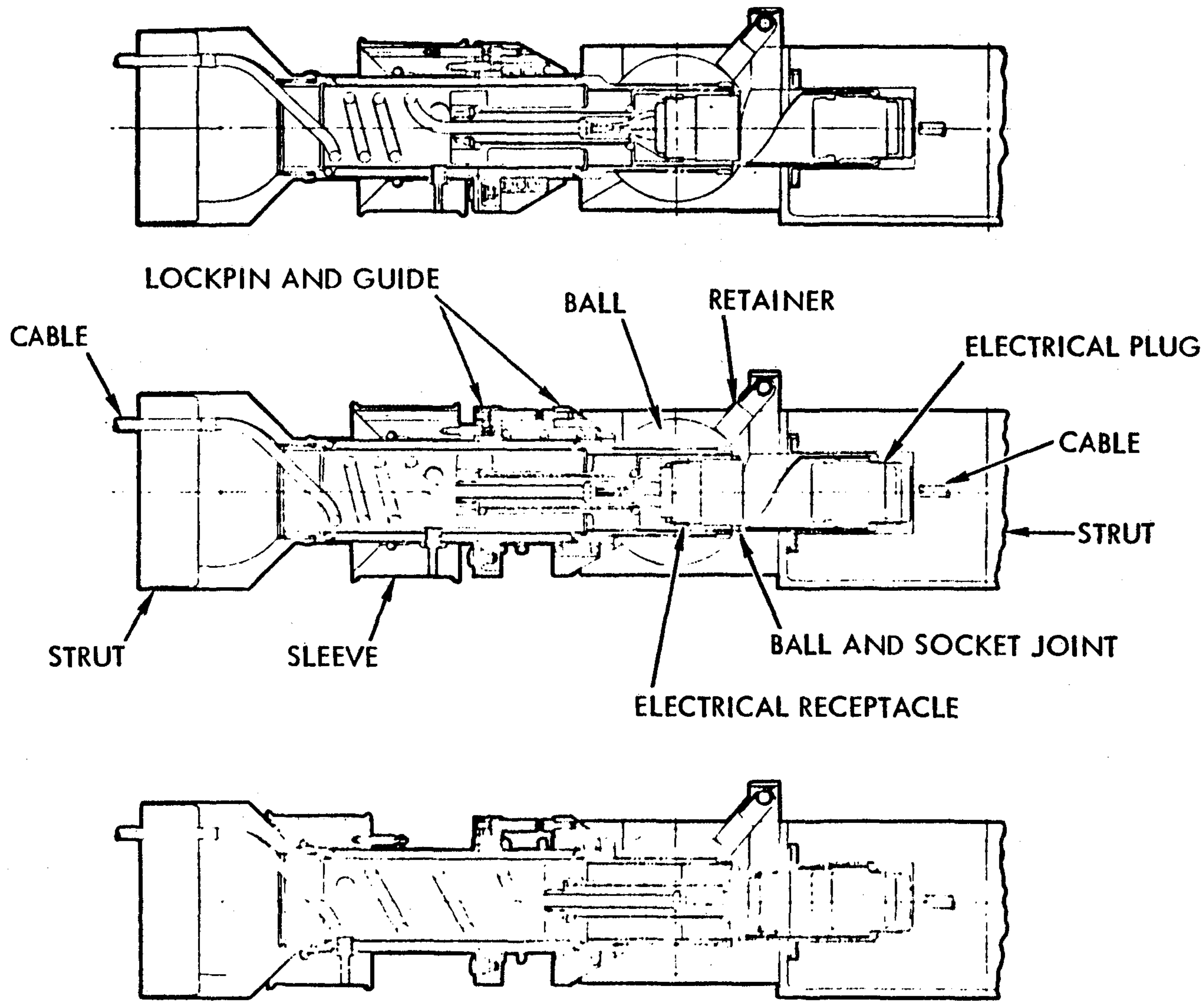


Concept - Integrated Electrical/Structural Connector

To reduce the number of individual connections which must be made, a combined electrical/mechanical joint has been conceived. This incorporates a multi-pin electrical connector with the mechanical ball joint shown previously. The ball joint is engaged first and locked into the socket. Then, the electrical connection is made with the aid of the lockpin and guide to assure proper alignment and indexing.

This design concept was developed at Rockwell under this NASA Contract NAS1-15322, but a different task assignment entitled, "Test Article Concept for Integrated Fiber Optics Cable Connector."

CONCEPT - INTEGRATED ELECTRICAL/STRUCTURAL CONNECTOR



Connector Component Trades/Selection

The connector component is a critical (enabling) technology item with respect to successful assembly and erection of space platforms. Due to the amount of services (power, data, fluid) that have to be interfaced between each segment of the distribution lines and the payload-to-platform, the best operational procedures for any potential design concept is indeterminate at this current stage of development. It is possible that the requirements are such that it will limit automated remote connections being performed via the RMS. The degree, if required, of EVA assistance must be determined and the implication of assembly scheduling and design impact must be understood fully. Therefore, at this present time, no definite approach has been selected regarding space assembly of connectors.

CONNECTOR COMPONENT TRADES/SELECTION

COMPONENT	ALTERNATES	SELECTION	TECH LEVEL
<u>ELECTRICAL</u>			
POWER	REMOTE		1
	EVA ASSIST		1
C/C	REMOTE		1
	EVA ASSIST		1
	ELECTRICAL		2
	FIBER OPTICS	ADVANTAGE NOT APPARENT FOR THIS APPLICATION	-
<u>FLUID</u>	REMOTE		1
	EVA ASSIST		1

NOTE: Primary trades in each case involve determining (1) degree of EVA assist, if any, which results in maximum reliability and minimum cost, degree of integration practical in single connector assembly; and (2) workable structural mounting/connector makeup concept.

Connector Component Technology Candidates

Two possible technologies are to develop designs for both electrical (power and signal) and fluid connector. Extensive ground simulation testing should be undertaken to identify the degree of automation possible, and the EVA involvement necessary. The disconnection of fluid connectors with zero residual must be a design requirement since it is important to many potential platform users to have zero contamination.

CONNECTOR COMPONENT TECHNOLOGY CANDIDATES

ELECTRICAL

OBJECTIVE

- RELIABLE MULTIPLE CONNECTORS CAPABLE OF REMOTE CONNECT/DISCONNECT

TECHNOLOGY ASSESSMENT

- SPACE-RATED CONNECTORS EXIST BUT ARE NOT DESIGNED FOR REMOTE CONNECT/DISCONNECT NOR FOR PIN MULTIPLES AND POWER LEVELS CONTEMPLATED

APPROACH

- DEVELOP DESIGNS SPECIFIC FOR PLATFORM REMOTE APPLICATION—MUST INCLUDE MULTIPLE POWER AND SIGNAL INTERFACE CONNECTORS

EXPECTED RESULTS

- DEVELOPMENT AND DESIGN VERIFICATION OF FAMILY OF DC-RF SINGLE/MULTI-PIN CONNECTORS FOR REMOTE ACTIVATION WITH MINIMUM EVA

FLUID

OBJECTIVE

- RELIABLE, ZERO-LEAKAGE CONNECTORS CAPABLE OF REMOTE CONNECT/DISCONNECT

TECHNOLOGY ASSESSMENT

- SPACE-RATED CONNECTORS EXIST WHICH ARE SAFE AND RELIABLE, BUT NONE ARE DESIGNED FOR REMOTE CONNECT/DISCONNECT—SOME WORK ON QUICK CONNECTS DONE FOR INTEGRATED ORBITAL SERVICE SYSTEMS STUDY (MARTIN/FAIRCHILD)

APPROACH

- DEVELOP DESIGNS SPECIFIC TO PLATFORM REMOTE APPLICATION
- PERFORM CONNECTS & DISCONNECTS WITH RMS SIMULATOR
- REMOTE TESTS IN VACUUM CHAMBER FOR ZERO LEAK AND RESIDUAL DISPERSAL ON DISCONNECTS

EXPECTED RESULTS

- DEVELOPMENT AND DESIGN VERIFICATION OF FREON CONNECTORS FOR REMOTE ACTIVATION WITH MINIMUM EVA

Electrical/Fluid Connector Technology Program Planning

A top level funding/scheduling is shown for the overall area of connector technology. These values do not include the flight component or its subsequent flight qualification.

ELECTRICAL/FLUID CONNECTOR TECHNOLOGY PROGRAM PLANNING

	1979	1980	1981	1982	1983	1984	1985	1936
			PHASE C/D GO-AHEAD			PLATFORM LAUNCH		
ELECTRICAL/FLUID CONNECTORS		400K	330K	500K	750K			
EXPAND CURRENT DESIGN CONCEPTS								
ADD REMOTE & EVA-ASSIST CAPABILITY								
HUMAN FACTORS STUDIES								
EXPERIMENTAL HARDWARE								
PROTOTYPES								
RMS SIMULATOR TESTS								
GROUND VACUUM CHAMBER OPERATIONAL TEST								
FLIGHT COMPONENT QUALIF.								

4.9 RENDEZVOUS AND DOCKING

Revisits of the Shuttle orbiter with the platform/utilities module must contend with the problem of rendezvous, terminal braking, and final berthing of the orbiter with the platform. It appears likely that the platform/utilities module will be placed in a quiescent mode of operation, the arrays feathered and/or partially retracted, at a stable attitude.

The orbiter will be "flown" to a close rendezvous position, with the platform target, always in direct line of sight (LOS) of the orbiter crew. The final terminal rendezvous and braking, if required, must be precision-controlled with respect to final velocity and position, to allow successful capture by the remote manipulator system (RMS).

The problems are concerned with the approach of a large, flexible structure that should not be contaminated or disturbed by the terminal-braking plume impingement, the placing of a slow-moving RMS to be positioned correctly to capture the platform, and the attenuating and then berthing of the flexible platform using the limited force (moment) capability of the RMS.

Rendezvous and Docking with Platform

There are many basic issues and constraints affecting the problem of safely flying the orbiter to rendezvous, final closure, and docking with large area flexible space platform systems. A prime consideration is the need for a fail-safe trajectory approach wherein the orbiter will automatically avoid the platform if no additional maneuvers or operations are performed. Throughout the approach, the orbiter orientation must provide continuous line-of-sight visibility with the target platform which must be in a stable attitude.

The current practice of "hard" docking cannot be tolerated with the lightweight flexible structures envisioned for the erectable platforms. This indicates the need for "soft" docking at very low approach velocity and with a minimum of terminal braking to avoid plume impingement onto the platform and equipment.

Final docking at these slow approach velocities will be using the RMS to engage with the target and perform the final velocity attenuation. After capture the RMS must maneuver onto the appropriate platform docking ports. The positioning of the docking ports and engagement fittings for capture by the RMS are strongly influenced by the approach trajectory and terminal braking procedures.

RENDEZVOUS AND DOCKING WITH PLATFORM

- FAIL-SAFE APPROACH TRAJECTORY
- PLATFORM IN STABLE ATTITUDE PRIOR TO ORBITER RENDEZVOUS
- PLATFORM IN VIEW FROM THE ORBITER AT ALL TIMES DURING APPROACH
- SOFT DOCKING - VERY SLOW APPROACH VELOCITY
- MINIMAL ORBITER BRAKING MANEUVERS
- MINIMAL PLUME IMPINGEMENT ONTO PLATFORM & EQUIPMENT
- RMS ATTENUATION FOR FINAL MANEUVERING ONTO DOCKING PORT
- WHERE DOCKING PORTS ARE POSITIONED ON LARGE SPACE PLATFORM FOR CLEAR APPROACH PATH

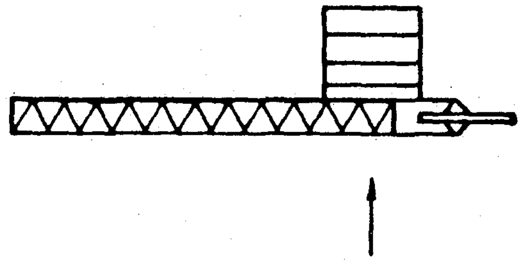
Rendezvous Approach Techniques

There are numerous methods for rendezvous approach; two typical approaches are shown. The "catch up" maneuver is accomplished in the orbit plane and thrusts toward the platform from an initial standoff distance (a ballistic path is assumed for the fail-safe trajectory). A second approach is for the orbiter to be at a different altitude than the target, and the orbiter performs a radial thrust maneuver for its translation to the target. It is important that the thrust maneuvers be performed such that the plume direction is away from the target to alleviate contamination and/or tip-off problems.

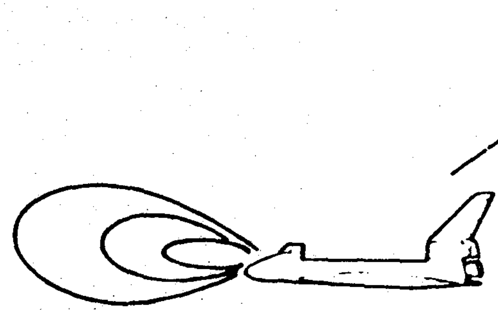
Both of the above approaches assumed that the target platform and orbiter were both in plane to one another. Another approach technique is to employ out-of-plane maneuvers and have the orbiter cross the target platform plane twice every orbit for the rendezvous.

All of the approach trajectories shown on the next series of charts are evaluated at a 250 nmi (463 km) altitude.

RENDEZVOUS APPROACH TECHNIQUES

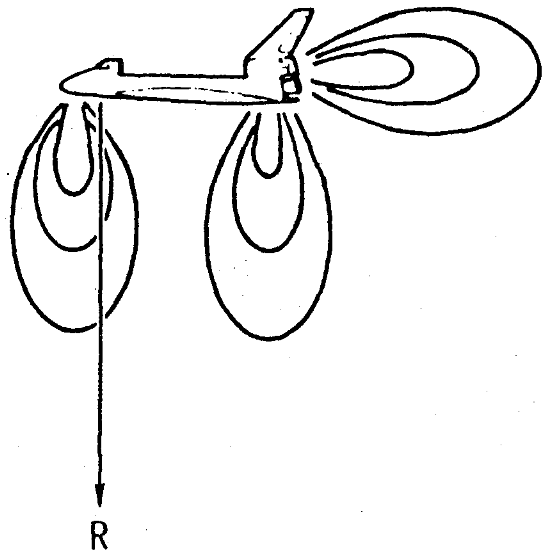


PILOT HOLDS R POSITION AND TRANSLATES UPWARD; ORBITAL MECHANICS EFFECTS PROVIDE BRAKING FORCE



PILOT HOLDS HORIZONTAL AND RADIAL STANDOFF; THRUSTS TOWARDS PLATFORM FOR CATCH UP ALONG FAIL-SAFE TRAJECTORY

1 CATCH UP



2 RADIAL THRUSTING

"Catch Up" Rendezvous Ballistic

A typical series of approach trajectories for the catch-up rendezvous maneuver are shown for an initial stand-off distance of 500 ft (152 m) behind the target and about 310 ft (94.5 m) below. For the RMS nominal capture profile, an initial radial velocity of 0.5877 ft/sec (0.1791 m/s) is required to maneuver the orbiter toward the target so that when they are at closest approach, their relative velocity is zero.

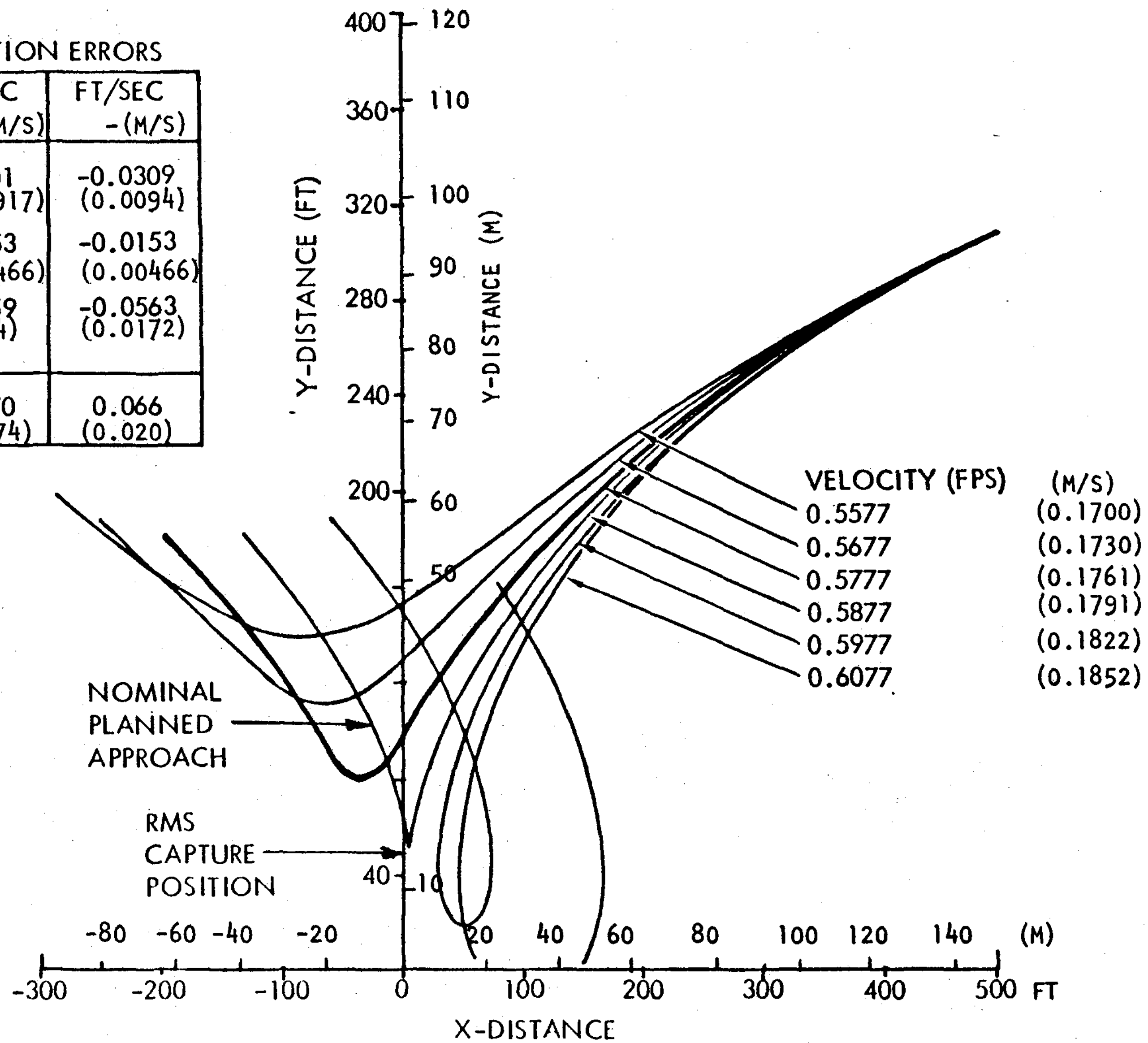
Other profiles shown are for errors in the initial starting velocity. The results show the extreme sensitivity to velocity errors as small as 0.01 ft/sec (0.003 m/s) will result in spatial position errors at encounters of about 30 ft (9.1 m/s)

Included in the table are estimates of the cutoff velocity errors that can be expected from the orbiter RCS. It is obvious that these errors are too large to control a ballistic approach from a standoff distance of 500 ft (152 m). Therefore, the catch-up rendezvous with a single thrust correction to achieve essentially zero velocity at capture is impossible with the current operations of the orbiter's RCS.

"CATCH UP" RENDEZVOUS BALLISTIC
Effect of Velocity Errors

ORBITER ΔV EXECUTION ERRORS

DIRECTION	FT/SEC +(M/S)	FT/SEC -(M/S)
X	0.0301 (0.00917)	-0.0309 (0.0094)
Y	0.0153 (0.00466)	-0.0153 (0.00466)
Z	0.0459 (0.014)	-0.0563 (0.0172)
RSS	0.0570 (0.0174)	0.066 (0.020)

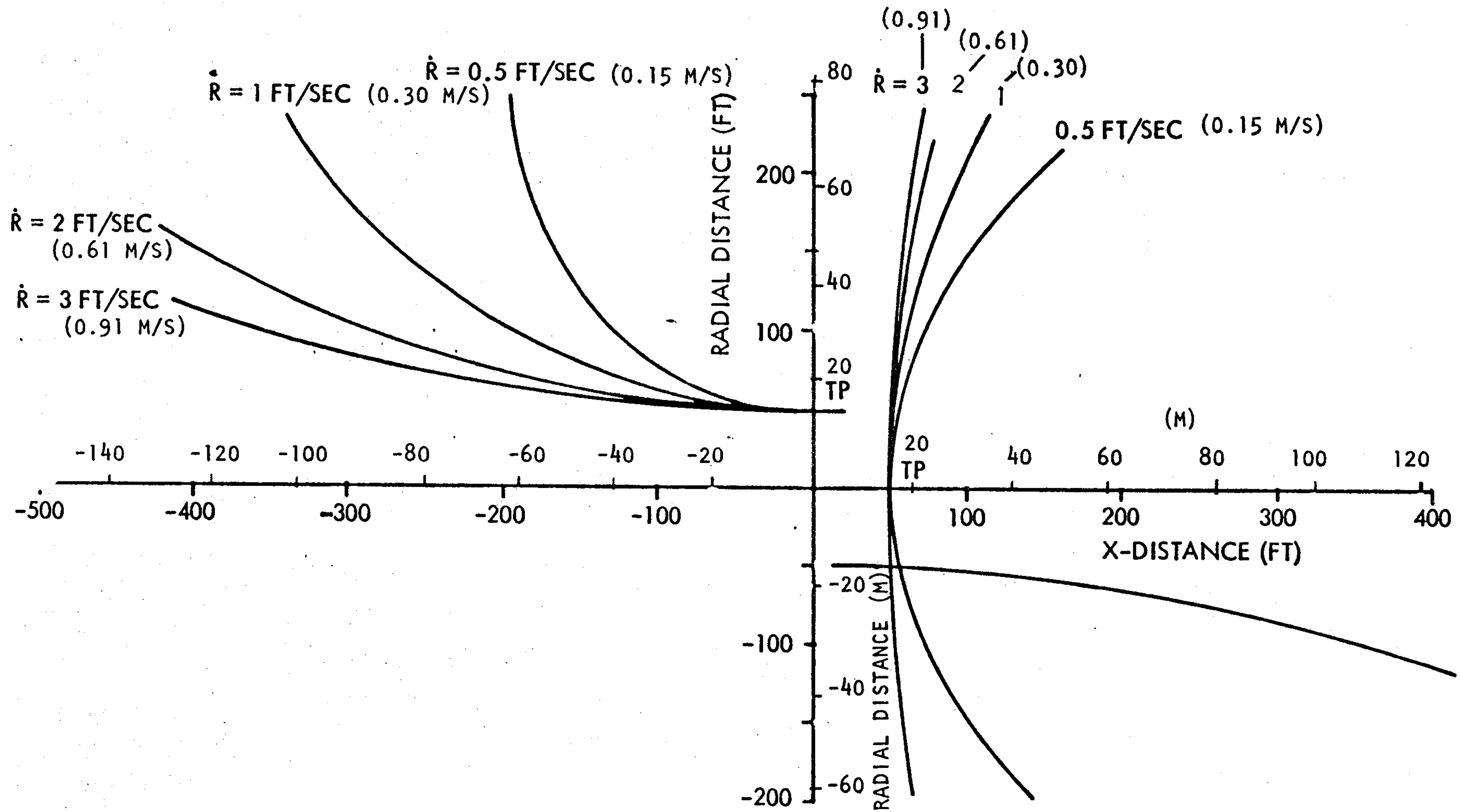


"Catch-Up" Rendezvous With Terminal Braking

A variation of the "catch-up" approach is to allow a terminal thrust maneuver just prior to capture. This approach is shown here for a series of maneuvers where the target is in front or behind the orbiter and there is only a horizontal velocity component at capture, or the target is below and the terminal velocity is completely vertical with no horizontal component.

Shown are the loci of starting positions for various initial velocities ranging from 0.5 to 3 ft/sec (0.15 to 0.91 m/s). Each trajectory will result in a closest approach to the target of 50 ft (15 m).

