REPORT NO. GDC-GPP-79-006 (I)
CONTRACT NAS8-33527
JUNE 1980

GEOSTATIONARY PLATFORM SYSTEMS CONCEPTS DEFINITION STUDY

FINAL REPORT
VOLUME II TECHNICAL
BOOK 3 OF 3

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Prepared by

GENERAL DYNAMICS
Convair Division
&
COMSAT

for the

National Aeronautics and Space Administration
GEORGE C. MARSHALL SPACE FLIGHT CENTER
Huntsville, Alabama

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Submitted to
GEORGE C. MARSHALL SPACE FLIGHT CENTER
National Aeronautics and Space Administration
Marshall Space Flight Center, Alabama 35812

Prepared by
GENERAL DYNAMICS CONVAIR DIVISION
P.O. Box 80847
San Diego, California 92138

and

COMMUNICATIONS SATELLITE CORPORATION
COMSAT Laboratories
Clarksburg, Maryland 20734

GEOSTATIONARY PLATFORM SYSTEMS
CONCEPTS DEFINITION STUDY
FINAL REPORT

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William T. Carey

William T. Carey, Chief
Applications Group, PS06
George C. Marshall Space Flight Center
Huntsville, Alabama

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Dr. Robert M. Bowman
Study Manager
General Dynamics Convair

Dr. Denis Curtin
Subcontract Manager
COMSAT Corporation

PREFACE

In today's world of expanding communication, military, and science satellite services, the geostationary orbit is rapidly becoming an extremely valuable and limited earth resource. Nations demand specific positions or "slots" in the orbit corresponding to their geographic longitude, seeking to maximize their territorial coverage and satellite performance. Sovereignty becomes an issue, with several nations at different latitudes and one longitude competing for the common longitudinal slot in the orbital arc. Common carriers within a developed nation demand equal rights for the best slots. Competition has been strong in the developed nations, and the developing nations are now voicing their concern.

At geosynchronous altitude, independent satellites operating at the same frequency must be separated by about 4 degrees of longitude to prevent RF interference (30 dB separation), dictated by the large beam widths of the small affordable ground antennas now in use. About 90 "slots" therefore exist around the world, with about 12 over the U. S. and our northern and southern neighbors.

The frequency spectrum is also a valuable and limited resource that is rapidly approaching saturation, particularly in those regions of low noise and freedom from atmospheric attenuation.

Both resources are now allocated worldwide by the International Telecommunications Union operating through subservient multinational and national agencies. Reallocation cannot solve our basic orbital arc and frequency saturation problems. Recent studies have shown projected traffic demands which will saturate both the geostationary orbital arc and the optimal frequency spectra in the near future. In the U. S. alone, current domestic satellite capacity is about 100 transponders. Projections indicate a five-fold increase in traffic demand for voice, data, and TV distribution in the next 10 years (by 1990); ten-fold by the year 2000. If video and audio conferencing expand as projected, the jump may be to 20 to 50 times the present traffic by 1990 and the year 2000, respectively.

Motivation for the rapid adoption of satellite communications services is primarily economic. Satellite communications provide lower service cost for certain fixed applications, economy of flexibility, and appreciable cost savings over terrestrial operation for mobile services direct to the users. Savings can be increased still further if the cost, complexity, and size of ground stations can be reduced by application of advanced communications and support technologies to a few satellites with expanded capabilities.

What is the solution to our orbital arc and frequency spectrum saturation problems, a solution that also lends itself to reduction of user costs?

One viable solution is the aggregation of many transponders, large antennas, and connectivity switches on board a small number of large orbital facilities. Such facilities, or platforms, can provide common power and housekeeping services to a number of coexistent communications systems, making maximum use of a single orbital slot. Large antennas with multiple spot beams and good isolation, bandwidth reduction, polarization diversity, and system interconnectivity can provide an equivalent transponder capacity over the U. S. at least an order of magnitude greater than the projected traffic demand for the year 2000.

In the public interest, NASA has initiated a program to encourage development of such geostationary platforms, anticipating the need for increased communications and other services in the near decades, at lower costs. In the past two years, initial NASA studies[1] have established the need and requirements for, and the feasibility of these platforms. NASA's George C. Marshall Space Flight Center has been authorized to carry out in-depth studies of geostationary platforms.

This report documents the results of the Geostationary Platform Initial Phase A Study, performed by General Dynamics Convair Division of San Diego with COMSAT Corporation of Clarksburg, Maryland, as subcontractor, under direction of the Marshall Space Flight Center. The performance period was from 1 June 1979 to 30 June 1980.

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[1] "Large Communications Platforms Versus Smaller Satellites," Future Systems, Inc., Report No. 221 February 1979, prepared for NASA HQ.

"Geostationary Platform Feasibility Study," Aerospace Corp., Report No. ATR-79(7749)-1, 28 September 1979, prepared for NASA/MSFC.

"Geostationary Platforms Mission and Payload Requirements Study," 30 October 1979, prepared for NASA/MSFC.

"18/30 GHz Communications System Service Demand Assessment," 30 June 1979, parallel studies by Western Union and ITT for NASA/LeRC.

"18/30 GHz Communications Service System Study," June 1979, parallel studies by Ford Aerospace & Communications Corp., and by Hughes Aircraft Co. for NASA/LeRC.

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GLOSSARY

ACOSS	Active Control of Space Structures
ACS	attitude control system
AFC	automatic frequency control
AIL	Avionics Integration Laboratory
APS	auxiliary power subsystem
APSK	amplitude and phase shift keying
APU	auxiliary power unit
ASE	airborne support equipment
ASU	acquisition and synchronization unit
BER	bit error rate
BFN	beam forming network
BOL	beginning of life
BOSS	baseline optical surveillance system
BSM	baseband switch matrix
CADSI	communications and data systems integration
C&W	caution and warning
CCC	central communications control
CER	cost estimating relationship
C/I	carrier/interference
CITE	cargo integration test equipment
CMG	control moment gyro
COMSAT	Communications Satellite Corporation
CONUS	Contiguous United States
CPS	customer premise services (same as DTU)
CQPSK	coherent quadriphase shift keying
CRT	cathode-ray tube
CSC	Computer Sciences Corporation
CTE	coefficient of thermal expansion
DARPA	Defense Advanced Research Projects Agency
DCU	distribution control unit
DDT&E	design, development, test and evaluation
DEU	Display Electronics Units
DHS	data handling system
DMS	Data Management System
DMSP	Defense Meteorological Satellite Program
DNSP	Defense Navigation Satellite Program
DoD	Department of Defense
DOD	depth of discharge
DoE	Department of Energy
DORA	double rolled array
DPS	Data Processing system
DQPSK	differential quadriphase shift keying
DSCS	Defense Satellite Communications System
DSN	deep space network

DTU	direct to user (same as CPS)
EHF	extra high frequency
EIRP	effective isotropic radiated power
EMC	electromagnetic compatibility
EMU	extravehicular mobility unit
EOL	end of life
EPC	electronic power conditioner
EPS	electrical power subsystem
ESA	European Space Agency
EVA	extravehicular activity
FCC	Federal Communications Commission
FDMA	frequency division multiple access
FDM/FM	frequency division multiplex/frequency modulation
FET	field effect transistor
FMECA	Failure Modes and Effects Criticality Analysis
FPR	flight performance reserve
FPT	flat plate trough
FRUSA	flexible rolled up solar array
FSI	Future Systems, Inc.
FSS	frequency selective subreflector
GDC	General Dynamics Convair Division
GDTTSS	General Dynamics Tetrahedral Truss Structure Computer Program
GEO	geostationary orbit
GFE	government-furnished equipment
GFP	government-furnished property
GN&C	guidance, navigation and control
GOICM	Ground Operations and Integration Cost Model
GP	Geostationary Platform
GPC	General Purpose Computer
GRARR	Goddard range and range rate
GSE	ground-support equipment
GSFC	Goddard Space Flight Center
G/T	gain-to-noise temperature ratio
HALO	High Altitude Laser Optics
HPA	high power amplifier
HVT	high volume trunking
ICL	intra-constellation link
I/F	interface
IF	intermediate frequency
IFSM	intermediate frequency switch matrix
IM	intermodulation
IMPATT	type of power-transmitting diode
IMU	inertial measurement unit
IOC	Initial Operational Capability
IOTV	Interim OTV

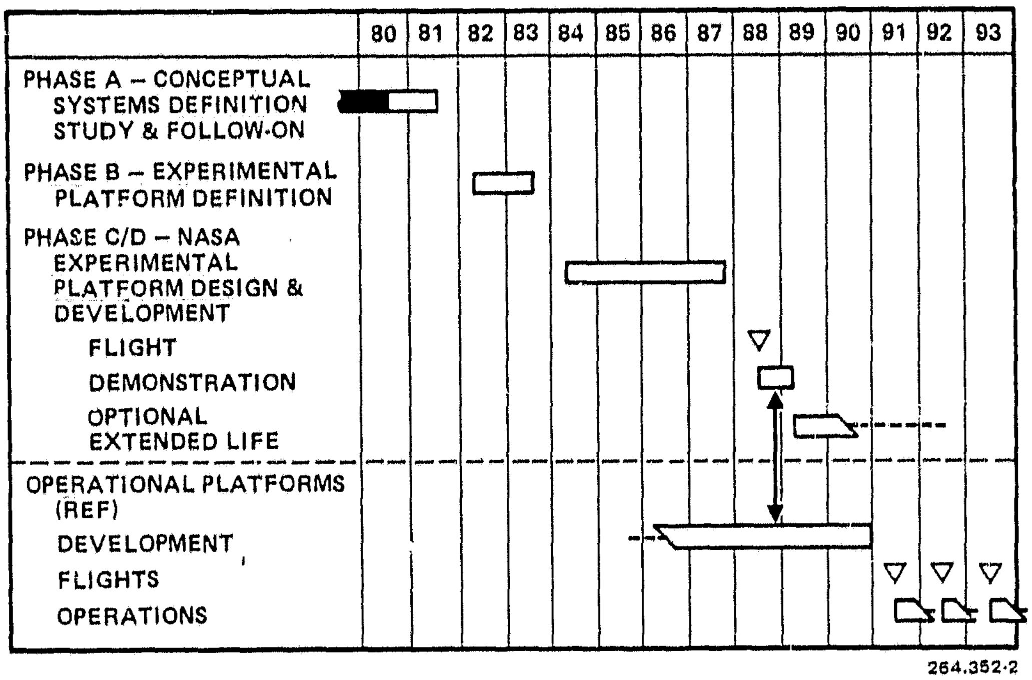
IPL	inter-platform link
IR	infrared
ISP	specific impulse
ITU	International Telecommunications Union
IUS	inertial upper stage
IVA	intravehicular activity
JPL	Jet Propulsion Laboratory
JSC	Lyndon B. Johnson Space Center
km	kilometer
KSC	John F. Kennedy Space Center
LaRC	Langley Research Center
LASS	Large Advanced Space System
LCC	life cycle cost
LEO	low earth orbit
LeRC	Lewis Research Center
LH_2	liquid hydrogen
LLLTV	low light level television
LO_2	liquid oxygen
LSS	large space structures
MBA	multibeam antenna
MBFRA	multiple beam frequency reuse antenna
MCC	Mission Control Center (at JSC)
MCDS	Multifunction Control and Display System
MDM	multiplexer-demultiplexer
MMP	Mission Mass Properties (program)
MMS	Multimission Modular Spacecraft
MMU	manned maneuvering unit
MOSFET	metal oxide silicon field effect transistor
MSFC	Marshall Space Flight Center
MSM	microswitch matrix
MTBF	Mean Time Between Failures
MUX	multiplex
NASA	National Aeronautics and Space Administration
nm	nautical miles
NPV	net present value
OBR	onboard regenerator
OLS	optical line scanner
OMJ	ortho-mode junction
OMS	orbital maneuvering subsystem
OOA	On-Orbit Assembly
OPF	Orbiter Processing Facility
OSR	optical solar reflector
OSS	orbital servicing system
OSS	Office of Space Science (NASA)
OSTA	Office of Space and Terrestrial Applications
OTV	orbital transfer vehicle

PCM	Pulse Code Modulation
PCR	payload changeout room
PDI	Payload Data Interleaver
PEP	power extension package
PETA	Parabolic Expandable Truss Antenna
P/L	payload
POP	perpendicular to orbit plane
PSP	payload signal processor
Q	quality factor (energy stored/energy lost)
QPSK	quadriphase shift keying
RAU	remote acquisition unit
RCS	reaction control subsystem
RF	radio frequency
RFI	radio frequency interference
RIU	remote interface unit
RMS	remote manipulator system
ROI	return on investment
RTOP	Research and Technology Operating Plan
SADA	solar array drive assembly
S/C	spacecraft
SCF	Satellite Control Facility
SCR	silicon controlled rectifier
SEP	solar electric propulsion
SGLS	space ground link subsystem
SPST	single pole single throw
SRT	supporting research and technology
SS	solid state
SSLCC	Space System Life Cycle Cost
SS-TDMA	satellite-switched time division multiple access
SSUS	spin-stabilized upper stage
STDN	space tracking and data network
STS	Space Transportation System
T&C	telemetry and command
TDMA	time-division multiple access
TDRS	Tracking and Data Relay Satellite
TDRSS	Tracking and Data Relay Satellite System
TMS	Teleoperator Maneuvering System
T/R	transmit/receive
TRIAC	type of regulator tube
TSO	time-sharing option
TT&C	telemetry, tracking and command
T/W	thrust-to-weight ratio
TWT	traveling wave tube
TWTA	traveling wave tube amplifier
ULP	ultra-lightweight panel
USB	unified S-band

UV	ultraviolet
VCHP	variable conductance heat pipe
VPF	vertical processing facility
WAP	Work Authorization Plan
WBS	Work Breakdown Structure
Wp	weight of propellant
Wt	weight
X_o	X-axis of Orbiter
X_p	X-axis of payload
Y_o	Y-axis of Orbiter
Y_p	Y-axis of payload
Z_o	Z-axis of Orbiter
Z_p	Z-axis of payload
ΔV	delta (incremental) velocity

SUMMARY

The George C. Marshall Space Flight Center (MSFC) has the responsibility within the NASA for the geostationary platform - to initiate conceptual studies, develop feasible concepts, coordinate user needs and technology requirements, and promote activities aimed at system hardware solutions to the projected service demands of the 1990s. The schedule, as shown here, provides for a National Aeronautics and Space Administration (NASA) experimental platform in 1988 to validate required technology, and operational platforms with launch dates in the 1990s.



Projected Development Schedule for Geostationary Platforms

On 31 may 1979, General Dynamics Convair was placed under contract to do the Initial Phase A Concepts Definition Study for the Geostationary Platform. NASA/MSFC's planned approach includes a review of communications, military and science payloads, and mission models, development and analysis of operational and experimental platform concepts, identification of communications and platform technology requirements, and development of supporting programmatic data. Primary objectives of the study are to select and conceptually define operational geostationary platforms based on time-phased mission and payload requirements, and to develop attendant costs, schedules, and supporting research and technology (SRT) requirements. This data will be used as a basis for definition of the NASA experimental geostationary platform, which will be the subject of follow-on studies, although some preliminary precursor work on the experimental platform was done during this initial phase of the study.

Six tasks were defined in the Statement of Work (SOW) for this study:

Task 1 - Further Define Candidate Missions and Payloads.

Task 2 - Define Candidate Approaches/Concepts and Conduct Analyses and Trades Leading to Selected Concepts.

Task 3 - Define Selected Approaches and Concepts.

Task 4 - Define Supporting Research and Technology and Recommended Space Demonstrations.

Task 5 - Define Requirements On and Interfaces With STS Hardware Elements.

Task 6 - Define and Develop Cost and Schedule Data.

This document, Volume II of the final report, summarizes the technical and programmatic work performed in satisfying Tasks 1 through 5 of the Statement of Work and Study Plan requirements for these tasks. It contains in-depth discussions of the study elements, engineering data, and system and programmatic trades generated during the study. Parts 1 and 2 of this volume address operational and experimental geostationary platforms, respectively. Extensive data tables and drawings are documented in the appendixes (Volume II Supplemental Data), where appropriate.

Task 6, Cost and Schedules Data, is treated separately (Volume III of the Final Report), per data procurement document instructions.

A summary of Task 1 through 5 results follows.

In Task 1, candidate geostationary platform missions and payloads were identified from COMSAT, Aerospace, and NASA studies. These missions and payloads were cataloged; classified with respect to communications, military or scientific uses; screened for application and compatibility with geostationary platforms; and analyzed to identify platform support requirements. Two platform locations were then selected (Western Hemisphere - 110°W, and Atlantic - 15°W), and payloads allocated based on nominal and high traffic models considering communications payloads only, and considering communications plus secondary [Department of Defense (DoD) and science] payloads. In all cases, candidate payload requirements and characteristics were defined on three-page candidate payload data summary forms (Appendix E).

In Task 2, candidate platform concepts were defined and analyzed, and trade studies performed leading to recommendation of selected concepts. Of 30 Orbit Transfer Vehicle (OTV) configuration and operating mode options identified from data supplied by NASA/MSFC, 18 viable candidates compatible with the operational geostationary platform missions were selected for analysis. Each was considered using four platform operational modes - 8 or 16 year life, and serviced or nonserviced, providing a total of 72 OTV/platform-mode options. Standard platform concepts were defined for each of the 72 options for both the nominal and the high traffic models, and payloads reallocated to these 144 options based on OTV performance capability and payload weight and power. For final trade study concept selection, a cost program was developed considering payload and platform costs and weight; transportation unit and total costs for the Shuttle and OTV; and operational costs such as assembly or construction time, mating time, and loiter time. Servicing costs were added for final analysis and recommended selection.

The 144 candidate concepts were screened and the nine best options for combinations of launch and operating modes, transfer vehicles, and evolutionary buildup modes were analyzed. Four were recommended and selected by NASA for further study. Alternative #1 was designated for definition in Task 3. Alternatives #2, 3, and 4 were deferred to the follow-on study for further defintion.

Task 3 defines concept Alternative #1 as a data base for further geoplatform analyses in this study, in sufficient detail to identify requirements for supporting research and technology, space demonstrations, GFE interfaces, costs, and schedules. Alternative #1 consists of six platforms in geostationary orbit (GEO) over the Western Hemisphere and six over the Atlantic, to satisfy the total payload set associated with the nominal traffic model. Each platform is delivered to low earth orbit (LEO) in a single shuttle flight, already mated to its LEO-to-GEO transfer vehicle and ready for deployment and transfer to GEO.

Although Alternative #4 was deferred to the follow-on study for further definition, it was looked at briefly in this initial study for comparison of configuration and technology requirements. Alternative #4 consists of two large platforms, one over the Western Hemisphere consisting of three docked modules, and one over the Atlantic (two docked modules), to satisfy a high traffic model. The modules are full-length orbiter cargo-bay payloads, mated at LEO to OTVs delivered in other shuttle flights, for transfer to GEO, rendezvous, and docking.

Alternatives #2 and 3, deferred to the follow-on study for definition, are respectively single-shuttle flight platforms docked at GEO and multiple-shuttle platforms in constellation at GEO.

Task 3 was expanded somewhat to include a preliminary feasibility study of an experimental platform to demonstrate communications and platform technologies required for the operational platforms of the 1990s. Six configurations were conceptually developed to consider a wide variation in payloads, structure, number of shuttle flights, and compatibility with available OTV performance characteristics. Results of this task (3A) are reported in Part 2 of this volume.

Task 4 identifies the SRT and space demonstrations required to support the 1990s Operational Platforms as typified by Concept Alternatives #1 and #4.

Task 5 identifies the requirements on and interfaces with STS hardware elements supporting the geostationary platform program, including the shuttle, orbital transfer vehicles, teleoperator, etc., to provide integrated support requirements to these programs.

The body of this volume concludes with a short preview of work to be accomplished on the follow-on study, in which operational platforms will be further characterized and concepts for an experimental geostationary platform further developed. Central to the further characterization of operational platforms will be the development of a multislot communications architecture using low-risk communications technology. Work on experimental geostationary platform concepts will concentrate on identifying affordable configurations compatible with potential upper stages.

SECTION 4

TASK 4: SUPPORTING RESEARCH AND TECHNOLOGY AND SPACE DEMONSTRATIONS

To place operational geostationary platforms in orbit in the 1990s, capable of supporting the high-capacity, expanded communications services needed in our near future, a significant advancement in both platform and communications technologies will be required. These technologies in most cases have been foreseen and in some instances are already in partial development. Others have surfaced as a result of the conceptual analysis effort in this study. To minimize program funding and schedule risks and to ensure proper operational program evolution, the more advanced technologies must be identified and defined, and plans developed to verify their operational validity. This section of the report addresses these tasks.

4.1 OBJECTIVE

The objective of this task is to identify and define supporting research and technology (SRT) needed to enable successful development of operational geostationary platforms of the 1990s. Accomplishment of this objective requires:

a. Identification of required technologies.

b. Comparison with current and planned capabilities, so that deficiencies can be identified.

c. Preparation of top-level recovery plan for each deficiency.

d. Preparation of an integrated recovery plan, including space experiments where necessary.

4.2 SCOPE

By the time we complete the follow-on to this initial study, we plan to identify SRT requirements for all four of the alternative concepts selected by NASA and designated Alternatives #1 through #4. The technologies involved fall generally into three main categories:

a. Technologies common to all platform concepts.

b. Technologies unique to constellation concepts (Alternatives #1 and #3).

c. Technologies unique to docked concepts (Alternatives #2 and #4).

Even though the only concept given in-depth treatment as a conceptual design in Task 3 has been Alternative #1, the technologies unique to such constellation concepts are still not well understood. The idea of a constellation or cluster is relatively new. Accordingly, the second of the above technology categories will need a great deal of continued attention in the follow-on. It is difficult, for example, to assess the sufficiency of station-keeping technology until requirements can be established for relative position maintenance of the constellation members. These requirements in turn depend on attainable performance in tracking intermodule links and on the details of the TDMA synchronization scheme for interpayload traffic. Such factors introduce compound unknowns that can only be rationally addressed in a dedicated study of considerable depth. There may even be "show-stopper" technologies associated with these new constellation technologies. Within the scope of this study, all that can be done is to identify some broad areas in which detailed technology analysis appears to be required.

For the other two categories, technology requirements are better understood. Even though in-depth conceptual design for the docked alternatives will not be accomplished until the follow-on study, some category 3 technologies have been included in the documentation, since GDC has been working these areas for several years. However, because of the additional work to be done in the follow-on, no attempt has been made to provide comprehensive coverage of these technologies at this time. Technologies common to all platform concepts are fairly well handled, although it can be expected that even this category will be expanded in the follow-on study.

4.3 METHODOLOGY

Identification of supporting research and technology requirements and planning for recovery of deficiencies follows a four-step process:

a. Identification and analysis of subsystems.

b. Development of technology requirements by examination of evolving concepts as geoplatform layouts and subsystem studies proceed.

c. Comparison to state of the art, as well as to the objectives of ongoing, related studies.

d. Evaluation for recovery.

For purposes of examination and analysis, technology requirements for geostationary platforms have been separated into two general groups:

a. Platform subsystems and operations.

b. Communications, including specialized communications/integration systems.

Subsystems within these areas have been analyzed and key geostationary platform technologies identified. Each technology has been categorized, evaluated with respect to state of the art and with reference to related studies, recovery plan scheduled, and the complete analysis documented on a three-page summary. These are included as tables within the subtasks that follow in this section of the report. Each table presents a single technology description, justification, and recovery plan. This separation of technologies does not preclude union of technologies into joint efforts where appropriate, for example when program funding and planning for the geoplatform can be better defined.

In each case, technologies have been defined with respect to specific platform requirements. For example, while an initial evaluation of power management and distribution technology for communications satellites indicates that it is already well in hand, our multihundred kW study with LeRC shows that the assumption may well be incorrect and the problem must therefore be worked with specific reference to geoplatform requiremements. The same is true of large solar arrays, which must deal specifically with requirements for long-term service at geostationary altitude.

Where prior studies dealing with the listed technologies have taken place, they are referenced. More detailed search for supporting data is recommended when advancement of any of the technologies is undertaken.

Some technologies, for example power management and distribution, are areas in which we have considerable familiarity by reason of ongoing contracts. Others, such as the development of a digital matrix switch, are more obscure since developments in this area are privately funded and proprietary to the developers.

Current and recent study efforts examined for state-of-the-art information are:

Orbital Power Module (MSFC/JSC)

SEPS (MSFC)

Space Operations Center (JSC Definitions)

20/30 GHz Study (LeRC)

Space Construction Systems Analysis (JSC)

Teleoperator and Servicing Studies (MSFC)

Clearly, these represent the front line of development, and few have proceeded to a limited hardware stage.

With the possible exception of a flight test program to demonstrate and evaluate a controlled, deployable structure, no special flight test requirements are foreseen prior to a platform demonstration flight. The experimental platform can be considered as a union of the critical technologies, each of which has been sufficiently proven on a preliminary basis through ground testing.

4.4 ANALYSIS AND RESULTS

4.4.1 PLATFORM SUBSYSTEMS. Figure 4-1 identifies the platform subsystems with which we are concerned. Tables 4-1 through 4-10 define technology requirements for the platform subsystems. These include:

a. Space construction.

b. Active control of large space structures.

c. Solar array.

d. Power management and distribution.

e. Secondary power source.

f. Increased performance RCS.

g. Thermal management.

h. Automatic docking and servicing.

A key early shuttle flight involves orbital deployment of a structural element and evaluation of both the structure in the orbital environment and its dynamic interaction with the orbiter control system. This experiment could provide additional value to the geoplatform effort if it is extended to obtain experience with hardware installation and removal, emphasizing both the RMS and astronaut participation. A definition study for an orbital experiment will be initiated by NASA/JSC in mid-1980. Preliminary estimates indicate that a flight date in 1984 is reasonable, and would be valuable to plans for an experimental platform flight in 1987 or 1988.

Table 4-10 outlines technology requirements in the area of docking and servicing. This task needs development in more detail during follow-on studies.

The general subject of servicing is a significant technology for future operations. It is directed toward achieving long life for platform systems by means of consumables replenishment, repair and replacement, and equipment update. Advancements in robotics, fluids transfer in orbit, remote sensing, remote rendezvous and docking, and mechanisms are required. To enable servicing, redesign of basic subsystems will be necessary from the standpoint of packaging for robotics handling, and for ease of making itnerconnects.

4.4.2 COMMUNICATIONS. The continuing rapid growth of communications message traffic (voice, data, and video) requires an expansion in utilization of existing and new satellite communications frequency bands before the 1990 decade.

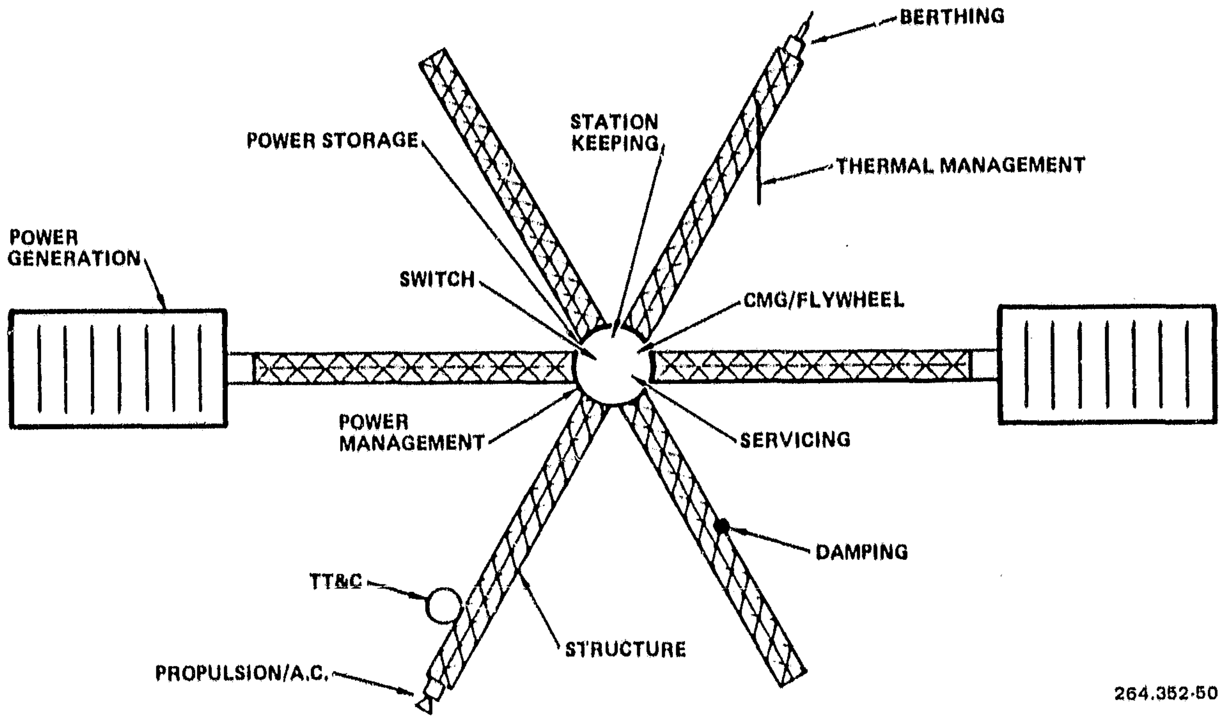


Figure 4-1. Platform Subsystems

The satellite frequency bands currently in use at 4 and 6 GHz are already crowded and new operational systems must be implemented at 12 and 14 GHz. Market studies sponsored by NASA LeRC indicate a possible saturation of 6/4 and 14/12 GHz bands by 1990.

Current NASA-sponsored communications R&D program efforts are aimed at developing the technology needed for utilization of the 30/20 GHz bands. Much of this technology will also serve to expand capacity at the lower frequency bands. Narrow beams for trunking applications will permit large volume traffic between single ground terminals located at carriers' facilities in major communications centers. Scanning or fixed spot beams for customer premise service can be configured for multiple reuse of the frequency band to conserve the spectrum. New spacecraft antenna techniques must be developed whereby a large number of independent fixed and/or scanning beams can be radiated from a single geostationary satellite. These multibeam antenna systems require technological development in the areas of large offset-fed parabolic reflectors, accurate reflector surface contours, reduced thermal distortion, and active real-time beam control.

(Continued on Page 4-36)

Table 4-1. Space Construction

DEFINITION OF TECHNOLOGY REQUIREMENT

1. TECHNOLOGY REQUIREMENT (TITLE): Space Construction Page 1 of 3

2. TECHNOLOGY CATEGORY: Platform Subsystems

3. OBJECTIVE/ADVANCEMENT REQUIRED: Controlled deployment of a space structure, structure evaluation, and evaluation of interaction with Orbiter control.

4. CURRENT STATE OF ART: Requirements yet to be defined.

5. DESCRIPTION OF TECHNOLOGY:
A deployable beam, proven and analyzed through ground tests, will be deployed in orbit. The beam will be designed to requirements for the geoplatform and other large space systems. Shuttle operations for deployment will be evaluated. Structural and thermal performance will be assessed. An important part of this technology is evaluation of control interaction between the Orbiter and large flexible objects that are attached to it. For this reason, the beam should be of a sufficient length and flexibility to require control itself.

6. RATIONALE AND ANALYSIS:

Although extensive ground testing will precede the flight of any prospective large space system, testing remains to be done in orbit on a simple structure that will bridge the gap between what can be learned and predicted on the ground, to gain the confidence and establish procedure that will be needed to deploy the first of the large space systems.

Table 4-1. Space Construction, Contd

DEFINITION OF TECHNOLOGY REQUIREMENT
1. TECHNOLOGY REQUIREMENT (TITLE): Space Construction Page 2 of 3
7. TECHNOLOGY OPTIONS: Prior evaluations of space construction technologies have virtually ruled out space fabrication and/or erection as viable options in the time frame of interest. On the other hand, deployable structures already have a substantial background, and show ready applicability to a number of systems in the near term.
8. TECHNICAL PROBLEMS: 1. Structure: Development of compression molded composite fittings to enable fabrication of thermally stable structures. 2. Dynamics: Development of devices that can accomplish both disturbance of structure for testing and control for damping.
9. POTENTIAL ALTERNATIVES: Metal fittings can be used but probably will not achieve optimum thermal performance. Control devices can be separate from test devices, but in either case development will be needed.
10. PLANNED PROGRAMS OR UNPERTURBED TECHNOLOGY ADVANCEMENT: NASA JSC plans "Orbiting Space Construction Definition Study" directed toward above technology beginning in CY80.
11. RELATED TECHNOLOGY REQUIREMENTS: Dynamic control of space structures.

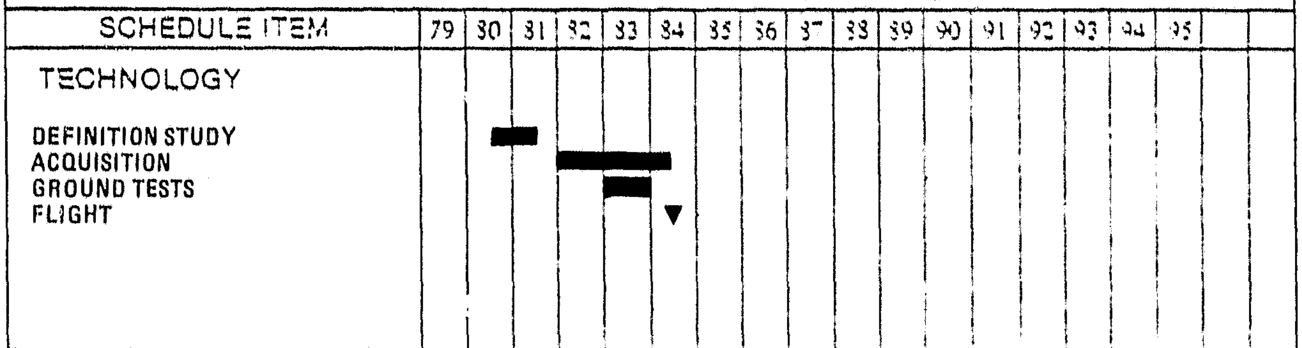
Table 4-1. Space Construction, Contd

DEFINITION OF TECHNOLOGY REQUIREMENT — No.

1. TECHNOLOGY REQUIREMENT (TITLE): SPACE CONSTRUCTION — Page 3 of 3

12. TECHNOLOGY REQUIREMENTS SCHEDULE:

CALENDAR YEAR



SCHEDULE ITEM	79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	95		
FUNDING LEVEL (In $M, 1980 dollars)		0.15	0.25	1.5	5.0	3.1													

13. USAGE SCHEDULE:

	79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	95	TOTAL
TECHNOLOGY NEED DATE						▼												
NUMBER OF LAUNCHES						1												

14. REFERENCES

NAS9-15718 SPACE CONTRUCTION SYSTEM ANALYSIS (ROCKWELL).
SAMSO TR-78-128 DoD/STS ON ORBIT ASSEMBLY CONCEPT DESIGN STUDY, FINAL IRAD REPORT.

15. LEVEL OF STATE OF THE ART:

1. Basic phenomena observed and reported
2. Theory formulated to describe phenomena
3. Theory tested by physical experiment or mathematical model
4. Pertinent functions or characteristic demonstrated, e.g., material, component
5. Component or breadboard tested in relevant environment in laboratory
6. Model tested in aircraft environment
7. Model tested in space environment
8. New capability derived from a much lesser operational model
9. Reliability upgrading of an operational model
10. Lifetime extension of an operational model

Table 4-2. Active Control of Large Space Structures

DEFINITION OF TECHNOLOGY REQUIREMENT

1. TECHNOLOGY REQUIREMENT (TITLE): Active Control of Large Space Structures — Page 1 of 3

2. TECHNOLOGY CATEGORY: Platform Subsystems

3. OBJECTIVE/ADVANCEMENT REQUIRED: Active and passive structural damping theory and hardware (mostly hardware).

4. CURRENT STATE OF ART: Diverse theories are emerging that are not tailored to specific applications. Space-worthy hardware for implementation is essentially nonexistent.

5. DESCRIPTION OF TECHNOLOGY:

The modal characteristics of a large platform will be extremely complex and computer-aided design techniques will be needed for the control system. In addition, practical space-worthy sensors and actuators are required to perform the sensing and control of flexible space structures.

6. RATIONALE AND ANALYSIS:

The large lightweight structure will have a new regime of long period modes of oscillation. When these oscillations are excited by station-keeping thrusters, thermal shock, and/or TMS operations, accurate pointing cannot be maintained without active damping. Also, static shape control systems and rigid body attitude control systems generally cannot tolerate lightly damped modes inside their bandwidths, independent of excitation.

Table 4-2. Active Control of Large Space Structures, Contd

DEFINITION OF TECHNOLOGY REQUIREMENT

1. TECHNOLOGY REQUIREMENT (TITLE): Active Control of Large Space Structures — Page 2 of 3

7. TECHNOLOGY OPTIONS:

1. Modify existing control theory for communications platform application.
2. Identify control component requirements, select, fabricate, development test, and life test.

8. TECHNICAL PROBLEMS:

1. Complexity of modal model.
2. Coupling of modes by control system action.
3. Modal observability at suitable control component locations.
4. Bandwidth of control components.
5. Control and observation spillover.

9. POTENTIAL ALTERNATIVES:

More rigid, much heavier structure.

10. PLANNED PROGRAMS OR UNPERTURBED TECHNOLOGY ADVANCEMENT:

1. USAF/DARPA ACOSS program is addressing optical precision problem on smaller structures.
2. Various IRAD programs.

11. RELATED TECHNOLOGY REQUIREMENTS:

Modeling techniques for large space structures and for damping mechanism, e.g. viscous, hysteristic.

Table 4-2. Active Control of Large Space Structures, Contd

DEFINITION OF TECHNOLOGY REQUIREMENT	No.
1. TECHNOLOGY REQUIREMENT (TITLE): ACTIVE CONTROL OF LARGE SPACE STRUCTURES	Page 3 of 3

12. TECHNOLOGY REQUIREMENTS SCHEDULE:

CALENDAR YEAR

SCHEDULE ITEM	79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	95		
TECHNOLOGY																			
REQUIREMENTS AND THEORY MODIFICATION																			
COMPONENT DESIGN																			
COMPONENT DEVELOP. TEST AND FABRICATION																			
FLIGHT CONSTRUCTION DEMO-STRATION COMPONENT LIFE TEST						▼								▽	LAUNCH OPERATIONAL PLATFORM				
FUNDING LEVEL (In $1,000, 1980 dollars)			500	250	250	200													

13. USAGE SCHEDULE:

	79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	TOTAL
TECHNOLOGY NEED DATE						▼											
NUMBER OF LAUNCHES						1*								1			

14. REFERENCES

*: ORBITER-BASED EXPERIMENT.

GDC 1980 IRAD 907, 908
CONTRACT F30602-80-C-0164

15. LEVEL OF STATE OF THE ART:

1. Basic phenomena observed and reported
2. Theory formulated to describe phenomena
3. Theory tested by physical experiment or mathematical model
4. Pertinent functions or characteristic demonstrated, e.g., material, component
5. Component or breadboard-tested in relevant environment in laboratory
6. Model tested in aircraft environment
7. Model tested in space environment
8. New capability derived from a much lesser operational model
9. Reliability upgrading of an operational model
10. Lifetime extension of an operational model

Table 4-3. Solar Array

DEFINITION OF TECHNOLOGY REQUIREMENT

1. TECHNOLOGY REQUIREMENT (TITLE): Solar Array Page 1 of 3

2. TECHNOLOGY CATEGORY: Platform Subsystems

3. OBJECTIVE/ADVANCEMENT REQUIRED: Multikilowatt solar arrays for long term service at geosynchronous orbit.

4. CURRENT STATE OF ART: SEPS, low kilowatt arrays for communications satellites.

5. DESCRIPTION OF TECHNOLOGY:

Perform development work and analysis directed toward achieving a capability to prepare design specifications for solar arrays in the range of 10 kW to 100 kW for communications platform service.

Dynamic behavior structural analysis, deep thermal cycles, particle irradiation, electrostatic charging effects, cell specifications, electrical network, and deployment mechanics.

6. RATIONALE AND ANALYSIS:

Power generation at high kW levels (over 10 kW) at geosynchronous altitude has no precedent. Highly reliable arrays that meet requirements for dynamic stability, plasma charge suppression, minimum degradation rate, efficiency, deep thermal cycles, and weight will be required. Currently SEPS and orbital power module studies provide a base for moving into the new technology area. Selection of alternatives to silicon solar cells is not ruled out.

Table 4-3. Solar Array, Contd

DEFINITION OF TECHNOLOGY REQUIREMENT

1. TECHNOLOGY REQUIREMENT (TITLE): Solar Array — Page 2 of 3

7. TECHNOLOGY OPTIONS:

1. Rigid versus membrane arrays.
2. GA-AS versus silicon
3. Concentrators
4. Large area cells
5. FRUSA, SEPS, ULP, etc. deployment techniques.

8. TECHNICAL PROBLEMS:

1. Production of high efficiency GA-AS not yet in quantity.
2. Automated assembly of large arrays incorporating cell and substrate advancements.
3. Deep thermal cycles long-term effect on large arrays.

9. POTENTIAL ALTERNATIVES:

1. Straightforward projection of silicon technology to large systems.
2. Provide advancements such as GA-AS cells and concentrators.

10. PLANNED PROGRAMS OR UNPERTURBED TECHNOLOGY ADVANCEMENT:

1. GA-AS development.
2. Orbital power module development.
3. Space operations center studies.

11. RELATED TECHNOLOGY REQUIREMENTS:

1. Power control and distribution.
2. Thermal management.
3. Pointing/attitude control.

Table 4-3. Solar Array, Contd

DEFINITION OF TECHNOLOGY REQUIREMENT	No.
1. TECHNOLOGY REQUIREMENT (TITLE): SOLAR ARRAY	Page 3 of 3

12. TECHNOLOGY REQUIREMENTS SCHEDULE:

CALENDAR YEAR

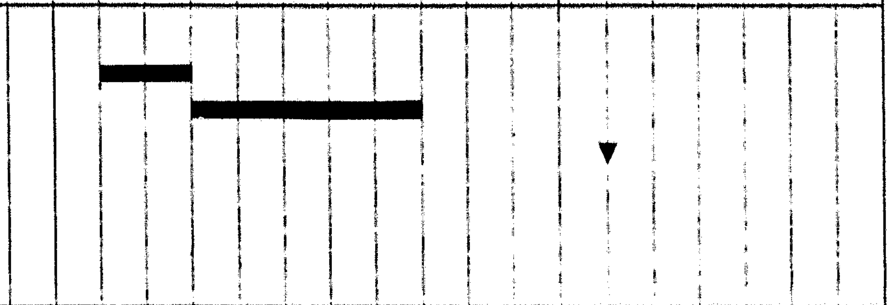
SCHEDULE ITEM	79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	95

TECHNOLOGY

DEFINITION

DEVELOPMENT

FLIGHT



FUNDING LEVEL (In $M, 1980 dollars)	79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	95
			0.2	0.3	1	2	3	2									

13. USAGE SCHEDULE:

	79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	TOTAL
TECHNOLOGY NEED DATE									▼								
NUMBER OF LAUNCHES													1				

14. REFERENCES

MSFC, JSC ORBITAL POWER MODULE STUDIES, POWER EXTENSION PACKAGE STUDIES
LeRC MULTI-HUNDRED kW POWER MANAGEMENT STUDY

15. LEVEL OF STATE OF THE ART:

1. Basic phenomena observed and reported
2. Theory formulated to describe phenomena
3. Theory tested by physical experiment or mathematical model
4. Pertinent functions or characteristic demonstrated, e.g., material, component
5. Component or breadboard tested in relevant environment in laboratory
6. Model tested in aircraft environment
7. Model tested in space environment
8. New capability derived from a much lesser operational model
9. Reliability upgrading of an operational model
10. Lifetime extension of an operational model

Table 4-4. Power Management System

DEFINITION OF TECHNOLOGY REQUIREMENT

1. TECHNOLOGY REQUIREMENT (TITLE): Power Management System Page 1 of 3

2. TECHNOLOGY CATEGORY: Platform Systems

3. OBJECTIVE/ADVANCEMENT REQUIRED: System development for high frequency AC, resonant power distribution with integral payload user isolation.

4. CURRENT STATE OF ART: Central controller providing nonisolated low voltage DC to payloads/users.

5. DESCRIPTION OF TECHNOLOGY:

Higher voltage, (100-200) high frequency AC power management system that provides:

1. Distributed, local conditioning and control at each user interface.
2. Power system isolation.
3. Simple, reliable interface connections.
4. Versatility to accommodate many payloads with differring requirements.

6. RATIONALE AND ANALYSIS:

Platforms of this type are designed to provide services to many different payloads and users. In general, their power requirements are widely different and most require good isolation between one another and any common power system. The high-frequency AC power system provides this in a way that is both cost and weight competitive with the simplest DC approach. Its versatility and noncontact interfaces provide for simple payload changes when required.

Table 4-4. Power Management System, Contd

DEFINITION OF TECHNOLOGY REQUIREMENT

1. TECHNOLOGY REQUIREMENT (TITLE): Power Management System Page 2 of 3

7. TECHNOLOGY OPTIONS:

1. Thyristor implementations with frequencies in the high audio range (10-20 kHz), using current state-of-the-art components.
2. High frequency (50-100 kHz) implementations using improved design switching devices (bipolar transistors or power FETs) for greater improvements in size and weight.

8. TECHNICAL PROBLEMS:

No significant technical problems.

9. POTENTIAL ALTERNATIVES:

1. Continue present low voltage DC approach.
2. Develop higher voltage DC system, 100 or 200.

Both with DC-DC converters for user/payload isolation.

10. PLANNED PROGRAMS OR UNPERTURBED TECHNOLOGY ADVANCEMENT.

1. AC systems being evaluated at LeRC.
2. DC controller being developed at MSFC (p^3 - 25 kW).

11. RELATED TECHNOLOGY REQUIREMENTS:

Some component development required: rotary transformer, transformer disconnects, AC coaxial power bus, RPC development.

Table 4-4. Power Management System, Contd

DEFINITION OF TECHNOLOGY REQUIREMENT	No.
1. TECHNOLOGY REQUIREMENT (TITLE): POWER MANAGEMENT SYSTEM	Page 3 of 3

12. TECHNOLOGY REQUIREMENTS SCHEDULE:

CALENDAR YEAR

SCHEDULE ITEM	79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	95		
TECHNOLOGY																			
COMPUTER MODEL																			
SYSTEM DEVELOPMENT																			
BREADBOARD AND TEST																			
FUNDING LEVEL (In $1,000, 1980 dollars)																			
PROGRAM			200	250															

13. USAGE SCHEDULE:

																	TOTAL
TECHNOLOGY NEED DATE																	
NUMBER OF LAUNCHES																	

14. REFERENCES

15. LEVEL OF STATE OF THE ART:

1. Basic phenomena observed and reported
2. Theory formulated to describe phenomena
3. Theory tested by physical experiment or mathematical model
4. Pertinent functions or characteristic demonstrated, e.g., material, component
5. Component or breadboard tested in relevant environment in laboratory
6. Model tested in aircraft environment
7. Model tested in space environment
8. New capability derived from a much lesser operational model
9. Reliability upgrading of an operational model
10. Lifetime extension of an operational model

Table 4-5. Power Management System Control

DEFINITION OF TECHNOLOGY REQUIREMENT
1. TECHNOLOGY REQUIREMENT (TITLE): Power Management System Control — Page 1 of 3
2. TECHNOLOGY CATEGORY: Platform Systems
3. OBJECTIVE/ADVANCEMENT REQUIRED: Onboard semiautonomous power management system controller with backup ground control.
4. CURRENT STATE OF ART: Current space systems use combined ground control power management system.
5. DESCRIPTION OF TECHNOLOGY: The geostationary platform provides power for distribution to a number of communications payloads and the platform subsystems. An energy storage system for eclipse is charged by the solar array system. The power management system has to control battery charge cycling, eclipse operation, dual power bus loading and failure circumvention. Generalized there is load, power, reliability, and configuration management as elements of the power management system.
6. RATIONALE AND ANALYSIS: Present space systems use some ground control of power management requiring intermittent ground operations. The advent of μ processor technology, lightweight reliable VMOS switching, and larger payload capacity permits consideration of an on-board power management system without excessive weight and size penalty. On-board control permits temporary loss of the ground station command link. Proper design of the on-board control will permit ground station takeover of the power management system in the event of failure of a function of the on-board control. Semiautonomous power management can be designed to maximize the efficiency of the system and the life of components. Configuration control can be designed to mitigate the effects of failed components.

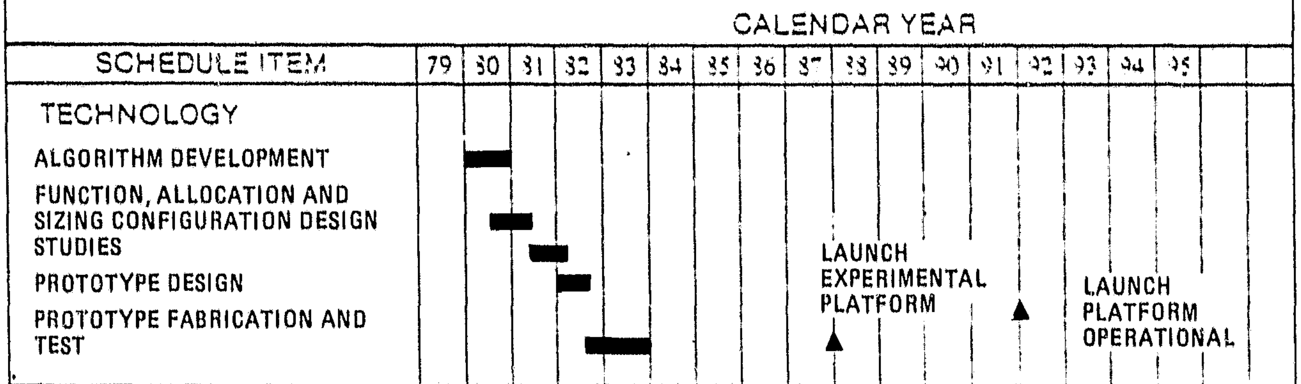
Table 4-5. Power Management System Control, Contd

DEFINITION OF TECHNOLOGY REQUIREMENT
1. TECHNOLOGY REQUIREMENT (TITLE): Power Management System Control — Page 2 of 3
7. TECHNOLOGY OPTIONS: 1. Microprocessor or special purpose logic design. 2. Common microprocessor for battery monitor and charge rate controller, and for power management, etc., or separate microprocessors for various functions. 3. Payload current limiting by payload regulators and/or by the power management system. 4. Distributed data and command bus system or centralized nonbus system.
8. TECHNICAL PROBLEMS: 1. Development of algorithms that will maximize efficiency of solar arrays, energy storage, and switching components, and the life of energy storage components. 2. Sizing of computational and control hardware and software for each power management system element. 3. Interface of microprocessors, data busses, and control elements. 4. round control takeover in event of malfunction.
9. POTENTIAL ALTERNATIVES: 1. Continue present power management system approach using combined ground control. 2. Simple on-board control without efficiency and life optimization. No extensive on-board computational capability.
10. PLANNED PROGRAMS OR UNPERTURBED TECHNOLOGY ADVANCEMENT DIS studies.
11. RELATED TECHNOLOGY REQUIREMENTS: 1. Integrated thermal management system. 2. Payload power usage and cycling. 3. Attitude control anomalies.

Table 4-5. Power Management System Control, Contd

DEFINITION OF TECHNOLOGY REQUIREMENT	No.
1. TECHNOLOGY REQUIREMENT (TITLE): POWER MANAGEMENT SYSTEM CONTROL	Page 3 of 3

12. TECHNOLOGY REQUIREMENTS SCHEDULE:



FUNDING LEVEL (In $1,000, 1980 dollars)	79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	95		
		100	200	400	400														

13. USAGE SCHEDULE:

	79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	TOTAL
TECHNOLOGY NEED DATE					▲												
NUMBER OF LAUNCHES									1					1			

14. REFERENCES

15. LEVEL OF STATE OF THE ART:

1. Basic phenomena observed and reported
2. Theory formulated to describe phenomena
3. Theory tested by physical experiment or mathematical model
4. Pertinent functions or characteristic demonstrated, e.g., material, component
5. Component or breadboard-tested in relevant environment in laboratory
6. Model tested in aircraft environment
7. Model tested in space environment
8. New capability derived from a much lesser operational model
9. Reliability upgrading of an operational model
10. Lifetime extension of an operational model

Table 4-6. Power Management Component Technologies

DEFINITION OF TECHNOLOGY REQUIREMENT

1. TECHNOLOGY REQUIREMENT (TITLE): Power Management Component Technologies - AC — Page 1 of 3

2. TECHNOLOGY CATEGORY: Platform Systems

3. OBJECTIVE/ADVANCEMENT REQUIRED: Sublevel component technologies in support of AC system development as described below.

4. CURRENT STATE OF ART: Components not developed.

5. DESCRIPTION OF TECHNOLOGY:

1. Rotary transformer for rotary joint.
2. Transformer disconnect for user interface.
3. Coaxial AC power transmission line.
4. High voltage, high current AC remote power controllers (RPC).
5. RPC data interface hardware.

6. RATIONALE AND ANALYSIS:

These sublevel devices are components needed to support the final design of a flyable, high voltage, high frequency AC power system.

Table 4-6. Power Management Component Technology, Contd

DEFINITION OF TECHNOLOGY REQUIREMENT
1. TECHNOLOGY REQUIREMENT (TITLE): Power Management Component Technology — Page 2 of 3
7. TECHNOLOGY OPTIONS: 1. Armature or flat design. 2. Multipole, multiwinding designs. 3. Size, shape, material, flexibility. 4. Thyristor, bipolar transistor, power FET output switching. 5. Wired or optical.
8. TECHNICAL PROBLEMS: None specifically identified.
9. POTENTIAL ALTERNATIVES: 1. Slip rings with conventional transformers. 2. Conventional connectors. 3. Twisted pair. 4. Electromechanical relays. 5. Single wire commands and data returns.
10. PLANNED PROGRAMS OR UNPERTURBED TECHNOLOGY ADVANCEMENT: Some to-be-defined programs at LeRC.
11. RELATED TECHNOLOGY REQUIREMENTS: AC system development.

Table 4-6. Power Management Component Technology, Contd

DEFINITION OF TECHNOLOGY REQUIREMENT — No.

1. TECHNOLOGY REQUIREMENT (TITLE): POWER MANAGEMENT COMPONENT TECHNOLOGY — Page 3 of 3

12. TECHNOLOGY REQUIREMENTS SCHEDULE:

CALENDAR YEAR

SCHEDULE ITEM	79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	95		
TECHNOLOGY																			
DEVELOPMENT			▬																
PROTOTYPE AND TEST			▬	▬															
FUNDING LEVEL (In $1,000, 1980 dollars)																			
PROGRAM			150	200															

13. USAGE SCHEDULE:

TECHNOLOGY NEED DATE																TOTAL
NUMBER OF LAUNCHES																

14. REFERENCES

15. LEVEL OF STATE OF THE ART:

1. Basic phenomena observed and reported
2. Theory formulated to describe phenomena
3. Theory tested by physical experiment or mathematical model
4. Pertinent functions or characteristic demonstrated, e.g., material, component
5. Component or breadboard-tested in relevant environment in laboratory
6. Model tested in aircraft environment
7. Model tested in space environment
8. New capability derived from a much lesser operational model
9. Reliability upgrading of an operational model
10. Lifetime extension of an operational model

Table 4-7. Secondary Power Source

DEFINITION OF TECHNOLOGY REQUIREMENT

1. TECHNOLOGY REQUIREMENT (TITLE): Secondary Power Source Page 1 of 3

2. TECHNOLOGY CATEGORY: Platform Subsystems

3. OBJECTIVE/ADVANCEMENT REQUIRED: Long life, high efficiency operation of replenishable NiH_2 battery system. Long life, nonreplenishable fuel cell and electrolyzer system.

4. CURRENT STATE OF ART: Space experiments with nonreplenishable NiH_2 battery packs. Operational space use of fuel cells. Lab use of electrolyzers.

5. DESCRIPTION OF TECHNOLOGY:

The geostationary platform requires a secondary power source that can be replaced in space for long term operation. The ultimate goal may be nonreplenishable 16-year life system. The NiH_2 battery system is a near-term solution, while the fuel cell and electrolyzer system is a long-term solution with potential lighter weight and higher storage capacity. For long life and efficiency, processor charge control algorithms using pressure, temperature, and/or voltage sensing needs to be developed.

6. RATIONALE AND ANALYSIS:

The mass of the secondary power source is a major portion of the EPS mass. Present Ni-CD batteries are limited in energy storage per unit mass. NiH_2 batteries have a substantial energy/mass efficiency improvement. Proven life of NiH_2 batteries is low to date, so a replenishable 8-year life system is initially proposed for the geoplatform, with separable thermal, electrical, and mechanical interfaces. The H_2-O_2 fuel cell is rapidly becoming a high power-to-weight ratio device. Electrolyzer development for a combined system is required to provide a lighter weight, higher power capability for future geoplatforms.

Table 4-7. Secondary Power Source, Contd

DEFINITION OF TECHNOLOGY REQUIREMENT

1. TECHNOLOGY REQUIREMENT (TITLE): Secondary Power Source Page 2 of 3

7. TECHNOLOGY OPTIONS:

1. NiH_2 battery. 1) Fluidic or nonfluidic thermal disconnect; 2) self-contained radiator; 3) DC connector or AC contactless transformer; 4) Quick release or bolt mounting; 5) Rail or pin support; 6) Ground monitor and control of charge; 7) Autonomous battery change control; 8) Voltage pressure, and/or temperature charge sensing; 9) Partially failed cell circumvention; 10) Eclipse emergence battery clamping of bus voltage.
2. Cell and electrolyzer. a. H_2-O_2, H_2-CL_2, H_2-BR; b. Combination unit or separate units; c. Solid polymer electrolyte or matrix acqueous alkaline.

8. TECHNICAL PROBLEMS:

1. Disconnect of fluid coolant lines.
2. Module removal by teleoperator.
3. Change algorithms for long life and high efficiency.
4. CL or BR handling.
5. Regenerative fuel cell reversal time duration.

9. POTENTIAL ALTERNATIVES:

1. Nuclear.
2. Momentum storage wheels.
3. Lightweight Ni-CD battery.
4. AG-H_2 battery.
5. Lithium, sodium batteries.
6. Molten carbonate fuel cell.
7. Solid oxide fuel cell.
8. Phosphoric acid fuel cell.

10. PLANNED PROGRAMS OR UNPERTURBED TECHNOLOGY ADVANCEMENT:

1. Nonreplenishable NiH_2 battery system for space use without sophisticated change control techniques for a 5-year life.
2. Nonregenerative lightweight fuel cell for expendable and unmanned OTV application.

11. RELATED TECHNOLOGY REQUIREMENTS:

1. Thermal management.
2. Teleoperator replenishment mechanisms.
3. Docking port and latches.
4. Power management and distribution.
5. Fluid tankage and plumbing systems.

Table 4-7. Secondary Power Source, Contd

DEFINITION OF TECHNOLOGY REQUIREMENT	No.
1. TECHNOLOGY REQUIREMENT (TITLE): SECONDARY POWER SOURCE	Page 3 of 3

12. TECHNOLOGY REQUIREMENTS SCHEDULE:

CALENDAR YEAR

SCHEDULE ITEM	79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	95		
TECHNOLOGY DISCONNECT INTERFACE STUDY AND ANALYSIS BATT. CHARGE ANAL AND TEST REPLENISHMENT BATT MODULE PROTOTYPE DESIGN, FAB AND GROUND TEST FUEL CELL TRADE STUDIES PROTOTYPE FUEL CELL DESIGN PROTOTYPE FUEL CELL FAB/TEST									LAUNCH EXPERIMENTAL PLATFORM				LAUNCH OPERATIONAL PLATFORM						
FUNDING LEVEL (In $1,000, 1980 dollars)			300	700	300	300		100	200	500	1.5K	500	200						

13. USAGE SCHEDULE:						NiH$_2$ BATTERY									FUEL CELL			
TECHNOLOGY NEED DATE						▼						▼					TOTAL	
NUMBER OF LAUNCHES									1								1	

14. REFERENCES

FORDYCE – "TECHNOLOGY STATUS – BATTERIES AND FUEL CELLS"
RITTERMAN – "CYCLING CHARACTERISTICS OF NICKEL-HYDROGEN CELLS
E.BETZ – "THE FIRST YEAR IN ORBIT FOR THE NTS-2 NI-H$_2$ BATTERY"
NUTTALL – "SOLID POLYMER ELECTROLYTE FUEL CELL AND H$_2$O ELECTROLYSIS STATUS REVIEW"
STEDMAN – "FUEL CELLS FOR 1980-1985 SPACE MISSIONS"

15. LEVEL OF STATE OF THE ART:

1. Basic phenomena observed and reported
2. Theory formulated to describe phenomena
3. Theory tested by physical experiment or mathematical model
4. Pertinent functions or characteristic demonstrated, e.g., material, component
5. Component or breadboard-tested in relevant environment in laboratory
6. Model tested in aircraft environment
7. Model tested in space environment
8. New capability derived from a much lesser operational model
9. Reliability upgrading of an operational model
10. Lifetime extension of an operational model

Table 4-8. Increased Performance RCS/Propulsion Subsystem

DEFINITION OF TECHNOLOGY REQUIREMENT

1. TECHNOLOGY REQUIREMENT (TITLE): Increased Performance RCS/Propulsion Subsystem — Page 1 of 3

2. TECHNOLOGY CATEGORY: Platform Subsystems

3. OBJECTIVE/ADVANCEMENT REQUIRED: Attain long life, highly reliable RCS/propulsion subsystem that will reduce overall platform system life cycle costs and/or increase revenue.

4. CURRENT STATE OF ART: Isp ≈ 200-220 sec; 7-10 year life; high reliability achieved through 2:1 redundancy factor.

5. DESCRIPTION OF TECHNOLOGY:

The baseline RCS/propulsion subsystem is a monopropellant hydrazine system with Isp = 230 sec. Long life and high reliability are achieved through a minimum development program for component refinements and redundancy of critical elements. Improvement in performance can be obtained through thermal augmentation, or substitution of pulsed plasma or ion propulsion devices. Technology required is development directed toward achieving capability to prepare design specifications for an advanced system capable of serving the geoplatform.

6. RATIONALE AND ANALYSIS:

The conventional N_2H_4 system was initially chosen because it provides adequate performance and is operationally proven. However, improved performance propulsion subsystems could lower the platform system operating costs by reducing the propellant resupply requirements and/or could increase the platform payload capabilities.

Table 4-8. Increased Performance RCS/Propulsion System, Contd

DEFINITION OF TECHNOLOGY REQUIREMENT

1. TECHNOLOGY REQUIREMENT (TITLE): Increased Performance RCS/Propulsion System — Page 2 of 3

7. TECHNOLOGY OPTIONS:

Three increased performance propulsion systems that have been demonstrated should be considered for geostationary platform applications, i.e.:

1. Thermally augmented N_2H_4 thrusters (Isp $\doteq$ 300 sec).
2. Pulsed plasma thrusters (Isp $\doteq$ 1500 sec).
3. Ion electric thrusters (Isp $\doteq$ 2000-4000 sec).
4. Magnetoplasma-dynamic thrusters.

One or more of the above options could be demonstrated on the experimental platform.

8. TECHNICAL PROBLEMS:

The four options all require increased electrical power input over the baseline N_2H_4 system. Option 1 requires 4000-6000 watts per pound of thrust. Options 2, 3 and 4 require 100,000 to 200,000 watts per pound of thrust. All options need further development for geoplatform application, especially to increase impulse capability, i.e., life.

9. POTENTIAL ALTERNATIVES:

A bipropellant system (Isp $\doteq$ 300 sec) could be applied, but further development of long-term oxidizer storage vessels and other components would be required. Also, a means to eliminate contamination products would be required.

10. PLANNED PROGRAMS OR UNPERTURBED TECHNOLOGY ADVANCEMENT:

Option 1 will be demonstrated on Intelsat V. Option 2 was demonstrated on LES-9 and is undergoing further development by USAF. Option 3 has been carried to 30-centimeter diameter by NASA, LeRC and development will continue if SEPS is approved as a FY 81 or 82 new start. Option 4 is now under development by JPL, AFRPL, and Princeton.

11. RELATED TECHNOLOGY REQUIREMENTS:

Power management and distribution.

Table 4-8. Increased Performance RCS/Propulsion Subsystem, Contd

DEFINITION OF TECHNOLOGY REQUIREMENT No.

1. TECHNOLOGY REQUIREMENT (TITLE): INCREASED PERFORMANCE RCS/PROPULSION SUBSYSTEM

12. TECHNOLOGY REQUIREMENTS SCHEDULE:

CALENDAR YEAR

SCHEDULE ITEM	79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	95	
TECHNOLOGY																		
OPTION 1																		
DESIGN		▬	▬															
PROTOTYPE GROUND TEST				▬	▬				▼ LAUNCH EXPERIMENTAL PLATFORM					▼ LAUNCH OPERATIONAL PLATFORM				
FUNDING LEVEL (In $1,000, 1980 dollars)		50	500	500	400													

13. USAGE SCHEDULE:

	79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	95	TOTAL
TECHNOLOGY NEED DATE					▼													
NUMBER OF LAUNCHES									1					1				

14. REFERENCES

15. LEVEL OF STATE OF THE ART:

1. Basic phenomena observed and reported
2. Theory formulated to describe phenomena
3. Theory tested by physical experiment or mathematical model
4. Pertinent functions or characteristic demonstrated, e.g., material, component
5. Component or breadboard tested in relevant environment in laboratory
6. Model tested in aircraft environment
7. Model tested in space environment
8. New capability derived from a much lesser operational model
9. Reliability upgrading of an operational model
10. Lifetime extension of an operational model

Table 4-9. Thermal Management

DEFINITION OF TECHNOLOGY REQUIREMENT

1. TECHNOLOGY REQUIREMENT (TITLE): Thermal Management Page 1 of 3

2. TECHNOLOGY CATEGORY: Platform Subsystems

3. OBJECTIVE/ADVANCEMENT REQUIRED: Systems analysis and technical approach to handling the thermal problem on a geostationary platform.

4. CURRENT STATE OF ART: Heat pipe/passive heat rejection subsystems on small satellites are well developed.

5. DESCRIPTION OF TECHNOLOGY:

Develop analytical and technical approach to management of thermal heat rejection on a high power platform that is characterized by the separate requirements of the mission packages attached to it, and the requirements of the platform itself.

6. RATIONALE AND ANALYSIS:

Desire for simplicity and reliability tends toward recommendation of passive heat rejection approaches. For a high power system like a geo-communications platform it is not clear that this approach can be continued. Some mixture of passive and active systems is indicated, particularly if centers of high energy concentration and rejection are predicted. A systems approach to this problem, based on analysis of sample configurations of platforms, can yield methodical and practical answers to questions of concentration, viewing, area, duty cycle, etc.

Table 4-9. Thermal Management, Contd

DEFINITION OF TECHNOLOGY REQUIREMENT

1. TECHNOLOGY REQUIREMENT (TITLE): Thermal Management Page 2 of 3

7. TECHNOLOGY OPTIONS:

Distributed versus concentrated heat rejection systems, and combinations of the two.