"CATCH UP" RENDEZVOUS WITH TERMINAL BRAKING



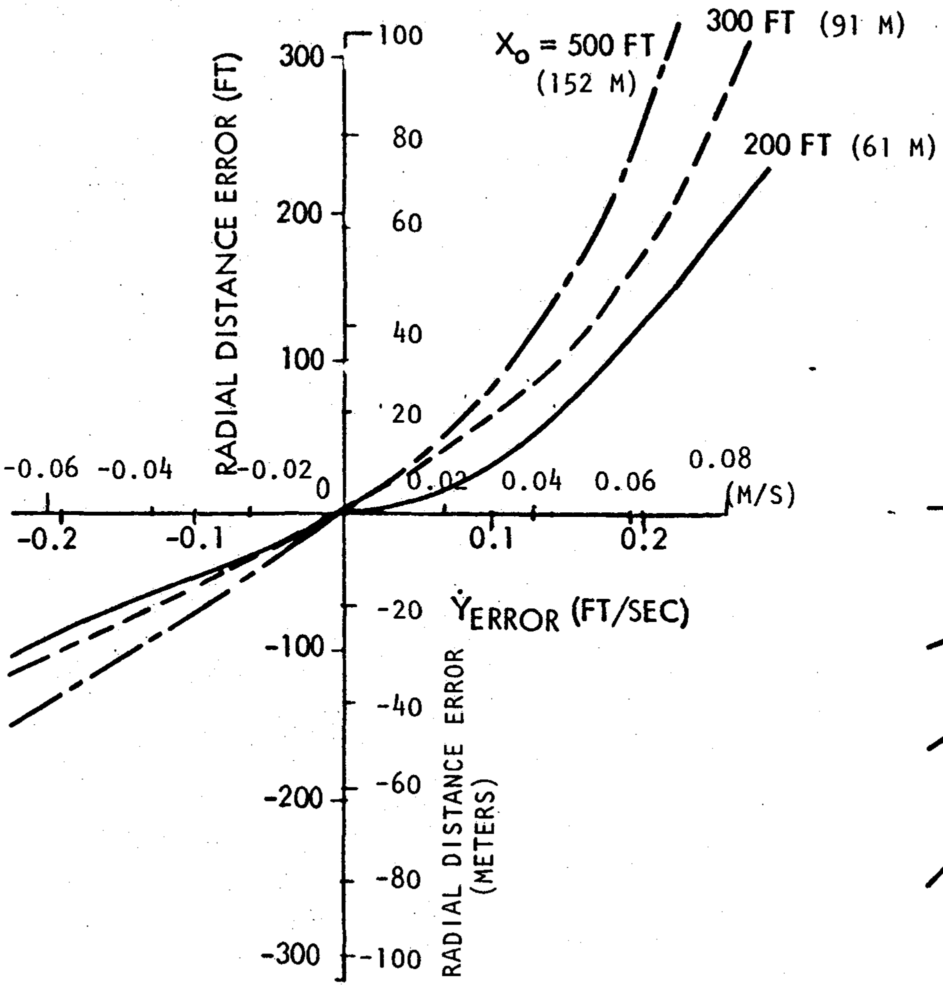
Position Sensitivity To Initial Velocity Errors

Although there is a terminal rendezvous velocity, the approach trajectories are still acceptable to initial position, pointing and velocity errors. The velocity errors have the most significant effect on the final radial distance error. The curves show that for the low rendezvous velocity of 0.5 ft/sec (0.152 m/s) the positional errors are too large for successful capture if the initial velocity error V >0.05 ft/sec (0.0152 m/s). As expected, the errors are reduced as the stand-off distance of the initial maneuver is reduced.

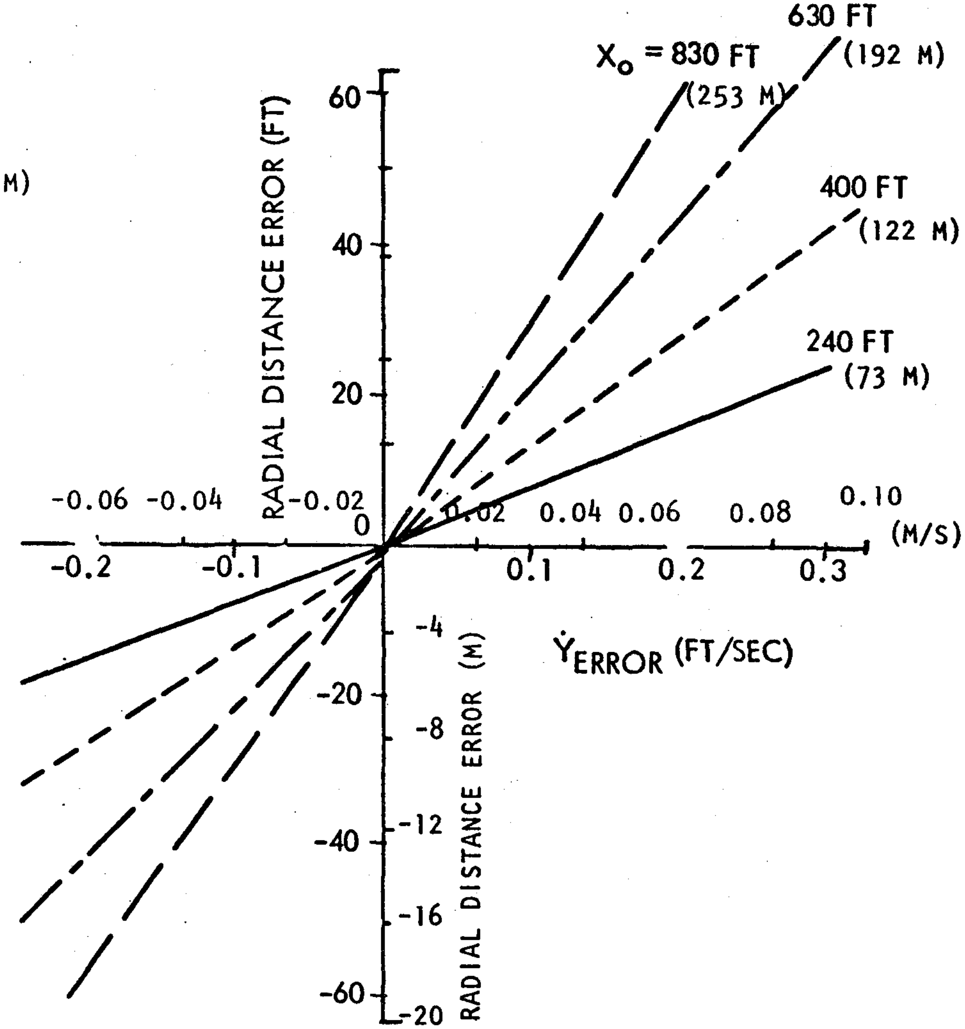
Radial distance errors within acceptable bounds can be obtained if the rendezvous velocity is increased. The chart shows the results for a rendezvous velocity of 3.0 ft/sec (0.91 m/s) where less than 50 ft (15.2 m) radial error is possible with velocity errors as large as 0.15 ft/sec (0.046 m/s). The problem with this approach is the high closing rate (3.0 ft/sec, 0.91 m/s) between the target and the orbiter which makes for difficulty in pointing and maneuvering the RMS end effector for successful capture.

POSITION SENSITIVITY TO INITIAL VELOCITY ERRORS

X_o = INITIAL SEPARATION DISTANCE



RENDEZVOUS VELOCITY = 0.5 FT/SEC (0.152 M/S)



RENDEZVOUS VELOCITY = 3.0 FT/SEC (0.91 M/S)

Trajectory Sensitivity

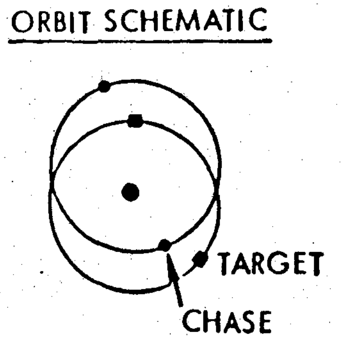
This chart highlights the trajectory sensitivity problem. Target miss distance coefficients (feet per fps) are shown for a representative in-plane closure path. For example, even with a relatively "close in" final maneuver (200 ft, 61 m) and a slow closing velocity (1.5 fps, 0.45 m/s), the miss coefficients are 35 feet per fps (10.7 m per m/s) and 130 feet per fps (39.6 m per m/s) for $\dot{x}$ and $\dot{z}$ errors, respectively.

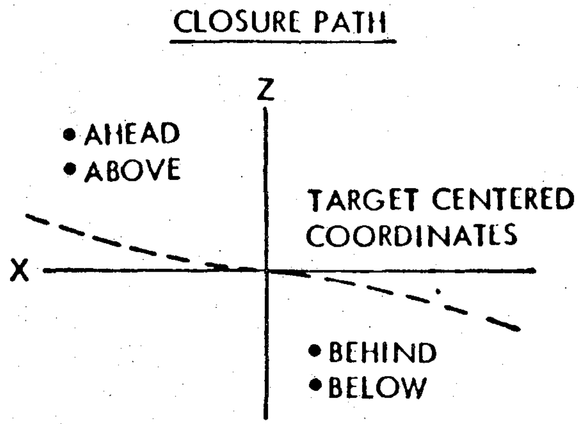
There are many other possible approach paths; but all have high sensitivities to maneuver errors, with sensitivity increasing at lower approach speeds. Previous investigations have indicated the orbiter minimum ΔV capability may be up to 0.45 fps (0.137 m/s) for certain conditions. This large minimum ΔV value (0.45 fps, 0.137 m/s) is due to rotation/translation coupling, minimum thruster impulse characteristics, and quantization increments designed into the flight control system software.

Thus, it is concluded that all approach trajectories will require significant proximity RCS firings which can cause plume effects on the target vehicle.

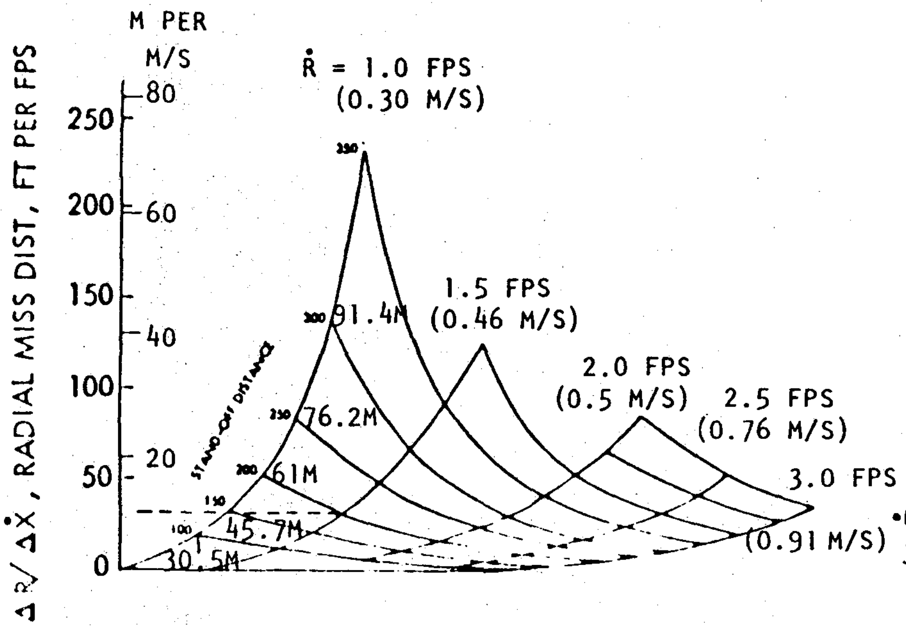
Higher approach speeds may reduce the plume impingement problem through reduced trajectory sensitivity.

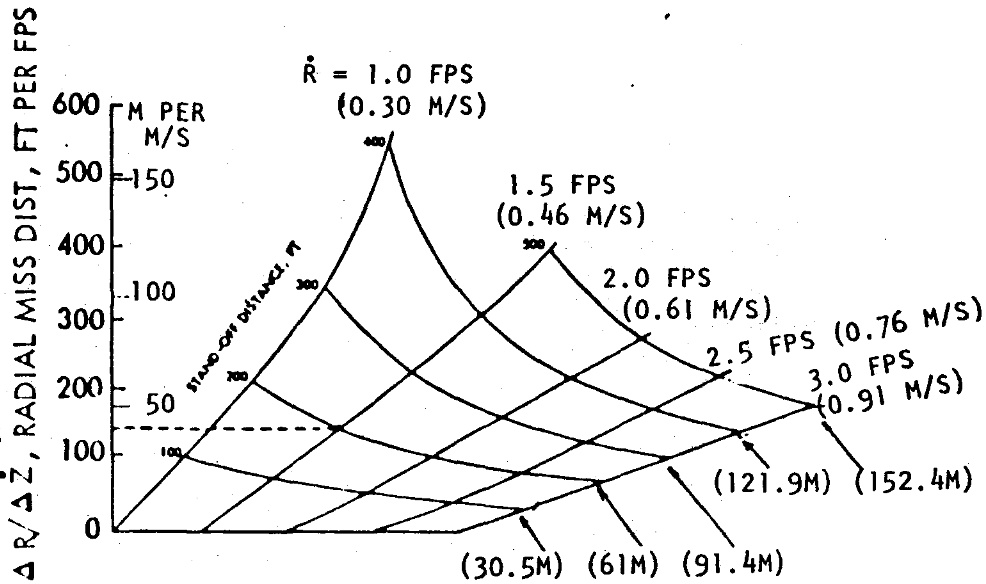
TRAJECTORY SENSITIVITY





- MANY POSSIBLE APPROACH PATHS
- LOW APPROACH SPEEDS RESULT IN HIGH ERROR SENSITIVITY
- ORBITER MIN ΔV'S UP TO 0.45 FPS (0.137 M/S)
- THUS, ALL APPROACHES REQUIRE PROXIMITY RCS FIRING
- APPROACH ΔV'S >3 FPS (0.91 M/S) PROBABLY REDUCE PLUME IMPINGEMENT PROBLEM





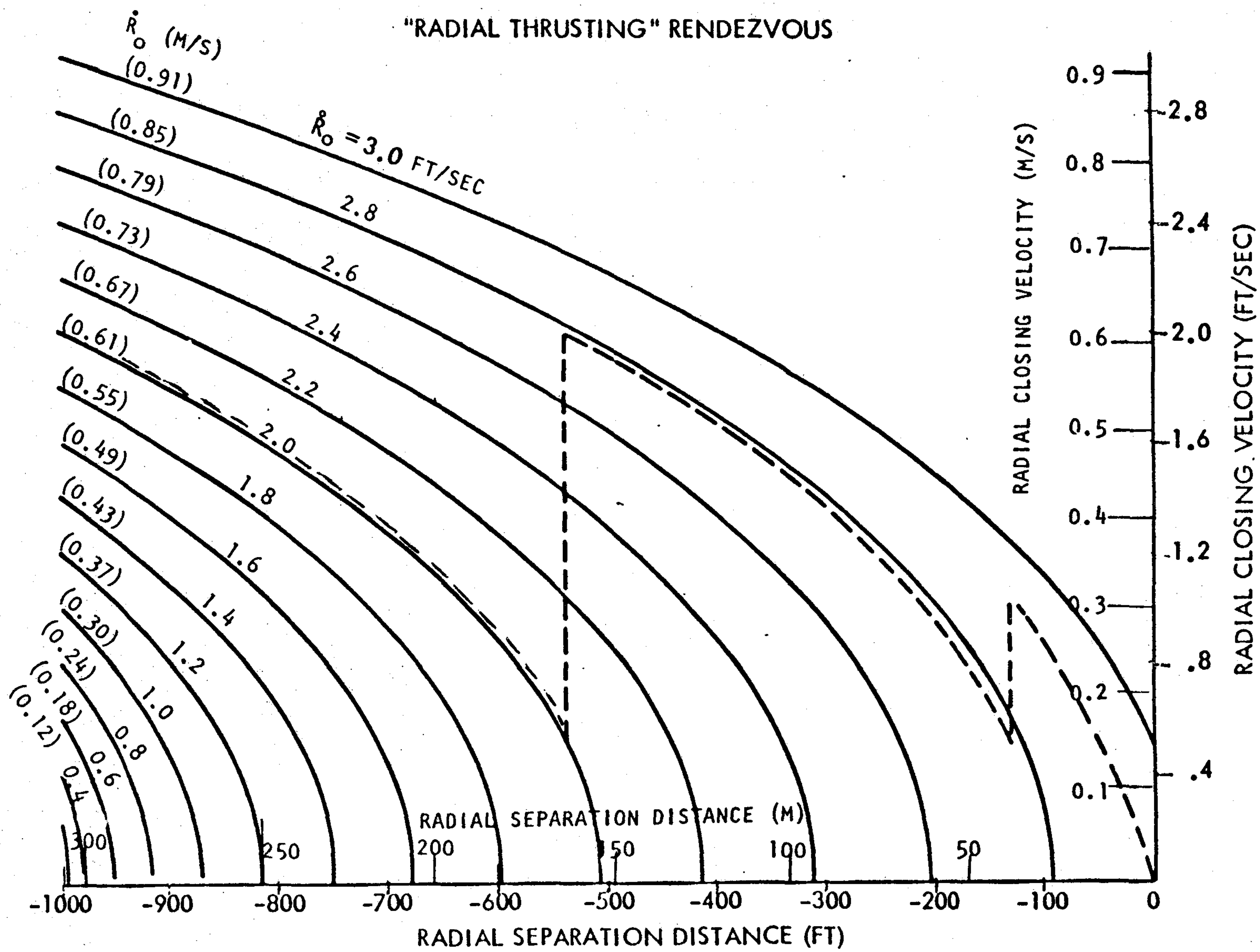
Radial Velocity/Distance Profiles

A second approach for rendezvous is to employ a radial thrusting maneuver which gradually rises the orbiter's orbit toward that of the target. The benefits of this approach result from the fail-safe aspect of target backing away from target if no action is taken, and the thrust plume being directed away from the target for the radial velocity increment ($\dot{R}_O$). The chart shows the radial separation distance for a series of initial starting conditions. For example, if the two objects are separated 1000-ft (305-m) radial distance and a radial velocity of 2.0 ft/sec (0.60 m/s), the orbit will close to within 510 ft (155.3 m) of the target, have zero radial closing velocity at this point, and then back away.

Shown is the trajectory for a typical three-thrust maneuver where the radial velocity is allowed to decay to 0.5 ft/sec (0.15 m/s) before the next thrust is applied. This could result in a zero radial separation distance and closure velocity.

RADIAL VELOCITY/DISTANCE PROFILES

"RADIAL THRUSTING" RENDEZVOUS

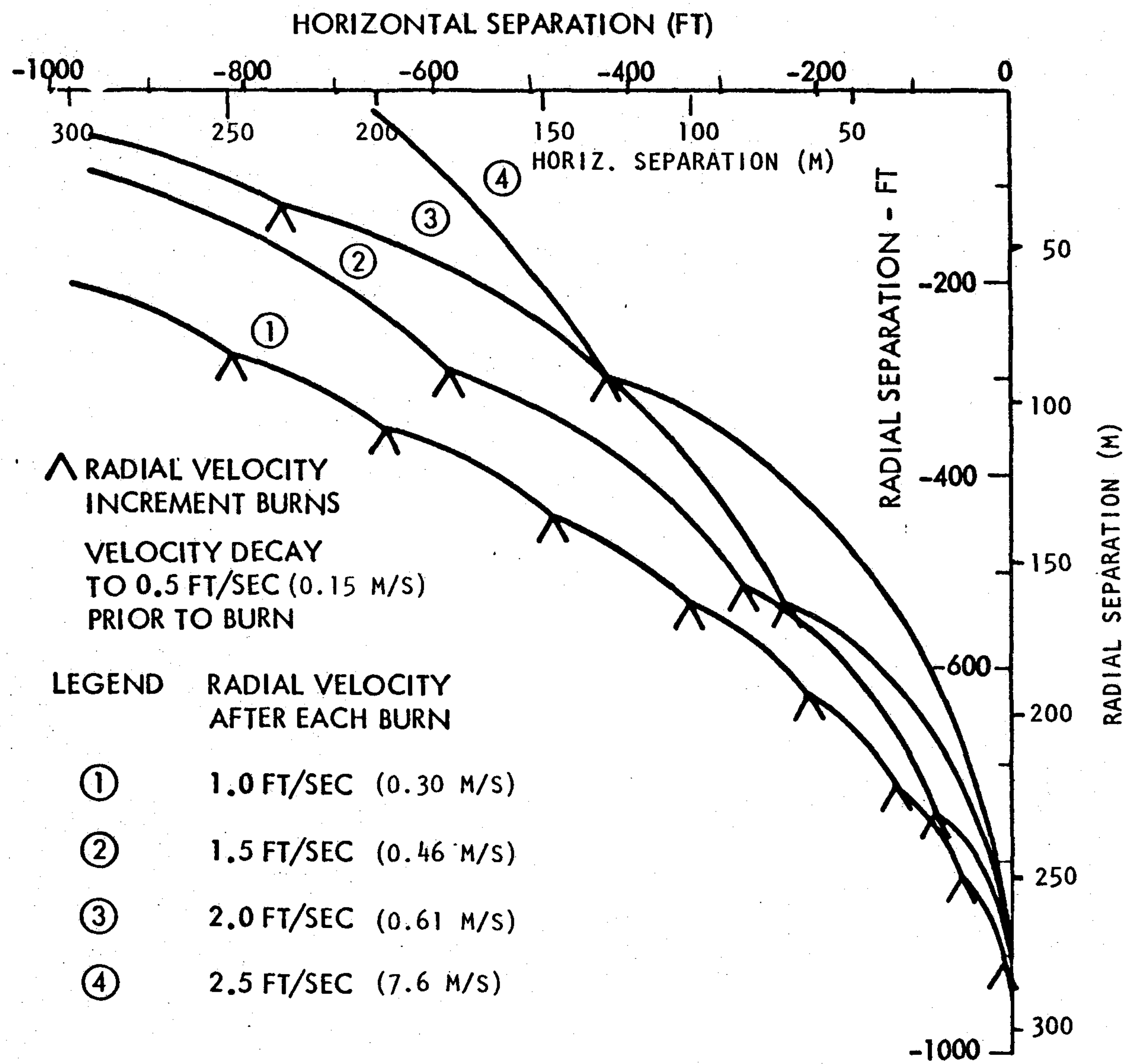


Multi-Thrust Approach Profile

The chart on the opposite page shows the horizontal separation for the multi-thrust radial approach. Curve 2 has a typical three-thrust (1.5 ft/sec, 0.46 m/sec) maneuver and achieves zero radial velocity at closest approach, but there is a significant horizontal velocity and displacement. The latter can be account for by initially positioning the orbiter not directly below the target, but about 1000 feet (305 m) ahead of the target.

Both the horizontal velocity and displacement can be reduced by imparting a small initial horizontal velocity. Unfortunately, any horizontal thrusting will have adverse effects, both from potential plume impingement and greater sensitivity to initial maneuver errors.

MULTI-THRUST APPROACH PROFILE

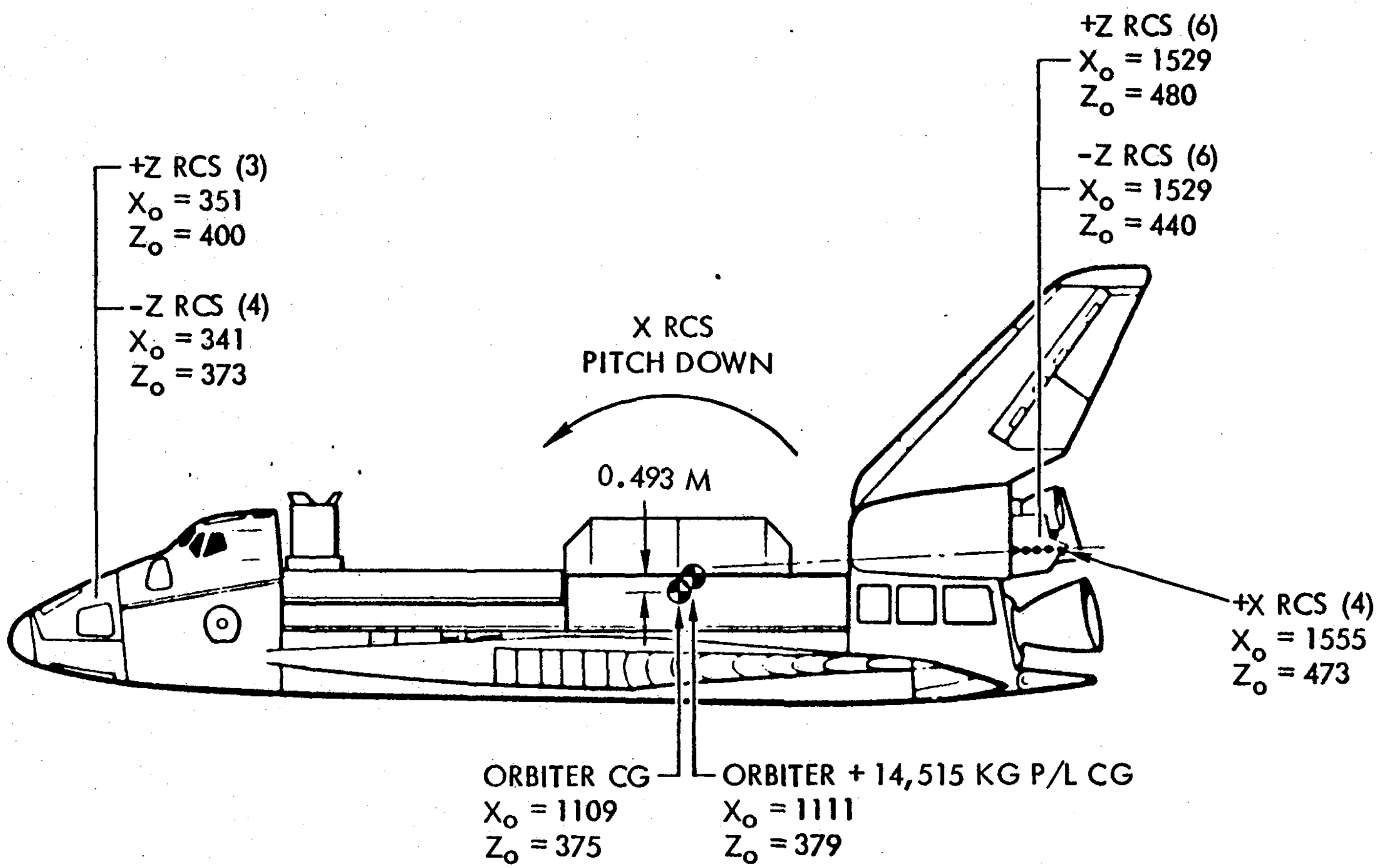


RCS Braking Maneuver

This chart depicts the pertinent thruster locations and configuration geometry important to +X RCS braking maneuvers. Typical orbiter center-of-gravity locations are shown to result in up to a 19.4-inch (0.493-meter) vertical offset from the 10-degree canted +X RCS thrust centerline. To balance the pitch-down moment associated with this type of thrusting maneuver, the Z-axis thrusters must be operated with a 1.39-percent duty cycle.

This is a simplistic model which basically neutralizes static thrust moments and does not include attitude deadband considerations, minimum impulse bit sizes, nor coupling between the different control axes. However, it is a good first-order approximation of the amount of Z-thruster firing required for steering during RCS X-axis braking maneuvers. The duty cycle factor is relatively sensitive to vertical center-of-gravity location.