8. TECHNICAL PROBLEMS:

Reliable, lightweight devices for high Q heat rejection.

9. POTENTIAL ALTERNATIVES:

1. Heat pipes.
2. Fluid transfer.
3. Individual payload (integral) heat rejection systems.

10. PLANNED PROGRAMS OR UNPERTURBED TECHNOLOGY ADVANCEMENT:

LTV contracted to LeRC for thermal management of a space operations center. Methodology may be adaptable to geoplatform.

11. RELATED TECHNOLOGY REQUIREMENTS:

Table 4-9. Thermal Management, Contd

DEFINITION OF TECHNOLOGY REQUIREMENT	No.
1. TECHNOLOGY REQUIREMENT (TITLE): THERMAL MANAGEMENT	Page 3 of 3

12. TECHNOLOGY REQUIREMENTS SCHEDULE:

CALENDAR YEAR

SCHEDULE ITEM	79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	95		
TECHNOLOGY																			
SYSTEMS ANALYSIS																			
RECOMMENDED HARDWARE DEVELOPMENT																			
LAUNCH OPERATIONAL PLATFORM													▼						
FUNDING LEVEL (In $M, 1980 dollars)			0.15	0.25	0.3	0.4	1	1.5	1	0.5	0.5								

13. USAGE SCHEDULE:

	79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	95	TOTAL
TECHNOLOGY NEED DATE											▼							
NUMBER OF LAUNCHES													1					

14. REFERENCES

LeRC STUDY (VOUGHT CORP) "THERMAL CONTROL OF A LOW EARTH ORBIT OPERATIONS CENTER"

15. LEVEL OF STATE OF THE ART:

1. Basic phenomena observed and reported
2. Theory formulated to describe phenomena
3. Theory tested by physical experiment or mathematical model
4. Pertinent functions or characteristic demonstrated, e.g., material, component
5. Component or breadboard-tested in relevant environment in laboratory
6. Model tested in aircraft environment
7. Model tested in space environment
8. New capability derived from a much lesser operational model
9. Reliability upgrading of an operational model
10. Lifetime extension of an operational model

Table 4-10. Automated Remote Docking and Servicing

DEFINITION OF TECHNOLOGY REQUIREMENT

1. TECHNOLOGY REQUIREMENT (TITLE): Automated Remote Docking and Servicing — Page 1 of 3

2. TECHNOLOGY CATEGORY: Operations

3. OBJECTIVE/ADVANCEMENT REQUIRED: Develop capability for remote automated operations and servicing.

4. CURRENT STATE OF ART: No current U.S. capability.

5. DESCRIPTION OF TECHNOLOGY:

Remote automated docking and servicing requires advances in:

1. Automatic soft-docking and latching devices with integral service couplings.
2. Remote sensing and targeting.
3. Equipment exchange robotics.
4. Fluids replenishment subsystems.
5. Command and control software and subsystems.

6. RATIONALE AND ANALYSIS:

Large investments required for geoplatforms call for service life far beyond that of conventional satellites. Means of equipment replacement, consumables replenishment, and repair must be developed. Very large platforms built up of smaller transportable sections will require docking and latching devices to accomplish physical connection. Generally, service connections will be made simultaneously across the interconnects. The least costly way of accomplishing the objectives is believed to be through robotics technology.

Table 4-10. Automated Remote Docking and Servicing, Contd

DEFINITION OF TECHNOLOGY REQUIREMENT

1. TECHNOLOGY REQUIREMENT (TITLE): Automated Remote Docking and Servicing — Page 2 of 3

7. TECHNOLOGY OPTIONS:

1. Remote automated operations - ground controlled.
2. Remote automated operations - autonomous.
3. Manned assist at GEO - requires development of MOTV.

8. TECHNICAL PROBLEMS:

High technology throughout all aspects of problem.

9. POTENTIAL ALTERNATIVES:

Design for nonservice, requiring heavier spacecraft designed for greater redundancy and long life.

10. PLANNED PROGRAMS OR UNPERTURBED TECHNOLOGY ADVANCEMENT:

1. Teleoperator development.
2. RMS preprogrammed activity.
3. NASA (JSC) plans for orbital construction demonstration (Reference Table 4-1).

11. RELATED TECHNOLOGY REQUIREMENTS:

1. Secondary power source.
2. Space construction.

Table 4-10. Automated Remote Docking and Servicing, Contd

DEFINITION OF TECHNOLOGY REQUIREMENT	No.
1. TECHNOLOGY REQUIREMENT (TITLE): AUTOMATED REMOTE DOCKING AND SERVICING	Page 3 of 3

12. TECHNOLOGY REQUIREMENTS SCHEDULE:

CALENDAR YEAR

SCHEDULE ITEM	79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	95		
TECHNOLOGY																			
STUDY CONTENT DEFINITION																			
DEFINITION STUDIES (5)																			
HARDWARE DEVELOPMENT																			
OPERATIONAL FLIGHT																			
FUNDING LEVEL (In $M, 1980 dollars)			.100	.625	.625	1	4	6	6	3	2	2							

13. USAGE SCHEDULE:

	79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	TOTAL
TECHNOLOGY NEED DATE																	
NUMBER OF LAUNCHES													1				

14. REFERENCES

MARTIN – TELEOPERATOR DEVELOPMENT
COMSAT CORP – REMOTE SERVICING ASSESSMENT
MARTIN – INTEGRATED ORBITAL SERVICING STUDY FOR LOW COST PAYLOAD PROGRAMS (NAS8-30820)

15. LEVEL OF STATE OF THE ART:

1. Basic phenomena observed and reported
2. Theory formulated to describe phenomena
3. Theory tested by physical experiment or mathematical model
4. Pertinent functions or characteristic demonstrated, e.g., material, component
5. Component or breadboard-tested in relevant environment in laboratory
6. Model tested in aircraft environment
7. Model tested in space environment
8. New capability derived from a much lesser operational model
9. Reliability upgrading of an operational model
10. Lifetime extension of an operational model

Multibeam systems require dynamic interconnectivity between receiving and transmitting beams. The means to accomplish this is another key area of technology that requires development of both IF and baseband switch matrices. These switches must interconnect large numbers of high capacity communication channels in real time.

Operational implementation of customer premise service requires onboard processing and routing of traffic from a very large number of small earth terminals. Alternative approaches employ baseband switch matrices and/or baseband digital processors. On-board processing will have advantages in:

a. Isolation of uplinks and downlinks.

b. Error detection and control.

c. Flexible response to traffic demands.

d. Message routing by "order wire" or "packet" control.

Other technological areas under development by NASA include low-noise receivers and multimode TWT and solid-state amplifiers. Multimode amplifier operation is needed to implement downlink power control for rain fade compensation at frequencies above 10 GHz.

It should be noted that although the above technology development programs are aimed at implementation of a 30/20 GHz satellite communication system, most of the techniques are equally applicable to communication in lower frequency bands.

Development efforts are also pushed in the areas of intersatellite communication by microwave or optical links, and by bandwidth-efficient and power-efficient modulation and demodulation techniques.

Tables 4-11 through 4-20 summarize the communications technology initiatives derived during this study, the majority of which concide with the results of earlier studies.

Table 4-11 presents the high-speed, high-capacity satellite switch matrix technology, which may be considered the core of the geostationary platform communications development.

Tables 4-12 through 4-14 address advanced antenna technology, including phased array development, which is presently evolving along several lines as a result of military funding.

Table 4-15 discusses multibeam frequency reuse antenna feed assemblies, a technology requirement on a par with the switch matrix. MBFRA feed assemblies will be required on the platform to provide high frequency reuse, low interbeam coupling, beam scanning and beam reconfigurability, and beam-tracking capability.

(Continued on Page 4-73)

Table 4-11. High Speed, High Capacity, Satellite Switch Matrix

DEFINITION OF TECHNOLOGY REQUIREMENT

1. TECHNOLOGY REQUIREMENT (TITLE): High Speed, High Capacity, Satellite Switch Matrix — Page 1 of 3

2. TECHNOLOGY CATEGORY: Specialized communications/integration device.

3. OBJECTIVE/ADVANCEMENT REQUIRED: On-board message switching for multiple spot beam satellite communications.

4. CURRENT STATE OF ART: Laboratory models of 8 × 8 RF matrix switches have been developed.

5. DESCRIPTION OF TECHNOLOGY:

A multiple-beam communications satellite requires means for interconnecting receive and transmit beams in a manner that matches traffic demand. This interconnection requirement can be met by a suitably designed switch matrix. Uplink receivers are connected to the input ports of the switch matrix; downlink transmitters are connected to the output ports. Opening and closing of the cross-point switches is controlled by a processor that is programmed to operate in accordance with the observed traffic patterns. The switch must operate at speeds in excess of 1 MHz with nanosecond switching times.

6. RATIONALE AND ANALYSIS:

The current methods of satellite switching employ RF switches that are large and heavy and require substantial operating power. The switches themselves have high insertion loss and need matched transmission lines for network interconnection. To date only a 16 × 16 port RF switch matrix has been fabricated. A 100 × 100 port switch employing similar technology would weigh about 1000 kg and consume 100 watts.

A considerable reduction in weight and power requirements could be obtained by developing a baseband switch that utilizes LSI techniques. Since the trend in space communication system design is towards all digital operation with time division multiple access and on-board processing, a base-band matrix switch offers the most practical approach to meeting the requirement for high speed, high capacity, minimum weight and power, and overall system flexibility.

Table 4-11. High Speed, High Capacity Matrix Switch, Contd

DEFINITION OF TECHNOLOGY REQUIREMENT

1. TECHNOLOGY REQUIREMENT (TITLE): High Speed, High Capacity Matrix Switch — Page 2 of 3

7. TECHNOLOGY OPTIONS:

1. Develop large, heavy switching matrix from discrete space-qualified components.
2. Develop integrated switching matrix using LSI techniques and space qualify complete unit.

8. TECHNICAL PROBLEMS:

1. Reliability - the reliability of the switches and a means of effective redundancy must be established.
2. Isolation.
3. Insertion loss.
4. Switching time.
5. Size - the units tend to be large.
6. Weight - the units tend to be heavy.

9. POTENTIAL ALTERNATIVES:

None presently known.

10. PLANNED PROGRAMS OR UNPERTURBED TECHNOLOGY ADVANCEMENT:

NASA Lewis Research Center has provided parallel study contracts for the development of prototype RF and baseband matrix switches and baseband processors.

11. RELATED TECHNOLOGY REQUIREMENTS:

1. Develop techniques for production of space-qualified LSI devices that use high-speed diodes or dual gate FETs.
2. Develop distribution control unit (DCL) to control switching matrix.
3. Develop acquisition synchronizer unit to synchronize ground station with switch sequence.

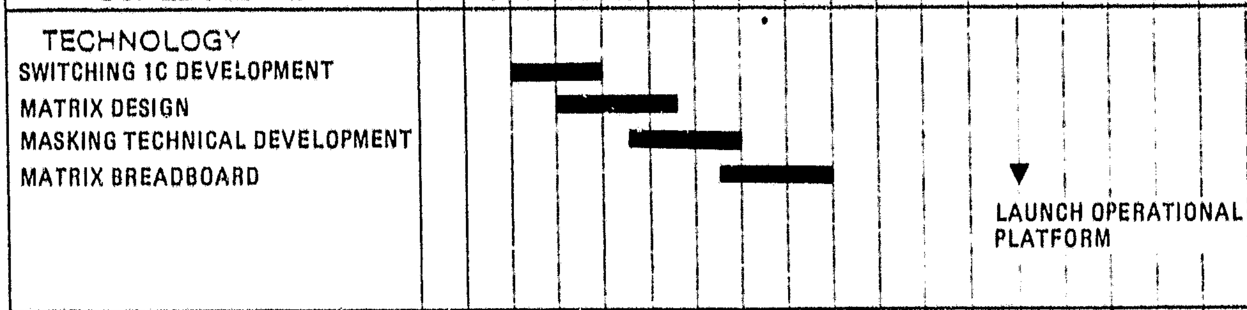
Table 4-11. High-Speed, High Capacity Matrix Switch, Contd

DEFINITION OF TECHNOLOGY REQUIREMENT	No.
1. TECHNOLOGY REQUIREMENT (TITLE): HIGH-SPEED, HIGH CAPACITY MATRIX SWITCH	Page 3 of 3

12. TECHNOLOGY REQUIREMENTS SCHEDULE:

CALENDAR YEAR

SCHEDULE ITEM	79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	95		



SCHEDULE ITEM	79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	95		
FUNDING LEVEL (In $M, 1980 dollars)			0.2	0.5	1	1.5	1.5	1	0.5										

13. USAGE SCHEDULE:

	79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	95	TOTAL
TECHNOLOGY NEED DATE														1				
NUMBER OF LAUNCHES																		

14. REFERENCES

(1) L. POLLACK, "TECHNOLOGIES FOR FUTURE SATELLITES," JOURNAL OF SPACECRAFT AND ROCKETS, VOL. 17, NO. 1, JANUARY-FEBRUARY 1980.

(2) X. ROZEC AND F. ASSAL, "MICROWAVE SWITCH MATRIX FOR COMMUNICATIONS SATELLITES," ICC PROCEEDINGS, VOL. III, JUNE 1976.

(3) Y. ITO, ET. AL., "ANALYSIS OF A SWITCH MATRIX FOR AN SS-TDMA SYSTEM," PROCEEDINGS OF IEEE, VOL. 65, NO. 3, MARCH 1977.

15. LEVEL OF STATE OF THE ART:

1. Basic phenomena observed and reported
2. Theory formulated to describe phenomena
3. Theory tested by physical experiment or mathematical model
4. Pertinent functions or characteristic demonstrated, e.g., material, component
5. Component or breadboard-tested in relevant environment in laboratory
6. Model tested in aircraft environment
7. Model tested in space environment
8. New capability derived from a much lesser operational model
9. Reliability upgrading of an operational model
10. Lifetime extension of an operational model

Table 4-12. Improvement of Deployable Antenna Reflector Surfaces

DEFINITION OF TECHNOLOGY REQUIREMENT

1. TECHNOLOGY REQUIREMENT (TITLE): Improvement of Deployable Antenna Reflector Surfaces

2. TECHNOLOGY CATEGORY: Antenna Systems

3. OBJECTIVE/ADVANCEMENT REQUIRED: 1) High surface accuracy, 2) randomized surface control point locations, 3) surface shaping, 4) larger size, 5) low intermodulation production introduction, and 6) improvement of packaging/deployment technique.

4. CURRENT STATE OF ART: Small deployable surfaces with regularly spaced control locations for operations below 10 GHz and limited scan angles.

5. DESCRIPTION OF TECHNOLOGY:

Future space antenna systems will have a larger number of beams and will be scanned further from the antenna axis to provide significant improvements in frequency reuse and also improve carrier to interference ratios. These future improvements in antenna performance place significant operational requirements on the antenna system (reflector, feed assembly, and active components). Particular emphasis should be placed on the development of both single and dual offset deployable reflectors.

A research program on phased arrays should also present some interesting results. When used to form multiple beams, it will probably be shown that as many BFNs as beams are needed. In a tradeoff versus an optical system, the phased array will probably prove to be more complex, weigh more, and require more power. (Continued on Page 4)

6. RATIONALE AND ANALYSIS:

Antennas applied in space at present have limited frequency reuse capability. Future communications traffic will require significant increases in the reuse of the frequency band, and therefore increased antenna dimensions, improved antenna design, and modified fabrication techniques will be necessary. Larger reflector apertures are necessary to obtain a larger number of narrower width beams. More accurate surfaces are required to allow greater scan angles with very low sidelobe levels. Reflector shaping will permit larger scan angles with reduced scan gain loss and beam broadening.

Table 4-12. Improvement of Deployable Antenna Reflector Surfaces, Contd

DEFINITION OF TECHNOLOGY REQUIREMENT

1. TECHNOLOGY REQUIREMENT (TITLE): Improvement of Deployable Antenna Reflector Surfaces — Page 2 of 3

7. TECHNOLOGY OPTIONS:

Large communications antennas with very high frequency reuse can be obtained from the following antenna types:

1. Large deployable single reflector systems.
2. Large deployed dual reflector systems.
3. Large deployed or assembled lens antennas.
4. Large deployable or assembled phased array or limited scan phased array antennas.

8. TECHNICAL PROBLEMS:

1. Antenna feed dimensions are very large for large scan angles.
2. Reflector surface control and surface accuracy.
3. Increased scan angles required.
4. Narrowed beamwidths.
5. Low sidelobe patterns for low interbeam coupling.
6. High polarization orthogonality.

9. POTENTIAL ALTERNATIVES:

1. Multibeam phased array antennas.
2. Bootlace and TEM lens antennas.

10. PLANNED PROGRAMS OR UNPERTURBED TECHNOLOGY ADVANCEMENT

11. RELATED TECHNOLOGY REQUIREMENTS:

1. Antenna feed and beam forming networks.
2. Frequency selective and polarization selective surfaces.
3. Graduated absorber lined reflectors.
4. Lens antennas.
5. Phased array antenna.

Table 4-12. Improvement of Deployable Antenna Reflector Surfaces, Contd

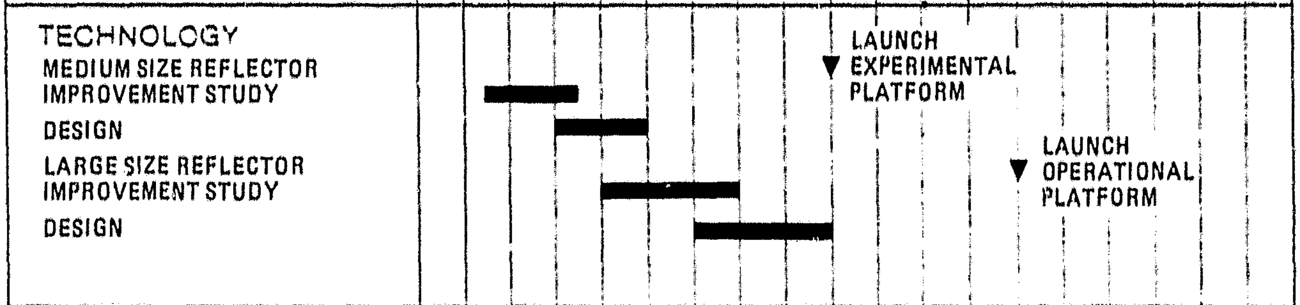
DEFINITION OF TECHNOLOGY REQUIREMENT — No.

1. TECHNOLOGY REQUIREMENT (TITLE): IMPROVEMENT OF DEPLOYABLE ANTENNA REFLECTOR SURFACES — Page 3 of 4

12. TECHNOLOGY REQUIREMENTS SCHEDULE:

CALENDAR YEAR

SCHEDULE ITEM	79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	95



	79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	95
FUNDING LEVEL (In $1,000, 1980 dollars)		50	50	100	450	150	300	300	100								

13. USAGE SCHEDULE:

	79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	TOTAL
TECHNOLOGY NEED DATE					▼				▼								
NUMBER OF LAUNCHES									1					1			

14. REFERENCES

15. LEVEL OF STATE OF THE ART:

1. Basic phenomena observed and reported
2. Theory formulated to describe phenomena
3. Theory tested by physical experiment or mathematical model
4. Pertinent functions or characteristic demonstrated, e.g., material, component
5. Component or breadboard-tested in relevant environment in laboratory
6. Model tested in aircraft environment
7. Model tested in space environment
8. New capability derived from a much lesser operational model
9. Reliability upgrading of an operational model
10. Lifetime extension of an operational model

5. DESCRIPTION OF TECHNOLOGY: (Continued)

Six areas of reflector system improvements are described herein that are required to obtain future performance needs of spacecraft and platform communications antennas. The first item provides a significant improvement of surface accuracy of large deployable surfaces. The surface accuracy of both mesh covered and solid surface reflectors need to be improved for enhanced frequency of operation (50 GHz). These improved surfaces must be maintained for earth based antenna testing and for their intended space application.

The second item provides a randomized positioning of the mesh reflector contour control points and is similar to the first item. Uniform lattice spacing of the control points increases the sidelobe levels in particular regions of space to unacceptable levels. These sidelobes are named grating lobes. Randomization techniques will spread the grating lobe energy over a large region with a corresponding reduction of the sidelobe levels.

Item 3, surface shaping of the single and dual reflector system studies, is required to provide improved scan angle and scanned beam performance. Scan angle limitations and dual reflector systems are generally small (near 5 degrees). However, the scan gain and scanned pattern characteristics of the dual reflector antennas are superior to single reflector antenna systems for scan angles less than the limiting values. Surface shaping, including Schwarzschild shaping, is necessary to obtain earth coverage with a single antenna while maintaining the improved scan gain, polarization, and sidelobe performance characteristic of the dual reflector system. Offset single reflector antenna systems presently scan to about 12 beamwidths with degraded gain and sidelobe characteristics. Single reflector shaping is also required to improve scanned beam parameters since these antennas will still be used in applications with larger beamwidths and with applications where small feed dimensions are required.

The enlargement of reflectors having highly accurate shaped surfaces is the fourth item. Increased reflector dimensions permit narrower beamwidths, decreased sidelobe levels, and increased reuse. Both solid surface and mesh surfaces need to be enlarged and advanced techniques to support the surface need development to provide future requirements for large antennas.

Intermodulation products introduced by the reflector surface have to be reduced to and maintained at very low levels in large deployable reflector surfaces. Methods of preventing corroded or partially conductive interfaces between the reflective surface components and the support structure require evaluation and testing prior to space application.

The last item concerning reflector design is a development program to improve large antenna packaging and deployment techniques. Techniques to both deploy the antenna upon arrival at low earth orbit and to retract the antenna before transfer to GEO are required for many antenna types.

Table 4-13. Phased Array Antennas

DEFINITION OF TECHNOLOGY REQUIREMENT

1. TECHNOLOGY REQUIREMENT (TITLE): Phased Array Antennas Page 1 of 3

2. TECHNOLOGY CATEGORY: Antenna Systems

3. OBJECTIVE/ADVANCEMENT REQUIRED: Develop multibeam phased arrays for space communications applications that are space shuttle compatible.

4. CURRENT STATE OF ART: Large ground based and small space based phased arrays with one to several simultaneous beams.

5. DESCRIPTION OF TECHNOLOGY:

Develop the phased array, which has inherent high scan angle capability, for space communications applications. Arrays need broad bandwidth capability and the capability for operation with a large number of independent beams. Broadband elements and polarizers displaying very low interelement coupling are required. Corporate feed assemblies with included phase shifters have to be simplified to reduce production costs and allow interleaving of the separate beams of the multibeam system, since each beam input is interconnected through a corporate feed to each element in the array. Methods to reduce weight of the communications phased array are needed, and deployability of the large phased array (20,000 elements) is necessary.

6. RATIONALE AND ANALYSIS:

Only small phased arrays with limited numbers of beams are presently utilized in space. Large ground based phased arrays have demonstrated high scan capability with negligible scan gain loss at 9 degree scan angles and a moderate number of independent beams. Space communication systems require a large number of independent beams whose low sidelobe and low interbeam coupling parameters are preserved for earth edge scanned beams. Thus, beam characteristics near the boresight of the phased array will be maintained over the complete earth coverage. Additionally, antenna stabiliziation requirements are significantly reduced for the phased array antenna since electrical correction of beam pointing can be incorporated.

Table 4-13. Phased Array Antennas, Contd

DEFINITION OF TECHNOLOGY REQUIREMENT

1. TECHNOLOGY REQUIREMENT (TITLE): Phased Array Antennas Page 2 of 3

7. TECHNOLOGY OPTIONS:

Large communications antennas with very high frequency reuse can be obtained from the following antenna types.

1. Large deployable single reflector antenna with multibeam feed.
2. Large deployable dual reflector antenna with multibeam feed.
3. Large deployed or assembled lens antennas with multibeam feeds.
4. Large deployable or assembled component phased array or limited scan phased array antennas.

8. TECHNICAL PROBLEMS:

A very large number of elements are required in the phased array and therefore a low cost element design is mandatory. Microwave integrated circuit multibeam corporate feed structures or butler matrices are also required. The array must be sectioned for stowage aboard the Shuttle Orbiter and then assembled in space.

9. POTENTIAL ALTERNATIVES:

Multibeam bootlace or TEM lens antennas. Large single and dual reflector antennas with multibeam feed assemblies.

10. PLANNED PROGRAMS OR UNPERTURBED TECHNOLOGY ADVANCEMENT.

General Dynamics pursuing single layer lens development for DARPA.

11. RELATED TECHNOLOGY REQUIREMENTS.

1. Large aperture antenna design.
2. Antenna feed and beam forming networks.
3. Frequency selective and polarization selective surfaces.
4. Graduated absorber lined reflectors.
5. Bootlace and TEM lens antennas.

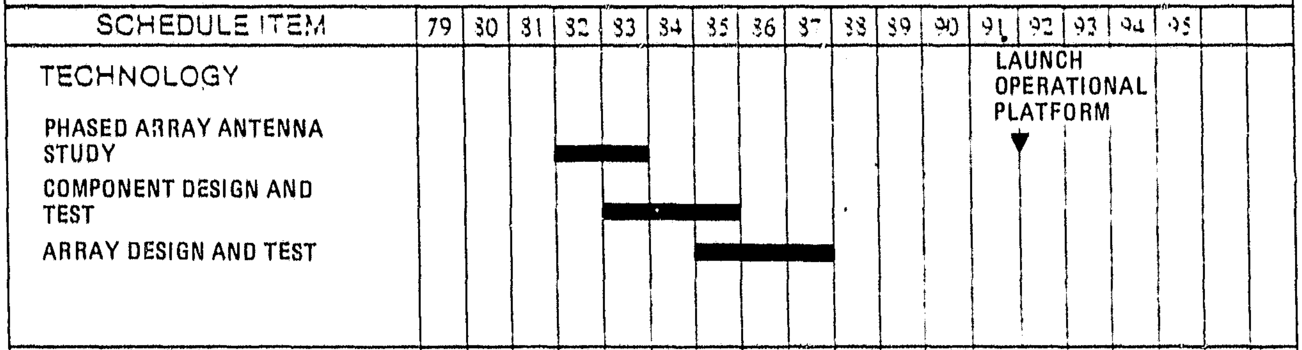
Table 4-13. Phased Array Antennas, Contd

DEFINITION OF TECHNOLOGY REQUIREMENT No.

1. TECHNOLOGY REQUIREMENT (TITLE): PHASED ARRAY ANTENNAS Page 3 of 3

12. TECHNOLOGY REQUIREMENTS SCHEDULE:

CALENDAR YEAR



SCHEDULE ITEM	79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	95		
FUNDING LEVEL (In $1,000, 1980 dollars)				100	200	500	1K	2K	1K										

13. USAGE SCHEDULE:

																	TOTAL
TECHNOLOGY NEED DATE									▼								
NUMBER OF LAUNCHES														1			

14. REFERENCES

15. LEVEL OF STATE OF THE ART:

1. Basic phenomena observed and reported
2. Theory formulated to describe phenomena
3. Theory tested by physical experiment or mathematical model
4. Pertinent functions or characteristic demonstrated, e.g., material, component
5. Component or breadboard tested in relevant environment in laboratory
6. Model tested in aircraft environment
7. Model tested in space environment
8. New capability derived from a much lesser operational model
9. Reliability upgrading of an operational model
10. Lifetime extension of an operational model

Table 4-14. Lens Antennas

DEFINITION OF TECHNOLOGY REQUIREMENT

1. TECHNOLOGY REQUIREMENT (TITLE): Lens Antennas Page 1 of 3

2. TECHNOLOGY CATEGORY: Antenna System

3. OBJECTIVE/ADVANCEMENT REQUIRED: Develop bootlace and TEM lenses for deployable space applications.

4. CURRENT STATE OF ART: Ground based testing programs to evaluate large lens antennas.

5. DESCRIPTION OF TECHNOLOGY:

Three principal types of antennas can be used for space communications applications: reflector, phased array, and lenses. The lens antenna has good scan capability. Further development is required to evaluate future space applications for the lens antenna; possbible hybrid lens applications require further study. A capability to deploy or assemble large lenses in space is required. The cost and weight have to be reduced, and the instantaneous lens bandwidth must be increased sufficiently to provide coverage of the transmit or receive bandwidths of the communications band before application of the lens antennas will be made for space communications.

6. RATIONALE AND ANALYSIS:

Lens antennas have very good scan angle capability, being nearly as good as phased array antennas and superior to reflector antennas. A number of improvements are required before the lens antenna will provide broadband, lightweight, and deployable operation. The lens antenna has much lower initial cost than the phased array and provides similar graceful degradation characteristics. Multibeam capability is provided by multiple feeds similar to reflector antennas in complexity and considerably more simple than the phased array, multiple beam antenna.

Table 4-14. Lens Antennas, Contd

DEFINITION OF TECHNOLOGY REQUIREMENT

1. TECHNOLOGY REQUIREMENT (TITLE): Lens Antennas Page 2 of 3

7. TECHNOLOGY OPTIONS:

Large communications antennas with very high frequency reuse can be obtained from the following antenna types:

1. Large deployable single reflector systems.
2. Large deployed dual reflector systems.
3. Large deployed or assembled lens antennas.
4. Large deployable or assembled phase array or limited scan phased array antennas.

8. TECHNICAL PROBLEMS: Lens antennas are made up of a large number of separate elements on both sides of the lens. The elements on one face are interconnected through TEM lines or waveguide to the corresponding element on the other lens face. The major technical problems for the lens antennas are associated with the individual elements. Interrelement coupling causes frequency dependent mismatches between the elements and the transmission lines and a corresponding reduced antenna bandwidth.

9. POTENTIAL ALTERNATIVES:

1. Multibeam phased array antennas.
2. Single and dual reflector antennas.

10. PLANNED PROGRAMS OR UNPERTURBED TECHNOLOGY ADVANCEMENT:

1. Space-based radar, DARPA.
2. Passive/active lens, DARPA.
3. Large deployable antenna, NASA/DARPA.

11. RELATED TECHNOLOGY REQUIREMENTS:

1. Antenna feed and beam forming networks.
2. Frequency selective and polarization selective surfaces.
3. Graduated absorber lined reflectors.
4. Single and dual reflector antennas.
5. Phased array antennas.

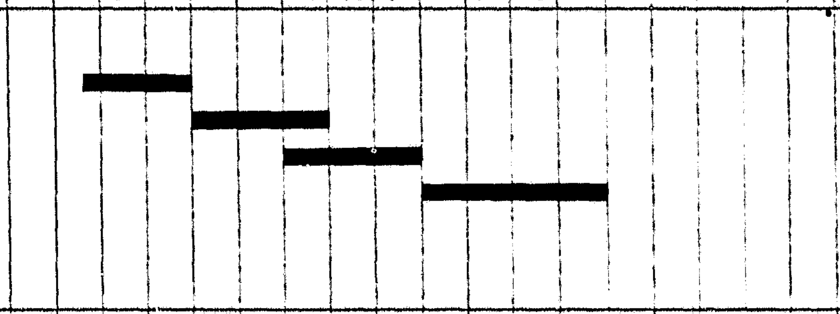
Table 4-14. Lens Antenna, Contd

DEFINITION OF TECHNOLOGY REQUIREMENT	No.
1. TECHNOLOGY REQUIREMENT (TITLE): LENS ANTENNA	Page 3 of 3

12. TECHNOLOGY REQUIREMENTS SCHEDULE:

CALENDAR YEAR

SCHEDULE ITEM	79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	95	
TECHNOLOGY																		
LENS TECHNOLOGY STUDY																		
LENS DESIGN																		
FABRICATION AND GROUND TEST																		
PHASE C, D																		
FUNDING LEVEL (In $1,000, 1980 dollars)		50	100	100	200	200	300	400	200									



13. USAGE SCHEDULE:

	79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	TOTAL
TECHNOLOGY NEED DATE													▼				
NUMBER OF LAUNCHES													1				

14. REFERENCES

15. LEVEL OF STATE OF THE ART:

1. Basic phenomena observed and reported
2. Theory formulated to describe phenomena
3. Theory tested by physical experiment or mathematical model
4. Pertinent functions or characteristic demonstrated, e.g., material, component
5. Component or breadboard-tested in relevant environment in laboratory
6. Model tested in aircraft environment
7. Model tested in space environment
8. New capability derived from a much lesser operational model
9. Reliability upgrading of an operational model
10. Lifetime extension of an operational model

Table 4-15. MBFRA Feed Assemblies

DEFINITION OF TECHNOLOGY REQUIREMENT

1. TECHNOLOGY REQUIREMENT (TITLE): Multibeam Frequency Reuse Antenna (MBFRA) Feed Assemblies

2. TECHNOLOGY CATEGORY: Antenna Systems (Communications Systems)

3. OBJECTIVE/ADVANCEMENT REQUIRED: Develop feed systems for very high frequency reuse, low interbeam coupling, beam scanning and beam reconfigurability, and limited beam-tracking capability.

4. CURRENT STATE OF ART: Limited frequency reuse on Intelsat V, reconfigurability and multielement DTU feed assembly breadboard demonstration.

5. DESCRIPTION OF TECHNOLOGY:

The technology development of the antenna feed assemblies is aimed at increasing reuse of the lower frequency communications bands, expanding into the 30/20 GHz and higher frequency bands, pointing antenna beams by scanning feeds electronically, increasing feed bandwidth for multiapplication feed capability, and reconfiguring beam to accommodate variations in orbital slot assignment and traffic volume.

Each component of the feed element needs improvement. Aperture design improvements are required in aperture distributions, more accurate matching, reduced interelement matching, and incorporation of interelement coupling cancellation circuits, increased frequency of operation, and increased bandwidths. The polarizer/transition requires improvements

(Continued on page 4)

6. RATIONALE AND ANALYSIS:

Feed systems for the 1990s time frame HVT and DTU will require greater capability for reconfigurability, improved beam shaping, and low interbeam coupling. The feed assemblies are large for large antennas and large scan angles; these assemblies will require deployability for HVT and space assembly techniques for the DTU (CPS) application. High-volume trunking will provide closely spaced cities in the NE CONUS with high interbeam isolation (both amplitude and polarization). Very high frequency reuse will be required in the DTU antenna systems. Beam reconfigurability will be required to provide for orbital slot changes and to accommodate traffic fluctuations and variation in compatibility introduced by the traffic changes. The advanced large antenna systems have beamwidths considerably smaller than the nominal platform stability. Thus, accurate beam tracking and pointing will be required in a flexible and reconfigurable feed system.

Table 4-15. MBFRA Feed Assemblies, Contd

DEFINITION OF TECHNOLOGY REQUIREMENT

1. TECHNOLOGY REQUIREMENT (TITLE): MBFRA Feed Assemblies

7. TECHNOLOGY OPTIONS:

Large communications antennas with very high frequency reuse can be obtained from the following antenna types:

1. Large deployable single reflector systems.
2. Large deployable dual reflector systems.
3. Large deployed or assembled lens antennas.
4. Large deployable or assembled phased array or limited scan phased array.

All but the last entry require an elaborate feed system to provide operational meters.

8. TECHNICAL PROBLEMS:

The spectrum and orbital slot resources are nearly saturated. Further traffic can be accommodated by greater frequency reuse and use of higher frequency allocations. The feed problems associated with higher reuse and higher frequencies are delineated in Section 5.

9. POTENTIAL ALTERNATIVES:

1. Multibeam phased arrays at lower frequencies.
2. Use larger number of antennas with simpler feed assemblies.
3. Not combine transmit and receive feed assemblies in a common reflector.

10. PLANNED PROGRAMS OR UNPERTURBED TECHNOLOGY ADVANCEMENT:

A considerable IRAD and CRAD development effort is underway in many facilities. Additional CRAD is required to tailor feed development at the higher frequencies and greater reconfigurability (30/20 GHz-NASA Lewis) (55 meter at 3 GHz reflector NASA-Lewis).

11. RELATED TECHNOLOGY REQUIREMENTS:

1. Single and dual reflector antennas.
2. Lens antennas.
3. Frequency selective and polarization selective surfaces.
4. Graduate absorber lined reflectors.

Table 4-15. MBFRA Feed Assemblies, Contd

DEFINITION OF TECHNOLOGY REQUIREMENT	No.
1. TECHNOLOGY REQUIREMENT (TITLE): MBFRA FEED ASSEMBLIES	Page 3 of 4

12. TECHNOLOGY REQUIREMENTS SCHEDULE:

CALENDAR YEAR

SCHEDULE ITEM	79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	95	
TECHNOLOGY																		
FEED DESIGN FOR IMPROVED REUSE AND LIMITED RECONFIGURABILITY		█	█	█	█													
HIGH PERFORMANCE FEED DESIGN INCLUDING BEAM POINTING						█	█	█	█									
PHASE CD										█	█	█	█					
FUNDING LEVEL (In $1,000, 1980 dollars)		100	100	200	200	300	300	300	300	300	400	1K	2K					

13. USAGE SCHEDULE:

	79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	TOTAL
TECHNOLOGY NEED DATE									▼				▼			
NUMBER OF LAUNCHES									1				1			

14. REFERENCES

15. LEVEL OF STATE OF THE ART:

1. Basic phenomena observed and reported
2. Theory formulated to describe phenomena
3. Theory tested by physical experiment or mathematical model
4. Pertinent functions or characteristic demonstrated, e.g., material, component
5. Component or breadboard-tested in relevant environment in laboratory
6. Model tested in aircraft environment
7. Model tested in space environment
8. New capability derived from a much lesser operational model
9. Reliability upgrading of an operational model
10. Lifetime extension of an operational model

in increased bandwidth, reduction in cross polarization levels, and smaller dimensions. The orthomode junction requires greater bandwidth, improved cross polarization isolation, and improved impedance matching. Desired bandwidths for future systems are in excess of 50 percent to permit receive/transmit functions of each communications system to be colocated. The colocated feed allows the use of a single common reflector, a common beam forming network and feed elements, common beam reconfigurability, and beam pointing or scanning capability in the antenna system.

The beam forming network for the communication antenna feed will be a highly complex device for the high frequency, high reuse, multifunction future applications. Significant improvements are required for the microwave integrated circuit (MIC) structure. A very large number of beams (in excess of 100 transmit and receive) with each beam exciting between six and nine feed elements arrayed in a cluster requires a large number of couplers, hybrids, variable power dividers, and variable phase shifter components to be assembled into the MIC beam forming network. Effort will be directed to decreasing insertion loss, increasing the operating frequencies and/or bandwidths, providing flexible computer control of the BFN, and providing a monopulse type tracking capability. The output of the monopulse tracking system is used to orient the receive and transmit beams through control of the BFN. The BFN justifies a considerable CRAD and IRAD expenditure as this single component is probably the limiting factor in frequency reuse and upper frequency of operation of the antenna system.

As a communications beam is scanned away from the antenna boresight, two degradations occur: the sidelobe levels increase and the cross polarization levels increase. To correct the amplitude and polarization errors of the scanned beam additional elements are excited near the main beam feed element. The phasing and amplitude weighting of the adjacent elements to obtain 35 dB sidelobe levels and 35 dB cross polarization for the highly scanned beams require additional study.

The feed assemblies are very large when large coverage angles are necessary with the large reflector antennas. As an example, the feed system for the 60 meter reflector designed for HVT CONUS coverage at C-band has a feed assembly dimension approximately 7 by 14 meters. An IRAD and CRAD effort is necessary to provide either space assembly techniques or space deployment techniques for these large feed assemblies. When dual reflector antenna systems are used to obtain their superior scan angle capability, the feed assemblies become larger and more difficult to package within the Space Shuttle Orbiter.

Table 4-16. Interplatform Links (IPLs)

DEFINITION OF TECHNOLOGY REQUIREMENT

1. TECHNOLOGY REQUIREMENT (TITLE): Interplatform Links (IPLs) Page 1 of 5

2. TECHNOLOGY CATEGORY: Communications

3. OBJECTIVE/ADVANCEMENT REQUIRED: Improved data capacity and improved control and pointing capability to eliminate data dropout of high data rate IPL.

4. CURRENT STATE OF ART: LES-8 and LES-9 IPL links with fairly broad beams and low data rate requirements.

5. DESCRIPTION OF TECHNOLOGY:

The areas requiring development/investigation are as follows:

1. Interplatform and intraplatform relative stationkeeping and the ability to track/point the antennas with very narrow bandwidths.
2. What missions will require interconnections?
3. What/how many frequencies should be assigned to the intraplatform link?
4. Data bandwidths requirements of the IPL.
5. Evaluation of laser capabilities for IPL applications.

6. RATIONALE AND ANALYSIS:

Mission XI, interplatform links, is actually a two-part problem. If one assumes that the platform is a single rigid structure, all frequency diversity interconnections can be effectively "hard wired" into place and we need concern ourselves only with the 32/25 GHz link between platforms in different orbital positions. (Alternative #4).

Alternatively, if the "platform" is a series of modules flying in some formation as to represent a "cluster" to the earth terminals, then one is faced with a dual problem. One must use an intraplatform (module-to-module) link to interconnect the different missions and/or frequency diversity approaches and use an interplatform link between clusters. This alternate or cluster approach has all the problems of the rigid platform vis-a-vis interplatform communications, compounded by the interplatform links, which are highly dependent upon the flight formation employed. (Alternative #1). (Continued on Page 4)

Table 4-16. Interplatform Links (IPLs), Contd

DEFINITION OF TECHNOLOGY REQUIREMENT

1. TECHNOLOGY REQUIREMENT (TITLE): Interplatform Links (IPLs)

7. TECHNOLOGY OPTIONS:

1. Use existing 32/25 GHz system with its limitation on data rates.
2. User higher frequency communication channel with larger available bandwidths.
3. Use optical communications link between system with an RF positioning link for near-in pointing of the optical system.

8. TECHNICAL PROBLEMS:

Using 10 percent of the total data rate of each platform of the interhemisphere pair as the IPL data bandwidth leads to very large bandwidth requirements. Multiple bands or optics systems will be required to accommodate these large bandwidths.

9. POTENTIAL ALTERNATIVES:

Use multiple satellite to ground links to provide interplatform links. This solution is very wasteful of spectrum resources.

10. PLANNED PROGRAMS OR UNPERTURBED TECHNOLOGY ADVANCEMENT:

11. RELATED TECHNOLOGY REQUIREMENTS:

Table 4-16. Interplatform Links (IPLs), Contd

DEFINITION OF TECHNOLOGY REQUIREMENT	No.
1. TECHNOLOGY REQUIREMENT (TITLE): INTERPLATFORM LINKS (IPLs)	Page 3 of 5

12. TECHNOLOGY REQUIREMENTS SCHEDULE:

CALENDAR YEAR

SCHEDULE ITEM	79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	95		
TECHNOLOGY																			
IPL-RF DESIGN																			
IPL-OPTICAL																			
FUNDING LEVEL (In $1,000, 1980 dollars)			50	50	50		100	100	100										

13. USAGE SCHEDULE:

TECHNOLOGY NEED DATE																	TOTAL
NUMBER OF LAUNCHES									1				1				

14. REFERENCES

15. LEVEL OF STATE OF THE ART:

1. Basic phenomena observed and reported
2. Theory formulated to describe phenomena
3. Theory tested by physical experiment or mathematical model
4. Pertinent functions or characteristic demonstrated, e.g., material, component
5. Component or breadboard-tested in relevant environment in laboratory
6. Model tested in aircraft environment
7. Model tested in space environment
8. New capability derived from a much lesser operational model
9. Reliability upgrading of an operational model
10. Lifetime extension of an operational model

Table 4-16. Interplatform Links (IPLs), Contd

5. DESCRIPTION OF TECHNOLOGY:

The resolution of the separate studies described in Section 5 above will determine the shape of the interplatform/intraplatform system and geometry.

Once the system geometry is established, the effect on the traffic handled by the IPL must be considered. J. H. Deal* indicated that SS-TDMA operation, for example, faces the following impairments:

1. Translation oscillator frequency stability in both platforms.
2. Doppler frequency offset due to relative satellite/platform motion.
3. Clock timing instabilities.
4. TDMA frame and burst synchronization.

Deal concludes that the above problems can be overcome with a "slaved" network approach which requires a special reference station and satellite equipment for control of the slaved SS-TDMA switch timing (see Figure). Similarly, FDM/FM has its related problems as do all other forms of data/analog transmission, all of which require further investigation.

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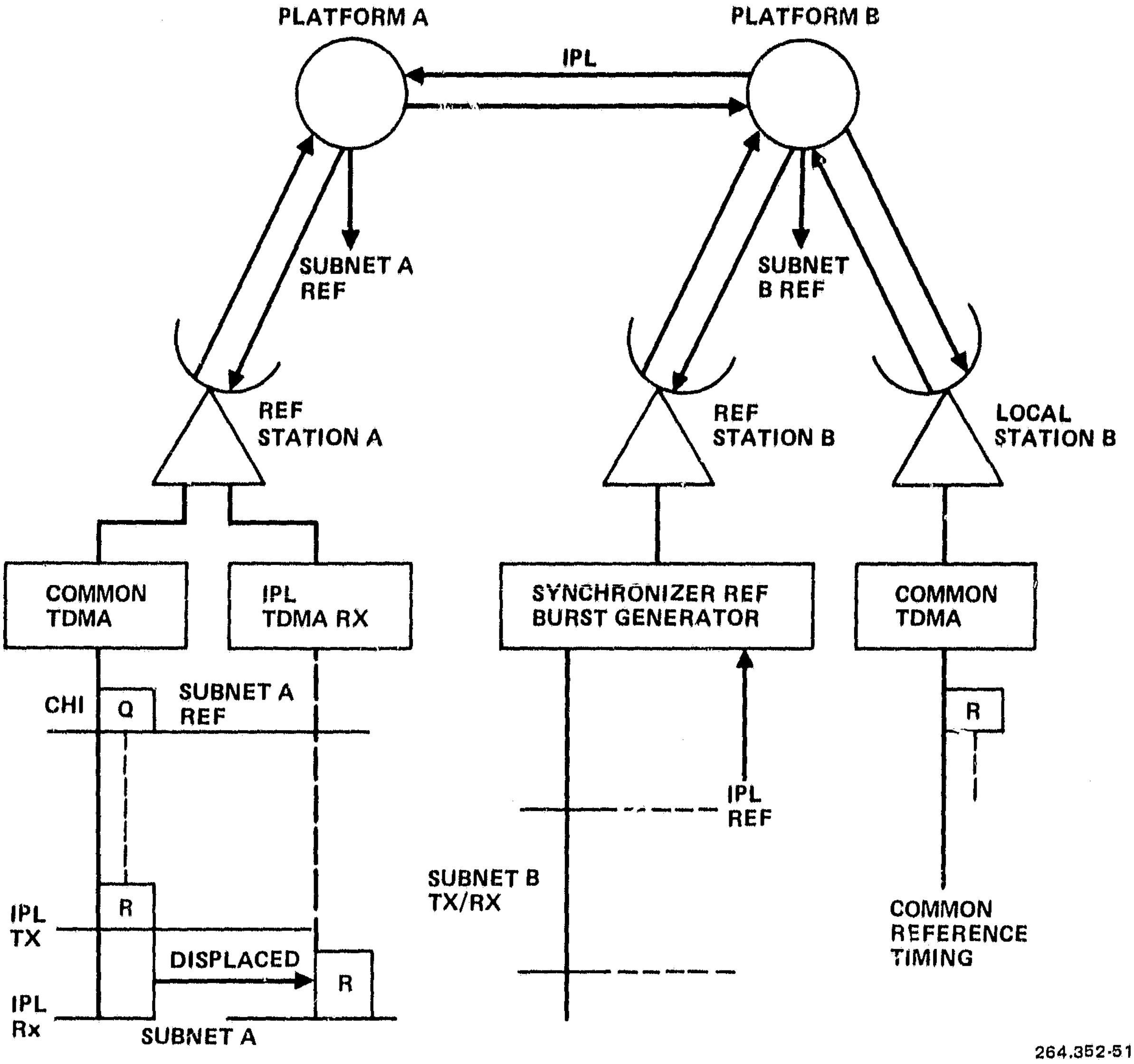
*Deal, J. H., "Digital Transmission Involving Intersatellite Links," International Conference on Digital Satellite Communications-4th, Montreal, Canada, October 1978.

Table 4-16. Interplatform Links (IPLs), Contd



SS - TDMA Slaved Subnetwork for IPL

Table 4-17. Intraconstellation Links (ICLs)

DEFINITION OF TECHNOLOGY REQUIREMENT

1. TECHNOLOGY REQUIREMENT (TITLE): Intraconstellation Links (ICLs) Page 1 of 4

2. TECHNOLOGY CATEGORY: Communications

3. OBJECTIVE/ADVANCEMENT REQUIRED: Significantly improved data capacity and improved higher dynamics tracking capability.

4. CURRENT STATE OF ART: LES-8 and LES-9 IPL links with fairly broad beams and low data rates.

5. DESCRIPTION OF TECHNOLOGY:

The areas requiring development/investigation are the following:

1. Provide basis for relative stationkeeping functions including:
   a. TDMA references and provisions for variable offsets in these references. Changes in TDMA references are introduced by variations in relative stationkeeping.
   b. Feasibility evaluation of splitting one TDMA payload among separate members of a constellation and maintaining synchronization.
   c. Evaluation of ranging measurement capability and determination of relative time references between platforms.

(Continued on Page 4)

6. RATIONALE AND ANALYSIS:

The ICL, although similar to the IPL, has distinctly different problems that require resolutions before their application to any type of constellation occurs. The data rates associated with the ICL are higher than for IPL and the comments applying to data rates in Table 4-16 apply to a greater extent in the ICL. If a central switch is utilized (particularly in the "string-of-perarls" configuration), the data rates configured are very high. IPL antenna beams in the cartwheel configuration from one platform can directly strike other constellation members as they revolve. he resulting EMI from direct illumination must be defeated by a complex IPL handover capability. Very accurate tracking acquisition must be obtained before handover, particularly at optical frequencies. When a master switch or a distributed switch is used, significant TDMA terminal synchronization and payload function handover problems may occur with changes in member stationkeeping dynamics and constellation variations.

Table 4-17. Intraconstellation Links (ICLs), Contd

DEFINITION OF TECHNOLOGY REQUIREMENT
1. TECHNOLOGY REQUIREMENT (TITLE): Intraconstellation Links (ICLs) Page 2 of 4
7. TECHNOLOGY OPTIONS: 1. Use existing 32/25 GHz system with its limitation of data rates. 2. Use higher frequency communication channel with larger available bandwidths. 3. Use optical communications links between systems with an RF positioning link for near-in pointing of the optical system.
8. TECHNICAL PROBLEMS: Using 25% of the total data rate of each platform in the constellation as the ICL data bandwidth leads to very large bandwidth requirements. Multiple bands or optic systems will be required to accommodate these large bandwidths.
9. POTENTIAL ALTERNATIVES:
10. PLANNED PROGRAMS OR UNPERTURBED TECHNOLOGY ADVANCEMENT.
11. RELATED TECHNOLOGY REQUIREMENTS:

Table 4-17. Intraconstellation Links (ICLs), Contd

DEFINITION OF TECHNOLOGY REQUIREMENT	No.
1. TECHNOLOGY REQUIREMENT (TITLE): INTRACONSTELLATION LINKS (ICLs)	Page 3 of 4

12. TECHNOLOGY REQUIREMENTS SCHEDULE:

CALENDAR YEAR

SCHEDULE ITEM	79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	95		
TECHNOLOGY																			
ICL-RF DESIGN			▬	▬	▬														
ICL-OPTICAL							▬	▬	▬										
FUNDING LEVEL (In $1,000, 1980 dollars)			50	50	50		100	100	100										

13. USAGE SCHEDULE:

TECHNOLOGY NEED DATE									▼				▼				TOTAL
NUMBER OF LAUNCHES									1				1				

14. REFERENCES

15. LEVEL OF STATE OF THE ART:

1. Basic phenomena observed and reported
2. Theory formulated to describe phenomena
3. Theory tested by physical experiment or mathematical model
4. Pertinent functions or characteristic demonstrated, e.g., material, component
5. Component or breadboard-tested in relevant environment in laboratory
6. Model tested in aircraft environment
7. Model tested in space environment
8. New capability derived from a much lesser operational model
9. Reliability upgrading of an operational model
10. Lifetime extension of an operational model

Table 4-17. Intraconstellation Links (ICLs), Contd

5. DESCRIPTION OF TECHNOLOGY: (Continued)

2. Evaluation of bandwidth requirements of the ICL for variations in constellation configuraiton.

3. Determine if a single constallation member should provide centalized switching capability for the interpayload links or if a distributed switching system with greater control complexity and more diffuse data transfer requirements should be used.

Table 4-18. Electromagnetic Compatibility/Interference

DEFINITION OF TECHNOLOGY REQUIREMENT
1. TECHNOLOGY REQUIREMENT (TITLE): Electromagnetic Compatibility/Interference — Page 1 of 4
2. TECHNOLOGY CATEGORY: Platform Technology
3. OBJECTIVE/ADVANCEMENT REQUIRED: Methods for the elimination of interference and design techniques to incorporate these methods of interference elimination are required for the GEO platform.
4. CURRENT STATE OF ART: Methods available for satellite with limited frequency reuse and limited number of radiating payloads.
5. DESCRIPTION OF TECHNOLOGY: Methods for the elimination of interference and design techniques to incorporate these methods of interference elimination are required for the geostationary platform. Three major classifications of interference occur in the platform: interplatform, interpayload, and intrapayload. The first class of interference is insignificant between platforms separated by orbital slot separations in the geostationary arc, but is very important for closely spaced satellites in close formation. Individual satellites arrayed in a time varying constellation introduce interference from sidelobe illumination and for some formation configurations mainlobe illumination of one satellite by other members of the constellation. The interference introduced by each of the other satellite constellation members is (Continued on Page 4)
6. RATIONALE AND ANALYSIS: Electromagnetic interference control methods have been worked out for satellites with limited frequency reuse and a limited number of radiating payloads. Future satellite constellations or platforms will have a significant increase in both frequency reuse capability and in the number of radiating payloads that can interfere with one another and within themselves. The possible sources of interference in a system as complex as the GEO platform are legion and each source must be considered separately. The interference coupling medias in the complex platform warrants investigation so that design drivers will be devised to present required performance in the final system.

Table 4-18. Electromagnetic Compatibility/Interference, Contd

DEFINITION OF TECHNOLOGY REQUIREMENT
1. TECHNOLOGY REQUIREMENT (TITLE): Electromagnetic Compatibility/Interference — Page 2 of 4
7. TECHNOLOGY OPTIONS: The spacecraft or platform must have the capability of operating with very low coupling between separate systems and within systems. The options are limited to platform design options. The primary options are formation flying of separate satellites and a single platform located within an orbital slot. The second option is the most tractable for EMC analysis and ground based testing.
8. TECHNICAL PROBLEMS: Provide sufficient isolation between separate channels of the multichanneled platform for required performance.
9. POTENTIAL ALTERNATIVES: Use optical processors and optical switching matrices to obtain greater inter-channel isolation. Use separate antennas for subcomponents of a payload to increase isolation between the payload subcomponents.
10. PLANNED PROGRAMS OR UNPERTURBED TECHNOLOGY ADVANCEMENT:
11. RELATED TECHNOLOGY REQUIREMENTS: 1. Antenna beam and polarization isolation. 2. Power conditioner/supply system decoupling. 3. Satellite switching and processor channel decoupling. 4. Payload channel data bus decoupling.

Table 4-18. Electromagnetic Compatibility/Interference, Contd

DEFINITION OF TECHNOLOGY REQUIREMENT	No.
1. TECHNOLOGY REQUIREMENT (TITLE): ELECTROMAGNETIC COMPATIBILITY INTERFERENCE	Page 3 of 4

12. TECHNOLOGY REQUIREMENTS SCHEDULE:

CALENDAR YEAR

SCHEDULE ITEM	79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	95		
TECHNOLOGY																			
DELINEATE EMC DESIGN PROCEDURES TO PROVIDE DESIRED INTERFERENCE LEVELS				━	━	━	━	━	━										
PHASE C, D INPUTS AND TESTING (TIME PERIOD)										━	━	━	━						
FUNDING LEVEL (In $1,000, 1980 dollars)				100	100	100	100	100	100										

13. USAGE SCHEDULE:

																	TOTAL
TECHNOLOGY NEED DATE													▼				
NUMBER OF LAUNCHES													1				

14. REFERENCES

15. LEVEL OF STATE OF THE ART:

1. Basic phenomena observed and reported
2. Theory formulated to describe phenomena
3. Theory tested by physical experiment or mathematical model
4. Pertinent functions or characteristic demonstrated, e.g., material, component
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6. Model tested in aircraft environment
7. Model tested in space environment
8. New capability derived from a much lesser operational model
9. Reliability upgrading of an operational model
10. Lifetime extension of an operational model

5. DESCRIPTION OF TECHNOLOGY: (Continued)

highly time dependent in both phase and amplitude. The rejection of in-band and intermodulation interference is difficult under these variable conditions. Further analysis and testing of the coupling between satellite constellation members will require experimental measurements with satellites equipped with antenna systems similar to the baseline configuration. A related interference occurs when a ground station antenna pattern simultaneously illuminates several satellites. The information direction to one satellite becomes a variable interference to an adjacent satellite. If a single large platform is used, the interference can be corrected since a fixed phase amplitude relation occurs.

Interpayload interference occurs within a satellite or platform between the many payloads present on the payload. Interference occurs between payloads when the separate payload channels are routed through common switching and processing components as well as payload peculiar components. Both electromagnetic, including optical, and acoustic coupling mechanisms are present. Intermodulation is also a high interference source depending on the material type and interconnects between components of the antenna reflectors, feeds, and the platform.

Intrapayload interface has sources similar to the interpayload sources, with the additional influence of the antenna system isolation. A primary source of interference is introduced by the reflector antenna feed assembly. The antenna feed has coupling between channels caused by overlapping of beams when high reuse of both uplink and downlink frequencies are used. Both the system's architecture and the antenna designs are combined to control the intrapayload interference levels.

The elimination of adjacent channel interference presents a major development problem for the filter technology. The DTU and HVT services presently envisioned on the platform have 40 MHz bandwidth transponders at Ka band. RF filters capable of separating the individual channels at the Ka band uplink frequency have bandwidths near 0.13 percent. These filters are very narrow and the control of adjacent channel levels will be difficult.

EMI testing of the payloads and interpayload switching and processing will be performed on the ground during the design C, D phases to validate performance capability. All of the systems comprising the platform will be tested together in the final stages of design for compatibility. When the platform components are isolated into a constellation of individual satellites, an added nonstationary variable is introduced that cannot be measured in the ground testing.

Table 4-19. Fiber Optics Data Transmission

DEFINITION OF TECHNOLOGY REQUIREMENT

1. TECHNOLOGY REQUIREMENT (TITLE): Fiber Optics Data Transmission

2. TECHNOLOGY CATEGORY: Communications System

3. OBJECTIVE/ADVANCEMENT REQUIRED: Develop small and low weight optical transmission lines for application between antenna feed assemblies and satellite switch and processor.

4. CURRENT STATE OF ART: Coaxial cable interconnects that are large and heavy cannot be used for large component separations.

5. DESCRIPTION OF TECHNOLOGY:

Fiber optics technology and optical processing and switching technology investigations are needed to reduce size and weight of the communications payloads on the GEO platform. The interface between the optical and electronic components of the system should be further refined to provide multiplex operation utilizing the full bandwidth capability of the optical transmission system. The broad bandwidth achieved in R&D laser diode systems has not been demonstrated in off-the-shelf devices. Therefore, emphasis should be placed on obtaining greater bandwidths from the fiber optics sources. Generally, long lifetimes are predicted for fiber optic systems; however, considerable testing will be required to determine MTBFs and required redundancy factors.

6. RATIONALE AND ANALYSIS:

Fiber optic transmission lines have large data bandwidth capability, relatively low transmission loss, and weigh little. These characteristics are desired for data transmission between the receiving/transmission feed assemblies and the satellite switching and processing assembly. Diode interfaces between the optical and electrical systems will be further reduced in size from their already small size. Methods to multiplex many data channels on a single fiber system to further reduce weight of the transmission system will be investigated. Switching and data processing systems will be either entirely optical or will be highly compatible with optical systems in the future.

Table 4-19. Fiber Optics Data Transmission, Contd

DEFINITION OF TECHNOLOGY REQUIREMENT

1. TECHNOLOGY REQUIREMENT (TITLE): Fiber Optics Data Transmission — Page 2 of 3

7. TECHNOLOGY OPTIONS:

1. Use coaxial cable systems with associated connectors and interconnect hardware - this leads to very bulky and heavy packaging.
2. Use waveguide assemblies - more difficult to deploy; bulkier and very heavy.
3. Use beamguide systems to translate the feed image near the processor - very limited beam scan capability and costly.

8. TECHNICAL PROBLEMS:

1. Capability for easy deployment of the optical fiber system will require special design considerations.
2. Interface locations between the electrical feed system and optical system trade studies.

9. POTENTIAL ALTERNATIVES:

Use coaxial cables or waveguides or a multiple beamguide system for the multibeam antenna.

10. PLANNED PROGRAMS OR UNPERTURBED TECHNOLOGY ADVANCEMENT:

Technology advancement will not be oriented for spacecraft applications without directed and funded programs.

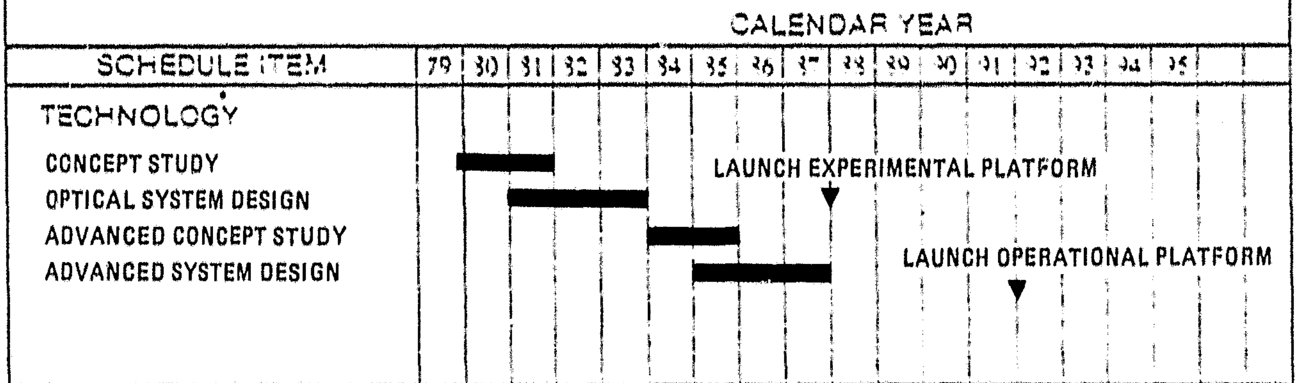
11. RELATED TECHNOLOGY REQUIREMENTS:

1. Optical processors.
2. Optical satellite switch assemblies.

Table 4-19. Fiber Optics, Contd

DEFINITION OF TECHNOLOGY REQUIREMENT	No.
1. TECHNOLOGY REQUIREMENT (TITLE): FIBER OPTICS	Page 3 of 3

12. TECHNOLOGY REQUIREMENTS SCHEDULE:



SCHEDULE ITEM	79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	95		
FUNDING LEVEL (In $1,000, 1980 dollars)		50	50	50	30	100	100	100	100										

13. USAGE SCHEDULE:

	79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	TOTAL
TECHNOLOGY NEED DATE									▼								
NUMBER OF LAUNCHES														1			

14. REFERENCES

15. LEVEL OF STATE OF THE ART:

1. Basic phenomena observed and reported
2. Theory formulated to describe phenomena
3. Theory tested by physical experiment or mathematical model
4. Pertinent functions or characteristic demonstrated, e.g., material, component
5. Component or breadboard-tested in relevant environment in laboratory
6. Model tested in aircraft environment
7. Model tested in space environment
8. New capability derived from a much lesser operational model
9. Reliability upgrading of an operational model
10. Lifetime extension of an operational model

Table 4-20. 30/20 GHz High Power Amplifiers

DEFINITION OF TECHNOLOGY REQUIREMENT

1. TECHNOLOGY REQUIREMENT (TITLE): 30/20 GHz High Power Amplifiers — Page 1 of 3

2. TECHNOLOGY CATEGORY: Specialized Communications/Integration Equipment

3. OBJECTIVE/ADVANCEMENT REQUIRED: Development of 300 watt TWTAs at 20 and 30 GHz.

4. CURRENT STATE OF ART: Developmental models of coupled cavity and helix tubes have been developed.

5. DESCRIPTION OF TECHNOLOGY:

Operation at Ka band will require power amplifiers capable of producing a minimum of 300 watts CW for uplink transmission.

6. RATIONALE AND ANALYSIS:

Operation at Ka band will require a minimum of 200 watts saturated power for the Customer Premises Service and 300 watts for the High Volume Trunking (Missions #1.2 and 2.2). The possibility of operating multicarrier per transponder exists and will require higher saturated outputs to accommodate backoffs to alleviate the carrier/interference ratio (C/I) problems.

Table 4-20. 30/20 GHz HPAs, Contd

DEFINITION OF TECHNOLOGY REQUIREMENT
1. TECHNOLOGY REQUIREMENT (TITLE): 30/20 GHz HPAs Page 2 of 3
7. TECHNOLOGY OPTIONS: Develop either coupled cavity tubes or, if possible, broadband helix tubes.
8. TECHNICAL PROBLEMS: 1. Output power: coupled cavity tubes have the higher power capability, but are narrow band. Helix tubes may not be realizable. 2. Reliability: nothing proven to date. 3. Production yield: only low power development models available.
9. POTENTIAL ALTERNATIVES:
10. PLANNED PROGRAMS OR UNPERTURBED TECHNOLOGY ADVANCEMENT:
11. RELATED TECHNOLOGY REQUIREMENTS: 1. Power supplies. 2. Manufacturing techniques. 3. Possible new magnetic materials.

Table 4-20. 30/20 GHz HPAs, Contd

DEFINITION OF TECHNOLOGY REQUIREMENT	No.
1. TECHNOLOGY REQUIREMENT (TITLE): 30/20 GHz HPAs	Page 3 of 3

12. TECHNOLOGY REQUIREMENTS SCHEDULE:

CALENDAR YEAR

SCHEDULE ITEM	79	80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	95		
TECHNOLOGY																			
LAB DEVELOPMENT			■	■	■														
TEST						■	■												
FLIGHT ARTICLE								■											
FUNDING LEVEL (In $1,000, 1980 dollars)				75	75	75	100	75	100										

13. USAGE SCHEDULE:

TECHNOLOGY NEED DATE					▼												TOTAL
NUMBER OF LAUNCHES										1							

14. REFERENCES

15. LEVEL OF STATE OF THE ART:

1. Basic phenomena observed and reported
2. Theory formulated to describe phenomena
3. Theory tested by physical experiment or mathematical model
4. Pertinent functions or characteristic demonstrated, e.g., material, component
5. Component or breadboard-tested in relevant environment in laboratory
6. Model tested in aircraft environment
7. Model tested in space environment
8. New capability derived from a much lesser operational model
9. Reliability upgrading of an operational model
10. Lifetime extension of an operational model

Tables 4-16 through 4-18 cover interplatform link (IPL) technology and electromagnetic compatibility requirements for the platform.