RCS BRAKING MANEUVER



$$Z_{RCS} \text{ DUTY CYCLE } = \frac{4 \times 19.4}{4 \times (1109-341) + 6(1529-1109)} \times 100$$

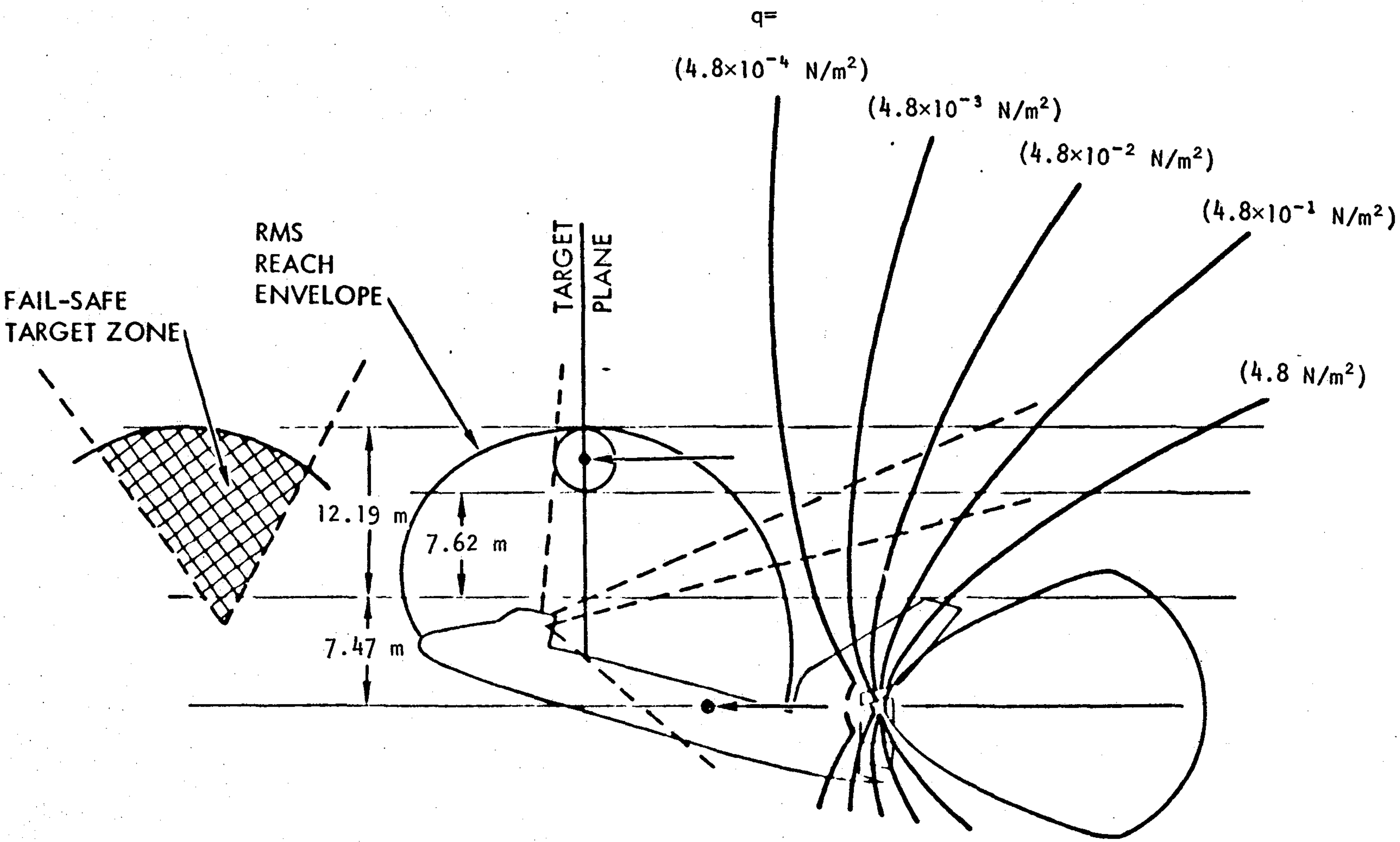
Z_{RCS} DUTY CYCLE = 1.39% DURING +X RCS FIRING

Fail-Safe Target Zone and Flowfield Contours OMS Engine

Approaching the target tail first will provide continuous LOS visibility. The orbiter in a slight tail down attitude will provide a maximum fail-safe target zone of 12.2 m depth on the orbiters center line. The target zone is dictated by the reach envelope of the RMS and the target capture should be made downstream from the target plane to allow for the forward motion of the target during the RMS attenuation procedure.

If any terminal braking is required, the orbiter's OMS engine or X-axis reaction jets can be employed to negate any relative motion in the X-direction. With the docking port positioned on the trailing edge of the platform, the resulting orbiter plume impingement can be effectively minimized. Shown are the flowfield contours for the OMS engine which are significantly smaller than the RCS X-axis jets in the region of the target plane.

FAIL-SAFE TARGET ZONE AND FLOWFIELD CONTOURS OMS ENGINE



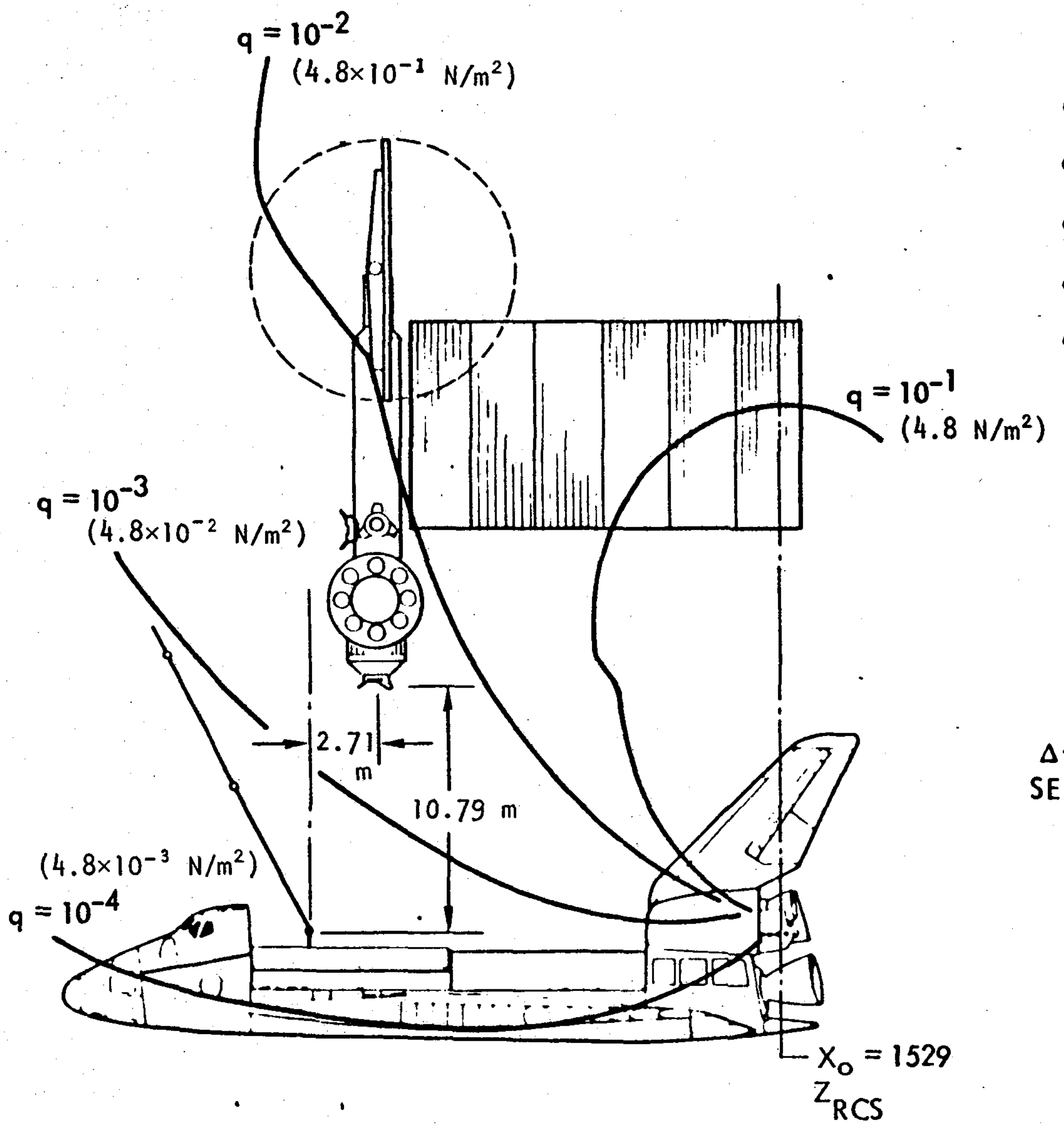
Plume Impingement Geometry

This chart illustrates relative orbiter initial service module positions and orientations for a "horizontal" approach path (relative velocity along the orbiter X-body axis). The service module is positioned for a 3 fps (0.915 m/s) "stopping" maneuver which would place the module directly over the RMS body station. This maximizes the RMS reach envelope to accommodate orbiter stopping maneuver tolerances. Dynamic pressure (q) contours for a single (Z-axis) RCS thruster are superimposed on this closure geometry.

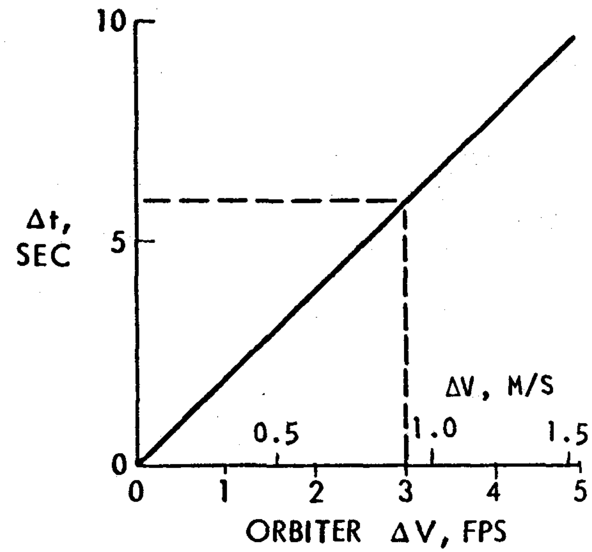
The 35.4 ft (10.79 m) vertical distance provides orbiter tail clearance while staying within the RMS reach. The 8.9 ft (2.71 m) horizontal dimension is the distance required to "stop" the orbiter (ΔV = 3 fps - 0.915 m/s) with a typical weight of 222,579 lb (100,960 kg) (four X-axis thrusters).

Although the Z-thruster duty cycle is 1.39 percent, the pressures contained in the vacinity of the target are three orders of magnitude larger than the main OMS engine plume, which might not need any Z-thruster duty cycle to perform the required "stopping" maneuver.

PLUME IMPINGEMENT GEOMETRY



- SINGLE THRUSTER PLUME DEF
- MULTIPLIED BY 6 (ALL AFT +Z THRUSTERS)
- DUTY CYCLE FACTOR (1.39%)
- AVG ΣF & ΣM ON OSM
- TYPICAL ORBITER WT: 86445 KG +14515 KG



Plume-Induced Motion

Based on the orbiter "stopping" geometry using its RCS thrusters depicted in the preceding chart, the plume-induced translation and rotational motion imparted to the configured service module of the space platform were calculated and are shown as a function of the final orbiter braking maneuver ΔV. These are the $\dot{d}$'s and $\dot{\theta}$'s induced by a 1.39% duty cycle of the Z-thrusters during aft X-thruster braking maneuvers.

Typical values are highlighted for an orbiter ΔV of 3 fps (0.915 m/s). For Case 1, service module vertical orientation, $\dot{d}$ = 0.05 fps (0.15 m/s) and $\dot{\theta}$ = 0.2 °/sec. For rough, but reasonable, time intervals of say 100 seconds, to track and grasp the platform with the RMS, the service module will have moved 5 ft (1.5 m) and rotated 20 degrees. This illustrates the dynamic nature of the RMS berthing problem and complications induced by plume effects. The above example conditions are within the force-of-torque limits of the RMS to arrest the platform motion, but the real problem would be to assure proper RMS joint alignment such that the continuing displacement of the service module during the RMS "motion arrest" operation is within the joint motion constraints. (Generalized motion would tend to force a given joint to deflect sideways, compared to its normal degree of freedom.)

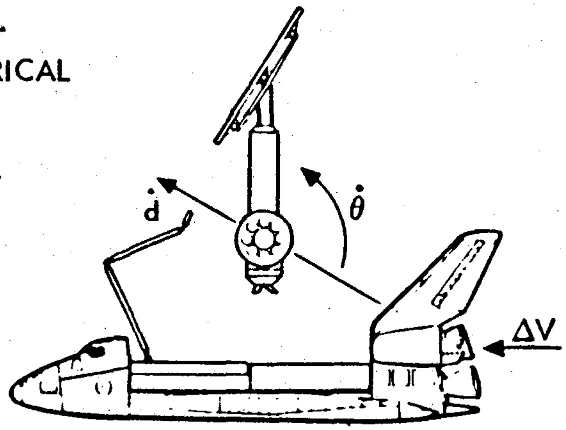
As shown in Case 2 on the chart, greater than order-of-magnitude reductions in the plume-induced service module motions can be achieved by controlling the terminal geometry. Further leverage on plume-induced motions is possible by controlling the orientation of the solar array ("feathering" the array to reduce frontal area in the direction of the plume stream lines), or retracting the array prior to docking.

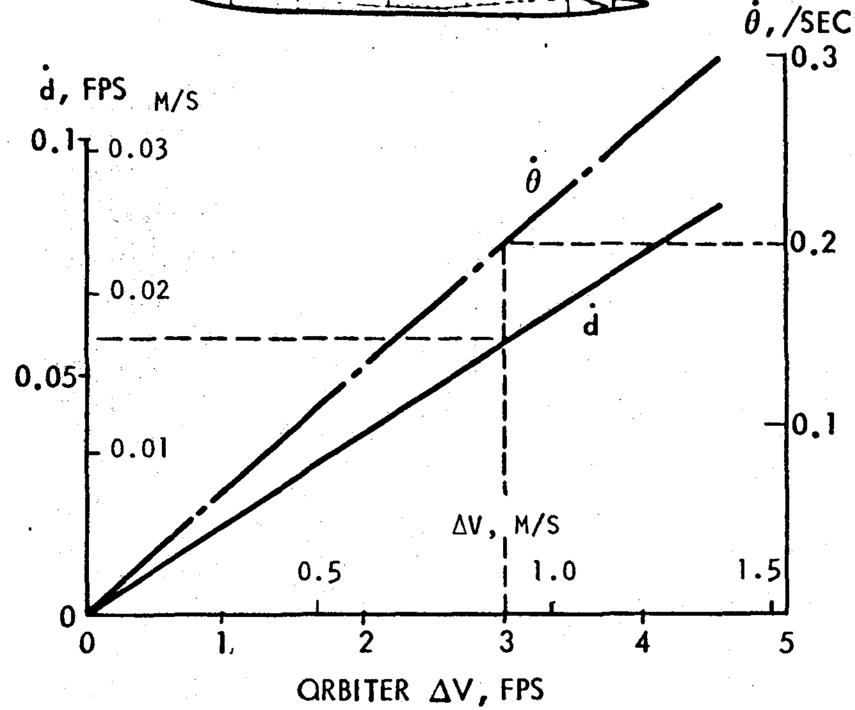
Thus, if the orbiter can be safely and accurately placed in position for a final braking maneuver, terminal geometry control can be used to reduce plume-induced motions to livable levels, with either the RCS or OMS engines.

PLUME INDUCED MOTION

CASE 1

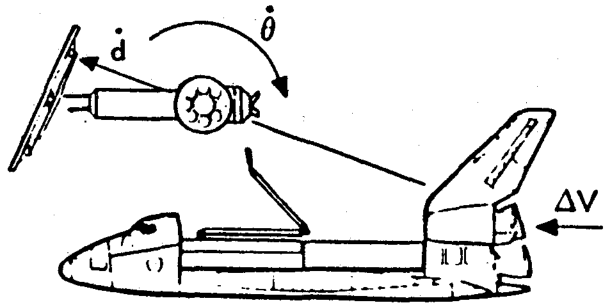
- SYMMETRICAL CONFIG
- VERTICAL ORIENT

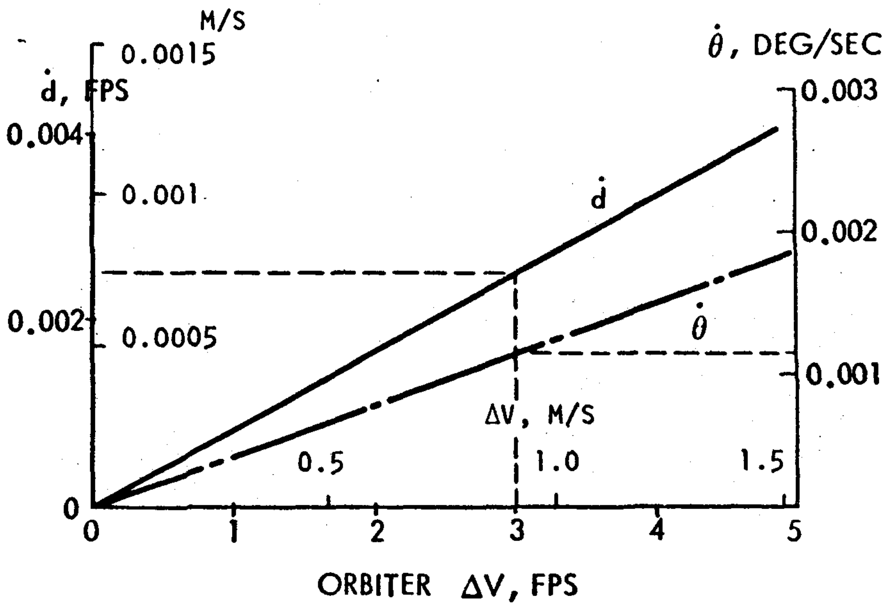




CASE 2

- SYMMETRICAL CONFIG
- HORIZONTAL ORIENT





Rendezvous and Docking Trades/Selection

This chart shows the various trades and selection that were made in the rendezvous and docking technology areas. It is recognized that these are enabling technologies that should be resolved early as they will have a profound impact on the on-orbit operations and design configuration arrangement for the docking ports, etc.

A radial rendezvous approach was selected as being the least sensitive to initial positional and velocity errors. Final capture and docking is accomplished by using the RMS to grap the platform in passing and gradually attenuating the relative motion between the platform and the orbiter.

During the assembly operation the platform is moved to the next work station by walking the platform along with the RMS and grasping hold at alternate docking ports or platform edge member supports.

Since this platform is not a manned habitat, the docking port can be a simple structural connection containing interface connection for power, signal, and fluids.

RENDEZVOUS AND DOCKING TRADES/SELECTION

COMPONENT	ALTERNATES	SELECTION	TECH. LEVEL
RENDEZVOUS	CATCH UP	TOO SENSITIVE TO INITIAL ERRORS	–
	NORMAL TO ORBIT PLANE	—	–
	RADIAL VECTOR APPROACH	NEEDS FURTHER INVESTIGATION	1
DOCKING PROCEDURES	NASA DOCKING MODULE	S.O.A.—BUT NOT WITH FLEXIBLE STRUCTURE OF ORBITER	–
	THRUST ATTENUATION	—	–
	EVA	—	–
	RMS ATTENUATION	METHOD SUGGESTED FOR RETRIEVAL OF PAYLOADS	1
	WALKING ALONG WITH RMS	REPOSITION HOLDS ALONG PLATFORM	1
DOCKING ATTACHMENT	NASA DOCKING PORT, RING ADAPTER	STANDARD SUGGESTED COMPONENTS	–
	UTILITY MODULE DOCKING PORT	POSITION USED FOR INITIAL ASSEMBLY AND UTILITIES SERVICING	1
	PLATFORM EDGE MEMBER SUPPORTS	ATTACHING TO FLEXIBLE STRUCTURE	1

Rendezvous and Docking Technology Candidates

The main technology development will be a recommended rendezvous approach trajectory that allows for a "soft" docking which is insensitive to initial position, pointing and velocity errors, is fail-safe in the event of failure to activate terminal maneuvers, and requires a minimum of braking thrust to alleviate target contamination and tip-off. Development must include both digital and graphical simulation to allow flying the orbiter to the platform. Throughout the approach there must be simulation of the platform's dynamics and field-of-view aspect of the target for the RMS operator.

Another technology will be the ability of activating the RMS to position itself relative to the approaching target and "snagging" the pick-up fitting of the platform with the RMS end effector. The amount of time allowed for this maneuver is dependent on the final approach velocities. Depth perception and estimation of relative ballistic path of the platform just prior to capture has to be ascertained by the RMS operator to effect successful capture. These technologies can be investigated using the RMS hydraulic simulator and a series of moving targets.

RENDEZVOUS AND DOCKING TECHNOLOGY CANDIDATES

RENDEZVOUS APPROACH

OBJECTIVE

- DEVELOP FAIL-SAFE ERROR INSENSITIVE APPROACH TRAJECTORIES WITH MINIMUM OF ORBITER BRAKING THRUST

TECHNOLOGY ASSESSMENT

- APOLLO "FAST" HARD-DOCKING WITH TERMINAL THRUSTING NOT ACCEPTABLE WITH LARGE-AREA FLEXIBLE STRUCTURE

APPROACH

- INVESTIGATE VARIOUS APPROACH CANDIDATES TO DETERMINE THEIR RESPECTIVE SENSITIVITY
- CONDUCT DIGITAL/GRAPHICAL SIMULATION FOR APPROACH DYNAMICS & VIEWING
- PERFORM GROUND ANALOG SIMULATION

EXPECTED RESULTS

- ACCEPTABLE "SLOW" RENDEZVOUS APPROACH— WITH OPTIMUM TARGET VIEWING AND COLLISION AVOIDANCE

DOCKING WITH RMS ATTENUATION

OBJECTIVE

- TO USE THE RMS TO GRASP THE TARGET AND GRADUALLY ATTENUATE THE RELATIVE MOTION OF THE ORBITER AND THE TARGET

TECHNOLOGY ASSESSMENT

- RMS DESIGN SPECS CALL FOR CAPTURING LARGE, RIGID, STABILIZED SATELLITES
- NO GROUND OR ON-ORBIT TESTING CONDUCTED WITH RMS TO SIMULATE DOCKING WITH LARGE FLEXIBLE STRUCTURE

APPROACH

- USE RMS SIMULATOR TO PERFORM ATTACHING & ATTENUATION OF CLOSING TARGETS
- DETERMINE RMS CAPABILITY OF RAPIDLY GRASPING VARIOUS SHAPED ATTACHMENT POINTS
- FIND THE EFFECT ON EFFECTOR DESIGN AND PLATFORM ATTACHMENT POINT REQUIREMENTS

EXPECTED RESULTS

- RMS WITH EFFICIENT EFFECTOR CAPABLE OF RAPIDLY GRASPING PLATFORM POINT AT SLOW RELATIVE MOTION TO ORBITER

Rendezvous and Docking Technology Candidates (Cont.)

There are several docking attachment ports positioned on the platform to allow for various docking positions during the different assembly and growth stages of the platform. Since the ports have only structural, power, signal and fluid interfaces they could consist of a lightweight docking ring which is attached to the flexible support structure. Due to the flexibility of the overall platform the docking has to be a "soft" maneuver. These requirements are different than those associated with the standard NASA docking adapter - considerably less force must be exerted during closure, otherwise the structure will distort. Actual ground simulation tests for candidate design must be conducted to investigate the effects of the flexibility of the backup support structure of the docking ring.

During assembly and on-orbit operations it is required to move the total platform relative to the orbiter. This is accomplished by walking along the platform using the RMS to gradually transfer the platform. Techniques have to be developed and verified by demonstrating that the flexible RMS can perform these maneuvers and fine position the platform for subsequent docking and attachment to the orbiter.

RENDEZVOUS AND DOCKING TECHNOLOGY CANDIDATES (CONT.)

DOCKING ATTACHMENT

OBJECTIVE

- LIGHTWEIGHT DOCKING RING FOR ADAPTATION TO PLATFORM STRUCTURE—RING TO HAVE STRUCTURAL AND UTILITY INTERFACES

TECHNOLOGY ASSESSMENT

- DOCKING PORTS PROVIDED ONLY STRUCTURAL INTERFACE

APPROACH

- IDENTIFY INTERFACE REQUIREMENT FOR REPETITIVE CONNECTS AND DISCONNECTS
- DEVELOP INTERFACE CONCEPTS (POWER & SIGNAL) AND PERFORM GROUND SIMULATION TESTS UNDER ZERO-GRAVITY WITH RMS AND/OR EVA

EXPECTED RESULTS

- DOCKING ADAPTER WITH UTILITY INTERFACES—POWER AND SIGNAL

WALKING ALONG PLATFORM WITH RMS

OBJECTIVE

- TO USE THE RMS TO REPOSITION THE PLATFORM RELATIVE TO THE ORBITER FOR TRANSFER TO ANY DOCKING RING ON PLATFORM

TECHNOLOGY ASSESSMENT

- RMS SPECS CALL FOR TRANSFER OF HEAVY OBJECTS INTO AND OUT OF THE CARGO BAY
- RMS STILL TO BE SPACE-QUALIFIED

APPROACH

- DEVELOP TRANSFER SPEEDS FOR PLATFORM DURING TRANSFER MANEUVERS
- USE RMS SIMULATOR AND DOCKING RING STRUCTURE TO PERFORM TYPICAL TRANSFER MANEUVERS OR FINE-POSITIONING

EXPECTED RESULTS

- PLATFORM TRANSFER PROCEDURE FOR ASSEMBLY & MAINTENANCE WORK ON THE SPACE APPLICATION PLATFORM

Rendezvous and Docking Technology Program Planning

An estimate is provided for the relative funding and scheduling of the four major technical areas associated with rendezvous and docking of the orbiter and the erectable space platform.