Table 4-19 summarizes the transmission technology needed for the platform. Coaxial cable interconnects are large and heavy and cannot be used for large component separations; small, low weight optical transmission lines are needed between satellite switch and processor, and the antenna feed assemblies.

4.5 CONCLUSIONS AND RECOMMENDATIONS

Figure 4-2 provides a scheduling overview of the principal technology tasks leading to an experimental platform flight in 1988. Each of the technological areas defined in Tables 4-1 through 4-20 require attention within the time frame that will provide for the 1988 capability, with the exception of servicing operations which could, if necessary, be deferred to the operational platforms. Table 4-21 is a tabulation of the technologies with regard to status and their order of priority for initiation.

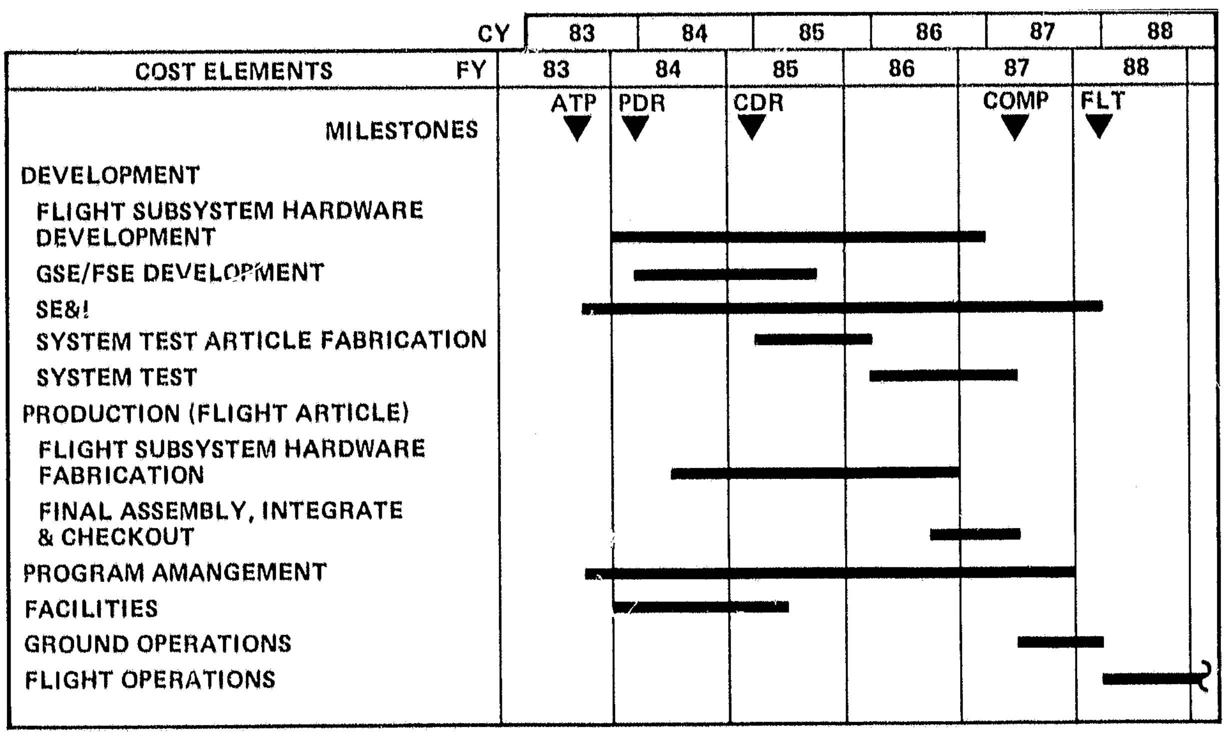


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Figure 4-2. Experimental Geostationary Platform Program Schedule

Table 4-21. Recommendations for Technology Advancement

	Begin Studies Now	Defer to Presently On-going Efforts	Begin Study When Experimental Platform is Defined	Begin Study When Operational Platform is Defined
1 Space Construction		X		
2 Active Control of LSS		X		
3 Solar Array for Geoplatform	X			
4 Power Mgmt System	X			
5 Power Mgmt System Control	X	X		
6 Power Mgmt Components	X			
7 Secondary Power Source		X		
8 Increased RCS Performance			X	
9 Thermal Management				X
10 Automatic Docking & Servicing		X		
11 Matrix Switch/Processor	X			
12 Deployable Antenna Surfaces		X		
13 Phased Array Antennas		X		
14 Less Antennas		X		
15 MBFHA Feed Assemblies	X			
16 Interplatform Links	X			
17 Intraconstellation Links	X			
18 Electromagnetic Compatibility	X			
19 Fiber Optics Data Transmission		X		
20 30.20 GHz HPAs	X			

Large solar arrays for long-term geostationary orbit service represent a technology that can probably benfit markedly from long-term studies. Power management and distribution are complex technologies and also need long-term R&D schedules. Definition studies in these areas should be initiated as soon as possible. Switch development is also a major task, but it may be desirable to lean heavily on independent development when the extent of the private efforts becomes clearer. Layout work on platform configurations has revealed feed modules to be a major packaging problem both in weight and geometry. Much work appears to be necessary to achieve better definition and to reduce the dimensions and masses of these devices.

Two other categories of effort are noted. Where effort is currently known to be underway, it is recommended that work specifically directed to geostationary platform requirements be temporarily deferred, at the same time making attempts to make requirements known to those conducting current studies. The remaining efforts are judged capable of being worked more effectively when the experimental platform has been defined well enough to provide specific requirements to the studies.

SECTION 5

TASK 5: STS INTERFACE REQUIREMENTS

The objective of this task is to identify the support and functional interface requirements imposed on space transportation system (STS) elements by the geostationary platform program.

The requirements on each STS element have been defined to a depth consistent with this study phase, segregated, and arranged to permit easy access by the supporting programs. Support and interface requirements for each STS element have been identified in this report section by subtask paragraph number: 5.1 - Orbiter, 5.2 - Orbital Transfer Vehicle (OTV), and 5.3 - Teleoperator and Servicing systems. Requirements on STS element subsystems such as structural, power, thermal, etc., have also been identified where applicable and the pertinent analyses and rationale included to support the recommendations.

Input data used to develop the interface requirements includes mission requirements generated in Task 1; selected platform and servicing concepts from Task 2; and platform subsystem definitions (structural interfaces, power, communications, thermal, fluid, etc.), transportation concepts (OTV), and logistics plans (timelines, vehicles, support requirements, and operations) from Task 3. Additional input data came from the NASA STS system element documents including Shuttle System Payload Accommodations (JSC 07700, Vol. XIV), Shuttle EVA Description and Design Criteria (JSC 10615); Manned Safety Requirements (JSC 11123), Overview Remote Satellite Services (Teleoperator) Program (9179), Orbital Transfer Vehicle Concept Definition Study (NAS8-33533), and data from NASA on Shuttle growth concepts.

The data was developed using the Western Hemisphere operational geostationary platform - Alternative #1, as a typical option. This data is also representative of the platforms for the Atlantic location. Data applicable to the other alternatives will be generated in the follow-on study as update tasks.

5.1 ORBITER

Platform requirements on, and interfaces with, the Orbiter stem from two missions - placing the platform at GEO, and providing service support for the platform during its lifetime. Requirements and interfaces for both these missions are contained in this section. Delivery of the platform is taken as the baseline mission, with different or additional requirements identified for the servicing mission.

5.1.1 <u>PERFORMANCE</u>. Ground rules for Alternative #1, the geostationary platform concept, require that the platform and OTV be launched in a single shuttle flight and that the servicing system and OTV also be launched in a single shuttle flight.

a. Platform delivery. The shuttle must be able to deliver 65,000 lb into a 160 n.mi. circular orbit at 28 1/2° inclination.

b. Servicing mission. For the ascent phase the shuttle must be able to deliver 65,000 lb into a 160 n.mi. circular orbit at 28 1/2° inclination. For the return-to-earth phase, the shuttle must be able to carry 15,000 lb back to earth.

5.1.2 <u>STOWAGE AND DEPLOYMENT</u>.

a. Platform delivery. Figure 5-1 shows a typical platform module (Alternative concept #1, Module 6) stowed in the Orbiter cargo bay, mated to its OTV as a single payload package. This arrangement accommodates a 26 ft long stowed platform module. The package is supported at its forward end by Orbiter attach points at Station 656.00. The aft end is supported by the OTV airborne support equipment at Stations 939.2 and 1269.6, as shown in Figure 5-2. The abort dump and vent line interfaces with the Orbiter are also shown. Umbilical lines for the other functions interface with the Orbiter at the locations shown in Figures 5-3, 5-4, and 5-5.

   The OTV will not have the stowed aerodynamic braking device for this mission. This mission will use the equipment available on the Orbiter aft flight deck; additional equipment requirements are TBD, to fit within the allowances shown in Figure 5-6.

   For checkout the OTV/platform must be rotated 75° from the horizontal to permit the deployed platform to clear the Orbiter, as shown in Figure 5-7. The ASE shown pivoted at Station 1269.6 (Figure 5-2) will not be able to rotate more than about 45°, as presently configured. Further analysis is needed on the ASE to allow rotation to 75°. This might mean moving the rotation point and OTV forward somewhat (approximately 1.5 ft to accommodate the rotation). The other interfaces with the shuttle would remain the same.

b. Servicing mission. The servicing system would be stowed in the cargo bay in much the same manner as the platform shown in Figure 5-1. The servicing system stowed length will be about 20 feet shorter than the platform stowed length, cantilevered off the forward end of the OTV. There will be no forward attachments from the servicing system to the Orbiter. The OTV installation in the shuttle will be the same as shown in Figure 5-2.

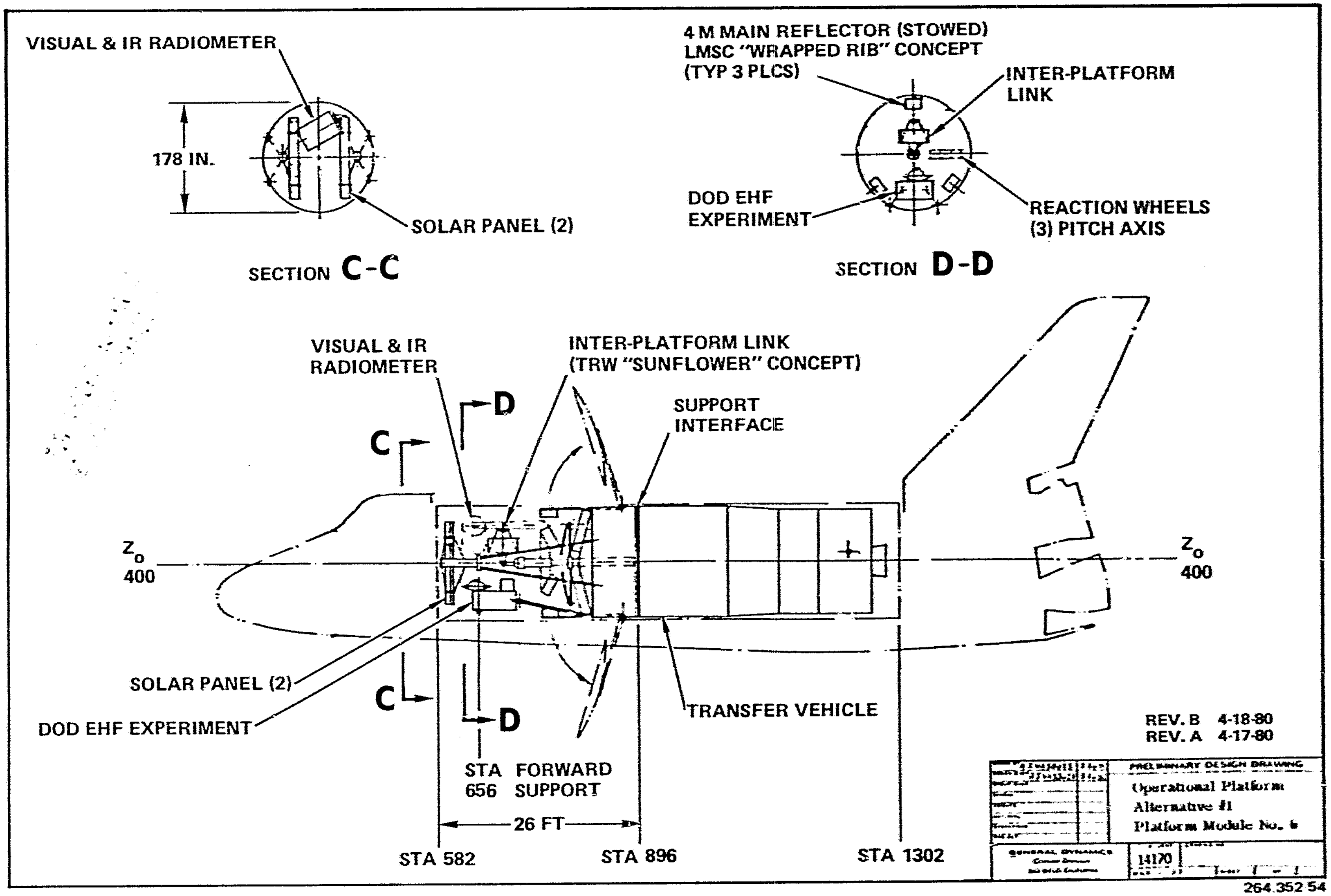

Figure 5-1. Typical Platform/OTV Payload Package in Orbiter Cargo Bay

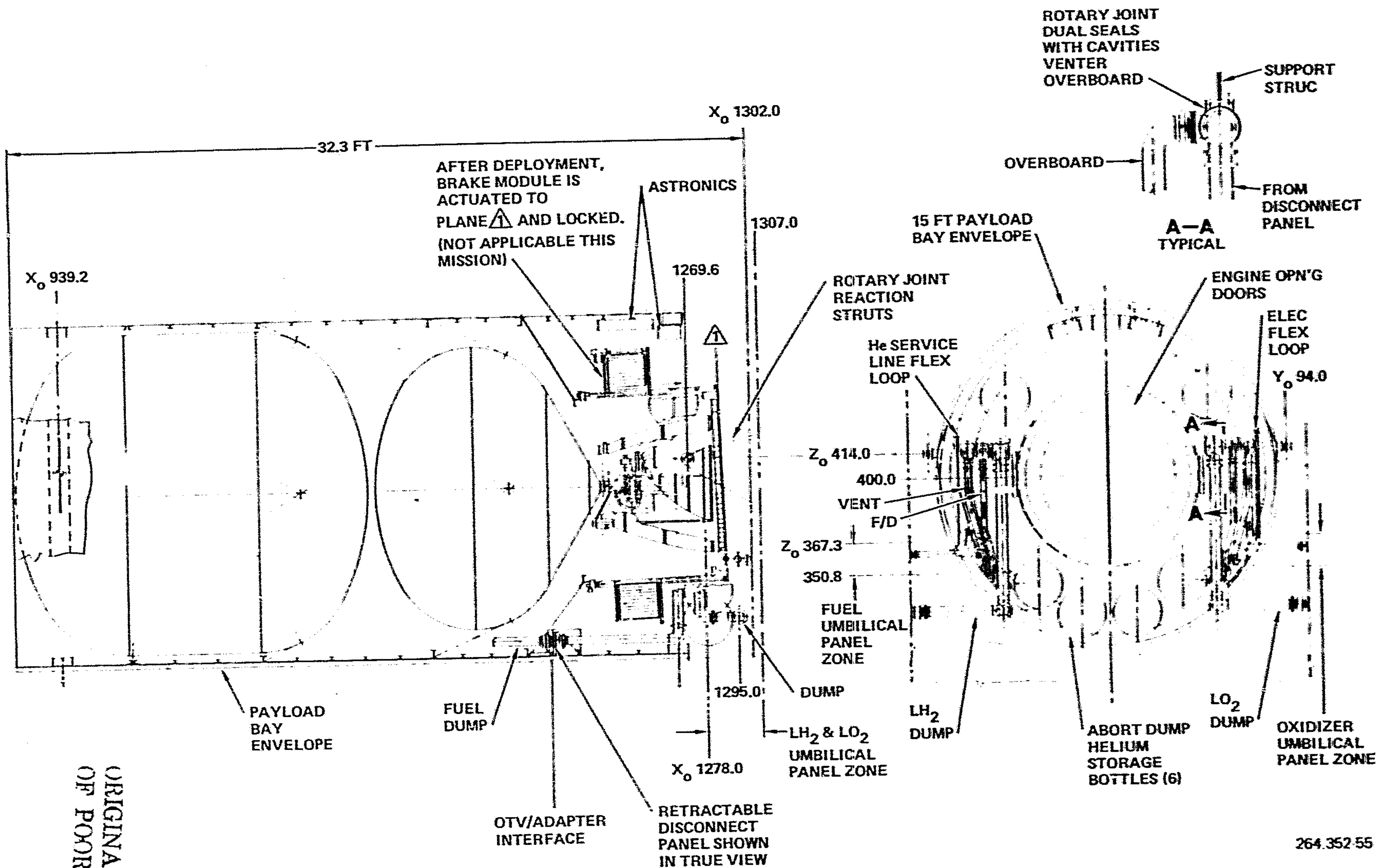

Figure 5-2. OTV Airborne Support Equipment

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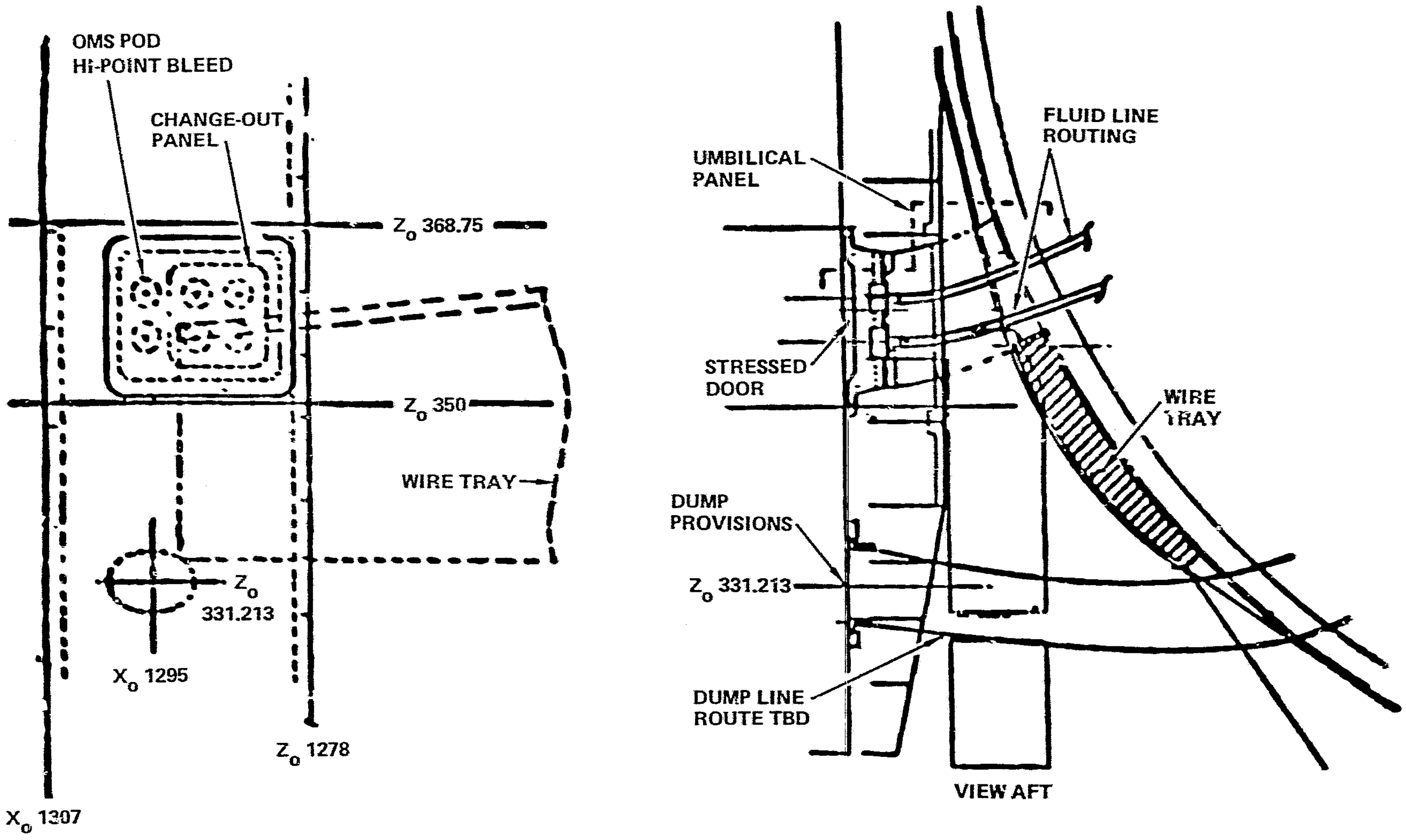

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Figure 5-3. Starboard (T-4) Payloads/OMS Delta-V Umbilical Panels and Dump Provisions (Routing Concepts)

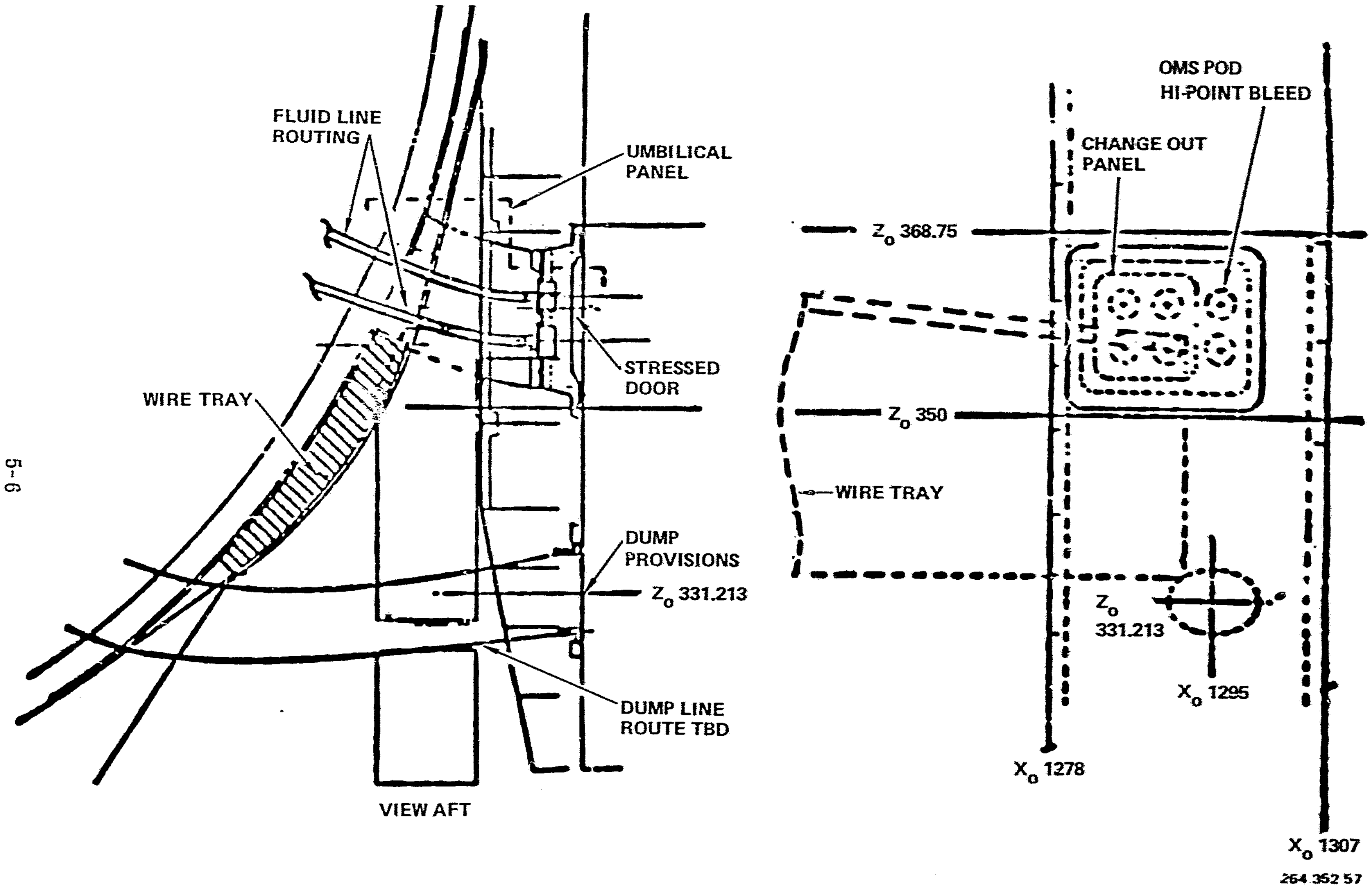


Figure 5-4. Port (T-4) Payload/OMS Delta-V Umbilical Panels and Dump Provisions

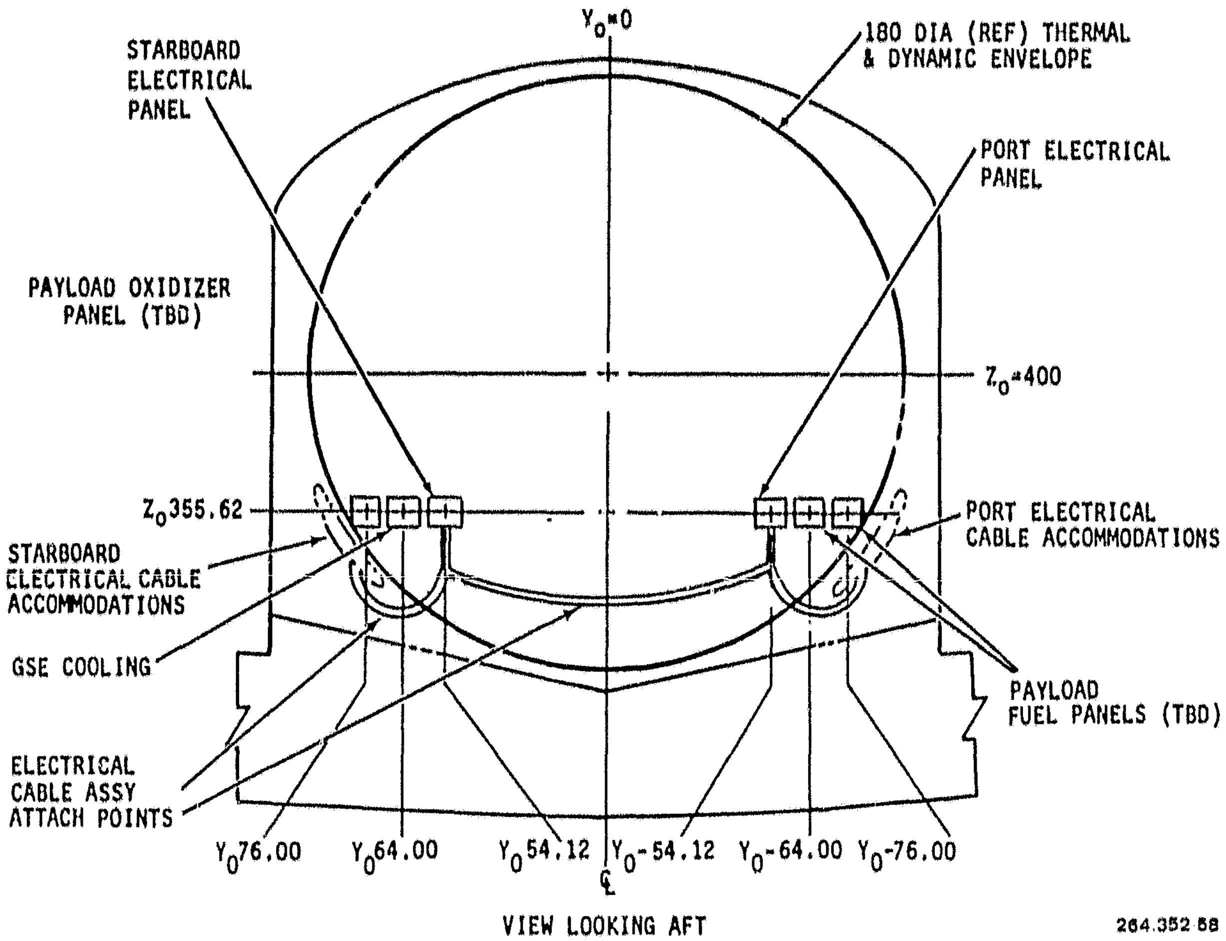
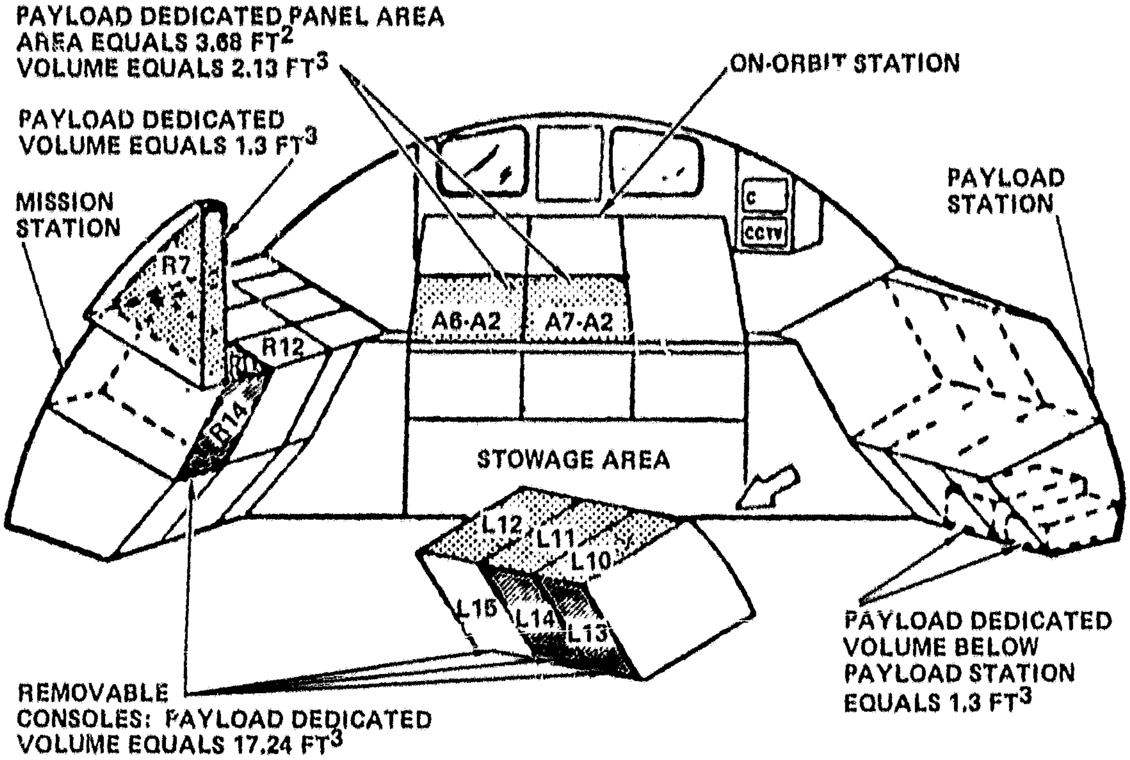


Figure 5-5. Shuttle Orbiter Payload Interface Locations - Xo 1307 Bulkhead



VIEW LOOKING AFT

NOTES:

1. LEGEND

PAYLOAD DEDICATED PANEL AREA EQUALS 16.85 FT^2.

ADDITIONAL PAYLOAD DEDICATED D&C PANELS ON INBOARD SURFACES OF THREE EQUIPMENT CONSOLES REQUIRE ALLOWANCE OF SIX (6) INCHES DEPTH OF NORMAL PANEL AREA. ALL COMPONENTS ON THESE SURFACES MUST BE FULLY RECESSED. ADDITIONAL PANEL SURFACE AREA IS 5.5 FT^2

2. TOTAL PAYLOAD DEDICATED VOLUME IS 21.64 FT^3

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Figure 5-6. Shuttle Orbiter Payload Physical Interface Locations - Aft Flight Deck General Arrangement

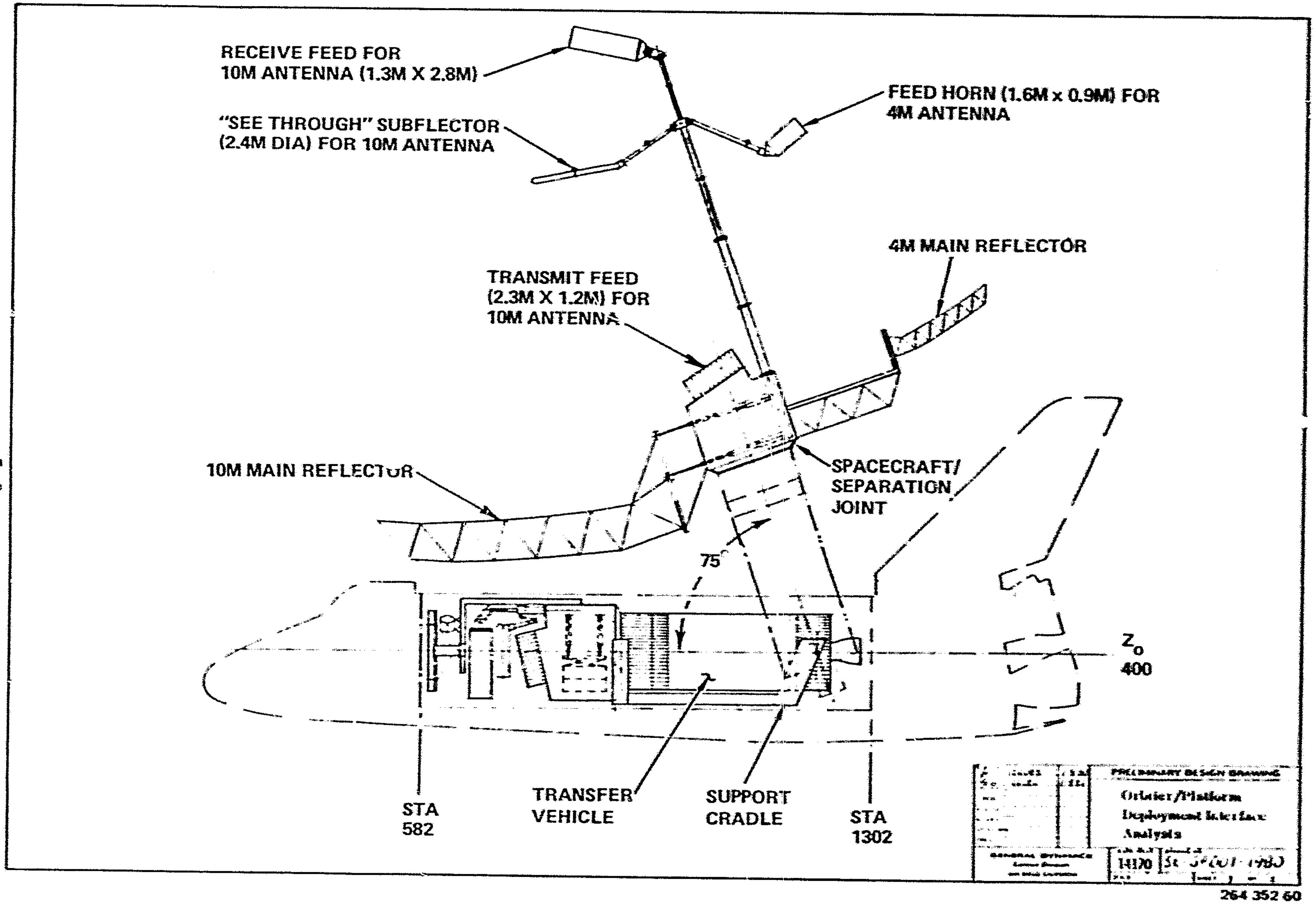

Figure 5-7. Platform Deployment and Checkout, Attached to Orbiter

For checkout the OTV/servicing system needs only 30° rotation from the horizontal; the ASE system shown in Figure 5-2 can accommodate this. The additional equipment required on the aft flight deck is TBD.

5.1.3 OPERATIONS.

a. Platform delivery. The shuttle has interfaces with the OTV/platform in three major mission phases: prelaunch, ascent to LEO, and payload deployment and checkout. The Orbiter has 160 hours to complete the turnaround from the last flight to liftoff for the next flight. Within this time period the OTV/platform must mate with the Orbiter, be checked out, tanked and be ready for launch.

The ascent to LEO phase takes approximately 1-1/2 hours, as shown in Figure 5-8. At the end of this time, the deployment and checkout phase for the OTV/platform can begin.

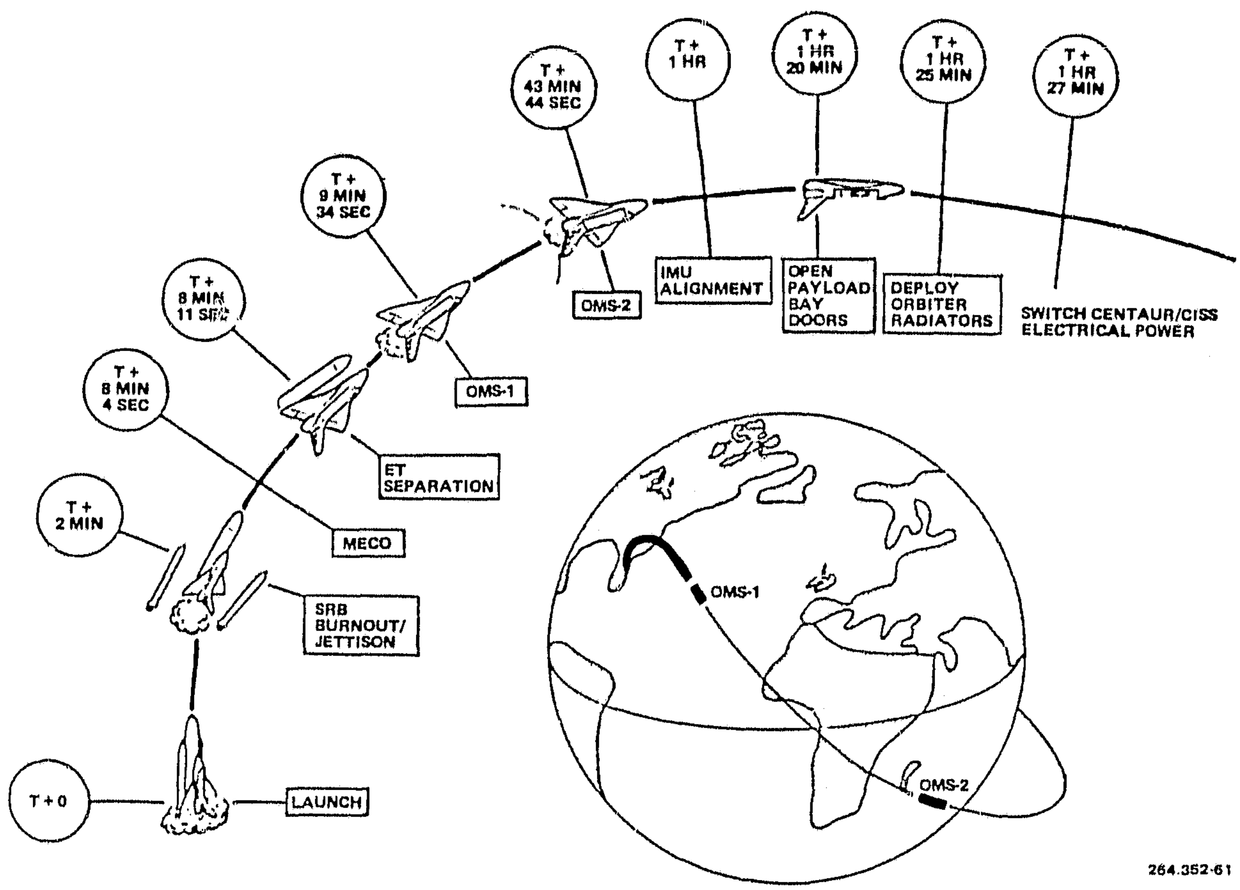

Figure 5-8. STS Flight Operations - Ascent Phase

The OTV/platform will first be rotated 75° out of the cargo bay. The elements of the platform will then be deployed, checked out, and adjusted as required. The platforms are designed to be automatically deployed without planned EVA assistance from the crew. However, if a contingency arises, then the crew will be used to overcome the difficulty in an EVA mode. The total LEO time including any contingencies for deployment and checkout will be less than seven days.

The platform/OTV fills the entire cargo bay. No allowance has been made for a four foot clear space at the forward end of the bay to permit EVA entrance into the bay to repair any difficulties that might arise and prevent cargo bay doors from closing. To meet the safety requirements in this area, the complete payload package (platform/OTV) is designed for jettison so the cargo bay doors can be closed.

b. Servicing mission. There are five major mission phases where the shuttle interfaces with the OTV/servicing system: 1) prelaunch, 2) ascent to LEO, 3) payload deployment and checkout, 4) payload rendezvous, docking, and stowage, and 5) return to earth.

The prelaunch time and ascent to orbit time for this mission are the same as for the platform delivery. The deployment and checkout operation is similar.

The OTV servicing system needs only to be rotated 30° from the horizontal for checkout. The operating elements of the servicing system will be activated and checked out. This will be an automatic operation, but if any difficulties arise the crew may be used in an EVA mode to solve the problem. Total time including any contingencies for deployment and checkout will be less than 48 hours.

It is assumed that the Orbiter will stay on orbit while servicing of the platform takes place at GEO instead of returning to earth after separation from the OTV/servicing system and sending a new Orbiter up to LEO later. The Orbiter will be the active element in rendezvous and docking with the OTV servicing system. After the OTV/servicing system has placed itself in the proper orbit for rendezvous with the Orbiter, the final rendezvous, docking, and stowage of the payload should take approximately 4-1/2 hours. The OTV/servicing system is captured with the remote manipulator system (RMS) and placed in the OTV ASE. The ASE performs the final stowage operation within the cargo bay. After the payload is stowed, Orbiter thermal conditioning and reentry phasing can take up to 15 hours before deorbit burn. From deorbit burn to touchdown will take about one hour.

5.1.4 SUPPORT SUBSYSTEM. This section identifies the requirements for the Orbiter subsystems for both missions.

5.1.4.1 Structural Interfaces.

a. Platform delivery. Figures 5-1 and 5-2 showed the structural interfaces between the platform/OTV and the Orbiter. The forward support for the platform is at Orbiter Station X = 656.0. The forward and aft structural supports for the OTV are shown in Figure 5-2. These are at Orbiter Station X = 939.2 and 1269.6. The platform/OTV is rotated 75° about a point at or forward of Station 1269.6 for payload deployment or jettison. Critical clearance for this operation involves the engine bell, upper helium storage bottles, and the OTV support adapter.

b. Servicing mission. Figure 5-2 shows the structural interface between the OTV and the Orbiter. The servicing system will be cantilevered off the front end of the OTV and will have no structural interface with the Orbiter. The servicing system/OTV does not have to be rotated more than 30° from the horizontal for deployment, checkout, and separation, so clearances are not as critical at the aft end as they are in the platform delivery mission.

The servicing mission requires an Orbiter payload deployment and retrieval system for retrieving the OTV/servicing system. The RMS will be used to capture the payload and position it in the OTV ASE so the ASE can stow the payload in the cargo bay for the return trip to earth. The standard end effector will be used.

5.1.4.2 Power.

a. Platform delivery. The platform/OTV will not require more than the 50 kW-hr of electrical power available from the Orbiter for deployment and checkout. After deployment the platform will go on internal power for the checkout operation. The electrical power interface will be as shown in Figure 5-5.

b. Servicing mission. The servicing system/OTV will not require more than the 50 kW-hr of electrical power available from the Orbiter for the deployment and checkout phase, and for the retrieval and stowage phase. The electrical power interface will be as shown in Figure 5-5.

5.1.4.3 Communications.

a. Platform delivery. For payload/OTV checkout while still attached to the Shuttle, the Shuttle must provide an S-band uplink of 32 kbps with 6.4 kbps command and a return link of 192 kbps with 128 kbps data. The umbilical interface with the Orbiter is shown on Figure 5-6.

b. Servicing mission. The same communications capability is required as for the platform delivery mission.

5.1.4.4 Propulsion/RCS.

a. Platform delivery. The allowable 4000 lb of RCS propellant for payload support is more than adequate to hold the required attitude during payload deployment and checkout and for the seven day mission (maximum).

b. Servicing mission. The allowable 4000 lb of RCS propellant for payload support is more than adequate for Shuttle attitude control during the checkout phase and to accomplish rendezvous and docking (approximately 2000 lb) to the payload during the return phase of the mission.

5.1.4.5 G&N.

a. Platform delivery. The on-orbit navigation accuracies that can be achieved by the Orbiter are more than adequate for the platform delivery mission.

For checkout while the platform is deployed and still attached to the Orbiter, the platform and its solar panels need to be pointed at the sun within ±15°. The platform will be on internal power while being checked out. Figure 5-7 shows the deployed platform in the checkout position attached to the shuttle. Checkout time may take up to 7 days, including contingencies.

b. Servicing mission. The on-orbit navigation accuracies that can be achieved by the Orbiter are more than adequate for the servicing mission. There are no pointing requirements from the servicing system during checkout.

For rendezvous and docking with the servicing system when it returns from GEO to LEO, the Orbiter will maintain the prescribed attitude for docking.

5.1.4.6 Fluid.

a. Platform delivery. The fluid interfaces between the platform/OTV and Orbiter are shown in Figures 5-2, 5-3, 5-4, and 5-5. These include the LH_2/LO_2 tanking, abort dump, and vent lines. The abort dump lines are sized to dump the LH_2/LO_2 in 300 seconds per the requirement.

b. Servicing mission. The fluid interfaces between the servicing system/OTV and Orbiter are the same as for the platform delivery mission.

5.1.4.7 Thermal.

a. Platform delivery. The platform has a self-contained thermal control system that is not required to interface either with the OTV or with the Orbiter. The OTV has a passive thermal control system that requires no interface with the Orbiter. The equipment on the aft flight deck can be cooled by the forced air from the Orbiter.

b. Servicing. There is no interface between the Orbiter thermal control system and the servicing system/OTV for either the ascent or descent mode.

5.1.4.8 Environmental Control.

a. Platform delivery. The platform/OTV will be designed to meet the Shuttle environment during prelaunch, ascent to LEO, and at LEO as described in the STS user handbook. No special provisions will be required from the Shuttle.

b. Servicing mission. The servicing system/OTV will be designed to meet the Shuttle environment for all modes of operation as described in the STS user handbook. No special provisions will be required from the Shuttle.

5.1.4.9 Lighting.

a. Platform delivery. The lighting shown in Table 5-1, located as shown in Figure 5-9, is required to perform the platform checkout while still attached to the Orbiter, and includes the RMS light.

Table 5-1. Orbiter Cargo Bay Lighting and Illumination

Payload Bay Floodlights	
Watts	200
Lumens/Watt	40 Minimum
Type	Arc Discharge
Beam	135° by 135°
Docking Floodlight	
Watts	200
Lumens/Watt	40 Minimum
Beam	120° by 120°
Overhead/Docking Light	
Watt	130
Lumens/Watt	12 Minimum
Beam	120° by 120°
RMS Light	
Watt	150
Lumens/Watt	12 Minimum
Beam	80°

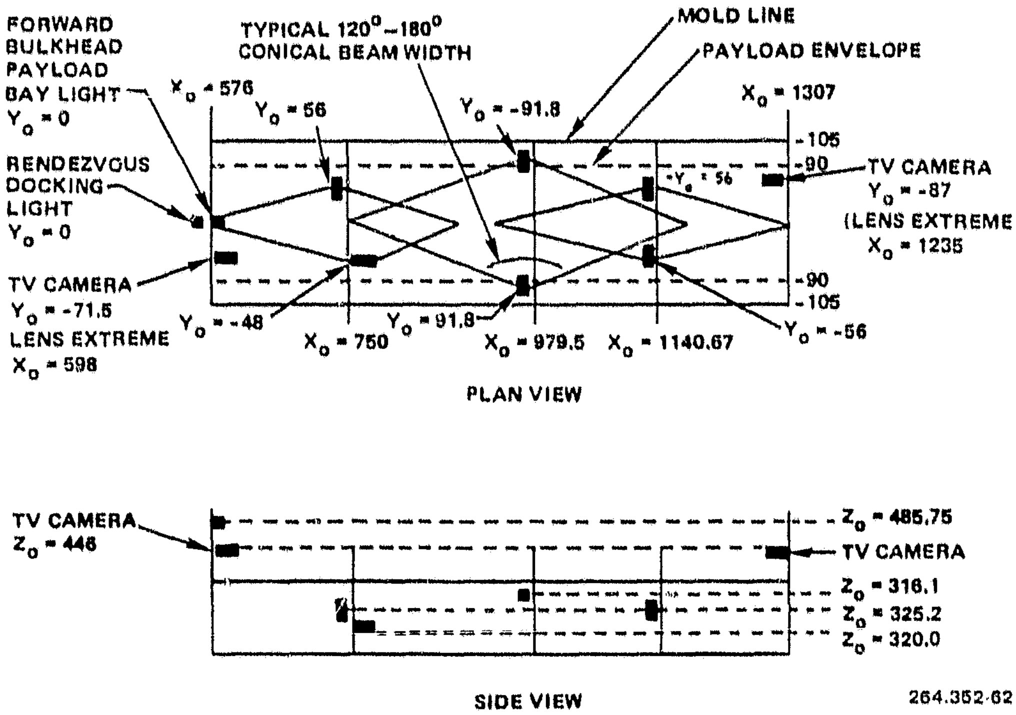

Figure 5-9. Cargo Bay Light and TV Camera Locations

b. Servicing mission. The same lighting is required as for the platform delivery mission.

5.1.4.10 Closed Circuit Television (CCTV) System.

a. Platform delivery. Five CCTV cameras are required for payload checkout. Two will be placed on the RMS as shown in Figure 5-10, one each on the forward and aft bulkheads, and one mounted at the No. 4 keel position.

b. Servicing mission. The same CCTV cameras are required as for the platform delivery mission.

5.1.4.11 Rendezvous System.

a. Platform delivery. There is no requirement for Orbiter rendezvous capability.

b. Servicing mission. The Orbiter rendezvous system is required to actively perform the rendezvous with the servicing system on its return from GEO.

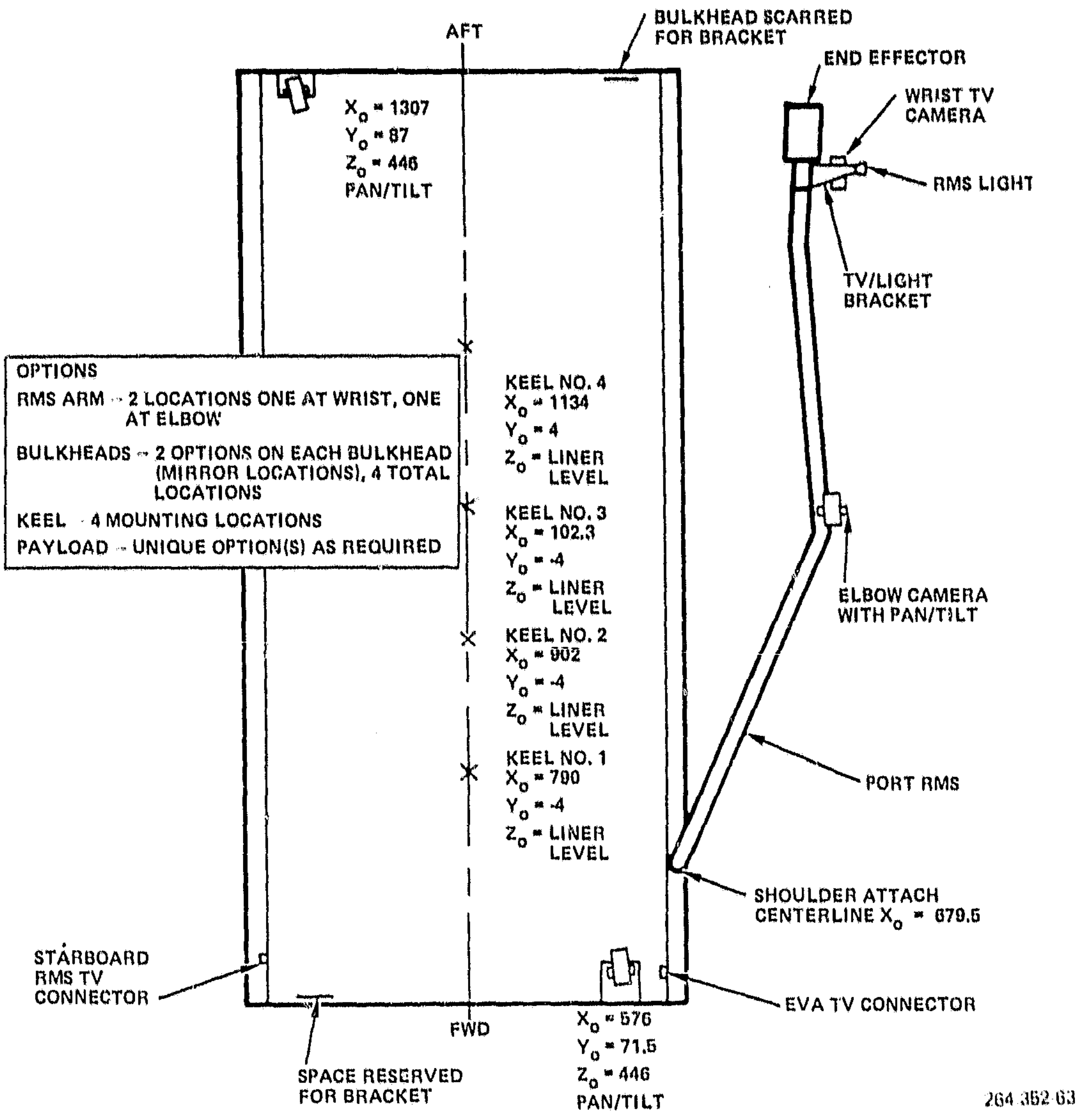

Figure 5-10. CCTV Camera Mounting Options

5.1.5 CREW. STS crew requirements for both normal and possible contingency operations during platform delivery missions and servicing missions are as follows:

a. Platform delivery. The platform has been designed for automatic deployment. For the nominal operation of deployment and checkout prior to separation from the Orbiter, four crewmen can perform the required tasks. The time on orbit for the nominal mission should be less than three days, using the equipment provided by the Orbiter plus additional special equipment to be placed on the aft flight deck in the areas shown in Figure 5-6.

For contingency operations where crewmen may be required to use an EVA mode to assist in platform deployment or to correct a malfunction that occurs during checkout, it is expected that the allowable two - 2 man/6 hour EVA operations using the two MMUs will be sufficient to correct the malfunctions. The total time on orbit for the contingency mode will be 7 days or less. For the EVA operation, the crewmen will use the middeck internal airlock.

b. Servicing mission. The servicing system has been designed for automatic deployment and checkout. For the nominal operation of deployment and checkout prior to separation from the Orbiter, four crewmen can perform the required tasks. The time on orbit for the nominal mission should be less than 24 hours. The crew will use the equipment provided by the Orbiter plus TBD additional special equipment to be placed on the aft flight deck in the areas shown in Figure 5-6.

For contingency operations where crewmen may be required to use an EVA mode to assist in correcting any malfunctions in the elements of the servicing system, it is expected that only one of the allowable two-2 man/6 hour EVA operations using two extravehicular mobility units will be sufficient to correct the malfunctions during the ascent mode and the other 2 man/6 hour EVA operation would be available to correct any malfunctions that might arise while properly restowing the payload for return to earth. The total orbital time for ascent including contingencies should be less than 48 hours. The total orbital stay time for the complete mission should be no greater than 7 days. For the EVA operation, the crewmen will use the middeck internal airlock.

5.2 ORBITAL TRANSFER VEHICLE (OTV)

Platform requirements on the OTV stem from two missions – placing the platform in its correct orbital position, and providing service support for the platform during its lifetime. Requirements for each of these missions are contained in this section. Delivery of the platform is considered the baseline mission, with different or additional requirements identified for the servicing mission.

This section identifies the performance, operations, and support required from the OTV to support both missions.

5.2.1 OTV PERFORMANCE. The ground rules for the Alternative #1 geostationary platform concept require that the platform and OTV be launched in a single Shuttle flight, and that the servicing system and OTV also be launched in a single Shuttle flight.

a. Platform delivery. The OTV must provide the change in velocity (ΔV) to transfer the geostationary platform from Shuttle low earth orbit (approximately 160 n.mi.) to an equatorial geosynchronous orbit at either 110°W longitude in the Western Hemisphere or 15°W longitude over the Atlantic. The change in velocity required is approximately 14,000 ft/sec. The platform must be positioned by the OTV within ±0.1 km of the required position. There will be a constellation of six platforms at each of the above locations and each platform must be placed within the above tolerance to properly function with the others. The OTV can be expendable but must have provisions to place it in a debris orbit above synchronous.

The payload weight to be carried will be a maximum of 6,895 kg. The maximum g-load imparted by the OTV on the platform must not exceed 0.07 g. The platform will be transferred from LEO to GEO in the deployed condition. Deployment of the platform will be checked out in LEO and any malfunction corrected before transfer to GEO. Tradeoff data show that weight of the platform increases significantly for transfer in the deployed mode if the g-loading exceeds 0.1 g to any extent.

b. Servicing mission. The OTV must provide sufficient change in velocity to transfer the geostationary platform servicing system (unmanned) from Shuttle low earth orbit (approximately 160 n.mi.) to an equatorial geosynchronous orbit at either 110°W longitude in the Western Hemisphere or 15°W longitude over the Atlantic. The change of velocity is approximately 14,000 ft/sec. The servicing system must be positioned by the OTV within ±0.1 km of the above required position. The OTV must deliver a maximum of 2267 kg to this position. There is no restriction on the g-loading for the servicing equipment. For this mission, the OTV will stationkeep near the center of the six-platform constellation while the servicing system detaches itself from the OTV and services the platforms as required. The OTV must stationkeep in this position during the servicing operation as called out in the operations section.

After servicing has been accomplished, the platform servicing system will redock with the OTV. The OTV will transfer the servicing system to a debris orbit to jettison expendables, then return the servicing system to LEO for rendezvous and dock with the Shuttle. The maximum return payload is 822 kg. The required change of velocity for this operation is approximately 14,500 ft/second. There is no restriction for the return flight g-load on the servicing system.

5.2.2 OTV STOWAGE AND DEPLOYMENT.

a. Platform delivery. The platform and its OTV will be stowed as a unit payload in the cargo bay. The forward end of the OTV will be attached to the aft end of the platform package to provide aft support for the platform while in the Orbiter, and to provide a thrust face during transfer from LEO to GEO. Forward support of the platform/OTV payload package during ascent to LEO will be provided by the Orbiter. The platform requires a stowed length of 26 ft within the cargo bay, leaving 34 ft forward of Orbiter Station 1302 allocated for OTV installation in the bay, including provisions for required rotation and deployment of the platform while still physically attached to the Orbiter. Figure 5-11 shows the stowage requirements for the OTV. Geostationary platform program guidelines require that the platform be deployed and checked out while still physically attached to the Orbiter in LEO before being transferred to GEO, to capitalize on STS man-assist capabilities. Figure 5-12 shows that the OTV/platform package must be rotated 75° out of the Orbiter to permit sufficient clearance between the Orbiter and elements of the platform when deployed. If a payload package is rotated about Orbiter Station 1269.6, a 30° angle is sufficient to permit separation from the Orbiter if the payload doesn't have any large elements to be deployed perpendicular to its centerline. However, if the OTV has to be rotated to 75° as shown in Figure 5-12, then the rotation point moves forward and shortens the allowable length for the OTV stage. The airborne support equipment must be able to support the aft end of the OTV, rotate it 75° with the platform attached, provide the avionic equipment to support checkout of the platform while still attached to the Orbiter, provide for abort dump of the cryogenic propellants if required during launch, and meet the other demands required of a Shuttle payload during launch to LEO. It must also be able to restow the OTV/platform in the cargo bay and support it on the way back to earth in case the platform does not meet checkout requirements for transfer to GEO.

b. Servicing mission. The servicing system and the OTV are stowed together in the cargo bay as a single payload package. The OTV is attached to the servicing system at the OTV forward end and provides the only support for the servicing system during shuttle boost to LEO and OTV transfer to GEO. The servicing system requires a stowed length of 20 ft within the cargo bay, leaving 45 ft forward of Orbiter Station 1302 for OTV installation including provisions for required rotation and checkout prior to separation from the Orbiter.

The servicing system must have checkout capability, including manipulator arms, while still attached to the Orbiter in LEO before being transferred to GEO. A rotation angle of 30° from the horizontal is all that is required for checkout and separation. The airborne support equipment must be able to support the aft end of the OTV, rotate it 30° with the servicing system attached, provide the avionic equipment to support checkout of the service system, provide for abort dump of cryogenic propellants, and meet the other demands required of a Shuttle payload during launch to LEO, receive the OTV and service system after separation from the Orbiter, position them back in the Orbiter, and support them adequately for the return mission to earth.

5.2.3 OPERATIONS.

a. Platform delivery. Figure 5-13 is a top level functional flow of the OTV mission phases and operational requirements for platform delivery, with the approximate times allocated for these operations.

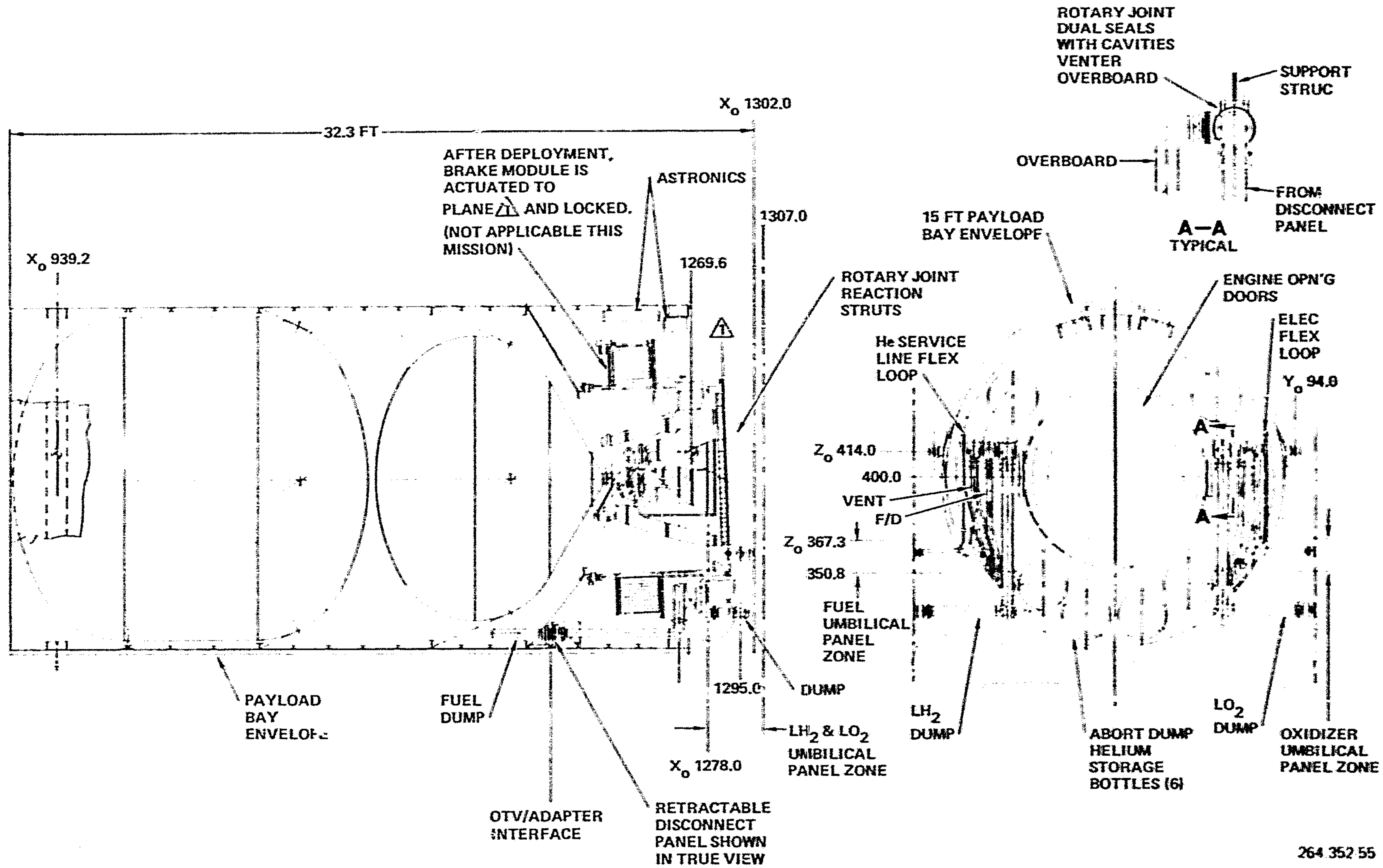

Figure 5-11. OTV Airborne Support Equipment

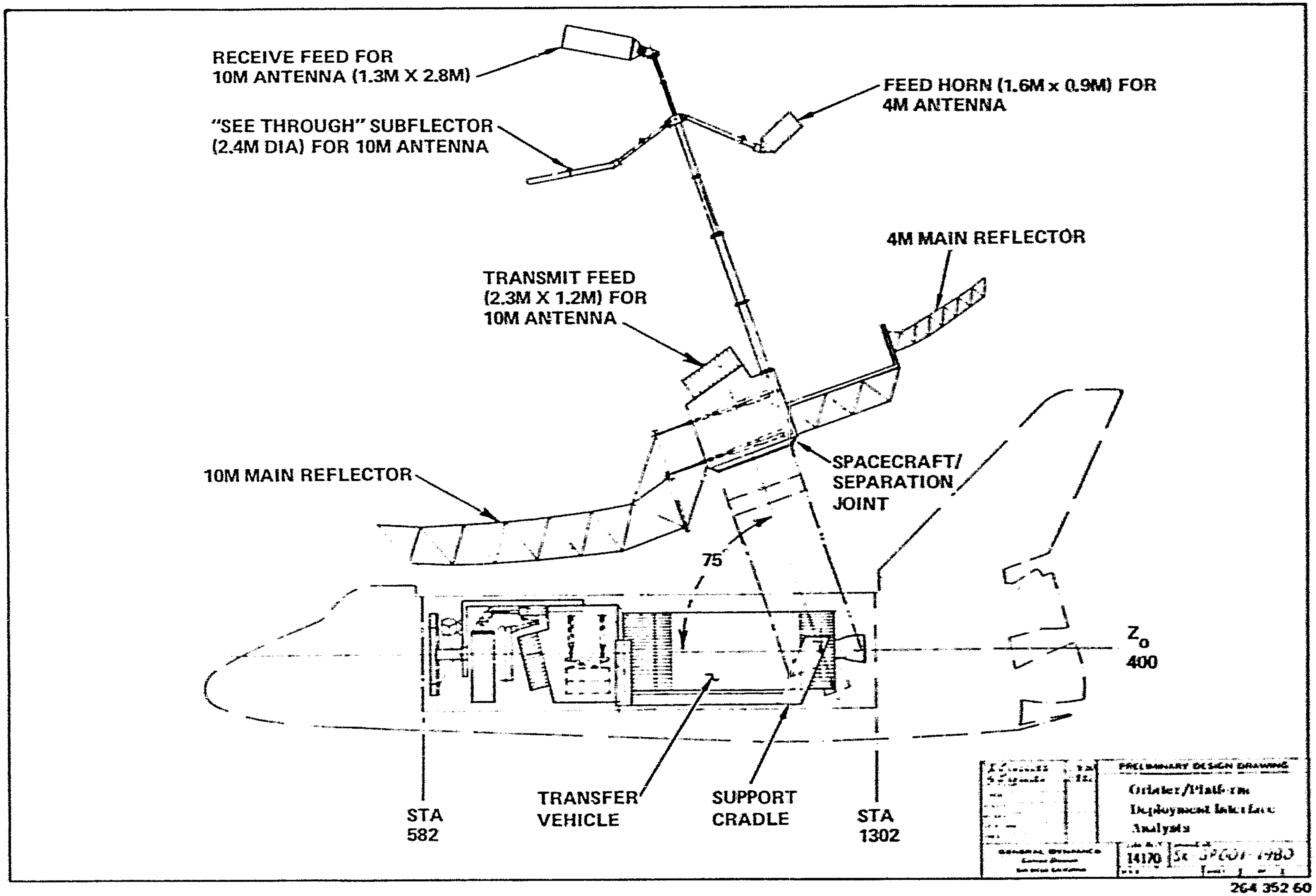

Figure 5-12. Platform Deployment and Checkout, Attached to Orbiter

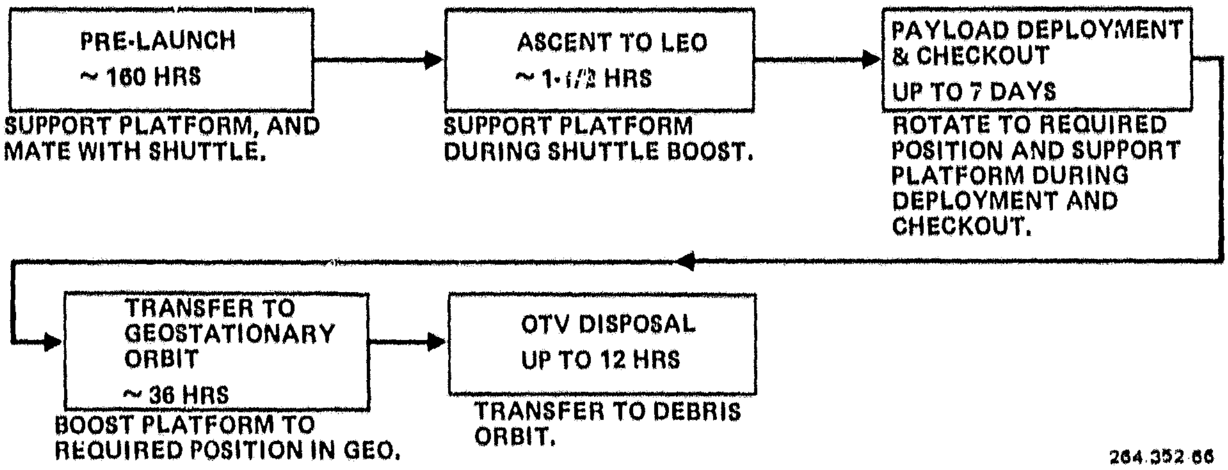

Figure 5-13. Platform Delivery Mission, Major Phases, and OTV Requirements

After separation from the Shuttle, the OTV will transfer its platform to geosynchronous equatorial orbit to join one of two six platform constellations: a Western Hemisphere constellation at 110°W longitude, and an Atlantic constellation at 15°W longitude. Operations are the same for both of these missions. The OTV delivers the platform to a required position (TBD) (see Section 5.2.4.6 for tolerances) and separates from the platform. The platform positions itself in the proper place in the constellation. The OTV is expendable and transfers itself to a debris orbit approximately 2000 n.mi. above GEO.

b. Servicing mission. Figure 5-14 is a top level functional flow of the OTV mission phases and operational requirements for platform servicing at both the Western Hemisphere and Atlantic locations, with the approximate times allocated for these operations.