RENDEZVOUS AND DOCKING TECHNOLOGY PROGRAM PLANNING

	1979	1980	1981	1982	1983	1984	1985	1936
			PHASE C/D GO-AHEAD			PLATFORM LAUNCH		
RENDEZVOUS APPROACH		100K	100K					
THEORETICAL METHODS								
GRAPHICAL SIMULATION & TESTS								
DOCKING WITH RMS ATTENUATION		100K	100K	100K				
EFFECTOR & ATTACHMENT REQ.								
GROUND TESTS WITH RMS SIM.								
DOCKING ATTACHMENT		200K	200K					
INTERFACE REQUIREMENTS								
DEVELOP INTERFACE CONCEPTS								
GROUND SIMULATION TESTS								
WALKING ALONG PLATFORM		100K	150K					
PLAN TRANSFER PROCEDURES								
BUILD PLATFORM MOCKUP MODEL								
USE RMS HYDRAULIC SIM. TEST								

4.10 VEHICLE OPERATIONS, CONSTRUCTION, AND PAYLOAD HANDLING

This section deals with the operations required to deploy the utilities module, assemble the platform and attach it to the utilities modules, and to install payloads onto the platform. On subsequent charts, the assembly and deployment procedure and timelines are discussed; equipment and fixtures required to support the operations are identified; and the necessary technology program is presented.

Erectable Platform Assembly Operations Analysis Objectives

The general objective of the assembly system analysis for the subject Erectable Space Sciences and Applications Platform Study is to identify an efficient orbiter-based assembly technique. The opposite chart lists topics indicating the importance of minimizing the interference with the orbiter schedule for the platform construction activity. The construction should use the then existing orbiter construction aids (RMS) to the extent feasible and planned extravehicular activity (EVA) when required to simplify design development.

Analyses will be required in order to optimize the balance between platform assembly aid development costs and the conservation of orbiter task time. The assembly system also should be designed so that it may provide utility for platform system maintenance on revisited flights and be useful for payload/pallet exchange operations. Finally, the platform assembly analysis being performed will lead to the identification of technology development required prior to design development of erectable platforms.

ERECTABLE PLATFORM ASSEMBLY OPERATIONS ANALYSIS OBJECTIVES

✓ PROVIDE COORDINATED PLANNING OF PLATFORM/PAYLOADS/ASSEMBLY SYSTEM TO RESULT IN MOST COST-EFFECTIVE PLATFORM PROGRAM

- MINIMIZE NUMBER OF ORBITER FLIGHTS
- MINIMIZE TIME ON ORBIT DURING ASSEMBLY
- MINIMIZE ORBITER TURNAROUND (GROUND OPERATIONS)
- SINGLE ORBITER MODIFICATIONS
- OPTIMUM COMPLEXITY OF ASSEMBLY AID DEVELOPMENT
  - ASSEMBLY FIXTURE
  - END EFFECTORS FOR RMS
  - OTHER MANIPULATOR REQUIREMENTS
- ASSEMBLY SYSTEM UTILITY FOR SYSTEMS MAINTENANCE AND PAYLOAD/PALLET EXCHANGE
  - DOCKING PROVISIONS
  - ORBITER "WALKING" OR PLATFORM TRANSLATION
- FLAG THE MORE CRITICAL TECHNOLOGICAL DEFICIENCIES IN SPACE ASSEMBLY OPERATIONS

Platform Assembly Analysis Guidelines

In order to provide a starting point for the platform assembly analyses, a brief summary of an assumed scenario of platform construction was generated. The opposite chart summarizes this assembly sequence according to orbiter flights. The plan is for a medium size (approximately 15 pallet capacity) platform. The first flight is estimated to provide packaging and assembly aids so that the platform utility module, propulsion module and platform structure can be completed. Subsequent flights will then add payloads to the platform to provide the operational system.

PLATFORM ASSEMBLY ANALYSIS GUIDELINES

FLIGHT 1 - ERECT UTILITY MODULE
- ATTACH PROPULSION MODULE
- CONSTRUCT PAYLOAD PLATFORM

FLIGHT 2 — INSTALL 5 PAYLOAD PALLETS (OR EQUIV.)

FLIGHT 3 - ADD 5 PAYLOAD PALLETS TO PLATFORM

FLIGHT 4 - ADD 5 PAYLOAD PALLETS TO PLATFORM

SUBSEQUENT FLIGHTS - PAYLOAD EXCHANGE, ENLARGE PLATFORM AND SUPPORTING SUBSYSTEMS AS REQUIRED, PROVIDE SYSTEMS REPAIR AND MAINTENANCE

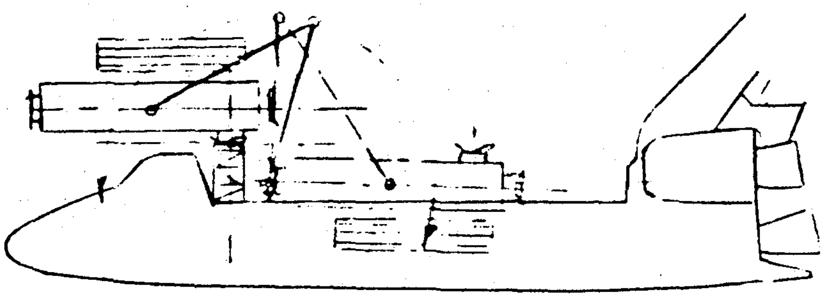
Utilities Module Deployment

The sketches summarize the first procedures of the first orbiter flight of the proposed erectable platform mission. All of the deployment and attachment procedures can be accomplished by the orbiter RMS. The special "off-to-the-side" docking module (assembly fixture) is assumed as the docking port for the utility module canister. The longest reach for the assembly process is that for attaching the propulsion module to the forward end of the solar array canister.

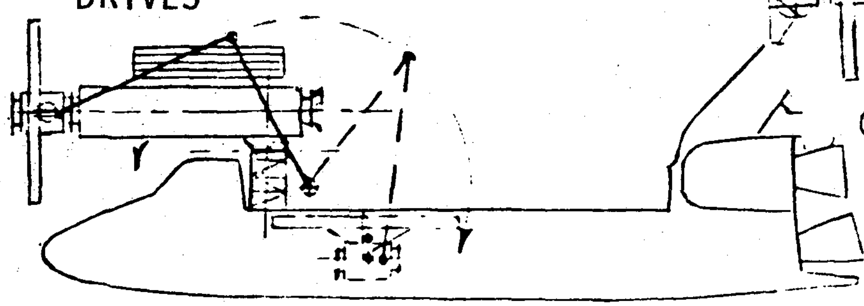
The docking ring has a turntable arrangement to allow attached modules to be rotated to positions favoring ease of viewing and installation of packages by the RMS.

UTILITIES MODULE DEPLOYMENT

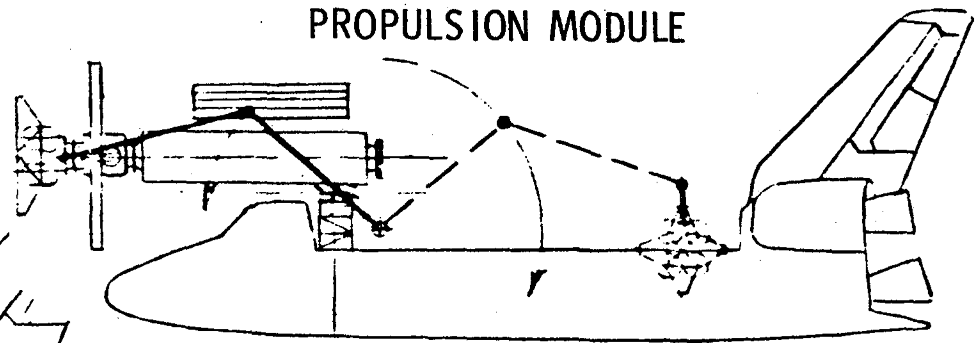
1. REMOVE CANISTER, ROTATE AND MOUNT ON ASSEMBLY FIXTURE.



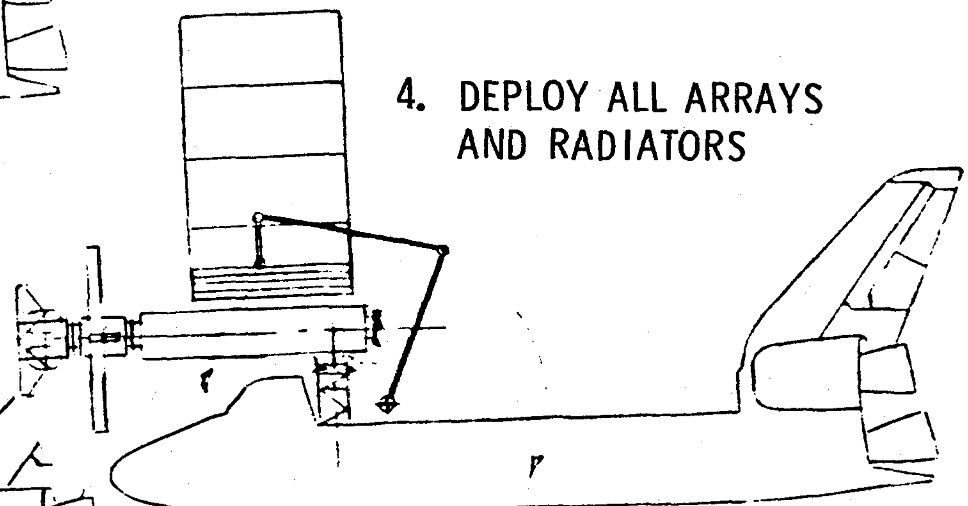
2. REMOVE AND MOUNT SOLAR ARRAY AND DRIVES



3. REMOVE AND MOUNT PROPULSION MODULE



4. DEPLOY ALL ARRAYS AND RADIATORS

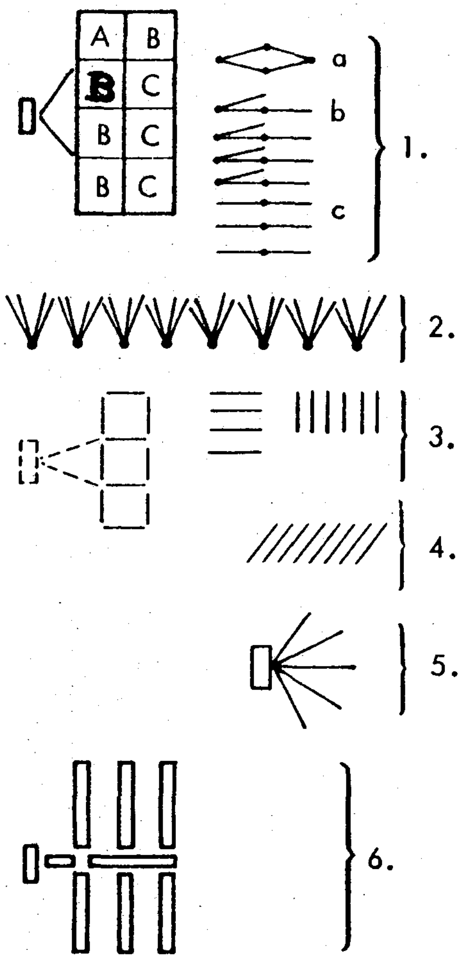


Platform Components for Assembly

Earlier studies had indicated the advantage of using a simple kit concept for assembly efficiency. Assuming that approach, this chart summarizes the types of kits selected for this structural assembly of the 2x4 cell pentahedral platform. The total count of sub-assemblies and pieces is 50. If individual struts and unions were assembled on orbit, the total count would increase to 116.

The upper left sketch indicates the number and types of kits required to construct the upper surface of the platform. The general configuration of the stored kits also is shown. This top surface provides 15 union hardpoint locations for potential single point attachment of 15 pallets. The other sketches provide similar summaries of the other structural components of the platform.

PLATFORM COMPONENTS FOR ASSEMBLY



1. THREE TYPES TOP SURFACE KITS	8 KITS
a. 4 STRUT, 4 UNION KIT (1) b. 3 STRUT, 2 UNION KIT (4) c. 2 STRUT, 1 UNION KIT (3)	
2. EIGHT INTERSURFACE KITS (8)	8 KITS
3. TEN LOWER SURFACE CONNECTING STRUTS	10 PIECES
4. EIGHT TOP SURFACE DIAGONAL STRUTS	8 PIECES
5. ONE UTILITY MODULE (UM) TO PLATFORM (P/F) STRUCTURAL ADAPTER KIT	1 KIT
6. EIGHT UTILITY DUCT SECTIONS (SHORT SECTION MAY BE PART OF UM TO P/F ADAPTER)	8 PIECES
7. SEVEN UTILITY DUCT CONNECTORS	7 PIECES
TOTAL STRUCTURAL SUBASSEMBLIES & SUPPORT ITEMS FOR ASSEMBLY SCHEDULING	50

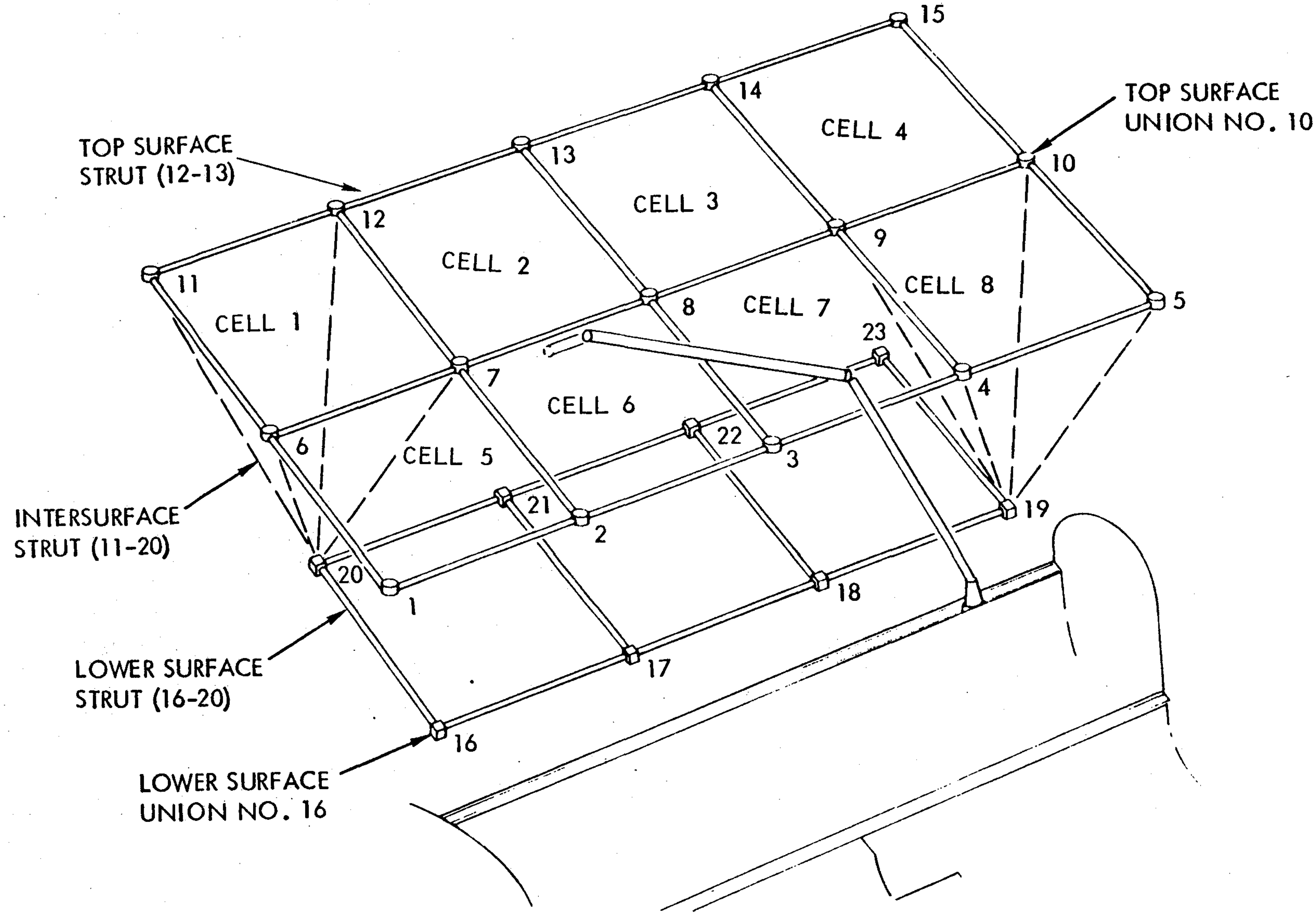
Platform Hardpoint and Strut Identification

The following series of charts illustrate and discuss a sequence of assembly operations for a selected assembly concept. To clarify the description of operations, an identification system for the platform structural elements was selected and is shown on the chart opposite.

The basic concept of the identification system is that of assigning sequential numbers to each of the 23 structural unions (hardpoints) of the platform. The individual struts can then be identified by a two-number system using the numbers of the two unions which are connected. The notation at the lower left of the chart thus identifies lower surface union 20, lower surface struts 16-20 and 21-20, and intersurface struts 11-20, 12-20, 7-20, and 6-20.

The next 20 charts then identify the gradual buildup of the platform structure in a series of operational steps. The final two charts in the series show the sequence for installing payloads on the platform. Special requirements either on structure design, construction aids, or operations activity also are listed in the descriptive text opposite the pictorial charts. These individual requirements will then be accumulated or summed to describe the operational needs imposed on specific pieces of hardware or operational procedures.

PLATFORM HARDPOINT AND STRUT IDENTIFICATION

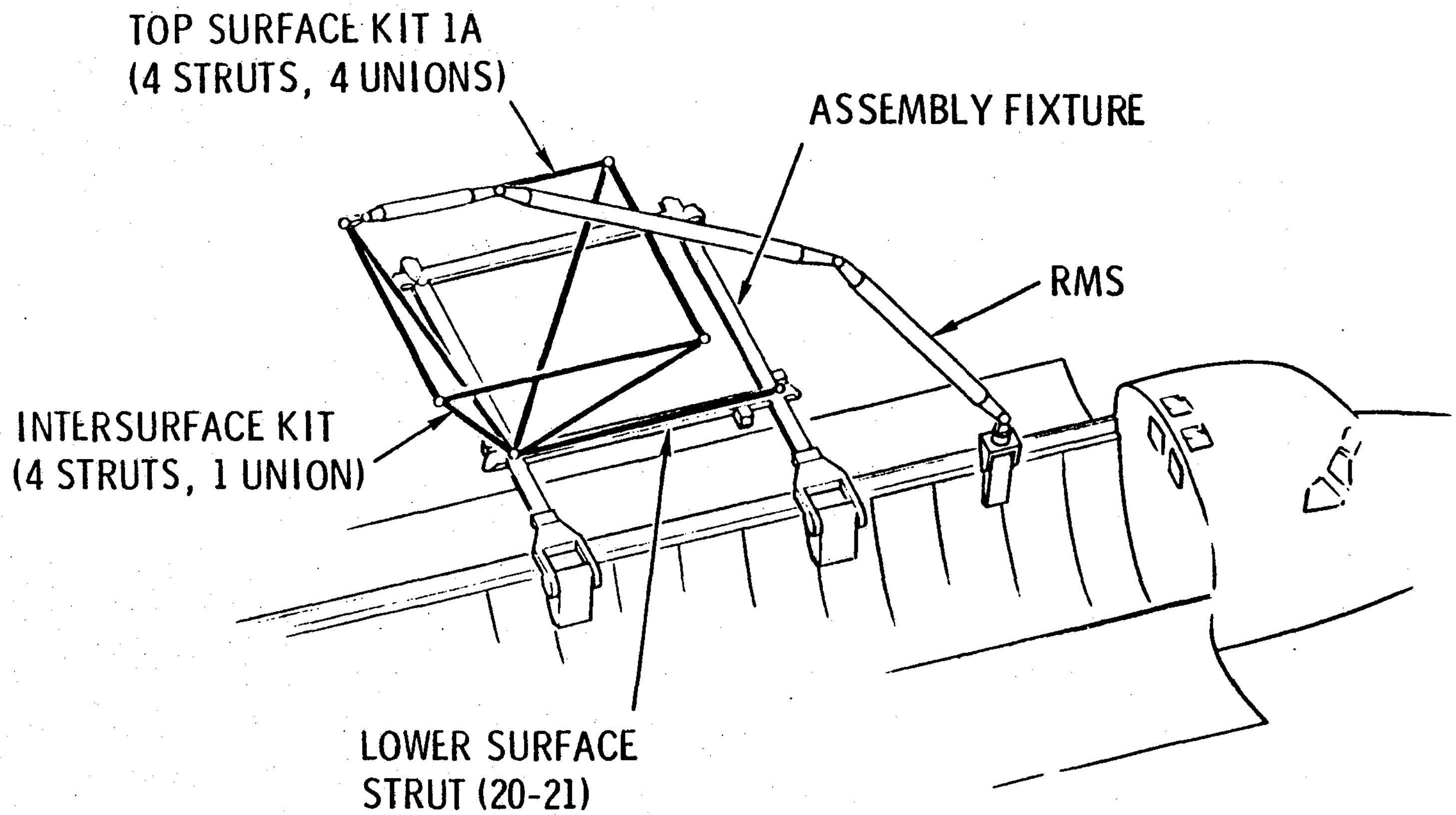


PLATFORM CELL 1 ASSEMBLY

STEP		REQUIREMENTS
1. INSTALL LOWER SURFACE STRUT (20-21) IN ASSEMBLY FIXTURE	1A.	RELEASE OF STRUT LAUNCH RESTRAINTS IN P/L BAY
	1B.	RMS END EFFECTOR GRIP ON STRUT
	1C.	GRAPPLE LOCATION ON STRUT
	1D.	STRUT GRIP JAWS ON ASSEMBLY FIXTURE WITH OPERATING CONTROL SYSTEM IN CABIN
2. INSTALL INTERSURFACE KIT (20) IN ASS'Y FIXTURE, DEPLOY STRUTS TO TOP SURFACE SPACING	2A.	RELEASE OF KIT LAUNCH RESTRAINTS IN P/L BAY
	2B.	2C (SAME AS 1B, 1C)
	2D.	UNION GRASP AND ALIGNMENT MECHANISM
	2E.	STRUT SPREADING MECHANISM (UMBRELLA SPRING?)
3. INSTALL TOP SURFACE KIT 1A, 4 UNION-TO-STRUT CONNECTS	3A.	(SAME AS 2A)
	3B.	END EFFECTOR GRASP KIT UNION (FOR ATTACH TO STRUT)
	3C.	END EFFECTOR GRIP OF TOP END OF STRUT DURING ATTACHMENT OF KIT UNION TO STRUT
	3D.	KIT STRUT SPREADING MECHANISM
4. INSTALL TOP SURFACE DIAGONAL STRUT (11-7)	4A.	4B, 4C (SAME AS 1A, 1B, 1C)
	4D.	CLEARANCE FOR CONNECTION OF 4TH STRUT END TO TOP SURFACE UNIONS 11 AND 7
	4E.	FINAL STRUT/UNION CONNECT SPACING ADJUSTMENT

(B-6)

PLATFORM CELL 1 ASSEMBLY



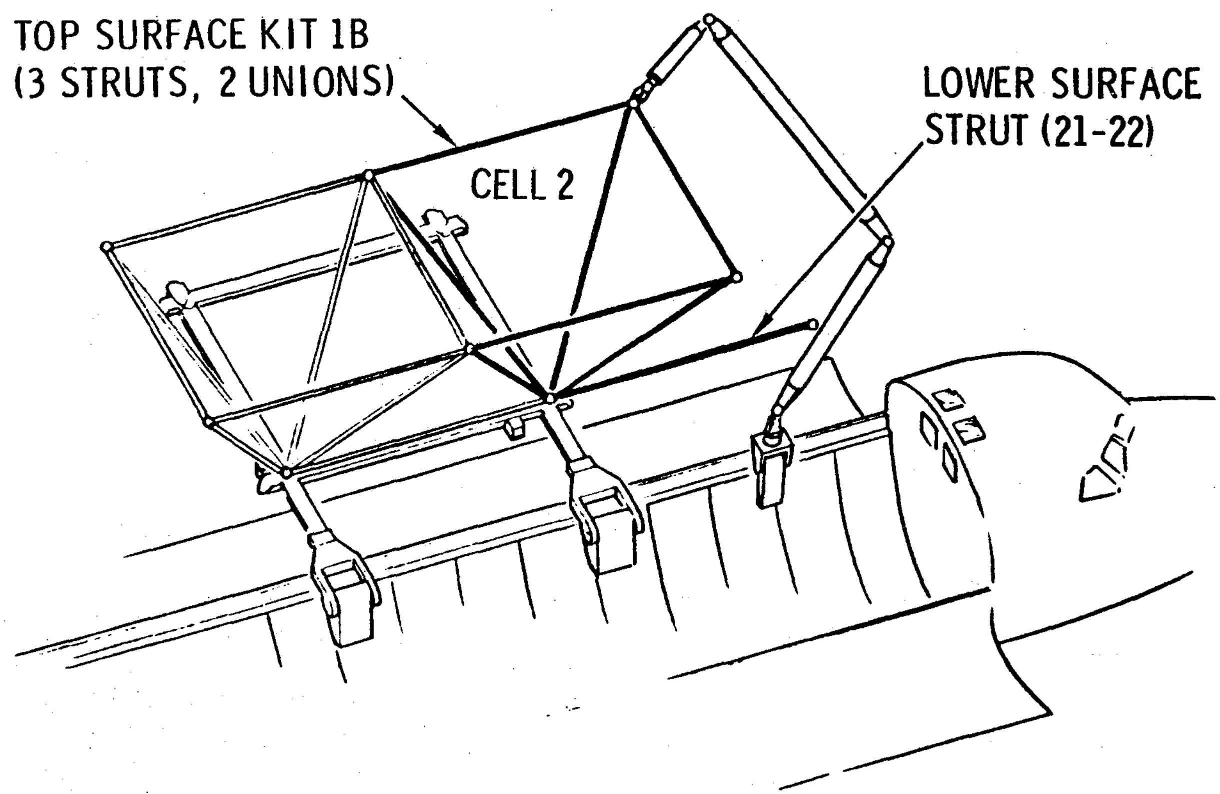
PLATFORM CELL 2 ASSEMBLY

STEP	REQUIREMENTS
1. INSTALL INTERSURFACE KIT (21) IN ASS'Y FIXTURE, DEPLOY STRUTS TO TOP SURFACE SPACING, ATTACH 2 STRUTS TO CELL 1	1A. THRU 1E (SAME AS REF CHART B6 ITEMS 2A THRU 2E) 1F. VISUAL CUES AND DISPLAYS FOR STRUT/UNION ATTACHMENT
2. INSTALL TOP SURFACE KIT 1B, 2 INTERSURFACE STRUT-TO-UNION CONNECTS, 2 TOP SURFACE STRUT-TO-UNION CONNECTS	2A. END EFFECTOR SUPPORT OF BOTH INTERSURFACE STRUT (21-13) AND UNION 13 DURING ATTACHMENT SAME FOR UNION 8 ATTACHMENT TO STRUT (21-8) 2B. STRUT SPREADING MECHANISMS (IN UNION) FROM KIT CONFIGURATION TO OPERATIONAL CONFIGURATION
3. INSTALL LOWER SURFACE STRUT (21-22)	3A. STRUT-TO-UNION LOCKED JOINT, UNION 21
4. INSTALL TOP SURFACE DIAGONAL STRUT (7-13)	4A. TO 4E (SIMILAR TO CHART B6, ITEMS 4A THRU 4E)

Note: Operations repeated until Cells 1 through 4 have been completed.

(B-7)

PLATFORM CELL 2 ASSEMBLY



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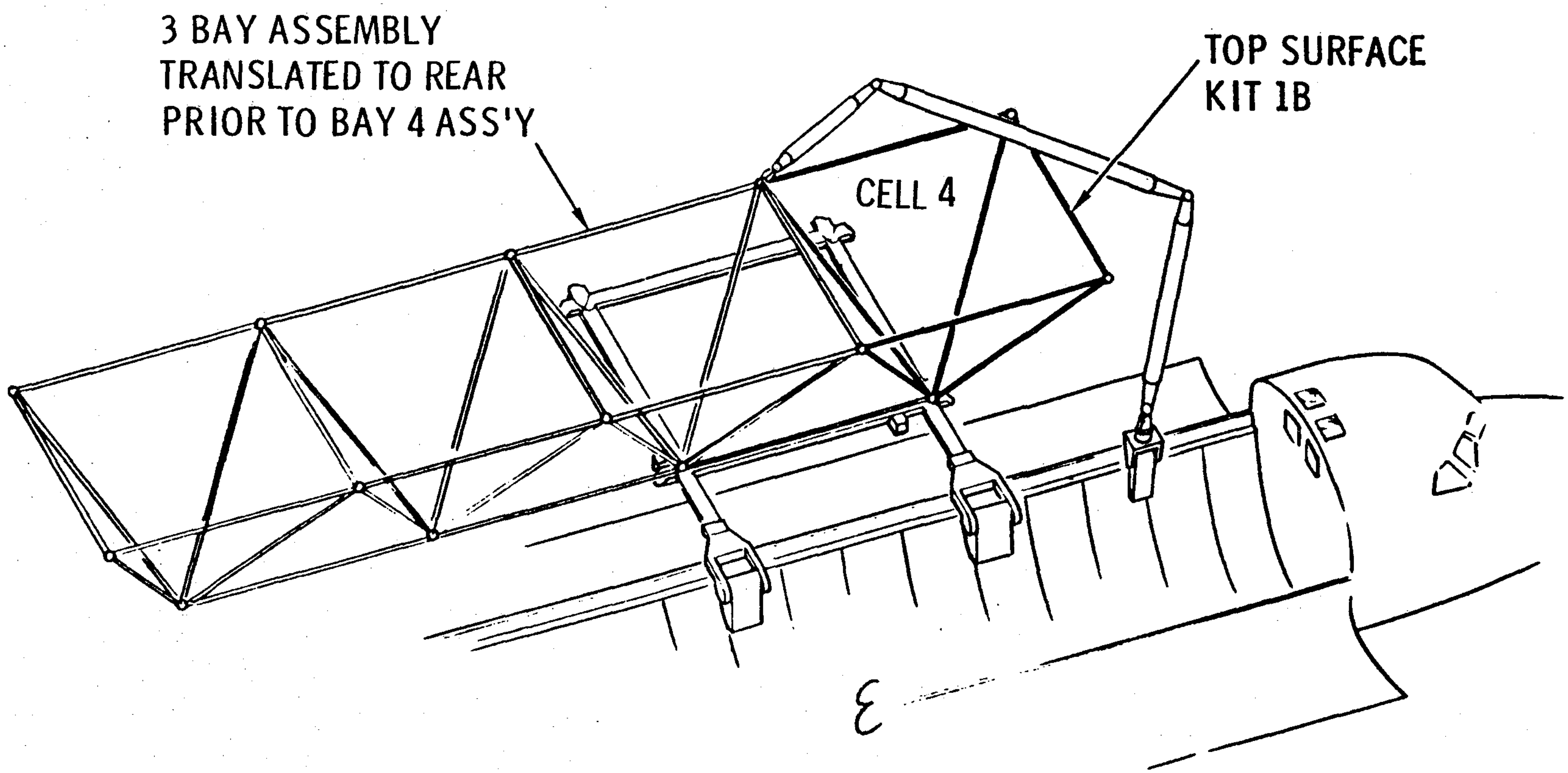
PLATFORM CELL 4 ASSEMBLY

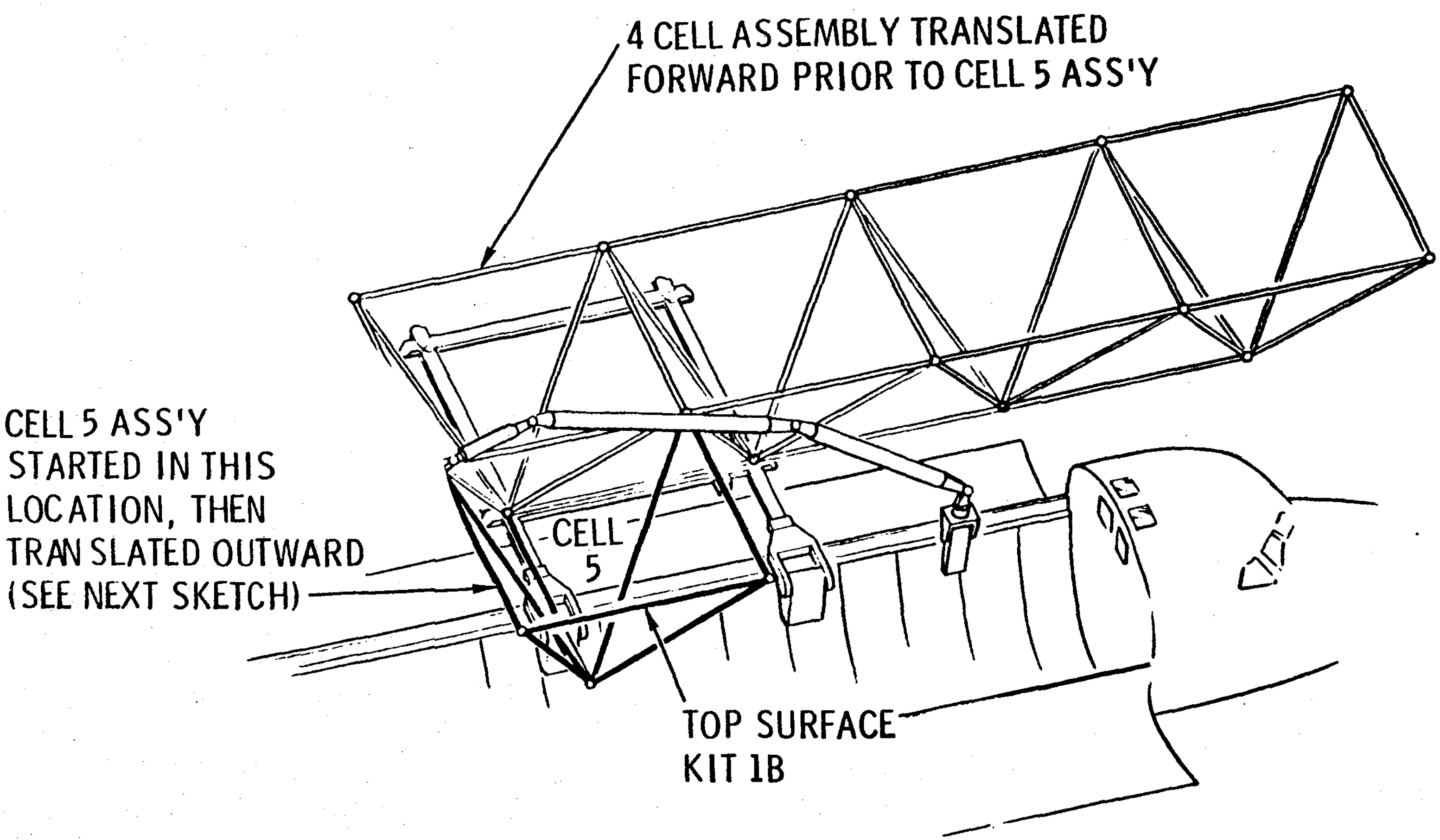


PLATFORM CELL 5 ASSEMBLY

STEP	REQUIREMENTS
1. MOVE 4 CELL ASSEMBLY 11 M FORWARD TO INITIAL POSITION	1A THRU 1C. (SIMILAR TO CHART B8, ITEMS 1A THRU 1C) 1D. PROVIDE FORWARD CLEARANCE FOR STRUCTURE TRANSLATE
2. INSTALL LOWER SURFACE STRUT (20-16)	2A. STRUT-TO-UNION LOCKED JOINT, UNION 20
3. TRANSLATE 4 CELL STRUCTURE OUTWARD (Y DIRECTION) 5.5 M (NOT SHOWN ON SKETCH)	3A. ASS'Y FIXTURE RELEASES UNION CLAMPS, STRUT CLAMPS AND TELESCOPING RAILS EXTEND ASS'Y OUTWARD (Y DIRECTION) 18'
4. INSTALL INTERSURFACE KIT 16	4A TO 4F (SIMILAR TO CHART B7, ITEMS 1A THRU 1F)
5. INSTALL TOP SURFACE KIT 1B	5A, 5B (SIMILAR TO CHART B7, ITEMS 2A AND 2B)
6. INSTALL TOP SURFACE DIAGONAL STRUT (1-7)	6A THRU 6E (SIMILAR TO CHART B6, ITEMS 4A THRU 4E)

(B-10)

PLATFORM CELL 5 ASSEMBLY

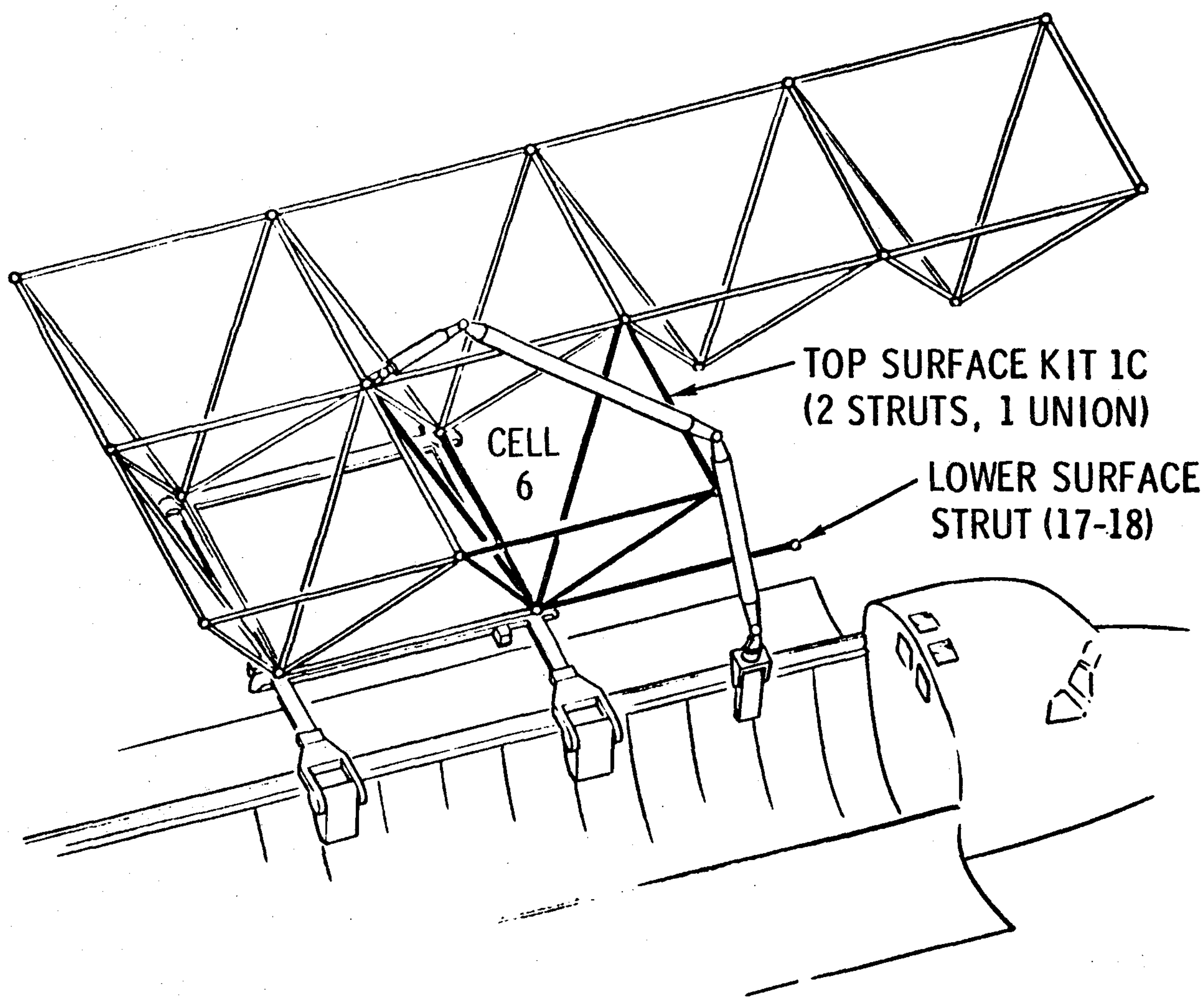


STEP	REQUIREMENTS
1. INSTALL LOWER SURFACE STRUT (16-17)	1A. STRUT-TO-UNION LOCKED JOINT, UNION 16
2. INSTALL LOWER SURFACE STRUT (17-21)	2A. STRUT-TO-UNION LOCKED JOINT, UNION 21
3. INSTALL INTERSURFACE KIT 17	3A TO 3F (SIMILAR TO CHART B7, ITEMS 1A THRU 1F)
4. INSTALL TOP SURFACE KIT 1C (2 STRUTS, 1 UNION)	4A. CONNECT UNION 3 TO STRUT (17-3), RMS 4B. CONNECT STRUT (2-3) TO UNION 2, RMS 4C. CONNECT STRUT (3-8) TO UNION 8, RMS
5. INSTALL TOP SURFACE DIAGONAL STRUT (7-3)	5A, 5B (SIMILAR TO CHART B7, ITEMS 2A AND 2B)
6. INSTALL LOWER SURFACE STRUT (17-18)	6A. STRUT-TO-UNION LOCKED JOINT, UNION 17

Note: Cells 7 and 8 are completed in an identical procedure.

(B-11)

PLATFORM CELL 6 ASSEMBLY

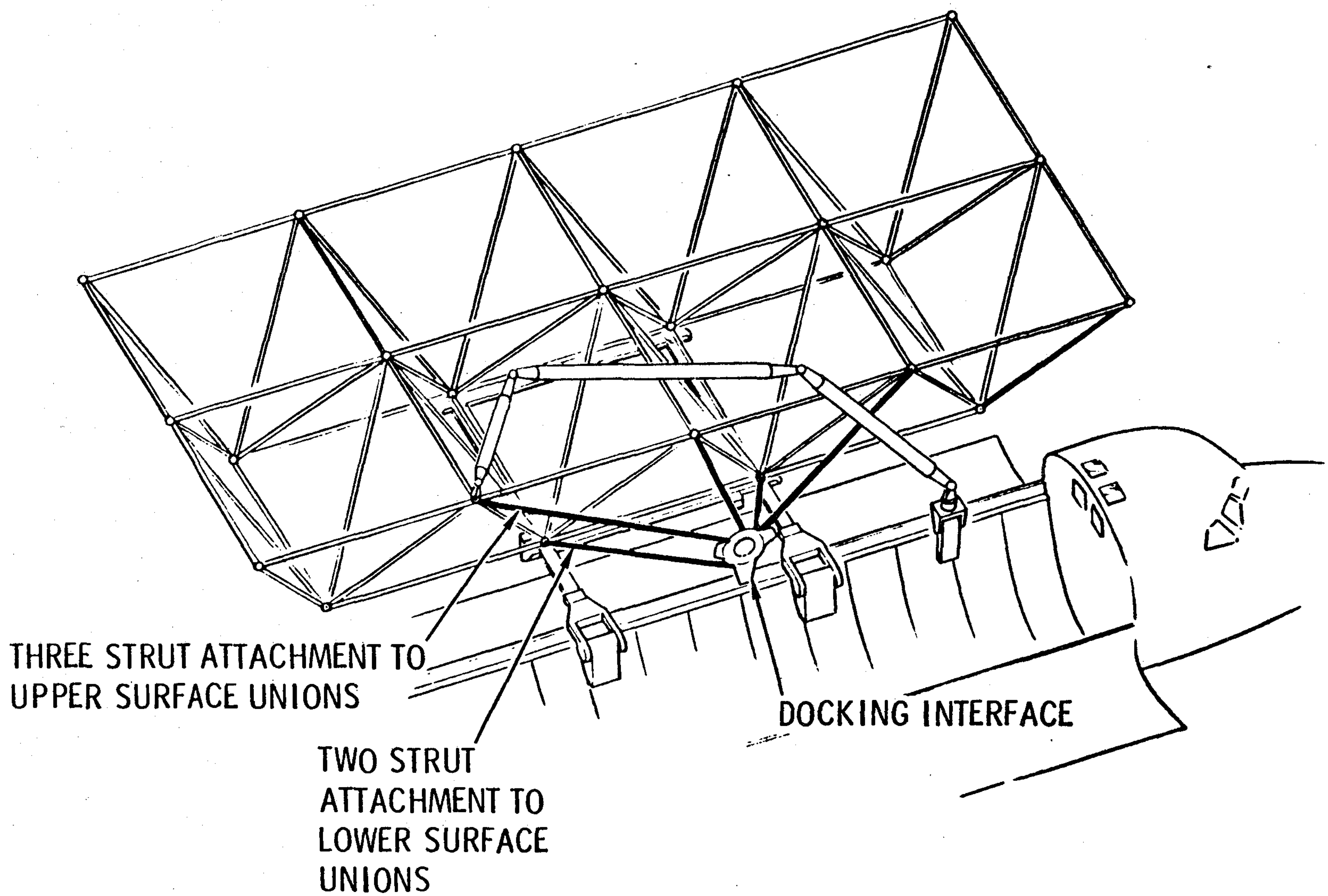


PLATFORM TO UTILITY MODULE ADAPTER ASSEMBLY

STEP	REQUIREMENTS
1. TRANSLATE STRUCTURE, CLAMP STRUT (17-18)	1A THRU 1C (SIMILAR TO CHART 8, ITEMS 1A THRU 1C)
2. REMOVE ADAPTER KIT FROM P/L BAY	2A. RELEASE ADAPTER FLIGHT RESTRAINTS 2B. PROVIDE RMS GRAPPLE POINT ON STRUT (U-17)
3. ATTACH STRUTS (U-17), (U-18) TO UNIONS 17 & 18	3A. VISUAL CUES AND DISPLAYS FOR STRUT/UNION ALIGN 3B. UNION WITH NON-STANDARD STRUT ATTACH SOCKET (8TH ATTACH POINT ON UNION)
4. ATTACH STRUTS (U-2), (U-3) AND (U-4) TO UNIONS 2, 3, AND 4	4A. VISUAL CUES & DISPLAYS 4B. UNIONS WITH NON-STANDARD STRUT ATTACH SOCKET ORIENTATION

(B-14)

PLATFORM TO UTILITY MODULE ADAPTER ASSEMBLY

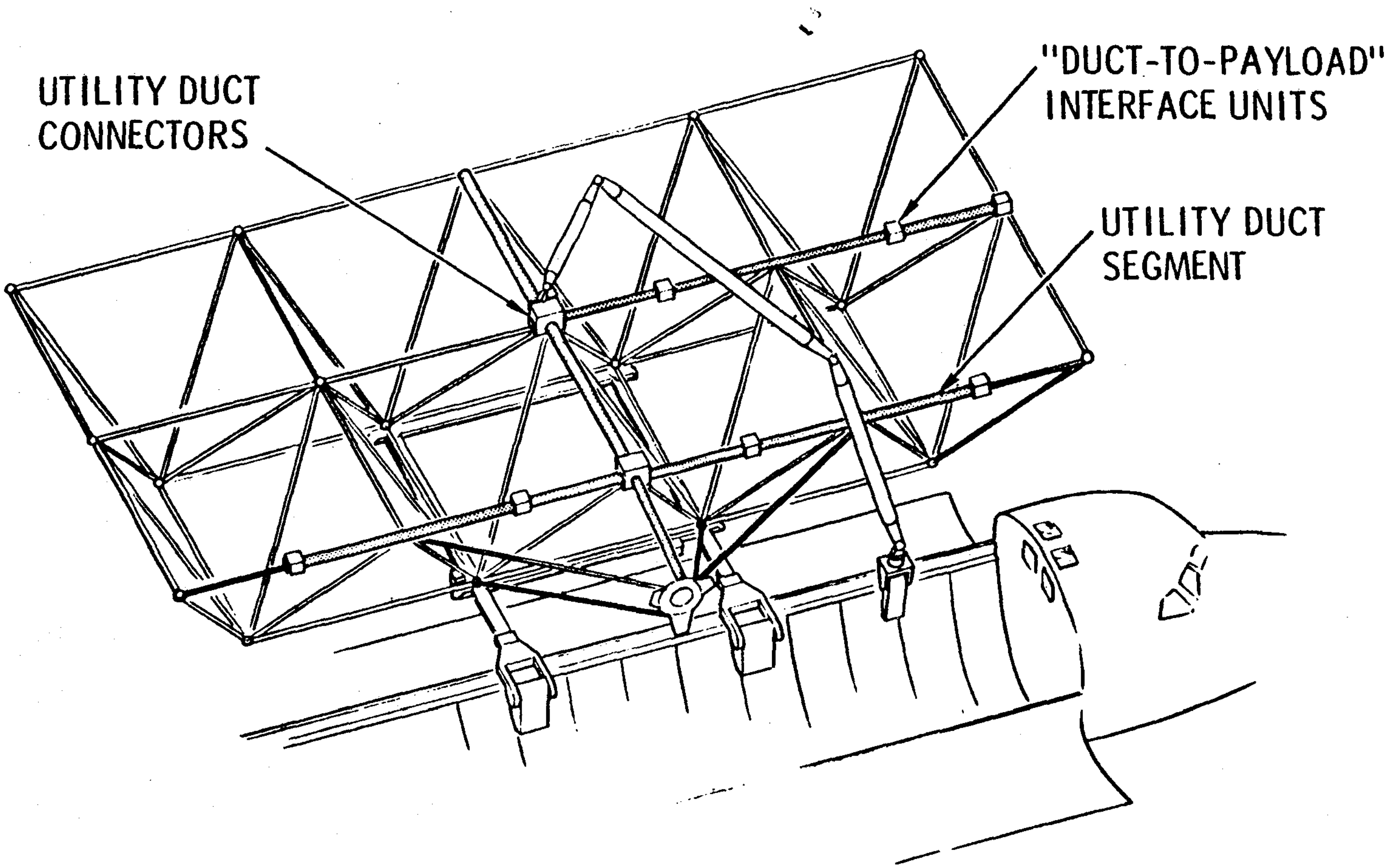


PLATFORM UTILITY DUCT INSTALLATION (PART I)

STEP	REQUIREMENTS
1. INSTALL DUCT SEGMENT (1-3)	1A. RELEASE LAUNCH RESTRAINTS FOR SEGMENT 1B. RMS GRAPPLE LOCATION ON DUCT SEGMENT 1C. DUCT TO STRUT ATTACHMENT CLAMPS AT CENTER
2. INSTALL DUCT SEGMENT (3-13)	2A THRU 2C (SAME AS 1A THRU 1C)
3. INSTALL DUCT SEGMENT (8-10)	3A THRU 3C (SAME AS 1A THRU 1C)
4. INSTALL DUCT SEGMENT (3-5)	4A THRU 4C (SAME AS 1A THRU 1C)
5. INSTALL UTILITY DUCT SEGMENT CONNECTORS AT UNION LOCATIONS 3 AND 8	5A. SPECIAL DUCT SEGMENT CONNECTORS FOR EACH DUCT-TO DUCT INTERFACE 5B. RMS GRAPPLE POINTS ON CONNECTORS 5C. EVA ACTIVITY FOR FINAL CONNECTION OPERATIONS AND INSPECTION 5D. EVA FOOT AND HAND RESTRAINTS FOR OPERATIONS
6. DUCT TO STRUCT ATTACHMENT AT END LOCATIONS 1, 5, & 13	6A. DUCT-TO-STRUT ATTACHMENT CLAMPS 6B. EVA FOR ATTACHMENT OPERATIONS 6C. EVA FOOT AND HAND RESTRAINTS FOR OPERATIONS