After separation from the Shuttle, the OTV will transfer the servicing system to GEO in the middle of a six platform constellation. The servicing system then separates from the OTV and services three of the platforms. The servicing system then returns and docks with the OTV. The OTV transfers the servicing system to a debris orbit approximately 2000 n.mi. above GEO where the expended bottles and batteries are jettisoned. The OTV then transfers back to the Shuttle orbit for rendezvous with the Orbiter. The Orbiter docks with the OTV and returns it and the servicing system to earth.

5.2.4 <u>SUPPORT SUBSYSTEMS</u>. This section identifies the requirements for the OTV subsystems including the airborne support equipment (ASE) for both missions. The requirements will be called out for the mission phases identified in Figures 5-13 and 5-14 as applicable.

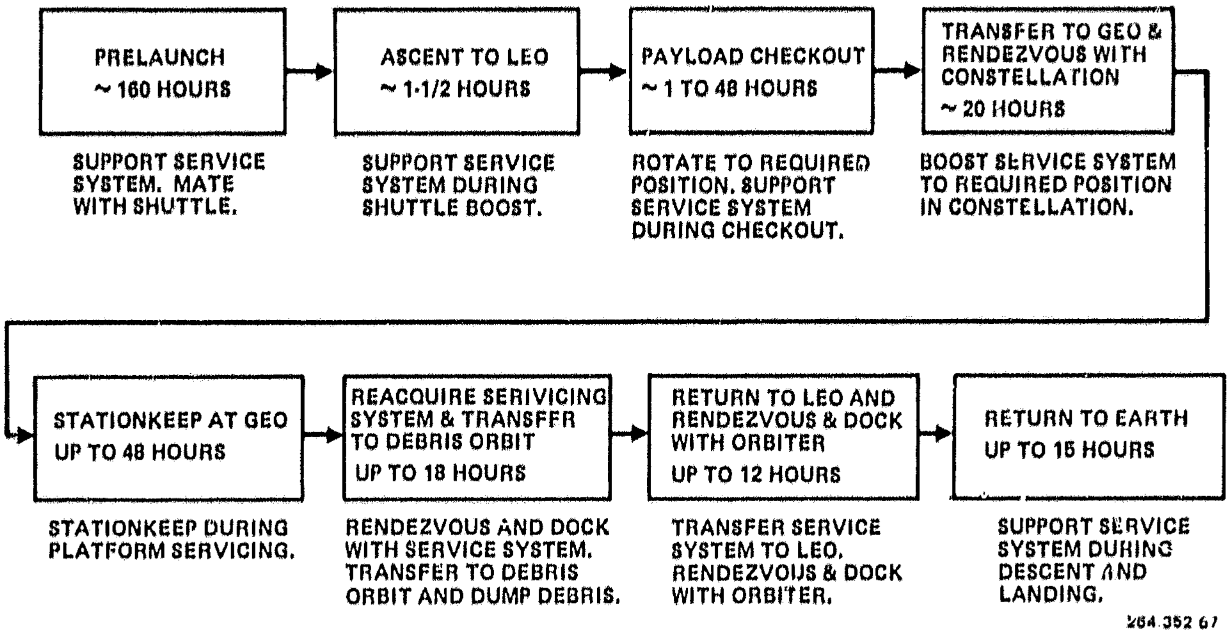

Figure 5-14. Service System Delivery Mission, Major Phases, and OTV Requirements

5.2.4.1 Structural.

a. Platform delivery. The OTV will provide aft support for the platform during mating of the payload on the ground, during the boost phase to LEO, during rotation out of the Orbiter for checkout prior to release from the cargo bay, and during the transfer phase and delivery to the required position at geostationary orbit. In all but the Shuttle boost phase the OTV structure will be the only support for the platform. During Shuttle boost the forward end of the platform will be supported in the Orbiter. The OTV must support the platform to withstand the launch and operational dynamic environment defined in the Space Shuttle System Payload Accommodations document JSC 07700, Vol. XIV, Revision F, Change 31, and the STS User Handbook. Figure 5-15 shows the OTV and forward support concept for the platform. The maximum weight of the platform is 6895 kg. The actual support load during Shuttle boost and abort landing is TBD.

The support system must be able to release the payload with minimum disturbing torques at geosynchronous orbit.

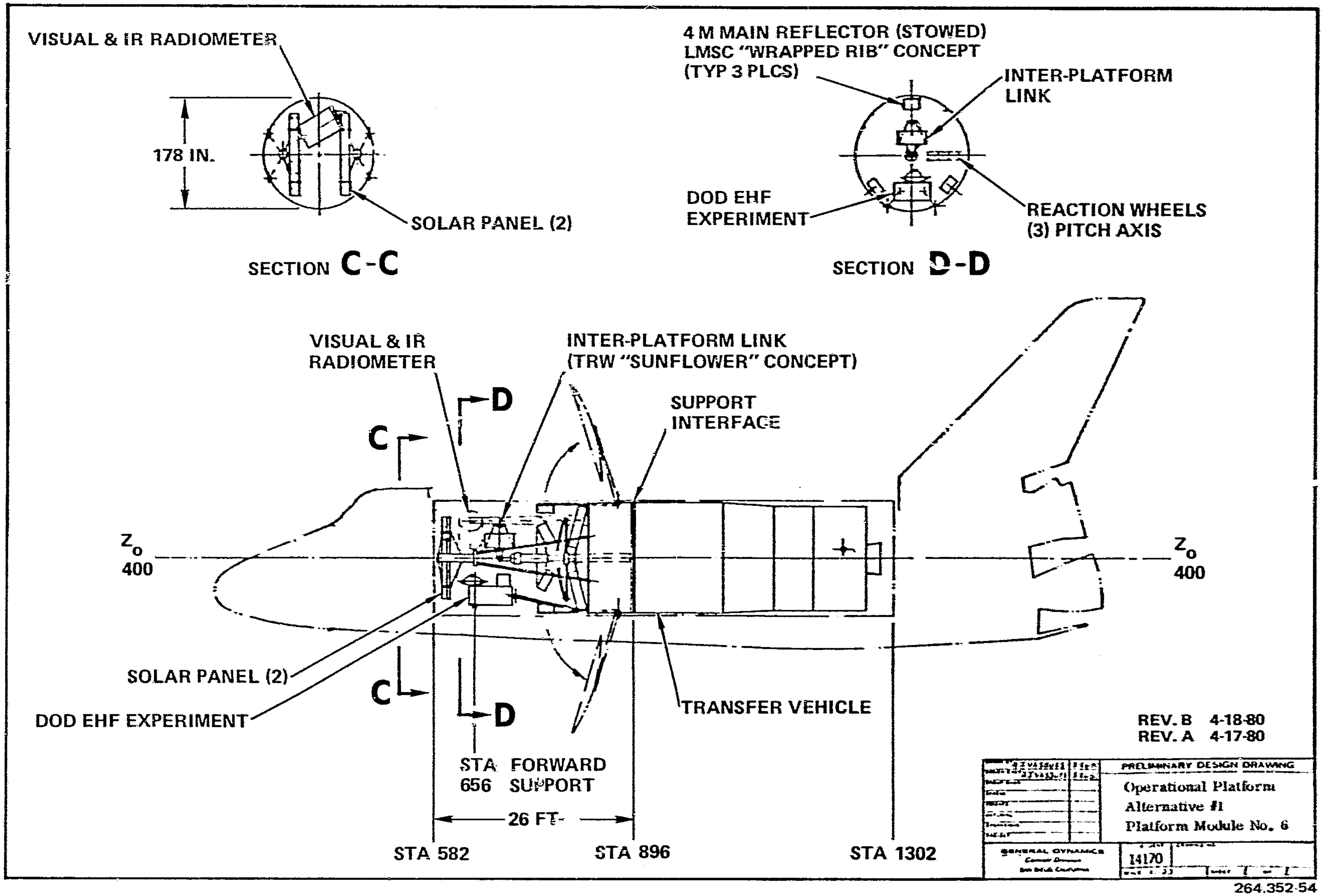

Figure 5-15. Platform Support Concept

b. Servicing mission. The OTV will provide support for the servicing system during mating of the payload on the ground, during the boost phase to LEO, during payload rotation out of the Orbiter for checkout prior to release from the Orbiter, during transfer to the required position at geostationary orbit, on the return flight to LEO including rendezvous, docking and stowage in the Orbiter, and during the return trip to earth. The OTV must support the servicing system to withstand the launch and dynamic environment defined in JSC 07700, Vol. XIV, and the STS user handbook. The maximum weight of the service system will be for the ascent mode and for the return mode. The support system must be able to release the payload without inducing any significant disturbing torques and must positively recapture the payload for the return flight.

5.2.4.2 Power.

a. Platform delivery. The interface between the OTV and payload will be 28 volts with an appropriate disconnect for separation. Power required from an external source for the payload will be supplied through the OTV. During prelaunch operations, 700 watts of power are required by the payload for approximately four hours to bring the gyros up to speed. From liftoff to LEO, through checkout, and through transfer to GEO, the payload will be on internal power and none will be required from the OTV.

b. Servicing mission. The interface between the OTV and payload will be 28 volts with an appropriate disconnect for separation. Power required from an external source for the payload will be supplied through the OTV. During prelaunch operations, 700 watts of power are required by the payload for approximately four hours to bring the gyros up to speed. From liftoff to LEO, through checkout, and through transfer to GEO, the payload will be on internal power and none will be required from the OTV. Likewise, on the return mission to the Shuttle and back to earth, the payload will not require power from the OTV.

5.2.4.3 Communications.

a. Platform delivery. The OTV must provide hard line communication links from the platform to the Orbiter up to the capacity of the Orbiter, as called out in the Space Shuttle Payload Data Handling and Communication Description and Performance document JSC 14241, November 1978. These will be used during checkout of the platform in LEO while still attached to the Orbiter. During transfer from LEO to GEO, data from the platform will be transmitted through the OTV communication system. The OTV communication system must provide an S-band uplink capable of handling 32 kbps with 6.4 kbps command and a return link of 192 kbps with 128 kbps data.

b. Servicing mission. The OTV must provide hardline communication links from the platform to the Orbiter up to the capacity of the Orbiter as called out in the Space Shuttle Payload Data Handling and Communication Description and Performance Document, JSC 14241, November 1978. These will be used during checkout of the service system while still attached to the Orbiter in LEO.

During transfer from LEO to GEO and back to LEO, data from the servicing system will be transmitted through the OTV communication system. The OTV communication system must have the same capability as on the platform delivery mission.

During servicing operations on the platforms, the servicing system will communicate with the ground through an RF link with the platform.

5.2.4.4 Main Propulsion.

a. Platform delivery. The OTV main engine(s) must provide approximately 14,000 ft/sec change in velocity to transfer the platform/OTV combination from LEO to the required position at GEO. The maximum weight of the platform will be 6,895 kg. In addition, the engine must provide an additional 415 ft/sec change in velocity to place the OTV into a debris orbit after delivering the platform. The total thrust of the engines must be between 1000 and 3000 lbf in order to minimize losses and not exceed a g-level of 0.07 during delivery of the platform. The maximum total number of burns required of the engine(s) will be eleven. The main engine(s) should be LH_2/LO_2 in order to meet the performance required within the volume constraints imposed on the OTV by the platform and the Shuttle cargo bay. The Isp should be greater than 415 seconds. The engine should weight no more than 500 lb and be no longer than 70 inches with the nozzle retracted.

b. Servicing mission. The main engine(s) must provide approximately 28,500 ft/sec change of velocity to transfer the servicing system/OTV combination from LEO to the required position at GEO, and return the combination to LEO for rendezvous and dock with the Shuttle if propulsive maneuvers are used throughout the mission. If an aero-braking maneuver is used on the return flight for retro into a phasing orbit for rendezvous with the Shuttle, then the engine(s) need only provide approximately 21,000 ft/sec change in velocity. The total thrust of the engines to minimize losses and not exceed a reasonable g-level during the mission must be between 10,000 and 20,000 lbf. The maximum total number of burns on any one mission will be 10 or less. The engine(s) are reusable and must be good for 10 missions between major overhauls and have an expected lifetime of 50 missions.

In order to meet the performance requirements:

1. The OTV main propellants should be LH_2/LO_2.
2. The Isp should be greater than 460 seconds.
3. The engine should weigh no more than 500 lb and be no longer than 70 inches with the nozzle retracted.

5.2.4.5 Attitude Control System (ACS).

a. Platform delivery. For this mission the ACS requirement on the OTV is approximately 75,000 lb-sec, based on an Isp of 230 sec. Table 5-2 shows the times and weight of N_2H_4 propellant to perform an eight-burn low-thrust delivery mission with subsequent two-burn disposal of the OTV in a debris orbit. GD/C has performed trade-off studies in the past on ACS monopropellants, bipropellants, and cyrogenic bipropellants (LH_2/LO_2) and has found that for this type of delivery mission N_2H_4 has a performance advantage up to a requirement of 250,000 lb-sec.

b. Servicing mission. For this mission, the ACS requirement on the OTV is approximately 61,000 lb-sec, based on an Isp of 230 sec. Table 5-3 shows the times and weight of N_2H_4 propellant to perform a nominal thrust servicing mission, stationkeep while the platform servicing takes place, and return to LEO to rendezvous and dock with the Orbiter. Debris disposal was not considered for ACS requirements presented in Table 5-3.

5.2.4.6 Guidance and Navigation (G&N).

a. Platform delivery. The OTV G&N system must be able to deliver a platform to a rendezvous point 96 km behind and 16 km below the Western Hemisphere or Atlantic constellation locations in geosynchronous equatorial orbit within ±8 km, ready for the final approach phase. The final approach phase must position the platform within ±0.1 km of its desired constellation location, with less than 3 cm/sec residual translational velocity, ±0.05°/sec rotation, and within ±1° of required attitude.

b. Servicing mission. The OTV G&N system must be able to deliver the servicing system to the same rendezvous point as for the platform. In the final approach phase, the OTV will then position the servicing system in the center of the constellation within ±1 m/sec required velocity and ±1° of required attitude for payload separation. After separation, the OTV must maintain an orientation with the engine pointed at the sun within ±5°. In preparation for redocking with the servicing system, the OTV must maintain an orientation 45° to the sun within ±1°, a translational velocity within ±3 cm/sec and ±0.05 deg/sec rotation. Previous studies (On Orbit Assembly Study - USAF 1979) covering docking of spacecraft at GEO have shown these tolerances to be acceptable and achievable.

Table 5-2. Total ACS Impulse for Low-Thrust OTV Mission, Platform Delivery

Event	Time, hrs:min	ACS ΔV, ft/sec	OTV/PL Weight lb	ACS Propellant Requirement N_2H_4, lb
Deploy OTV/Platform		10	58,000	88.0
Transfer, LEO to GEO				
Coast 1	0:50			0.5
Burn 1	0:26			1.2
Coast 2	1:18		54,000	0.8
Burn 2	0:25			1.1
Coast 3	1:30			0.9
Burn 3	0:24			1.1
Coast 4	1:48		51,000	1.1
Burn 4	0:26			1.2
Coast 5	2:18			1.4
Burn 5	0:28			1.3
Coast 6	3:00			1.8
Burn 6	0:30			1.3
Coast 7	4:42			2.8
Burn 7	0:32			1.4
Coast 8	4:42			2.8
Burn 8	1:20			3.6
Deploy Platform		40		128.0
OTV Transfer to Debris Orbit				
Coast 9	0:20			0.2
Burn 9	0:01			--
Coast 10	12:00			7.2
Burn 10	0:01			1.1
Rendezvous and Dock with Debris Depot (Optional)		15	6,000	76.0
				323.8

Total Impulse: ~75,000 lb/sec

Table 5-3. Total ACS Impulse for Round Trip OTV Platform Servicing Mission (No Disposal of Expended Components in Debris Orbit)

Event	Time, hrs:min	ΔV ft/sec	OTV/PL Weight lbs	ACS Propellant Requirement N_2H_4, lb
Deploy OTV/Payload		10	64,000	88.0
Transfer, LEO to GEO				
Coast 1	0:50			0.5
Burn 1	0:15			10.0
Midcourse ACS			35,000	4.2
Coast 2	5:27			3.3
Burn 2	0:07			4.7
Rendezvous, Deploy Payload and Stationkeep (TMS Operations)		40	15,000	128.0
Transfer, GEO to LEO				
Coast 3	11:40			7.0
Burn 3	0:02			1.3
Midcourse ACS			11,000	1.7
Coast 4	5:27			3.3
Burn 4	0:01			0.7
Coast 5	3:00		9,000	1.8
Burn 5	0:01			0.7
Coast 6	4:00		7,000	2.4
Attitude Hold at LEO	0:30			7.9
				265.5

Total Impulse: ~61,000 lb-sec

For the return flight to rendezvous with the Orbiter, the OTV G&N system must be able to place the OTV/servicing system payload in a circular orbit 16 km higher than the Orbiter and 96 km forward of the Orbiter, within ±8 km. During the final approach maneuver, the OTV will maintain the same orientation, velocity, and rotation requirements as stated above for the servicing system docking at GEO, but will be passive during the final closing and docking phase.

5.2.4.7 Environmental Control.

a. Platform delivery. There is no requirement for the OTV to provide environmental control in any form to the platform from prelaunch to delivery to GEO. The platform will require no environmental control before deployment while still attached to the Shuttle. After deployment and during transfer from LEO to GEO, the platform will provide its own environmental control.

b. Servicing mission. There is no requirement for the OTV to provide environmental control in any form to the servicing system from prelaunch to delivery to GEO or on the return mission to earth.

5.2.4.8 Airborne Support Equipment (ASE).

a. Platform delivery. The ASE must be able to provide aft support for the platform in the Shuttle through the OTV and must meet the load requirements for a Shuttle payload during launch to LEO. It must also provide the hard line avionics link from the platform through the OTV to the Shuttle for status monitoring and checkout before separation from the Shuttle.

   The ASE must provide rotation and support for the platform/OTV payload package at LEO for platform deployment and checkout prior to separation from the Orbiter. Figure 5-12 shows the OTV/platform package rotated 75° out of the cargo bay while still attached to the Orbiter, so that the elements on the platform will clear the Orbiter sufficiently when deployed. Analysis on the OTV study has shown that to achieve a rotation angle of 75° the OTV must be rotated about a point forward of Orbiter Station 1296.6. This shortens the available length for the OTV as opposed to the nominal rotation station of 1269.6, which can be used for payloads that have no large elements to deploy while still attached to the Orbiter. This may require an ASE design change to move the rotation point somewhat forward of Station 1269.6.

b. Servicing mission. The ASE must be able to provide aft support for the servicing system in the Shuttle through the OTV and must meet the load requirements for a Shuttle payload during launch to LEO. It must also provide the hardline avionics link from the servicing system through the

OTV to the Shuttle for status monitoring and checkout before separation from the Shuttle.

The ASE must provide rotation and support for the servicing system/OTV combination in LEO for checkout prior to separation from the Orbiter. A rotation angle of 30° from the horizontal is all that is required for checkout and separation.

The ASE must be able to receive the OTV/servicing system on the return mission, position them back in the Shuttle and support them adequately for the return mission to earth.

5.3 SERVICING SYSTEM (TELEOPERATOR)

This section identifies the performance, operations, and support requirements required from the teleoperator to service the platforms.

5.3.1 SERVICING SYSTEM PERFORMANCE. The ground rules for the servicing mission require that the servicing system and OTV be launched on a single 65K Shuttle flight and that the service system be delivered by the OTV to a position near the constellation of six platforms either at the Western Hemisphere location or the Atlantic location. The servicing system will then proceed from the OTV to individual platforms to perform the servicing operations. After servicing of the platforms is complete (for a typical mission three platforms will be serviced on each flight) the teleoperator will return to the waiting OTV. The OTV will then transfer the servicing system to a debris orbit approximately 2000 n.mi. above GEO. Expended propellant bottles and batteries will be jettisoned at this orbit. The OTV will then transfer the servicing system from the debris orbit back to the Shuttle orbit and rendezvous and dock with the Orbiter. The Orbiter will return the OTV and servicing system to earth.

During the servicing operation at GEO the teleoperator must be able to provide a nominal velocity change of 42 ft/sec to transfer from the OTV to one platform, go from the first platform to the second, go from the second to the third platform, and then return to the OTV.

5.3.2 SERVICING SYSTEM ENVELOPE AND MASS. The servicing system with the platform logistics cargo has an allowable envelope 176 inches in diameter by 20 ft long. The total allowable mass of the servicing system with the logistics cargo for the baseline mission is 2306 kg. The maximum mass of the logistics cargo is 1433 kg leaving a maximum allowable mass for the servicing system equal to 873 kg.

The servicing system must be able to accommodate propellant bottles up to 45 inches in diameter. A total of 6 of these must be accommodated on each mission. The total logistics supply volume to be accommodated is 350 ft^3.

The servicing system must be supported off the forward end of the OTV during Shuttle launch and transfer to LEO, and during transfer to and from GEO by the OTV.

5.3.3 SERVICING SYSTEM OPERATIONS. Table 5-4 describes the operations along with their required times for the servicing mission from liftoff in the Shuttle until touchdown. After the Shuttle reaches approximately 160 n.mi. orbit the OTV/servicing system will be rotated 30° from the horizontal in the cargo bay and checked out prior to separation. The OTV will transfer the servicing system to GEO and place it at the required position relative to the six-platform constellation. The servicing system will separate from the OTV and proceed to the first platform to be serviced. It will rendezvous and dock with this platform and remove and replace the depleted N_2H_4 bottle with a full one. The servicing system will then separate from this platform to maneuver and dock with the next platform. At the second platform it will also remove and replace the depleted N_2H_4 bottle. The servicing system will then maneuver and dock with the third and last platform and remove and replace three batteries. After this last servicing operation the servicing system will return and dock with the waiting OTV.

After the teleoperator redocks with the OTV, the OTV transfers the system to a debris orbit approximately 2000 n.mi. above GEO to jettison the expended bottles and batteries. The OTV transfers the servicing system from the debris orbit to the Shuttle orbit where it will rendezvous and dock with the Orbiter. The Orbiter returns the OTV/servicing system to earth.

5.3.4 SUPPORT SUBSYSTEMS. This section identifies the requirements for the servicing system subsystems.

5.3.4.1 Structural. The structure must be able to dock with and be supported by the forward end of the OTV during boost in the Shuttle and during LEO to GEO transfer and return. Figure 5-16 shows the forward end of the OTV. The interface with the OTV and the servicing system itself must withstand the launch and operational dynamic environment defined in the Space Shuttle System Payload Accommodations document JSC 07700, Vol. XIV, Rev. F, Change 32, and the STS user handbook. The docking and support mechanism on the servicing system forward of OTV Station 457, as shown in Figure 5-16, is chargeable to the servicing system. The structure aft of Station 457, to support the servicing system is chargeable to the OTV. There will be one male docking mechanism on the servicing system that will be used to dock with both the OTV and the platform. The female docking mechanism attached to the forward end of the OTV will be the same as the one on the platform. An interface disconnect panel accommodating data/communication links must be part of the docking/support system so that the servicing system can be checked out while still attached to the orbiter in LEO prior to transfer to GEO. Figure 5-17 has the features that should best meet the docking and support requirements for docking to both the platform and OTV.

Table 5-4. Servicing Flight Operations, Atlantic Constellation

Event	Start Time (hrs/min)	Event Time (min/sec)	Elapsed Time (hrs/min)
Lift-Off	0:00	–	0:00
SRB Separation	0:02	–	0:02
MECO	0:08	–	0:08
ET Separation	0:08	–	0:08
OMS - 1 burn (231 ft/sec)	0:09	1:50	0:11
Ascent Coast (50 by 150 n.mi.)	0:11	32:20	0:44
OMS - 2 Burn Circularization	0:44	1:45	0:46
Reconfigure Orbiter Software	0:48	12:00	1:00
Enable TMS Discretes	1:00	–	–
Orbiter IMU Alignments	1:00	20:00	1:20
Open Payload Bay Doors	1:20	5:00	1:25
Deploy Radiators/Activate Cooling	1:25	2:00	1:27
Transfer OTV Electrical Power	1:27	–	–
OTV Checkout and Systems Verification	1:28	30:00	1:58
TMS Checkout and Systems Verification	1:28	30:00	1:58
Update OTV Navigation	1:58	0:30	1:59
Reorient to OTV Deployment Attitude	1:59	15:00	2:14
Rotate OTV	2:14	4:30	2:19
Activate OTV-Orbiter RF Link	2:19	5:00	2:24
Final OTV/TMS Checkout	2:24	13:00	2:37
Separate OTV	2:37	–	–
OTV Separation Coast	2:37	4:00	2:41
Activate OTV	2:41	1:00	2:42
Coast to First Nodal Crossing	2:42	38:00	3:20
Coast to Phasing Orbit Burn	3:20	180:00	6:20
OTV Phasing Burn (third Nodal Crossing) Δv = 3025 ft/sec	6:20	5:56	6:26
Phasing Orbit Coast	6:26	135:00	8:41
OTV Transfer Orbit Insertion (Δv = 5037)	8:41	7:36	8:49
Coast to Midcourse	8:49	180:00	11:49
Midcourse Correction (Δv = 50 ft/sec)	11:49	0:21	11:49
Coast to GEO	11:49	135:00	14:04
OTV GEO Insertion Burn (15°W, 60 n.mi. range, Δv = 5825 ft/sec)	14:04	6:13	14:10
Search and Acquire Constellation	14:10	60:00	15:10
Initial Rendezvous Burn (48 ft/sec)	15:10	0:21	15:10
Coast	15:10	300:00	20:10
Perform Braking Burns (26 ft/sec)	20:10	120:00	22:10
Rendezvous (Center Position, 15°W)	22:10	–	–

Table 5-4. Servicing Flight Operations, Atlantic Constellation (Contd)

Event	Start Time (hrs/min)	Event Time (min/sec)	Elapsed Time (hrs/min)
Activate TMS	22:10	30:00	22:40
TMS Burn (2 ft/sec)	22:40	–	–
Coast	22:40	300:00	27:40
Braking Burns (4 ft/sec)	27:40	60:00	28:40
TMS Rendezvous with Platform 8	28:40	–	–
Await Lighting	28:40	0-14 hours	35:40 (avg)
Maneuver to Docking Position	35:40	60:00	36:40
Dock TMS with Platform 8	36:40	–	–
Remove Empty N_2H_4 Bottle 3	36:40	16:00	36:56
Install Full N_2H_4 Bottle 3	36:56	16:00	37:12
Operations Margin	37:12	180:00	40:12
Separate TMS/Platform 8	40:12	–	–
Maneuver to Transfer Position	40:12	30:00	40:42
TMS Transfer Injection (6 ft/sec)	40:42	–	–
Coast	40:42	120:00	42:42
Braking Burns (10 ft/sec)	42:42	60:00	43:42
TMS Rendezvous with Platform 9	43:42	–	–
Maneuver to Docking Position	43:42	60:00	44:42
Dock TMS with Platform 9	44:42	–	–
Remove Empty N_2H_4 Bottle 3	44:42	16:00	44:58
Install Full N_2H_4 Bottle 3	44:58	16:00	45:14
Operations Margin	45:14	180:00	48:14
Separate TMS/Platform 9	48:14	–	–
Maneuver to Transfer Position	48:14	30:00	48:44
TMS Transfer Injection (2 ft/sec)	48:44	–	–
Coast	48:44	300:00	53:44
Braking Burns (4 ft/sec)	53:44	60:00	54:44
TMS Rendezvous with Platform 12	54:44	–	–
Await Lighting	54:44	300:00	59:40
Maneuver to Docking Position	59:40	60:00	60:40
Dock TMS with Platform 12	60:40	–	–
Remove Battery 1	60:40	16:00	60:56
Install Battery 1	60:56	16:00	61:12
Remove Battery 2	61:12	16:00	61:28
Install Battery 2	61:28	16:00	61:44
Remove Battery 3	61:44	16:00	62:00
Install Battery 3	62:00	16:00	62:16
Operations Margin	62:16	180:00	65:16
Separate TMS/Platform 12	65:16	–	–
Maneuver to Transfer Position	65:16	30:00	65:46

Table 5-4. Servicing Flight Operations, Atlantic Constellation (Contd)

Event	Start Time (hrs/min)	Event Time (min/sec)	Elapsed Time (hrs/min)
TMS Transfer Injection (6 ft/sec)	65:46	–	–
Coast	65:46	120:00	67:46
Orient OTV	67:43	3:00	67:46
Braking Burns (10 ft/sec)	67:46	60:00	68:46
Rendezvous with OTV	68:46	–	–
Maneuver TMS to Docking Position	68:46	1 hour	69:46
Dock TMS with OTV	69:46	–	–
OTV Burn to Debris Orbit (Δv = 210, low thrust)	69:46	1:30	69:48
Coast to +2000 n.mi.	69:48	720:00	81:48
OTV Burn to Circularize (Δv = 205, low thrust)	81:48	1:20	81:49
Jettison Expended Bottles and Batteries	81:49	–	–
Coast to Nodal Crossing	81:49	0-12 hours	87:44
OTV Burn to Return Transfer Orbit (Δv = 5726 ft/sec)	87:49	3:14	87:52
Coast to Midcourse	87:52	180:00	90:52
Midcourse Correction (Δv = 50 ft/sec)	90:52	0:21	90:52
Coast to LEO	90:52	75:00	93:07
LEO Phasing Orbit Burn (Δv = 3741 ft/sec)	93:07	1:32	93:09
Phasing Orbit Coast	93:09	1.5-3 hours	95:24
LEO Circularization Burn (20 n.mi. above, 130 n.mi. in front, Δv - 4990, $\Sigma\Delta v$ = 8231 ft/sec)	95:24	1:25	95:25
Orbiter Rendezvous with OTV	95:25	129:00	97:34
Vent OTV LH_2	97:34	30:00	98:04
Vent OTV LO_2	98:04	30:00	98:34
Disable OTV RCS	98:34	–	–
Capture OTV with RMS	98:34	2:30	98:37
Return OTV to Cradle	98:37	17:30	98:54
Orbiter Thermal Conditioning and Reentry Phasing	98:54	<12-15 hours	110:54
Orbit Determination	110.54	1:30	110.56
Orbiter IMU Alignment	110:56	20:00	111:16
Close Payload Bay Doors	111:16	20:00	111:36
Orient to Deorbit Attitude	111:36	10:00	111:46
OMS Deorbit Burn (338 ft/sec)	111:46	2:00	111:48
Orient to Entry Attitude	111:48	10:00	111:58
Coast to Entry Interface	111:58	8:00	112:06
Entry Interface (400,000 ft)	112:06	–	–
Entry Flight Operations	112:06	24:16	112:30
TAEM-Landing Operations	112:30	5:01	112:35
Touchdown (4.7 days)	112:35	–	–