(B-15)

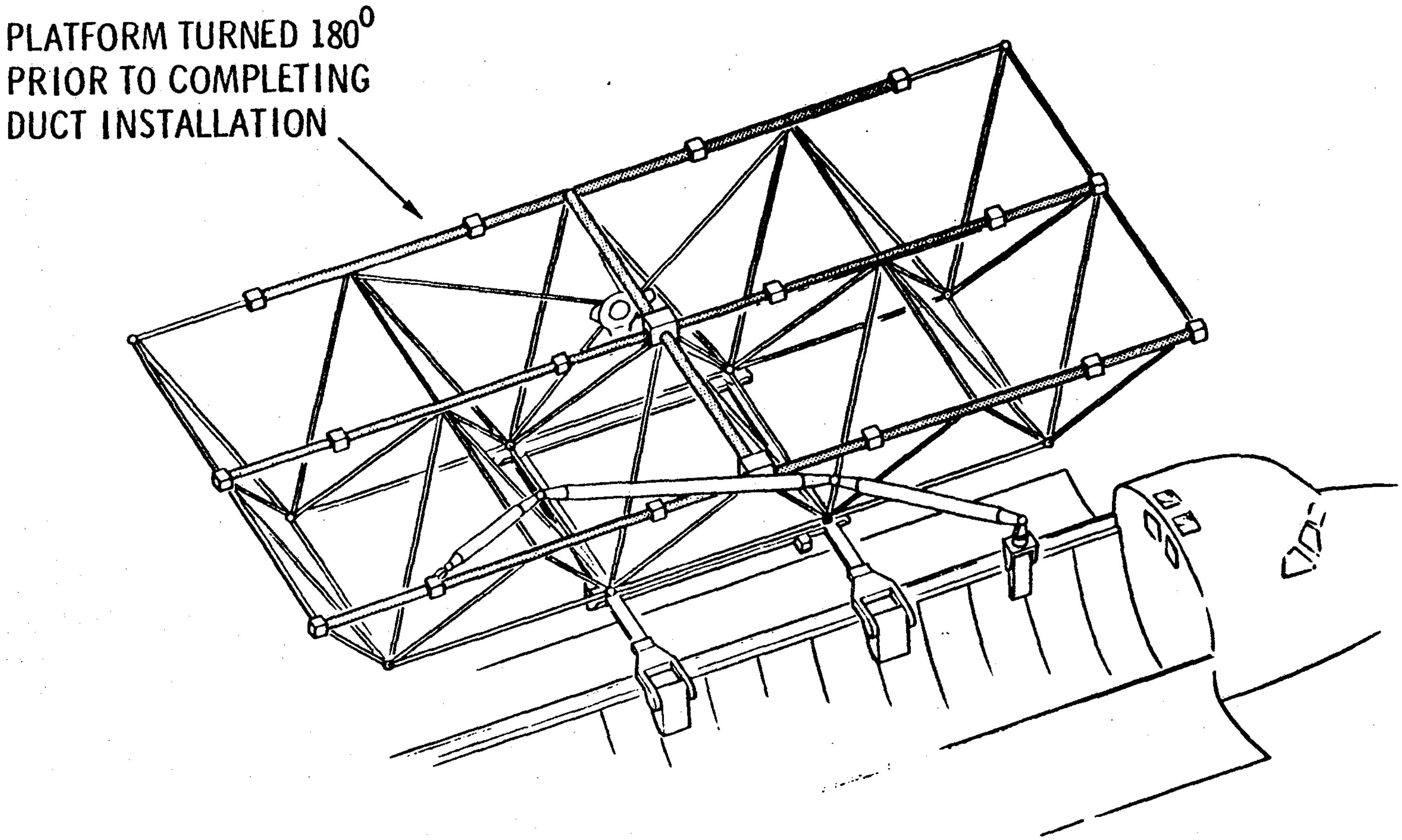
PLATFORM UTILITY DUCT INSTALLATION (PART 1)



STEP	REQUIREMENTS
1. ROTATE PLATFORM ASS'Y 180°, REINSTALL IN ASS'Y FIXTURE	1A. PLATFORM GRAPPLE FIXTURE NEAR C.G. 1B. VISUAL CUES FOR PLATFORM REINSTALL
2. INSTALL DUCT SEGMENT (15-13)	2A THRU 2C (SAME AS CHART B-15, 1A THRU 1C)
3. INSTALL DUCT SEGMENT (8-6)	3A THRU 3C (SAME AS 2A THRU 2C)
4. INSTALL DUCT SEGMENT (13-11)	4A THRU 4C (SAME AS 2A THRU 2C)
5. INSTALL UTILITY DUCT SEGMENT CONNECTORS AT UNION LOCATIONS 8 AND 13	5A THRU 5D (SAME AS CHART B-15, 5A THRU 5D)
6. DUCT TO STRUCTURE ATTACHMENT AT END LOCATIONS 15, 6, & 11	6A THRU 6C (SAME AS CHART B-15, 6A THRU 6C)

(B-16)

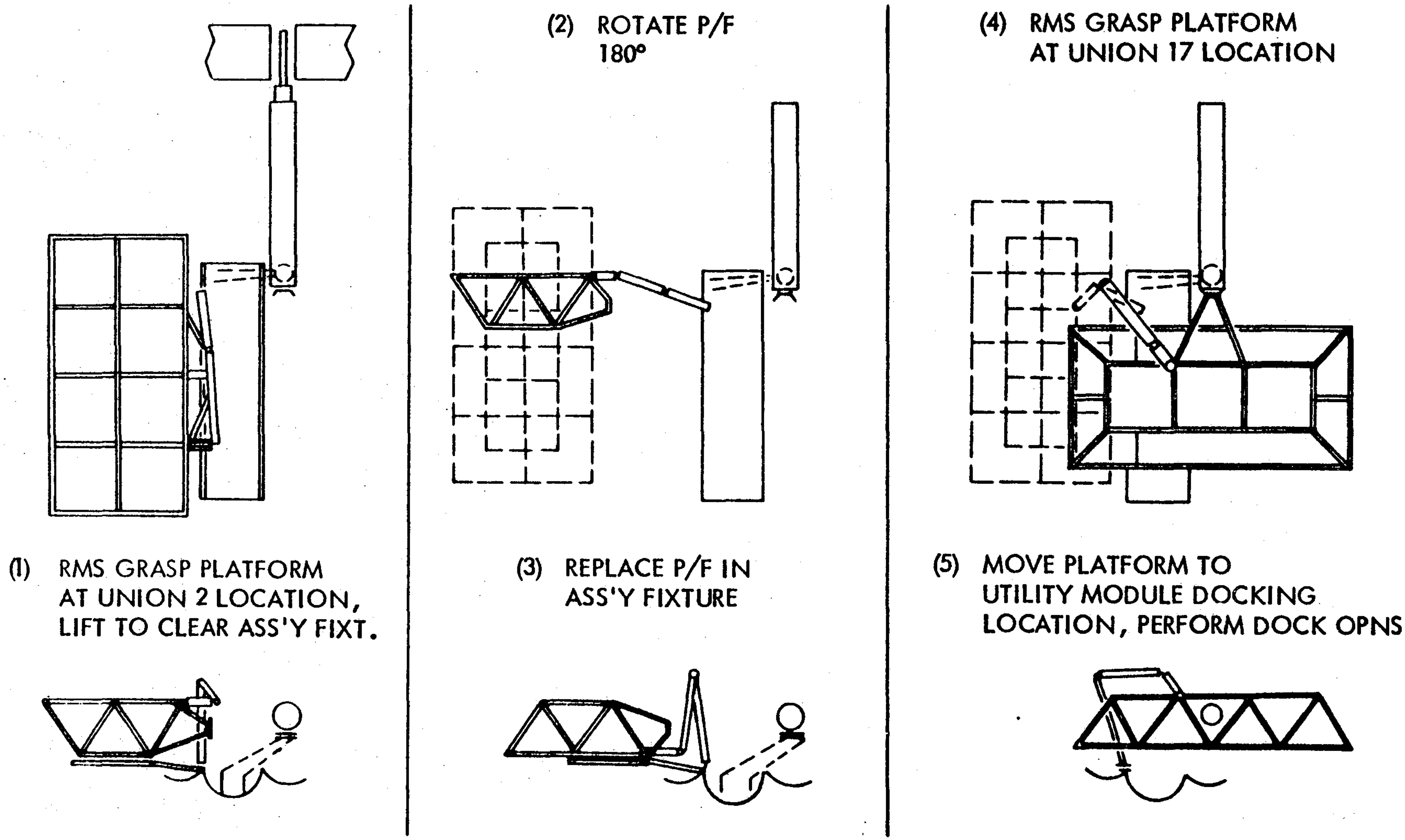
PLATFORM UTILITY DUCT INSTALLATION (PART 2)



PLATFORM ATTACHMENT TO UTILITY MODULE

1. REVERSE PLATFORM TO LOWER SURFACE "UP"	1A. RMS AND P/F GRAPPLE INTERFACE UNITS 1B. RMS CONTROL ROTATION AT END EFFECTOR (INERTIA LIMITS, ETC.) VISUAL CUES FOR REDOCKING P/F IN ASS'Y FIXTURE
2. TRANSLATE P/F TO POSITION TO DOCK WITH UTILITY MODULE	2A. RMS AND P/F GRAPPLE INTERFACE UNITS 2B. RMS-TO-PLATFORM COLLISION AVOIDANCE CONTROL
3. COMPLETE P/F TO UTILITY MODULE DOCK	3A. MECHANICAL DOCKING MECHANISM
4. COMPLETE P/F UTILITY CONNECTIONS	4A. ELECTRICAL POWER CONNECTORS 4B. COMMUNICATIONS/CONTROLS INTERFACE CONNECTORS 4C. FLUID LINE CONNECTORS

PLATFORM ATTACHMENT TO UTILITY MODULE

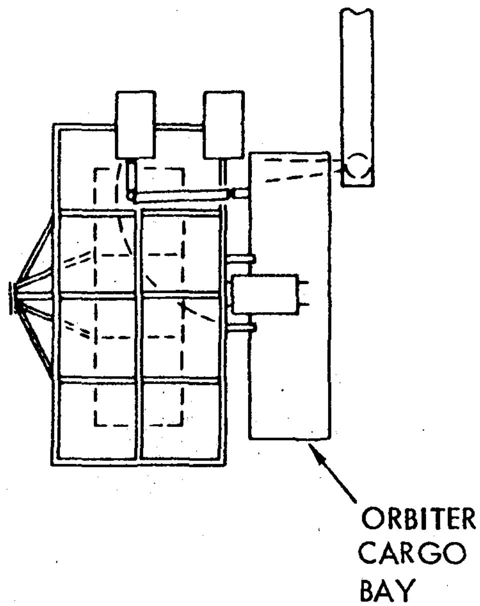


PAYLOAD - TO - PLATFORM INSTALLATION

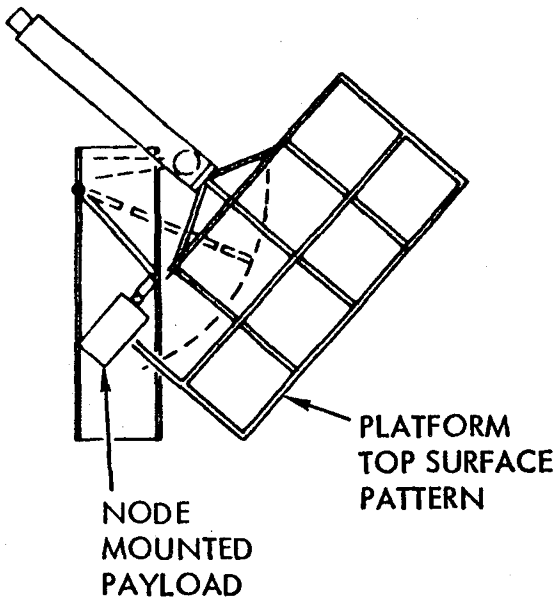
1. RMS GRASP P/L IN ORBITER CARGO BAY	1A. GRAPPLE FIXTURE ON PAYLOAD 1B. QUICK RELEASE OF PAYLOAD RESTRAINTS
2. MOVE P/L TO PLATFORM ATTACH LOCATION	2A. RMS CONTROL PROGRAM FOR HEAVY MASS, OFF-CENTER LOAD ACCELERATION/DECELERATION
3. FASTEN P/L TO PLATFORM	3A. VISUAL CUES FOR ACCURATE PLACEMENT 3B. PLATFORM LATCHING DEVICES TO SECURE P/L
4. MAKE UTILITY TO P/L CONNECTIONS	4A. ELECTRICAL POWER CONNECTOR 4B. DATA/COMMUNICATION LINE CONNECTORS 4C. FLUID LINE CONNECTORS 4D. AUTOMATED OR EVA CONNECTOR FASTENING
5. CHECKOUT P/L OPERATIONAL SYSTEMS	5A. C/O CONTROL CONSOLE IN ORBITER 5B. C/O SENSORS IN PAYLOAD SYSTEMS 5C. C/O CIRCUITS FROM PAYLOADS TO CONSOLE

PAYLOAD - TO - PLATFORM INSTALLATION

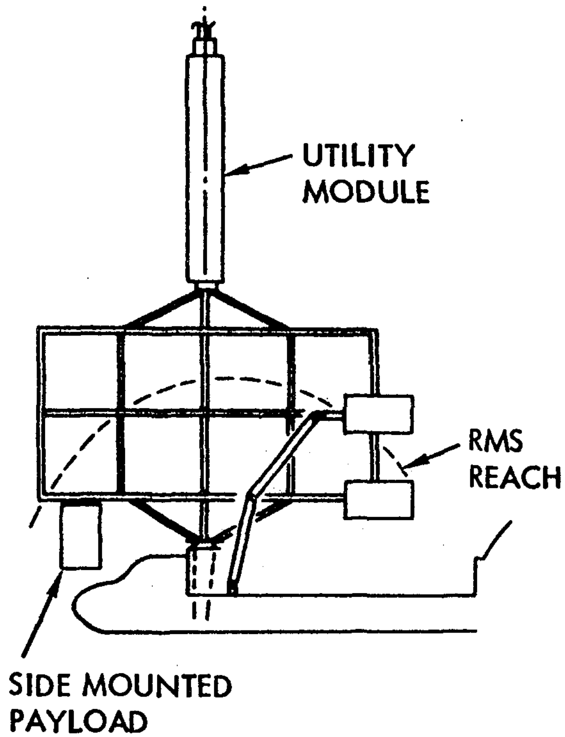
(1) PAYLOADS INSTALLED WITH PLATFORM DOCKED TO ASSEMBLY FIXTURE



(2) PAYLOADS INSTALLED WITH PLATFORM DOCKED TO UTILITY MODULE (HORIZONTAL)



(3) PAYLOADS INSTALLED WITH PLATFORM DOCKED TO UTILITY MODULE (VERTICAL)



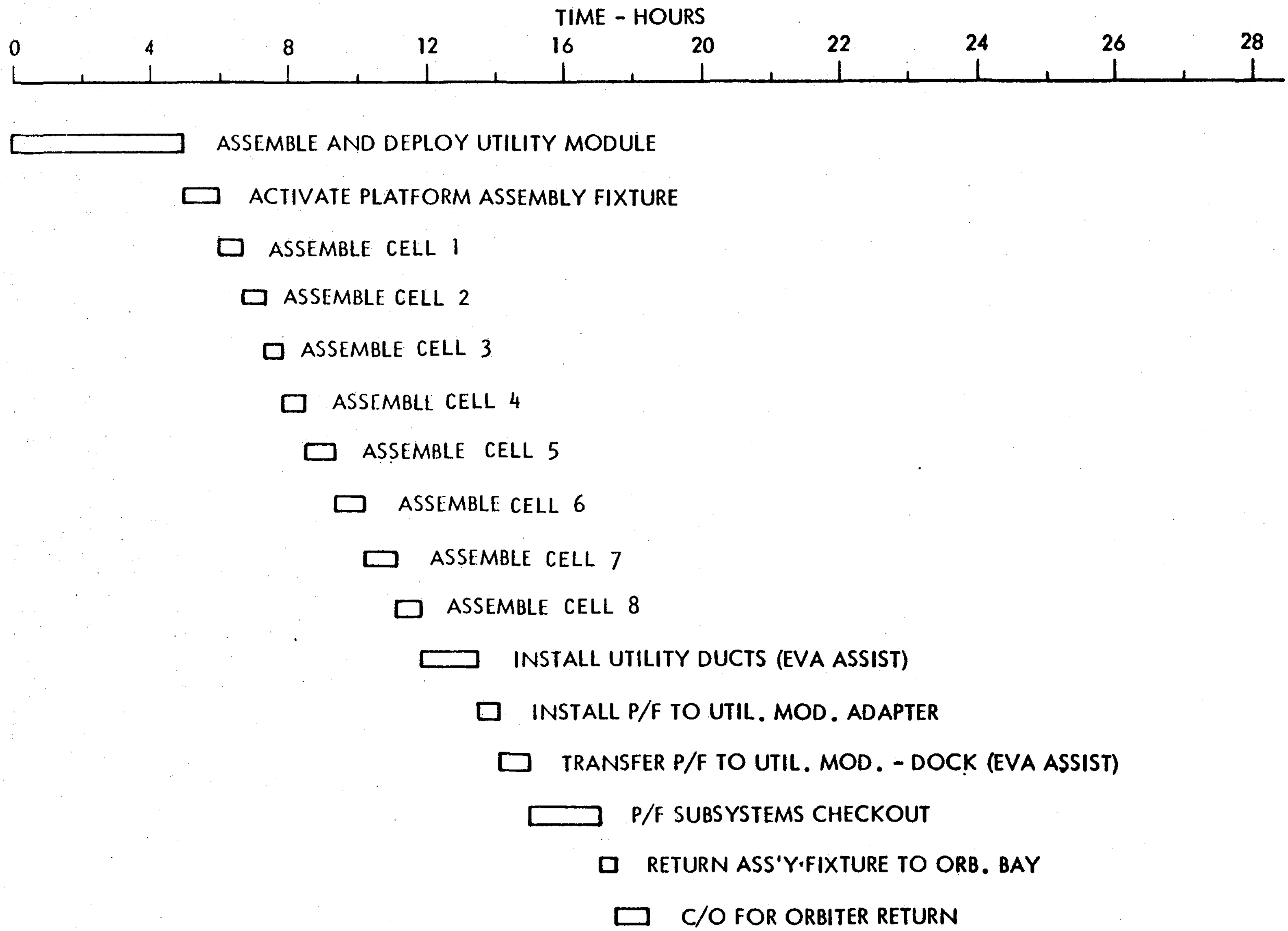
Platform Assembly Timeline

This chart summarizes an estimated assembly operations timeline for the first orbiter flight dedicated to the installation of the 8-cell pentahedral area platform. The deployment of the utility module had an allowance of five hours assigned. The next activities were time estimates generated for the various steps of the platform assembly sequence previously illustrated.

Final assembly activities were the utility duct installations and the joining of the platform to the utility module. A two-hour allowance for checkout of the platform subsystems was made. Other flight-terminating operations include return of the platform assembly fixture to the orbiter cargo bay, and finally orbiter checkout for the return to earth. The resulting total operating time estimate was approximately 18.5 hours. Elapsed time would be greater when allowances are made for crew meals, sleep, and other orbiter-related mission activities.

Time allocations for strut/union joining operations was based on test data obtained from the RMS simulator tests conducted at NASA/JSC using the Rockwell ball and socket concepts.

PLATFORM ASSEMBLY TIMELINE



Platform Assembly Requirements Summary

The next two charts summarize the platform assembly requirements which were identified in the previous construction sequence charts. The technology status for each item is shown in the right hand column. Here, as for previous subsystem technology summaries, a status of 1 indicates an enabling (critical) technology development, a number 2 status is one which would enhance (improve) the assembly feasibility while a number 3 status represents a requirement which can be expected to be met in the course of the routine engineering activities of a Phase B/C/D program.

PLATFORM ASSEMBLY REQUIREMENTS SUMMARY

	TECH. STATUS
STRUCTURAL MEMBERS*	
• STRUTS: GRAPPLE POINT REINFORCEMENT	3
• UNIONS: QUICK CONNECT & LOCK, DISCONNECT CAPABILITY	1*
CLOSE TOLERANCE JOINTS	3
• UTILITY DUCTS: GRAPPLE POINT REINFORCEMENT	3
PROTECTIVE END COVERS	3
• UTILITY CONNECTORS: GRAPPLE POINTS	3
QUICK FAIL-SAFE CONNECT/DISCONNECT	1*
RMS FUNCTIONAL DESIGN	
• ELBOW ROTATION CAPABILITY	2
• COMPUTER & SOFTWARE FOR COLLISION AVOIDANCE	1
RMS END EFFECTORS	
• GRAPPLE MECHANISM FOR VARIETY OF STRUCTURAL ELEMENTS	1
• END EFFECTOR WITH SUPPORTS FOR UNION DURING STRUT/UNION JOINT ASSEMBLY	2
ORBITER-TO-PLATFORM DOCKING GUIDANCE/CONTROL**	
• ORBITER-TO-UTILITY MODULE DOCKING	1
• ORBITER-TO-PLATFORM ASSEMBLY FIXTURE DOCKING	2

*See sections on Structures and Utilities Distribution for Technology Candidate Analyses

**See sections on Rendezvous and Docking Subsystem for Technology Candidate Analyses.

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PLATFORM ASSEMBLY REQUIREMENTS SUMMARY

	TECH. STATUS
ORBITER CARGO BAY HOLDING MECHANISMS	
• STRUCTURAL MEMBER LAUNCH RESTRAINT, QUICK-DISCONNECT MECHANISMS	3
• PLATFORM ASSY FIXTURE LAUNCH RESTRAINT MECHANISM	3
• PAYLOAD/PALLET QUICK-DISCONNECT MECHANISMS	2
PLATFORM ASSEMBLY FIXTURE AND TOOLING	
• BASIC PLATFORM ASSEMBLY FIXTURE DEVELOPMENT	1
• POWERED DEPLOYMENT OF PLATFORM ASSEMBLY FIXTURE	3
• POWERED OPERATIONS OF PLATFORM HOLDING AND TRANSLATION MECHANISMS	3
GENERAL	
• VISUAL ASSIST SYSTEMS	3
• ORBITER OPERATIONS CONTROL CONSOLE, DESIGN ANALYSIS & ORBITER INTERFACE	1
• ASSEMBLY OPERATIONS SIMULATION PROGRAMS	1
• FLIGHT EXPERIMENTS SUCH AS INVESTIGATED UNDER LANGLEY STUDIES; e.g., Technology Verification (TV)-2	2

Platform Assembly Technology Candidates

The next four charts provide brief summaries of technology development projects suggested to enable and enhance the platform assembly capabilities for the mid-1980 time period. Each summary lists a development activity objective, an estimate of the current technology status, the suggested approach for the subject technology development, and the expected results.

PLATFORM ASSEMBLY TECHNOLOGY CANDIDATES

RMS ELBOW ROTATION

OBJECTIVE

- ENHANCE RMS FLEXIBILITY FOR PERFORMING PART DELIVERY/RETRIEVAL OPERATIONS

TECHNOLOGY ASSESSMENT

- PRESENT RMS DESIGN HAS ELBOW JOINT LIMITED TO "PITCH" MOTION (-2° TO +160°)

APPROACH

- FURTHER SIMULATION, ANALYSIS, AND ASSESSMENT OF THE RMS MANEUVER "SIDE REACH" AND OTHER FUNCTIONS TO DETERMINE NEED FOR ELBOW ROTATION
- DETERMINE RMS DESIGN CHANGE MODIFICATIONS/ ADDITIONS
- IMPLEMENT CAPABILITY IN RMS FLIGHT ARTICLES FOR ORBITER-BASED PLATFORM ASSEMBLY

EXPECTED RESULTS

- SIMPLIFICATION OF SEVERAL RMS ASSEMBLY OPERATION MOTIONS

RMS COMPUTER SOFTWARE

OBJECTIVE

- PROVIDE COMPUTER-AIDED ASSISTANCE IN PREVENTING RMS/PLATFORM/ORBITER COLLISIONS DURING PLATFORM ASSEMBLY

TECHNOLOGY ASSESSMENT

- REQUIRED COMPUTER/SOFTWARE NOT PRESENTLY AVAILABLE

APPROACH

- ANALYZE COLLISION AVOIDANCE REQUIREMENTS FOR PLATFORM ASSEMBLY AND MISSION OPERATION MANEUVERS PERFORMED BY RMS
- DETERMINE APPROPRIATE COMPUTER FOR APPLICATION
- DEVELOP SOFTWARE FOR COLLISION AVOIDANCE PROGRAM AND INTEGRATE WITH RMS CONTROL SYSTEM

EXPECTED RESULTS

- GREATER RELIABILITY AND SAFETY OF PLATFORM ASSEMBLY OPERATIONS

PLATFORM ASSEMBLY TECHNOLOGY CANDIDATES

RMS END EFFECTOR (TRANSLATION)

OBJECTIVE

- PROVIDE RMS END EFFECTOR CAPABILITY TO HANDLE PLATFORM AND PAYLOAD COMPONENTS

TECHNOLOGY ASSESSMENT

- RMS END EFFECTORS BEING SUPPLIED ARE NOT SUITABLE FOR PLATFORM MISSIONS

APPROACH

- ANALYZE END EFFECTOR GRIP REQUIREMENTS FOR VARIETY OF PLATFORM-RELATED ARTICLES TO BE HANDLED, COORDINATE WITH COMPONENT DESIGNS
- PROVIDE DESIGN DETAILS INCLUDING RMS MECHANICAL AND CONTROL INTERFACE
- DEVELOP THE MORE CRITICAL END EFFECTORS REQ'D

EXPECTED RESULTS

- RMS AND END EFFECTOR COMPATIBLE WITH PLATFORM OPERATIONS

RMS END EFFECTOR (STRUT/UNION JOINING)

OBJECTIVE

- PROVIDE RMS END EFFECTOR WITH OPTIMIZED EFFECTIVENESS FOR STRUT/UNION JOINING

TECHNOLOGY ASSESSMENT

- AN RMS END EFFECTOR THAT CAN SUPPORT THE STRUCTURAL UNION WHILE STRUT/UNION JOINT CONNECTION IS BEING MADE IS NOT AVAILABLE BUT MAY INCREASE ASSEMBLY PRODUCTIVITY

APPROACH

- COMPARE TYPICAL ASSEMBLY TIMELINE ESTIMATES WITH/WITHOUT SPECIALIZED END EFFECTOR
- PERFORM GROUND TESTS OF MODEL
- DEVELOP THE MORE CRITICAL END EFFECTORS REQUIRED

EXPECTED RESULTS

- END EFFECTOR COMBINATION TO OPTIMIZE ERECTABLE PLATFORM-TYPE OPERATIONS

PLATFORM ASSEMBLY TECHNOLOGY CANDIDATES

ASSEMBLY FIXTURE

OBJECTIVE

- PROVIDE ASSEMBLY AID FOR ERECTABLE STRUCTURE CONSTRUCTION

TECHNOLOGY ASSESSMENT

- NEW PROGRAM FOR ERECTABLE STRUCTURE APPLICATION

APPROACH

- COORDINATE ASSEMBLY FIXTURE REQUIREMENTS WITH STRUCTURAL DESIGNS AND ASSEMBLY SEQUENCE ANALYSIS
- DESIGN ASSEMBLY FIXTURE
- PERFORM GROUND TESTS
- CONDUCT FLIGHT TESTS AS APPROPRIATE

EXPECTED RESULTS

- ASSEMBLY AID FOR EFFICIENT PRODUCTION OF ERECTABLE PLATFORM AND FOR POTENTIAL USE IN PAYLOAD INSTALLATION AND PAYLOAD EXCHANGE

CONTROL CONSOLE

OBJECTIVE

- PROVIDE OPERATOR CONTROL CONSOLE FOR ERECTABLE PLATFORM AUTOMATED ASSEMBLY AND MISSION OPERATIONS

TECHNOLOGY ASSESSMENT

- PRESENT ORBITER CONSOLE FOR RMS AND PAYLOAD BAY OPERATIONS HAS LIMITED ENLARGEMENT CAPABILITY

APPROACH

- ANALYZE ALL AUTOMATED ASSEMBLY AND PAYLOAD OPERATIONS PLANNED FOR PLATFORM SATELLITE—INCLUDES ASSEMBLY, PAYLOAD INSTALLATION AND EXCHANGE, PAYLOAD CHECKOUT, ETC.
- DESIGN ADDITIONS REQUIRED TO PRESENT ORBITER PAYLOAD CONSOLE
- COORDINATE WITH ORBITER PROGRAM FOR CONSOLE MODIFICATION
- CONDUCT SIMULATIONS OF COMPLETED UNIT

EXPECTED RESULTS

- ADEQUATE CONTROL SYSTEM FOR AUTOMATED ACTIVITIES AND SUPPORT OF THE PLATFORM PROGRAM

PLATFORM ASSEMBLY TECHNOLOGY CANDIDATES

SIMULATION PROGRAMS

OBJECTIVE

- TEST PRACTICALITY OF PROPOSED PLATFORM STRUCTURE DESIGNS, ASSEMBLY AIDS, AND ASSEMBLY SEQUENCES BY COMPUTER GRAPHICS AND SCALED OR FULL-SCALE GROUND TESTS OF PLATFORM CONSTRUCTION

TECHNOLOGY ASSESSMENT

- EXISTING COMPUTER CAPABILITIES CAN BE PROGRAMMED FOR THE PROPOSED APPLICATION. FACILITIES EXIST FOR GROUND TEST OF FULL-SCALE STRUCTURE COMPONENTS.

APPROACH

- DESIGN REQUIRED COMPUTER SOFTWARE SUBSYSTEMS
- TEST PROPOSED ASSEMBLY SCHEMES BY COMPUTER SIMULATION
- PERFORM GROUND TESTS ON SELECTED ASSEMBLY ALTERNATIVES

EXPECTED RESULTS

- IMPROVED CONFIDENCE IN PLATFORM ASSEMBLY ACTIVITIES

FLIGHT EXPERIMENTS

OBJECTIVE

- PROVIDE ON-ORBIT TESTS OF CRITICAL PLATFORM ASSEMBLY OPERATIONS

TECHNOLOGY ASSESSMENT

- GROUND TESTING CANNOT COMPLETELY SIMULATE ZERO-GRAVITY SPACE ENVIRONMENT; THEREFORE, ON-ORBIT TESTS SHOULD BE ACCOMPLISHED FOR SELECTED OPERATIONS MOST UNIQUE TO THE PLATFORM PROGRAM.

APPROACH

- ANALYZE ERECTABLE PLATFORM ASSEMBLY OPERATIONS TO DETERMINE HIGH-RISK DEVELOPMENTS
- DESIGN SIMPLE FLIGHT EXPERIMENTS REQUIRED TO TEST OPERATIONS FEASIBILITY
- DESIGN EXPERIMENT/ORBITER INTERFACE HARDWARE AND PERFORM FLIGHT TEST
- ANALYZE FLIGHT TEST RESULTS FOR PLATFORM SYSTEM MODIFICATION REQUIREMENTS

EXPECTED RESULTS

- HIGH CONFIDENCE IN PLATFORM PROGRAM FEASIBILITY

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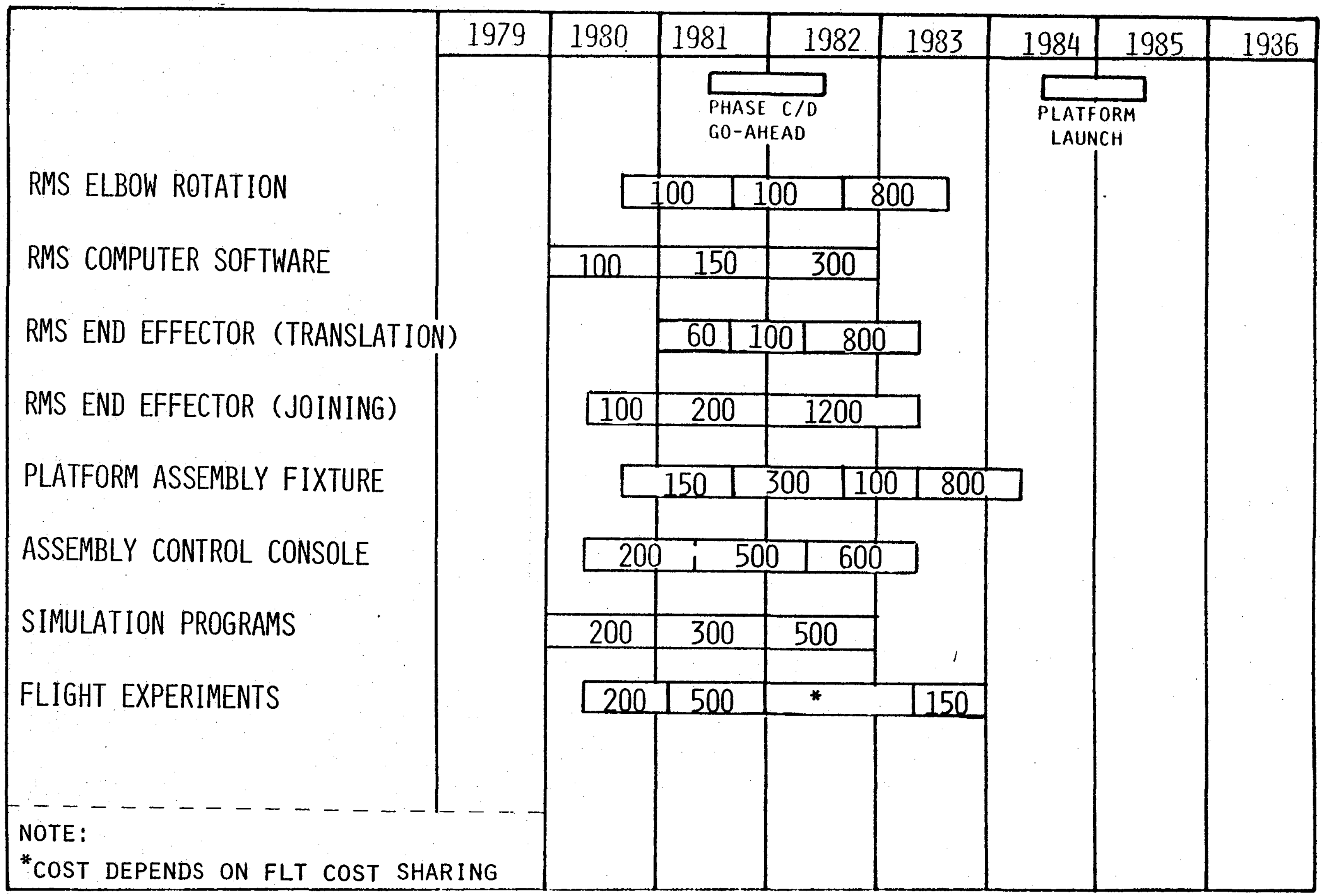
Platform Assembly Technology Program Planning

This chart presents a suggested time phasing and rough order of magnitude cost estimates for the previously summarized eight technology development tasks.

PLATFORM ASSEMBLY TECHNOLOGY PROGRAM PLANNING



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5.0 TECHNOLOGY ASSESSMENT SUMMARY MEASURE OF ADEQUACY

It was felt that some further insight into the current status of science and applications platform technology might be achieved with the evaluation of a "measure of adequacy" for each technological issue.

In the ideal case, one would desire to start the detailed development of a major system such as a platform when the status of each pertinent technological issue is rated as 1.0, i.e., completely adequate. As a practical matter, such a condition is highly improbable and, compared to the Apollo program, for example, only serves to illustrate the degree of risk or inadequacy that can be tolerated if a program is wanted badly enough. The application of such a criterion, serves to give some additional visibility as to which of the many technological issues pertinent to a major new program warrants special effort either before program development starts or concurrent with it.

The data on the next 3 pages show evaluations of the measure of adequacy of the technical areas already discussed in the various subsections of Section 4. Alongside each tic mark is the indicator of the area in which improvement is suggested, i.e., cost/kW, production capacity, life performance, space qualification, etc. The material on these charts is not meant to summarize or substitute for the detailed rating or judgment factors used throughout Section 4, but rather to add another broad assessment factor to them which may lie useful in establishing development program priorities.

TECHNOLOGY ASSESSMENT SUMMARY
MEASURE OF ADEQUACY

TECHNOLOGY/ELEMENT	0 0.1 0.2 0.3 0.4 0.5 0.6 0.7 0.8 0.9 1.0
POWER SUBSYSTEMS	
SOLAR CELLS (Si)	Δ Cost/kW; Δ Prod. Capacity
SOLAR CELLS (Ga)	Δ Cost/kW & Production Capacity
SOLAR ARRAY (Si)	Δ Cost / kW; Δ Production Capacity
SOLAR ARRAY DEPLOYMENT	Δ Performance
BATTERY—NiCd	Design, Cost, Perf., & Production Δ
BATTERIES—NiH_2	Δ Design, Cost, Production
POWER CONDITIONING/DISTRIB.	Design, Cost, Perf., & Production Δ
THERMAL CONTROL SUBSYSTEMS	
RADIATOR—HEAT PIPE/FREON HYB.	Δ Design, Prod., Perf., Cost
THERMAL COATINGS	Δ Development & Perf.
FREON PUMP	Life Performance Δ
FLUID LOOP COMPONENTS	Design, Cost, Perf., & Production Δ
C^3 SUBSYSTEMS	
COMMUNICATIONS EQUIPMENT	Design, Cost, Perf., & Production Δ
GPS AUTOMATED USER EQUIPMENT	Δ Design & Space Qualif.
TDRS OPERATIONAL IMPROVEMENT	Δ Operations Procedures for Sharing

TECHNOLOGY ASSESSMENT SUMMARY
MEASURE OF ADEQUACY

TECHNOLOGY/ELEMENT	0 0.1 0.2 0.3 0.4 0.5 0.6 0.7 0.8 0.9 1.0
STRUCTURAL SUBSYSTEMS	
STRUCT. MATERIALS & PROCESSING	Design, Test & Space Qualif. —Δ
JOINTS & JOINING	Design, Perf.,Test, Qual. Δ
PROPULSION SUBSYSTEMS	
HYDRAZINE CONTAMINATION	Performance —Δ
ON-ORBIT INSTALL. & INTERFACE	Analysis & Design Δ
RCS THRUSTER PIVOT ARMS	Design, Test, Qual.—Δ
100-500 LB THRUSTER	Life Qualification——Δ
RENDEZVOUS APPROACH	Analysis Δ
DOCKING W/RMS ATTENUATION	Δ Anal., Sim., Design, Perf.
DOCKING ATTACHMENT	Design, Perf. Δ
WALKING WITH RMS	Δ Anal., Sim., Design, Perf.
ATTITUDE & DIRECTIONAL CONTR. S/SYS.	
SYSTEM MODELING	Techniques & Methods—Δ
MODEL APPLIC./CONTROL THEORY	Model Reduction——Δ
LSS Control Simulation	Δ Analysis & Simulation
CMG STORAGE DEVICES	Design, Production, & Cost—Δ
GRAVITY-GRADIENT BOOM	Δ Anal., Design, & Qual.
ATT. CONTROL COMPONENTS	Design, Production & Qual.Δ

TECHNOLOGY ASSESSMENT SUMMARY
MEASURE OF ADEQUACY

TECHNOLOGY/ELEMENT	0 0.1 0.2 0.3 0.4 0.5 0.6 0.7 0.8 0.9 1.0
PAYLOAD ACCOMMODATION SUBSYST.	
MINI-PALLETS	Design, Development, & Qual. Δ
PLATFORM TABLES	Design, Development, & Qual. Δ
STRUCTURAL MOUNTING	Design, Dev. & Qual. Δ
LOAD-CARRYING DISCONNECT FITTING	Design, Development, & Qual. Δ
UTILITIES DISTRIB. SUBSYSTEMS	
ELECTRICAL	Design, Development, & Qual. Δ
FLUID	Δ Design, Development and Qual.
OPERATIONS-DERIVED SUBSYSTEMS	
RMS MODIFICATIONS	Design, Development, & Qual. Δ
RMS END EFFECTORS	Design, Dev., & Qual. Δ
RMS SOFTWARE	Δ Anal., Sim., Dev., & Test
CONTROL CONSOLE MODIFICATION	Design, Development, & Qual. Δ
ASSEMBLY FIXTURES	Δ Reqmts., Analysis, Design & Dev.
SIMULATION & FLIGHT TEST	Analysis, Development, & Test Δ

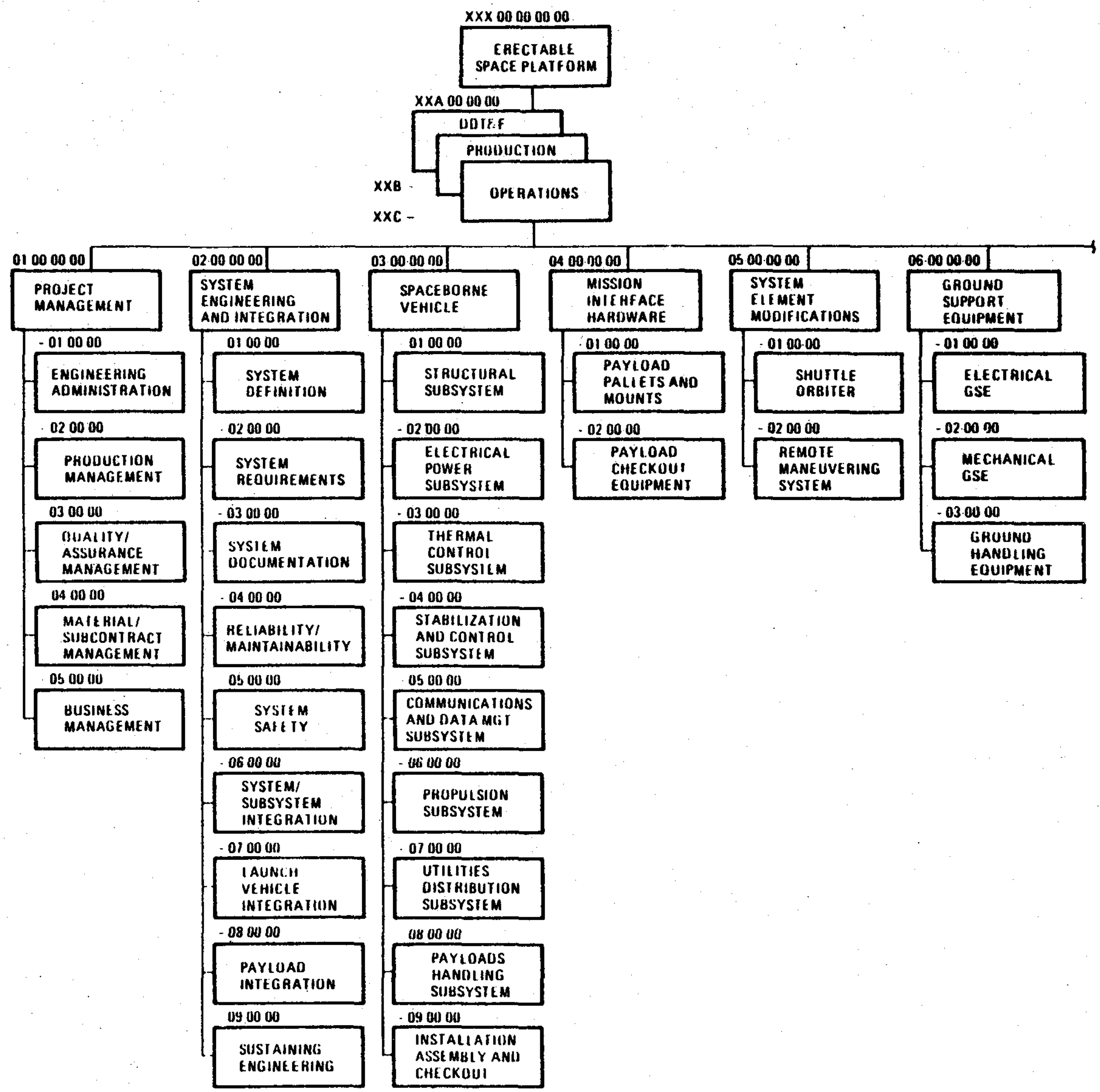
6.0 PRELIMINARY PLATFORM DEVELOPMENT PROGRAM PLAN

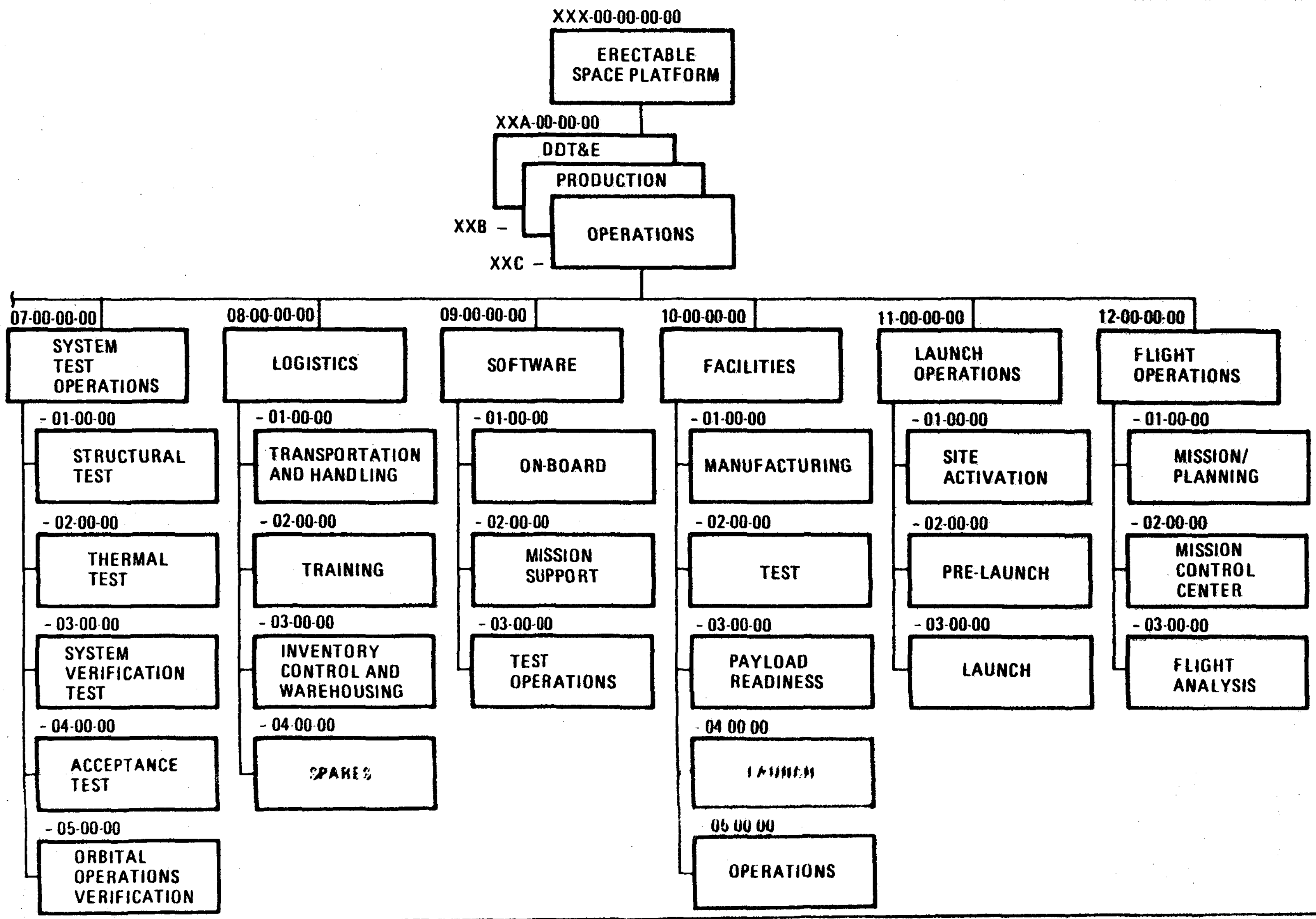
Some of the elements of an initial program plan are described. These include a Work Breakdown Structure, a top level program flow, a Master Schedule, and a rough order of magnitude cost assessment.

Preliminary Platform Development Program Plan

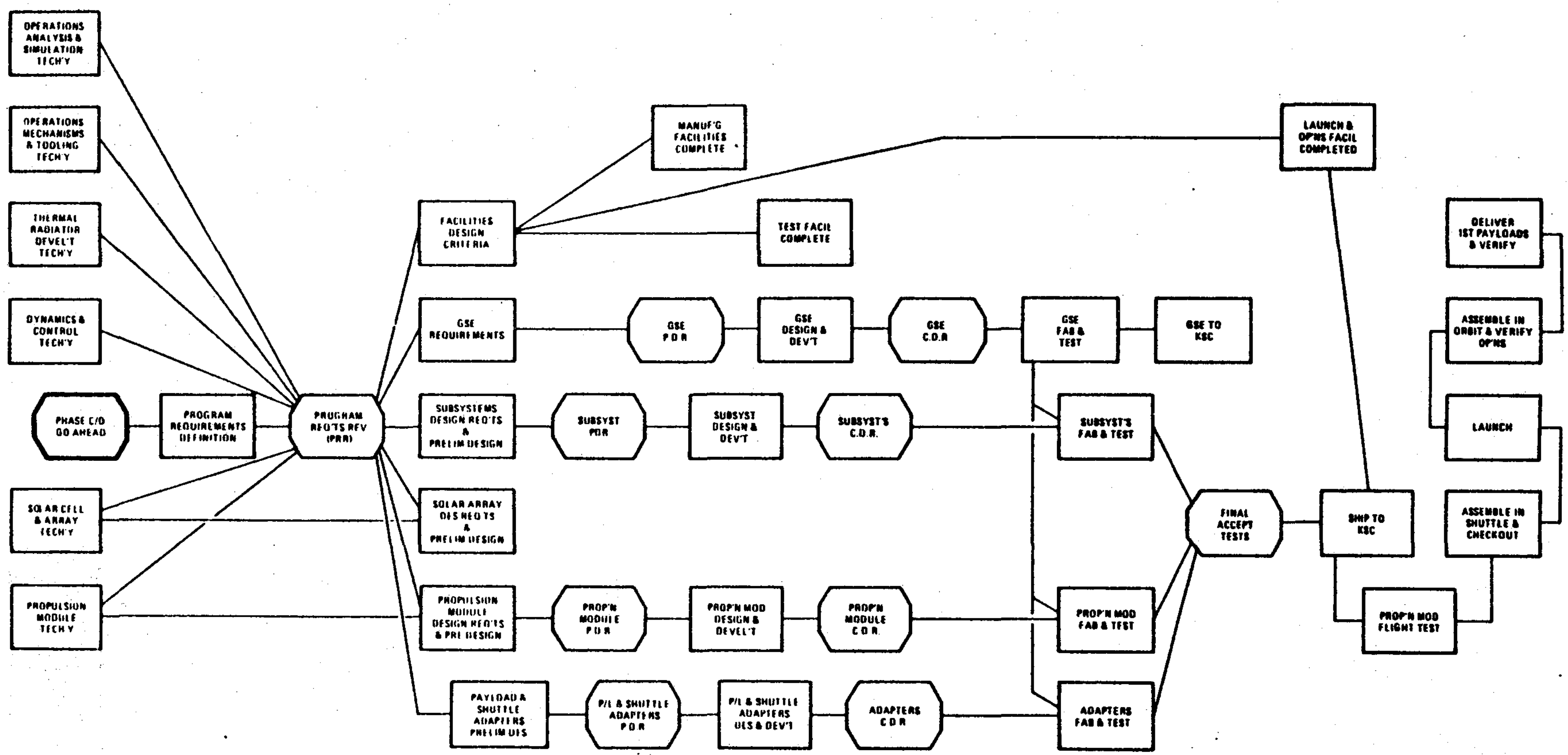
A very preliminary development plan for an erectable space platform for Science and applications was prepared. The next four pages outline an initial work breakdown structure (WBS), a program sequence, and a critical development schedule. It was assumed, for this plan, that the power module's solar arrays, propulsion module, and Shuttle orbiter modification programs would be treated by NASA as separate contracts, not under erectable science and applications platform module contractor's control. Development was assumed to be about 42 months from Phase C/D authority to proceed to platform launch.

INITIAL WORK BREAKDOWN STRUCTURE ERECTABLE SPACE PLATFORM DEVELOPMENT PROGRAM

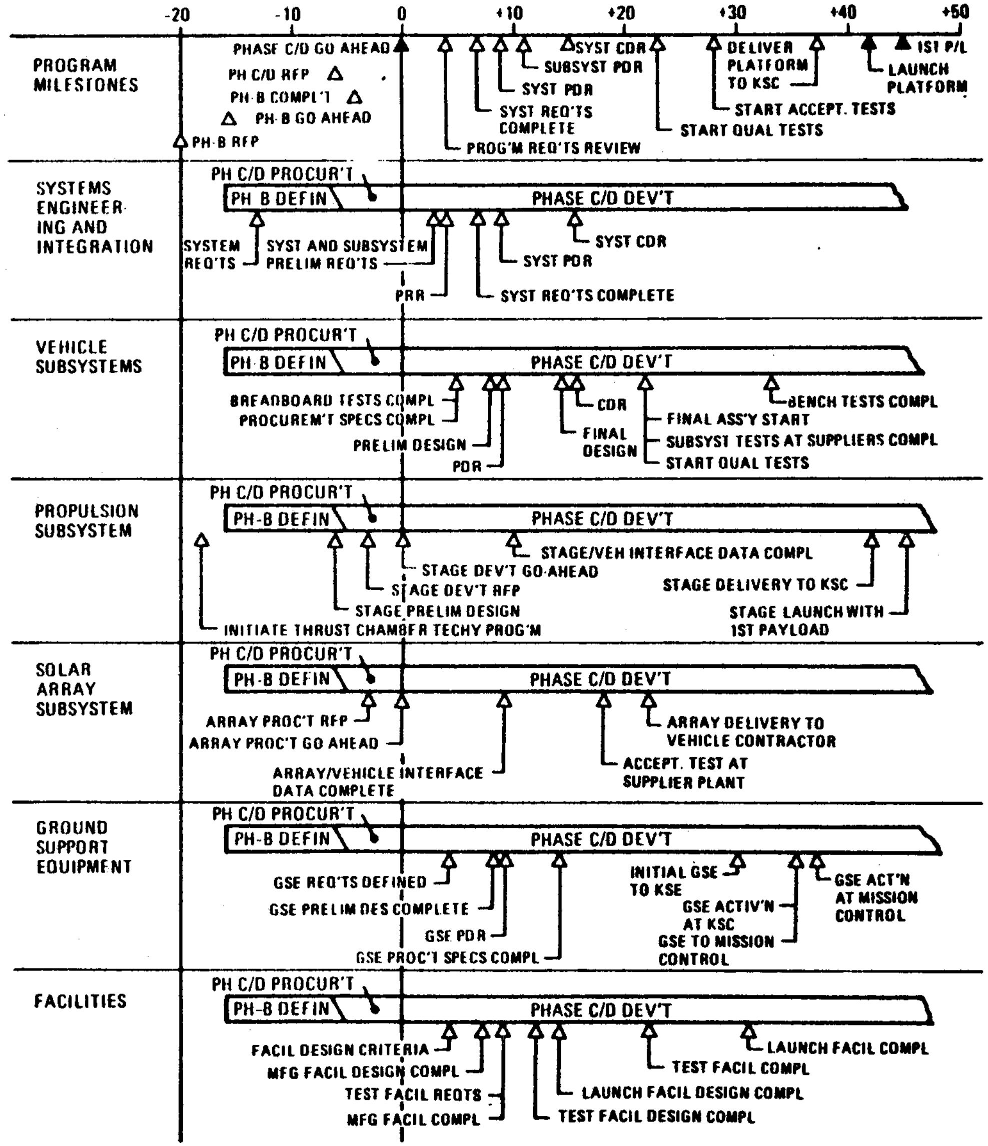



XXX-00-00-00-00
ERECTABLE SPACE PLATFORM
XXA-00-00-00
DDT&E
PRODUCTION
XXB –
OPERATIONS
XXC –
07-00-00-00
SYSTEM TEST OPERATIONS
- 01-00-00
STRUCTURAL TEST
- 02-00-00
THERMAL TEST
- 03-00-00
SYSTEM VERIFICATION TEST
- 04-00-00
ACCEPTANCE TEST
- 05-00-00
ORBITAL OPERATIONS VERIFICATION
08-00-00-00
LOGISTICS
- 01-00-00
TRANSPORTATION AND HANDLING
- 02-00-00
TRAINING
- 03-00-00
INVENTORY CONTROL AND WAREHOUSING
- 04-00-00
SPARES
09-00-00-00
SOFTWARE
- 01-00-00
ON-BOARD
- 02-00-00
MISSION SUPPORT
- 03-00-00
TEST OPERATIONS
10-00-00-00
FACILITIES
- 01-00-00
MANUFACTURING
- 02-00-00
TEST
- 03-00-00
PAYLOAD READINESS
- 04 00 00
05 00 00
OPERATIONS
11-00-00-00
LAUNCH OPERATIONS
- 01-00-00
SITE ACTIVATION
- 02-00-00
PRE-LAUNCH
- 03-00-00
LAUNCH
12-00-00-00
FLIGHT OPERATIONS
- 01-00-00
MISSION/ PLANNING
- 02-00-00
MISSION CONTROL CENTER
- 03-00-00
FLIGHT ANALYSIS

INITIAL PROGRAM SEQUENCE
ERECTABLE SPACE SCIENCES AND APPLICATIONS PLATFORM



MONTHS FROM PROGRAM START



Rough Order Magnitude (ROM) Cost Assessment

A ROM cost assessment shows that for the three elements that make up the payload platform facility as defined herein. The greatest cost element is the utilities (power) module followed by the propulsion module and the payload platform in that order. The platform itself is only a small portion of the total facility cost. The payload platform facility will carry from 5-10 times (quantity and/or size of payloads) that of a conventional satellite and it's initial total cost estimates are roughly in that order.

Real savings may be in the way experiments are handled and the flexibility offered by a platform. Further study must be accomplished before reporting details in such a sensitive area. First look, however, does not throw out a platform on the basis of cost.

APPENDIX

PAYLOADS CHARACTERISTICS SUMMARY

PAYLOAD NO.	CATEGORY OR DESIGNATION	YEAR AVAILABLE	WEIGHT (KG)	DIMENSIONS (M)	PALLETS (EQ.)	POWER (KW) AV/PK	TEMP. LIM./ CRYOGENICS	ORBIT ALT. (KM)	ORBIT INCL. (DEG)	VIEW/ACCURACY	DATA RATE Mb/s or as noted	SPECIAL REQUIREMENTS	PLATFORM COMPATIBILITY
01	SOL PHYS 1	1982	1000	-	1	0.5 0.75	25 ±5 C	≥400	HI	SOL±10 ARC-SEC	1.6	APPROX. 6 MONTHS MAINTENANCE	P-2
02	SOL PHYS 2A	1983	1000	1ϕ×1.5ℓ	.5	0.05	50 W COOLING 100 K	≥300	HI	SOL,±.5	1.6	SUN-SYNC TO 50% SUNLIGHT	P-2
03	SOL PHYS 2B	1983	500	1.2×1.2× 3ℓ	1	0.3	20±2 C	375	28	SOL, 3 ARC-SEC	0.5	STABILITY, 0.2 ARC-SEC/10 SEC	P-1, STD IPS REQUIRED
04	SOL PHYS 3	1986	280	0.5×0.5× 3	1	0.2	(T)	200	NOT CRIT.	SOL, 10 ARC-SEC	TV 8/5S	ALSO SMALLER PACKAGE ADD FILM/2 MO	P-2 (PROBABLE), STD IPS REQUIRED
05	SOL PHYS 4	1988	(T)	2 @ 1×1× 3 M	1	0.2	10±1 C	≥400	NOT CRIT.	SOL, 1 ARC-MIN.	TV 8/5S	ADD FILM/2 MO	P-2 (PROBABLE), STD IPS REQUIRED
06	SOL PHYS 5	1989	75	.3×.3×2	1	0.2	10±1 C	≥200	HI, NOT CRIT.	SOL, 10 ARC-SEC	TV		P-2, STD IPS REQUIRED
07	SCADM SOL DYNAM.	1986	1700		1	1.5	25±5 C	≥575	HI	SOL, 1 ARC-MIN.	0.5		P-2, STD IPS REQUIRED
08	PINHOLE CAMERA, 100 M	1985	500	1 1 2		0.5	25±10 C	≥200	HI	SOL±5°	0.5	100-M BOOM	"TALL POLE" DUE TO BOOM
09	PINHOLE CAMERA, 1 KM	1988	3000 + 400	20ϕ + 9ϕ		0.5	25±10 C	≥200	HI	SOL±5°	0.5	1-KM TETHER	"TALL POLE" DUE TO TETHER & SIZE
10	SOFT X-RAY	1989	1000	1×1×7ℓ	3	0.35	(T)	≥400	HI	SOL, <1 ARC-SEC	TV 8/5S		P-2, STD IPS REQ'D
11	SOLAR OPTICAL TELESCOPE	1984	6600	3.8 × 7.3ℓ	3	2.5	-	450	HI	SOL, <1 ARC-SEC	0.05		P-2, STD IPS REQ'D

P-1 = 28°, 400 KM, INERTIAL; P-2 = 90°, 575 KM, INERTIAL, SOLAR POINTING; P-3 = 57°, 400 KM, EARTH L.V.

PAYLOAD NO.	CATEGORY OR DESIGNATION	YEAR AVAILABLE	WEIGHT (KG)	DIMENSIONS (M)	PALLETS (EQ.)	POWER (KW) AV/PK	TEMP. LIM./ CRYOGENICS	ORBIT ALT. (KM)	ORBIT INCL. (DEG)	VIEW/ACCURACY	DATA RATE Mb/s or as noted	SPECIAL REQUIREMENTS	PLATFORM COMPATIBILITY
12	SPP 1 PART. ACC.	1982	500		0.5	T (USE .5) 2	25 ± 10C	ANY	>56	EARTH 1°	0.5 +TV	2 PAYLOADS—SEPAC + AEPI	P-3
13	SPP 2	1982	395	1×1.5	0.5	T (USE .5)	T	T	T	N/A	T		ASSUME P-1
14	SPP 3	1983	2000		1	0.25 0.25	T	300 TO 3000	50 TO 90	EARTH MAG FLD 0.2°	0.3		P-3
15	ATM SCI 1	1984	1000		1	1.5 2.3	(T) REGEN CRYO	300 TO 500	28 & 70	EARTH 3°		REGEN. LHe	P-1 & P-3 GIMBAL ON P-1
16	CLIR CRYO/LIMB	1985	500 W/1 MO LHe		1	0.12 0.6	3 TO 115 K LHe	400 ± 100	28 & 70	EARTH (1 ARC-SEC)	0.524	LHe	P-1 & P-3, STD IPS REQ'D
17	LIDAR	1986	1000		1	3 3	25 ± 10C	200 TO 400	28 & 70	EARTH ±0.5°	0.5		P-1 AND P-3, GIMBAL ON P-1
18	ATM GRAV. WAVE ANT.	1988	4000 (ASSUM)	100 φ ANTENNA		10 MW	T	250	56 NOT CRIT.	EARTH 5°	0.01		"TALL POLE" DUE TO 100 M φ ANTENNA, HIGH POWER PROBLEM
19	PART. BEAM INJ.	1986	"	100×100× 10 M		0.1 0.4	T	WANT WIDE RANGE	>56	EARTH MAG FLD ± 1°	0.2	LARGE SCREENS	"TALL POLE" DUE TO LARGE SCREEN DEPLOYMENT
20	RAD BELT DYNAM FAC	1985	500	1×0.5× 0.5	.5	0.3 0.3	25±10C	>400 IF CIRC.	HIGH	EARTH MAG FLD 1°	0.036	WOULD LIKE ECCENT. ORB. 400 IF CIR.	O.K. W/CIRC. ORB. ASSUME P-3
21	SUBSAT.	1986 (REUSED)	2000	4×3×3	2	(T)	T	T	T	T	T		ASSUME P-1
22	WAVE PART.	1984	100 1000 TETH-ER	$(.3)^3$ + DIPOLE		25 25	T	250 TO 500	≤55	GRAV STAB N/A	0.2	BOOM DIPOLE + TETHER FACILITY	"TALL POLE"—HIGH POWER AND TETHER

P-1 = 28°, 400 KM, INERTIAL; P-2 = 90°, 575 KM, INERTIAL, SOLAR POINTING; P-3 = 57°, 400 KM, EARTH L.V.

PAYLOAD NO.	CATEGORY OR DESIGNATION	YEAR AVAILABLE	WEIGHT (KG)	DIMENSIONS (M)	PALLETS (EQ.)	POWER (KW) AV/PK	TEMP. LIM./ CRYOGENICS	ORBIT ALT. (KM)	ORBIT INCL. (DEG)	VIEW/ACCURACY	DATA RATE Mb/s or as noted	SPECIAL REQUIREMENTS	PLATFORM COMPATIBILITY
23	CHEM REL	1986	3000	3φ × 1 ℓ	1	N/A	T	NOT CRIT.	NOT CRIT.	N/A	N/A		ASSUME P-1
24	MAG PULS.	1980 (≥1)	1000	(T) 1-KM ANT.		25	T	>2000	56	N/A	0.001	1-KM ANTENNA	"TALL POLE" ALTITUDE TOO HIGH
25	TETHER FAC.	1984 (1)	1000	2×2×4		0.12 0.1	T	225	28&70	GRAV N/A	0.2	50-M BOOM	"TALL POLE" DUE TO BOOM & TETHER
26	HE PALLET 1	1983 (2)	1000		1	0.4 0.6	22±5 C	400	28	INERTIAL 6 ARC-MIN	10 kb	GAS RESUP. 1-2 YR	P-1, STD IPS REQ'D
27	HE PALLET 2	1983 (≥1)	3000	3×3×4	2	0.2 0.5	0 TO 30 C	300 TO 500	≥70 NOT CRIT.	INERTIAL N/A	0.1		P-2
28	HE PALLET 3	1986 (1-2)	2× 1000		0.5	0.3 1	22±5 C	T	T	INERTIAL 6 ARC-MIN	30K	GAS RESUP.	P-1, STD IPS REQ'D
29	HE PALLET 4	1988 (2-10)	1000	2×2×1	1	0.12	5 TO 35 C	400	28	INERTIAL NONE	10K	0.1 ARC-MIN. POINT REF.	P-1
30	HE PALLET 5	1989 (2-10)	10,000	3φ×5ℓ	1	0.1	0 TO 30 C	400	28	INERTIAL 1	3K		P-1
31	HE PALLET 6	1984 (≥2)	1200	6 @ $(1.2)^3$	1	0.12	15±10C	400 ±100	≤28	INERTIAL NONE	4K		P-1
32	HE PALLET 7	1986 (≥2)	1000	1×1×3	0.5	0.35	20±10C	400 ±100	≤28	INERTIAL 1 ARC-MIN	25K		P-1, STD IPS REQ'D
33	HE PALLET 8	1987 (≥2)	250	1.3×1.3× 1.5	0.3	0.1	20±10C	400 ±100	≤28	INERTIAL 1°	0.25	GAS RESUP.	P-1
34	HE PALLET 9 & 10	1989 (2-10)	10,000	5φ×10ℓ	4	0.15	20±10C	400	≥28	INERTIAL 1°	T		P-1
35	ASTP I	1982 (≥1)	70 + 500	.5×2×1× + .5×5×1	1	1.0 4.0	T	400	28 & 50	INERTIAL ASSUME 1 ARC- SEC	T	2 SEP. P/L	P-1 & P-3 STD IPS REQ'D

P-1 = 28°, 400 KM, INERTIAL; P-2 = 90°, 575 KM, INERTIAL, SOLAR POINTING; P-3 = 57°, 400 KM, EARTH L.V.

PAYLOAD NO.	CATEGORY OR DESIGNATION	YEAR AVAILABLE	WEIGHT (KG)	DIMENSIONS (M)	PALLETS (EQ.)	POWER (KW) AV/PK	TEMP. LIM./ CRYOGENICS	ORBIT ALT. (KM)	ORBIT INCL. (DEG)	VIEW/ACCURACY	DATA RATE Mb/s or as noted	SPECIAL REQUIREMENTS	PLATFORM COMPATIBILITY
36	ASTP 2	1984 (1)	1000	1φ×2ℓ	1	0.2 4	T	300	28	INERTIAL .5 ARC-S	100 b	FILM & EVA	P-1, SUPER IPS REQ'D
37	ASTP 3	1987 (1)	3500		2	4 9		>300	28 TO 50	INERTIAL T	1	MAY REQUIRE 100 W CRYO FOR DETECTOR	P-1
38	ASTP 4	1983 (≥1)	3500		1	4	20±10C	300 TO 500	28 TO 50	INERTIAL 1 ARC-SEC	1		P-1, STD IPS REQ'D
39	ASTP 5	1985 (≥1)	3000		1	1.5	"	"	"	INERTIAL .1 ARC-SEC	1		P-1, SUPER IPS REQ'D
40	SIRTF	1985 (6 MO)	2500	9×4×4	3	5	10°K, LHe	>350	ANY	"	3	LHe	P-1, SUPER IPS REQ'D
41	SPACELAB-II SURVEY	1986 (2-10)	725 INCL. LHe	3.4×1.8× 1.0	0.5	0.8 0.31	-10 TO 25°C, LHe	300 TO 500	28	T	1	LHe	P-1
42	LAMAR	1987 (≥1)	2000	3×3×3	2	0.24	22±5 C	400	28	INERTIAL 6 ARC-MIN	50K		P-1, STD IPS REQUIRED
43	STAR LAB	1987 (2-10)	2000	1.5φ×4	2	1.3		300	28	INERTIAL 1 DEG	TO 3	0.02 ARC-SEC REF	P-1
44	ATM TELE. PL DET.	1988 (2-10)		1.5φ×20ℓ		4	T	300 TO 400	28	INERTIAL .001 ARC-SEC	1	EVA 1/YR	P-1, SUPER IPS REQ'D
45	LG. IR TEL.	1989 >1	16,000	T	T	T	CONST. TEMP. CRYO	400 TO 700	28 TO 50	INERTIAL 0.1 ARC-SEC	1	CRYO	P-1, SUPER IPS REQ'D
46	ADDITIONAL XRO INSTR.	1988 ≥5	800	$(1.5)^2 \times$ 2.2 + 1×1×1.5	1	1	20±10C	400± 100	<28	INERTIAL 5 ARC-SEC	.15		P-1, STD IPS REQ'D

P-1 = 28°, 400 KM, INERTIAL; P-2 = 90°, 575 KM, INERTIAL, SOLAR POINTING; P-3 = 57°, 400 KM, EARTH L.V.

PAYLOAD NO.	CATEGORY OR DESIGNATION	YEAR AVAILABLE	WEIGHT (KG)	DIMENSIONS (M)	PALLETS (EQ.)	POWER (KW) AV/PK	TEMP. LIM./ CRYOGENICS	ORBIT ALT. (KM)	ORBIT INCL. (DEG)	VIEW/ACCURACY	DATA RATE Mb/s or as noted	SPECIAL REQUIREMENTS	PLATFORM COMPATIBILITY
47	SVRF	1987 30-DAY	1000	3×3×1.5	1	4 12	300 C MAX	200 TO 500	ANY	ANY	T	$\leq 10^{-4}$g	UNSUITABLE, MFG.
48	MEC	1984 30-DAY	3000 3890	3.7ϕ×3	1	14 25	200 C MAX	180 500 200	28	ANY	T	$\leq 10^{-5}$g, 1 kW DURING ASCENT	UNSUITABLE, MFG.
49	MEM	1987 3 MO	12,000	4ϕ×5.4	1+	20	T	180 TO 500	28 TO 57	ANY	T		UNSUITABLE, MFG.
50	S-S/L-PM-S	1984 30-DAY	3000	5.8×2.3× 0.5	1+	14	10 TO 40 C	"	"	ANY	T	10^{-3} TO 10^{-5}g	UNSUITABLE, MFG.
51	GCP							GEO					UNSUITABLE, GEO
52	NAT. OCEAN SAT. SYST.	1984 3	3820	11×4×4	3	1.8 24	T	600 TO 800	85 TO 87	ANY 0.5°	T 120	HIGH DATA RATE 24 KW PEAK POWER	P-2, IF CAN ACCEPT ALT, INCL. & PEAK POWER
53	85-G							GEO					UNSUITABLE—GEO
54	SEV. STORM							GEO					UNSUITABLE—GEO
55	LANDSAT D	1981 T	2130	2.2ϕ×2.5	1	.5	T	705	98.14	T	T		P-2, IF CAN ACCEPT ALT. & INCL.
56	EARTH RES. ATM. PROC.	1984 1	1357	T	USE 1	1.8 3.4	T	180 TO 289	57	EARTH	T		P-3, IF CAN ACCEPT ALT.
57	SHUT. IMAGE RADAR	1985 1	808	11 M^3	1	2	T	180 270 225	28 90 57	EARTH 2.5	.12	UNFOLDING L-BAND ANT. CORNER MOUNT	P-3, IF CAN ACCEPT ALT.
58	SPACE METEOR. RADAR	1990 1	514	7 M^3	1	3.4 8.0	T	170 230 200	"	EARTH T	.12		P-3, IF CAN ACCEPT ALT.
59	LIDAR LTS	1982 T	300	7 M^3	0.5	3	T	ANY 200 300	30 TO 90 (USE 57)	EARTH 1	1		P-3, IF CAN ACCEPT ALT.

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PAYLOAD NO.	CATEGORY OR DESIGNATION	YEAR AVAILABLE	WEIGHT (KG)	DIMENSIONS (M)	PALLETS (EQ.)	POWER (KW) AV/PK	TEMP. LIM./ CRYOGENICS	ORBIT ALT. (KM)	ORBIT INCL. (DEG)	VIEW/ACCURACY	DATA RATE Mb/s or as noted	SPECIAL REQUIREMENTS	PLATFORM COMPATIBILITY
60	LIDAR LPS	1984 4 YR	300	0.5 M^3	0.25	2.8	T	ANY 200 300	30 TO 90	EARTH 1	1		P-3, IF CAN ACCEPT ALT.
61	CLIMATE RES. SAT.	1986 3	2000	5 4 4	2+	2.5	T	600 TO 900	56	EARTH T	T		P-3, IF CAN ACCEPT ALT.
62	PASSIVE MICROWAVE	1985 1.5	325	40 M^3	2	0.5	T	300 TO 900	57 TO 90	EARTH 6 ARC-MIN.	0.2	3.15M RAD OF ROTATION CORNER MOUNT	P-3, STD IPS REQUIRED
63	ATM SOLAR STUDY	1984 6 MO	624	8 M^3	0.5	1.6 2.1	T	200 300 500	28 TO 70 (USE 57)	SOLAR EARTH LIMB T	T		P-3
64	LAND/ATM PROFILE	1984 6 MO	1321	2.5 M^3	0.25	4.1 4.8	T	370 TO 400	45 TO 55	EARTH 17 ARC-SEC	T		P-3, STD IPS REQ'D
65	SYST. 85	1985 2	1000		1 ASSUMED	1	T	700 1600	98	EARTH T			P-2, IF CAN ACCEPT ALT. & INCLINATION, GIMBAL REQ'D

P-1 = 28°, 400 KM, INERTIAL; P-2 = 90°, 575 KM, INERTIAL, SOLAR POINTING; P-3 = 57°, 400 KM, EARTH L.V

SCREENING—PAYLOAD ELIMINATION

PAYLOAD NO.*	DESCRIPTION	PROBLEM
8	100-M PINHOLE CAMERA	100-M BOOM
9	1-KM PINHOLE CAMERA	1-KM TETHER
18	ATMOSPHERIC GRAVITY WAVE ANTENNA	100-M-DIA ANTENNA/HIGH PWR
19	PARTICLE BEAM INJECTION	100x100x10-M SCREENS
22	WAVE PARTICLE INTEGRATION	300-M TETHER
24	MAGNETIC PULSATIONS	1-KM ANTENNA
25	TETHER FACILITY	1-KM TETHER & 200-KM ALT.
47	SPACE VACUUM RESEARCH FACILITY	MANUFACTURING
48	MATERIALS EXPERIMENT CARRIER	MANUFACTURING
49	MATERIALS EXPERIMENTATION MODULE	MANUFACTURING
50	SHUTTLE/SPACELAB/POWER MODULE-SORTIE	MANUFACTURING
51	GEOSTATIONARY COMM. PLATFORM	GEOSYNCHRONOUS
53	SYSTEM 85 OPERATIONAL GEOSAT (85-G)	GEOSYNCHRONOUS
54	SEVERE STORM RESEARCH SATELLITE	GEOSYNCHRONOUS

*Assigned for purposes of this study

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