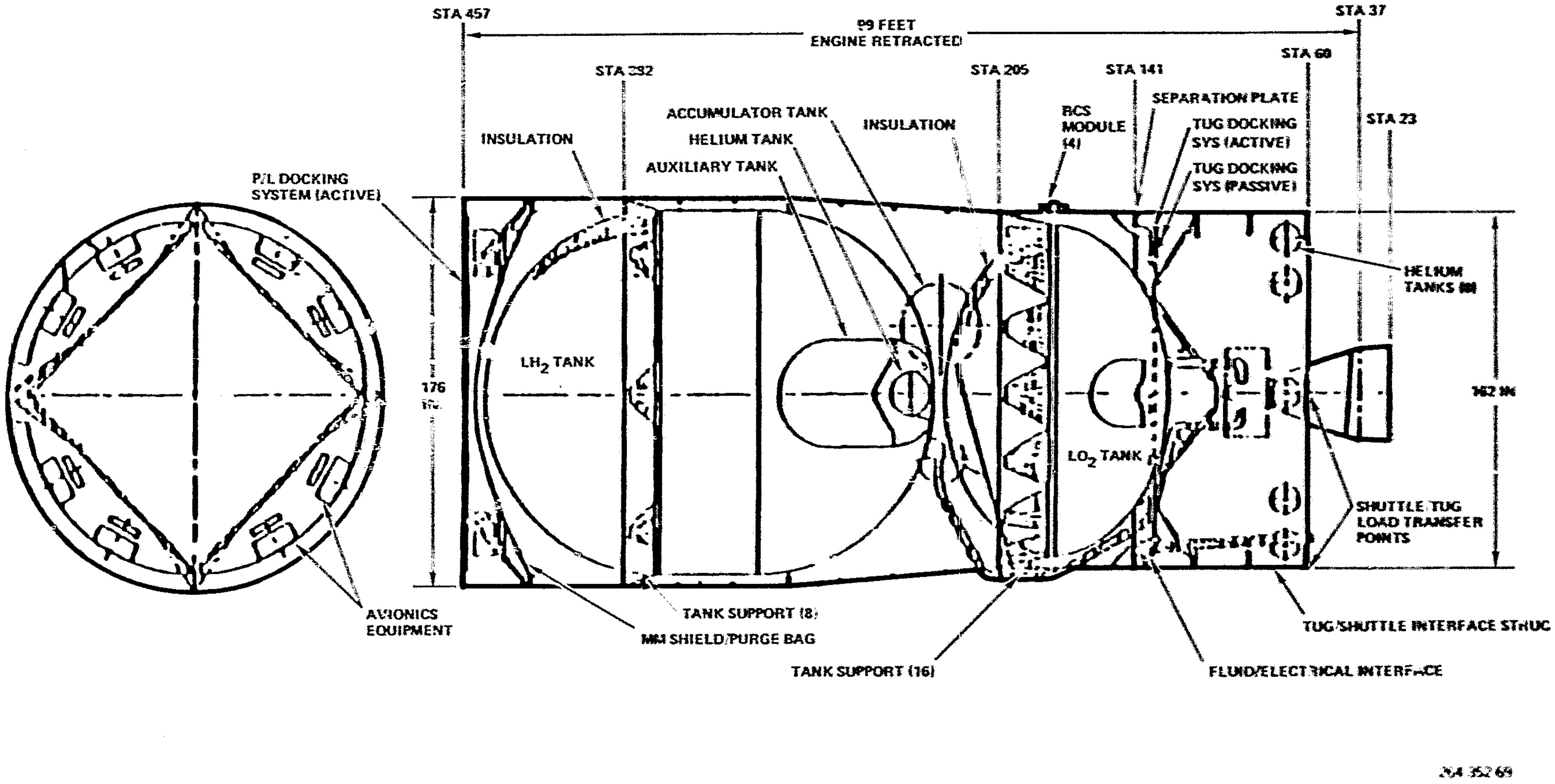

Figure 5-16. OTV Configuration

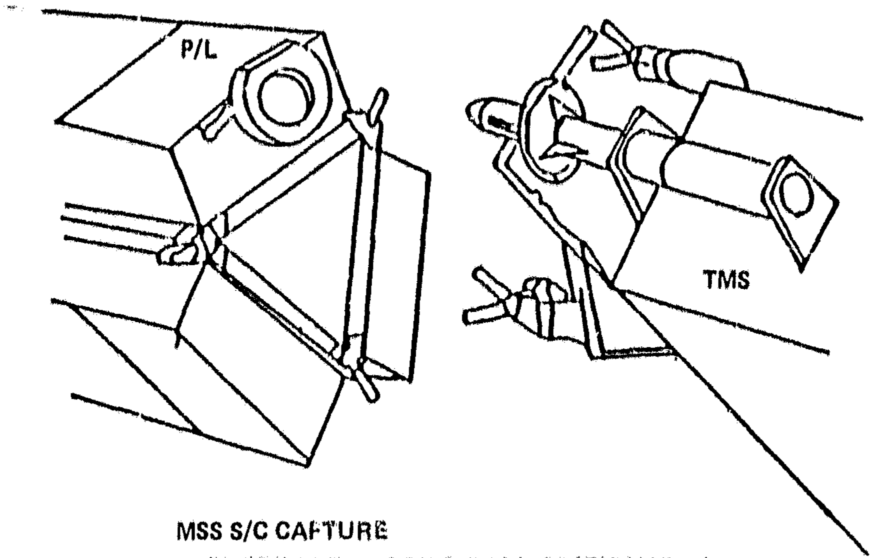

Figure 5-17. TMS/Payload Docking Approach

The servicing system must have a manipulator arm to transport the logistics resupply units from the teleoperator to the platform, similar to the arrangement shown in Figure 5-18. The nominal reach of the manipulator arm must be 20 feet from the docking interface to attach or detach the logistics supplies from their platform mounts. The manipulator arm must also be capable of attaching or detaching the logistics supplies from their stowed position on the teleoperator. The end effector on the manipulator arm can be similar to the RMS end effector 1.

The structure must be able to support the logistics supplier for both the normal and abort conditions in the Shuttle. It must also make provisions for supplies detachment and reattachment to the teleoperator by means of the manipulator arm.

5.3.4.2 Power. The logistics supplies and the platform have no power requirements from the servicing system. Table 5-4 portrays the events and their times for the servicing mission which can help establish the energy requirements to be supplied by the servicing system.

5.3.4.3 Avionics. There will be a disconnect panel at the interface between the servicing system and the OTV to provide hardline capability for service system checkout while still attached to the shuttle in LEO. It will also have a command capability to the servicing system for operations while attached to the OTV, such as jettisoning spent bottles and batteries at the debris orbit.

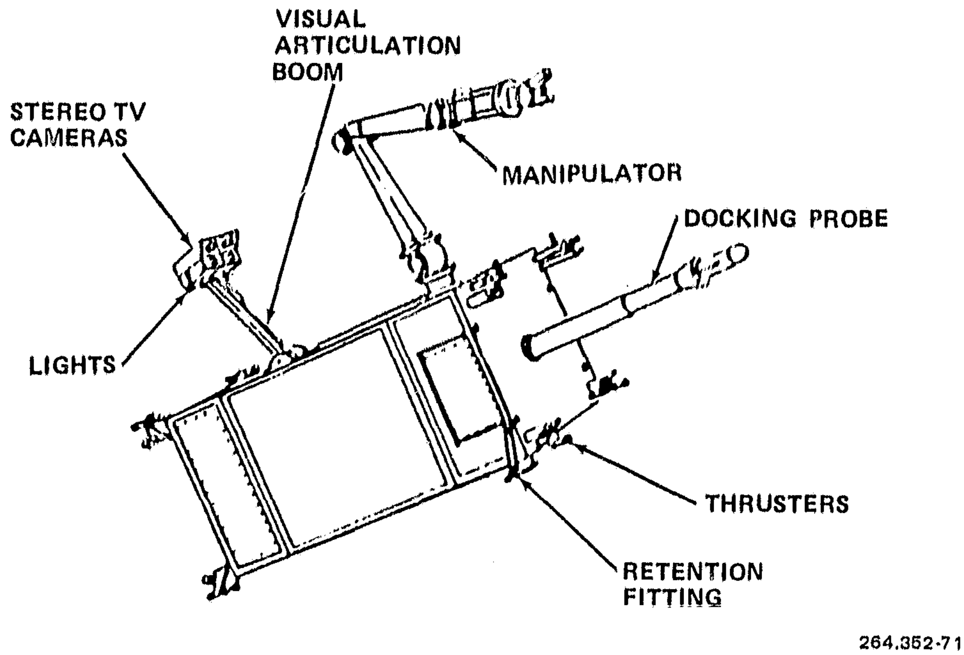

Figure 5-18. Manipulator Arm Concept

a. Command and control. After reaching GEO and being positioned in the middle of the constellation by the OTV, the commands and control for the servicing system will come from the ground through each platform to be serviced via RF link to the teleoperator. The communication link will provide the control for the servicing system during its servicing operation, including separation from the OTV, rendezvous and docking with the platform, removing and replacing logistics equipment from the teleoperator to the platform and back again, separating from the platform, and rendezvous and docking with the OTV.

b. Video. Stereo TV cameras are required as part of the servicing system in order to provide the depth of detail necessary to successfully control the removal and replacement of the logistics supplies on the platform. The cameras must be placed on an articulated boom in order to observe the several areas where the activities must be performed. Figure 5-18 illustrates this type of arrangement with the stereo TV cameras mounted on the articulated boom.

c. Lighting. The servicing system must provide artificial lighting to illuminate the interfaces between the manipulator arm and the logistics equipment as well as the interfaces between the logistics equipment and the teleoperator and the platform. Figure 5-18 illustrates a possible arrangement for the location of lights so that they can illuminate the required areas where the cameras are needed for control of the manipulator arm.

d. Rendezvous radar system. The servicing system requires a rendezvous radar system for rendezvous and docking with both the platform and the OTV. The capability must be such that this maneuver can be successfully completed remotely with command and control from the ground.

5.3.4.4 Propulsion/Reaction Control System. Functionally, the teleoperator propulsion system must be able to provide translation and rotation in three axes to perform the servicing mission.

The change in velocity requirement is approximately 42 ft/sec for translation transfer. Additional energy will be needed for stabilization and control.

In order to minimize exhaust product contamination of the platforms, it would be desirable to use cold gas as the propellant in the propulsion system.

5.3.4.5 Environmental Control. There are no environmental requirements imposed on the servicing system from the logistics resupply components, the platform, or the OTV. The resupply components must provide their own environmental control.

5.3.4.6 Stabilization and Control System. The teleoperator stabilization system must be able to maintain the required centerline orientation within ±2°, the required roll orientation within ±5° and velocity control within ±3 cm/sec. These numbers were derived from analysis performed on the on-orbit assembly study for the USAF and an initial analysis of docking disturbances with the platform.

5.4 CONCLUSIONS AND RECOMMENDATIONS

This is the first cut at the requirements imposed on the Orbiter, OTV, and teleoperator maneuvering system (TMS) by one of the alternative operational geostationary platforms concepts. From this initial analysis, it appears that these three transportation system elements can meet the platform requirements. Because of the conceptual nature of this study, the requirements have only been analyzed at the system level. More work and greater depth of detail will be needed in follow-on study phases for a more comprehensive definition of these requirements. In addition, the other alternative concepts must be analyzed to the system level to determine any additional requirements on the three transportation elements.

PART II

EXPERIMENTAL GEOSTATIONARY PLATFORMS

SECTION 6

TASK 3A: EXPERIMENTAL GEOSTATIONARY PLATFORMS

Early in this study, as operational geostationary platform concepts and their corollary technology requirements began to emerge (Part I of this report), NASA/MSFC anticipated the need for an experimental geostationary platform to demonstrate the advanced technologies, systems, and uses required to pave the way for the operational platforms of the 1990s. These technologies would advisedly be demonstrated early in the geostationary platform program to verify concept feasibility and justify further program planning.

As an adjunct to Task 3, a preliminary feasibility assessment of an experimental platform (Task 3A) was therefore authorized, to be performed and reported on early in Task 3, prior to definition of the selected operational platform concepts and without the benefit of Task 3 and Task 4 results. This section of the final report, Part II of Volume II, documents the results of this experimental platform feasibility study.

As presently conceived, the experimental platform would be placed in geostationary orbit in about 1988, probably over the Western Hemisphere at about 110°W. Upon completion of experiments and demonstrations relevant to CONUS interest, the platform could be moved to an Atlantic position (15°W) for continuation of demonstrations and validations related to international communications systems. Such planning is flexible and dependent on the ultimate choice of payloads selected for this platform during Phase B. Preferably, payloads would be limited to those that could be accommodated with the platform in a single Shuttle flight, payloads that would demonstrate payload and platform technologies that were feasible, practicable, promised the greates benefit overall, and that involved technical risks sufficiently above current satellite technology to warrant the use of public funds.

Successful proofing by the experimental platform of advanced communications systems and technologies and of platform deployment/assembly and control technologies would enable inclusion of such technologies in the design of the 1990s operational platforms, ultimately relieving the geostationary orbital arc and spectrum saturation problems, and lowering communications costs to the user.

6.1 MISSION OBJECTIVE

The primary objective in placing an experimental geostationary platform in orbit will be to demonstrate the technologies, systems, and uses necessary to pave the way for operational platforms of the 1990s. Specifically, the experimental platform (E/P) should:

a. Clearly demonstrate a significant step toward operational platforms in both payload and platform technologies, systems, and uses. Communications experiments should be directed toward more efficient use of the frequency spectrum, increased capacity, new services, and greater hardware capability. Platform experiments should demonstrate better packaging and deployment techniques, structural advances, modular buildup, servicing, orbital transfer techniques, and rendezvous and docking technology.

b. Demonstrate realistically and conservatively enough to instill confidence and interest in user participation.

c. Demonstrate technologies sufficiently advanced over current satellite capabilities to warrant the use of public funds, technologies that are attractive from the users standpoint but that involve risks that the users would be reluctant to fund.

As a corollary to demonstrating the technologies, systems, and uses necessary to pave the way for operational platforms, it is essential that the experiments be user-oriented, with results that clearly point to the benefits to be gained by adjusting to advanced communications and platform technologies.

6.2 SCOPE OF TASK

The basic purpose of this task is to assess the feasibility of an experimental platform to perform a preliminary evaluation of the practicability and capabilities of an E/P from the standpoint of technology, schedule, and cost. If the results appear encouraging, a more detailed analysis could be the subject of a follow-on study.

By direction, the effort expended on this task is limited to conceptual design and feasibility analysis. The task is not intended to produce a recommended design.

6.3 GROUND RULES AND GUIDELINES

Specific ground rules and guidelines constraining this task are as follows:

a. Launch in 1987 or 1988.

b. Investigate concepts requiring no more than two Shuttle flights, with a single Shuttle flight as a design goal.

c. LEO-to-GEO transfer vehicle to be included as part of the one or two Shuttle flight payloads.

d. Centaur, IOTV, and IUS to be considered for LEO-to-GEO transfer.

e. Platform payloads to satisfy the following mission objectives:

   1. Clearly demonstrate communications and platform technologies applicable to follow-on operational platforms.

2. Demonstrate multidiscipline operation and support capability.

3. Demonstrate servicing and modular growth capability to extent practical.

f. Basic systems (structure, power, etc.) must be stepping stones to operational platforms.

g. Platform/payloads weight to be compatible with transfer vehicle and Shuttle (65,000 lb) performance.

h. Solar array power.

i. Orbital location at 110°W, capable of moving to 15°W.

j. No platform-to-platform rendezvous and docking capability required.

k. Life:

1. One to two-year test and experiment phase, proofing advanced concepts.

2. Open-ended minimum five-year life available for operational experiments, exploring user reaction to new types of service, trial use of expanded capabilities, investigating acceptance, and other ideas related to advances in user applications.

6.4 INPUT DATA

Input data used to develop experimental platform concepts in this task include the results of preceding tasks, COMSAT data, NASA/MSFC data, and STS publications.

a. Payloads. Primary (communications) candidate payloads for operational geostationary platforms were basically derived from NASA studies (Aerospace Corp., F. E. Bond; COMSAT Corp., W. Morgan) and expanded in Task 1 as input data for Tasks 2 and 3. From this data base, experimental communications payloads were selected for this study.

Secondary (DoD and science) candidate payload listings were supplied by NASA/MSFC. These were updated and prioritized for experimental platform payload selection.

b. Platform concepts. Platform structural elements, systems, and configurations were derived from Task 3 - operational platform definitions.

c. SRT. All candidate payloads and technologies were evaluated with reference to Task 4 - requirements for supporting research and technology.

d. OTV. LEO-to-GEO orbital transfer vehicle data were taken from NASA/MSFC Technical Memo PD01-79-70, "STS Upper Stage Geosynchronous Payload Capability" and subsequent updates PD01-79-72 and PD01-80-16.

e. Shuttle. Shuttle requirements, interfaces and capabilities data were taken from JSC 07700, Vol. XIV, "Space Shuttle System Payload Accommodations," Revision F, Change 32.

6.5 STUDY PLAN

Having identified advanced technology requirements and the need for an experimental geostationary platform, it becomes necessary to evaluate the feasibility of such a platform. The methodology used in this preliminary feasibility assessment involves a three-step process:

a. Analyze operational and technical elements of the experimental platform concept:

    1. Identify candidate technologies.

    2. Identify candidate payloads.

    3. Analyze mission options.

    4. Consider structural concepts.

    5. Consider growth potential.

    6. Identify subsystem requirements.

b. Develop candidate platform configurations and characteristics, including:

    1. Payloads.

    2. Antenna and support subsystems.

    3. Power requirements.

    4. Platform weight.

c. Evaluate candidate concepts for feasibility.

In following this process, the intent is to investigate a range of concepts and options, including various combinations of primary (communications) and secondary (DoD and science) payloads with different payload densities, packaging requirements, and power requirements; capabilities with respect to single and multiple Shuttle flights; and orbital transfer vehicle options. The results are intended to give NASA not only an evaluation of feasibility of an experimental platform, but an insight into the capabilities, program interfaces, and simplicity/complexity of such a platform.

6.6 ANALYSIS AND RESULTS

6.6.1 <u>CANDIDATE TECHNOLOGIES</u>. Since technology demonstration is the primary objective of the experimental platform mission, all platform and communications technologies considered necessary for development of future operational platforms were first identified.

Table 6-1 lists the platform technologies. Some of these, such as multidiscipline payload support and growth by modularity, were dictated by the study ground rules and guidelines. Others are inherent to the deployment-at-LEO and transfer-to-GEO concept as distinguished from the space fabrication or assembly concepts. The remainder are related to large space structures in general, such as large structure dynamics and active stabilization.

Table 6-1. Platform Technologies to be Demonstrated for Future Operational Platform Use

Feasiblity of multidiscipline payload support
Growth by modularity
Postdeployment docking, payload addition, and servicing at LEO with the TMS
Docking, payload addition, and servicing at GEO with the TMS
Platform system and subsystem functioning at GEO
Integrated structure and payloads for deployment
Deployment and checkout at LEO
Low coefficient of thermal expansion (CTE) structures
Active stabilization
Large structure dynamics
Transfer of large structures from LEO to GEO
Autonomous stationkeeping, housekeeping, and redundancy management
Advanced component and subsystem flight qualification

The list is not prioritized. The platform technologies were given equal consideration for inclusion on the experimental platform concepts, limited primarily by feasibility or practicability with respect to the basic study constraints such as the single or two-Shuttle flight mission restriction.

Candidate communications systems technologies are listed in Tables 6-2 for platform-related communications technologies, and Table 6-3 for nonplatform related communications technologies. Both lists are compilations of advanced communications technologies identified by NASA centers, COMSAT, and General Dynamics Convair, and have not been prioritized. All have been given equal consideration in developing experimental communications payloads consistent with weight, volume and design constraints of the platform mission. As the experimental geostationary platform program progresses, technologies and payloads will undoubtedly be prioritized and selected by multiagency concurrence to derive the greatest benefit from the program.

Table 6-2. Advanced Communications Technology Candidates, Platform Related

---

Large reflectors

- Narrow beams
- Closely-packed beam isolation testing
- Beam reconfigurability of antenna coverage, C and Ku band
  - Use of variable power dividers (VPD)
  - Use of variable phase shifters (VPS)
- Multibeam frequency reuse antennas (MBFRA) - low sidelobe reflector
- High EIRP and receive G/T

On-board switching and processing

- Semistatic switch, 6/4 GHz - 8 by 8
- Dynamic switch (satellite-switched TDMA), 14/11 and 30/20 GHz - 16 by 16, 10 by 10, 25 by 25
- On-board regeneration

Common service support systems concepts

Connectivity to other payloads (RF or baseband)

Assessment of interference levels between platform payloads

Table 6-3. Advanced Communications Technology Candidates, Nonplatform Related

---

Advanced power amplifiers

- Solid-state amplifiers
- High-power traveling wave tubes (TWT)
- Multilevel traveling wave tube amplifiers (TWTA), 30/20 GHz

Independent satellite gain control for uplinks and downlinks

Satellite power control for uplinks and downlinks

Satellite signal fade monitoring

Earth station up-path power control

Earth station adaptive depolarization correction

Forward error correction coding

Digital techniques and modulation

- Delta-modification, linear predictive coding
- High bit-rate modems

The communications technologies, together with the platform technologies listed in Table 6-1, provide the basic requirements for platform payloads with which to implement the technology experiments.

It should be noted that most of the advanced technologies listed herein will be demonstrated on the experimental platform. Some require prior research and development. These, together with the technologies which cannot be demonstrated on the experimental platform, have been listed and planned for as discussed in Task 4, Part I of this report.

6.6.2 CANDIDATE PAYLOADS. Payloads for the experimental geostationary platform have been categorized by NASA/MSFC as either primary or secondary payloads. Primary payloads are defined as those necessary to demonstrate the technologies, systems, and uses for application on future operational geostationary platforms.

Secondary payloads are defined as those that can be accommodated on the platform to demonstrate multidiscipline payload support, and which also serve to fulfill DoD and the science community's test and experiment needs.

If practical, all payloads are preferred to have postexperiment utility, whether they are primary or secondary payloads. A lightning mapper, for example, having proven the validity of the concept, could continue to serve the user community for the duration of platform life, spotting most probable fire locations in the area of coverage.

6.6.2.1 Primary Payloads. Platform technologies are basically inherent in the design of the platform itself, and do not generally require "payloads" to demonstrate their validity. All of the platform technologies listed in Table 6-1, both hardware and operational have been accommodated in the platform concepts discussed later in this section of the report.

Candidate communications payloads have been selected by NASA and COMSAT in five frequency bands, to support demonstration of the communications technologies listed in Tables 6-2 and 6-3. These candidate payloads are summarized in Table 6-4. They include multibeam frequency reuse antennas in all bands except the interplatform links, frequencies from 1.5 to 32 GHz, antenna diameters from 1 to 10 meters, shaped and spot beams, reconfigurability, and multiple access switches.

The 30/20 experiment as shown in Table 6-4 has been defined by NASA with two alternatives:

a. Alternative #1 with two 4-meter reflectors and a 10 by 10 switch for trunking.

b. Alternative #2 with five additional 1-meter reflectors and a 25 by 25 switch for customer premise services (CPS).

Table 6-4. Candidate Communications Payloads

Payload	Antenna	Beam	Frequency (GHz)	Transmitter	Receiver
C-Band, Point-to-Point, Dual Polarized	5 to 10m	MBFR	4	X	
Hemispheric and Zoned Beams	4m	MBFR	6		X
Global Coverage	Horn		4	X	
Global Coverage	Horn		6		X
Ku-Band, Point-to-Point, Dual Polarized					
4 Spot Beams to U.S. East Coast	4m	MBFR	14/11	X	X
4 Spot Beams to W. Europe	4m	MBFR	14/11	X	X
10 by 10 SS/TDMA Matrix Switch					
L-Band, Sea Mobile					
Global Coverage	4-Helix		1.6/1.5	X	X
Global Coverage	Horn		6/5	X	X
Spot Beams to Specific Fishing Grounds and Sea Lanes	5m	MBFR	1.6/1.5	X	X
30/20 Experiment					
Alt 2 { Alt 1 { 10 Spot Beams to CONUS	4m	MBFR	30/20	X	X
2 Scanned Beams to CONUS	4m	MBFR	30/20	X	X
10 by 10 Switch					
25 Beam Coverage of CONUS	1m (5)	MBFR	30/20	X	X
25 by 25 Switch					
Interplatform Links					
Mechanical Gimbal Mount	2m	SB	25	X	
Mechanical Gimbal Mount	2m	SB	32		X

The payloads listed in Table 6-4 can be used for a multiplicity of advanced technology demonstrations, including frequency selective sub-reflectors and surfaces.

More detailed descriptions of the 6/4 and 14/11 GHz payload systems, characteristics, and technologies to be demonstrated are given in Figures 6-1 and 6-2, and in Table 6-5.

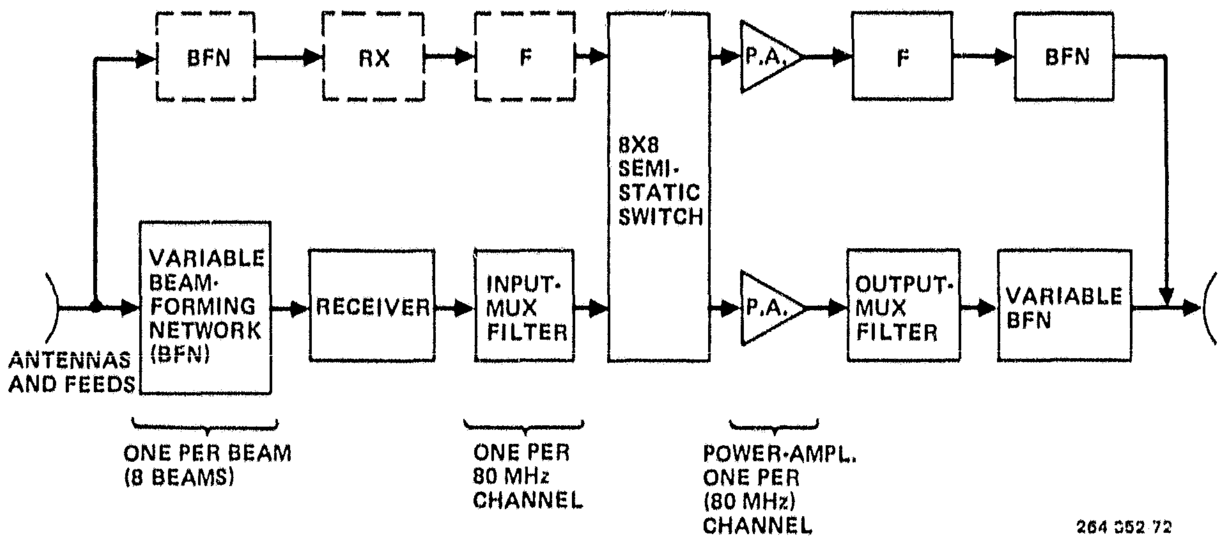

Figure 6-1. C-Band Communications System

The sea-mobile systems concept for the experimental platform is as follows:

a. L-band; 1.6/1.5 GHz (transmit/receive); RF bandwidth; 14 MHz.

b. Four beams.

c. Global beam coverage; four-helix array; TWTA.

d. Frequency reuse: 2 zone beams; 1 hemi beam, 6-meter multibeam reflector antenna with offset feed assembly; solid-state amplifiers, (or 6-meter phased array of helices).

e. Four transponders (7 MHz each).

f. On-board processing and switching.

g. Connectivity to and from shore stations via the C-band point-to-point communications payload.

h. Operational tests: (examples) communications from shore to ships in zones and hemi area coverages (and return); communications to small ships with smaller antennas (G/Ts and EIRPs).

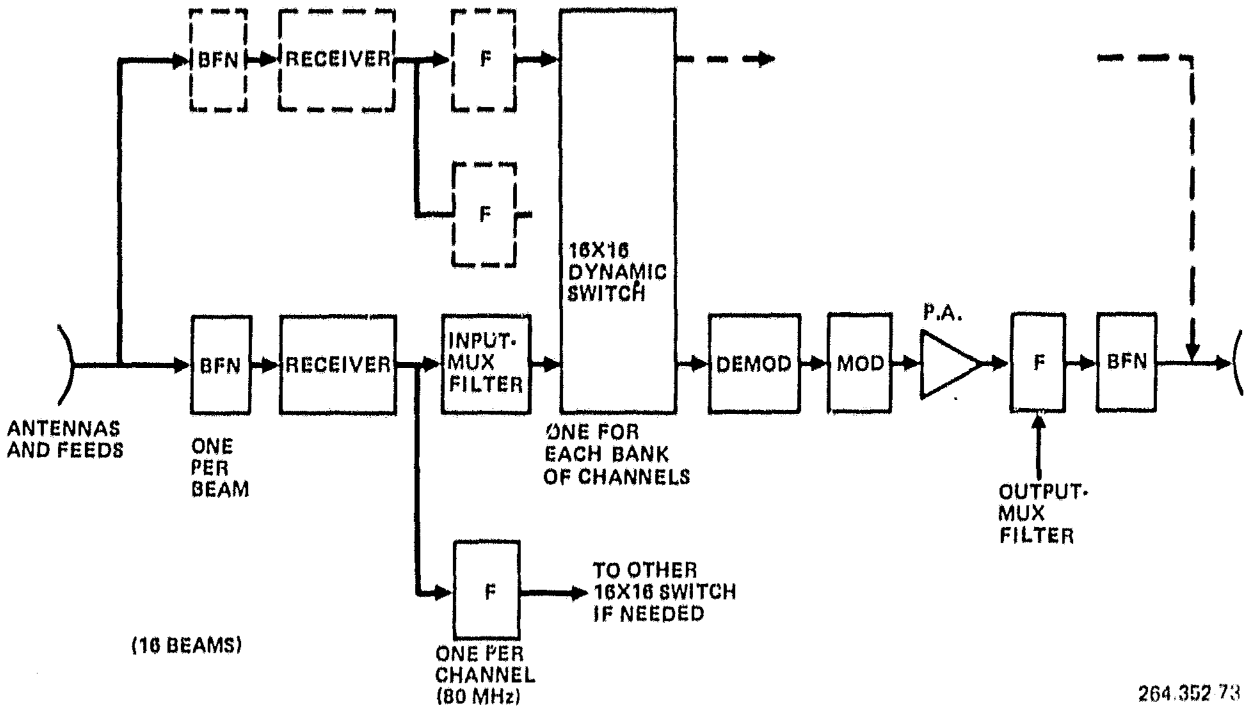

Figure 6-2. Ku-Band Communications System

Typical operational coverages from 15°W are shown in Figure 6-3. The system, including connectivity with the C-band system through the processor and RF switch, is shown in Figure 6-4.

The 30/20 GHz system concepts identified by NASA for the experimental platform are summarized here by characteristics. Definition of the two alternatives is not firm as yet:

a. Alternative #1: (CONUS)

1. Demonstrate heavy trunk communications.
2. 10 spots (0.3°); (link performance, etc.) (2 groups of 5 spots).
3. 10 by 10 dynamic switch.
4. SS-TDMA (500 Mbps).
5. C/N_0 (down) - 104.5 dB-Hz.
6. Link availability 99.9 percent (site diversity).
7. Two 4-meter dishes (one for each group).

Table 6-5. Candidate C-Band and Ku-Band Payloads

Fixed Service	6/4 GHz	14/11 GHz	Remarks
Antenna	5m (1°)	4m (0.5°)	Higher EIRP, G/T
Coverage	Area	Spot	
In-Orbit Beam-Shaping Reconfigurability	Extensive	Limited	
Variable Power Dividers/Phase Shifters (VPD/VPS)	Yes	No	
Low Sidelobe Reflector	Yes	Yes	
Number of Beams*	8	16	Frequency-Reuse (Dual-Pol)
Modulation/Access	FDM/FM/FDMA	SS-TDMA**	**120 Mbps
IF Switch	8 by 8 (Semistatic)	16 by 16 (Dynamic)	
Demodulation/Remodulation	No	Yes	
Baseband Switch	—	Possible	
Solid-State Amplifiers	Yes	Yes	
Tests			
Beam Shaping	Yes	No	
Beam-to-Beam Interference	Yes	Yes	
Power Control (E-S)	—	Yes	At Earth Station (E-S)
Adaptive Depol. Control (E-S)	—	Yes	At Earth Station (E-S)

**At 6/4 GHz: 2 Hemi + 6 Zones (Opposite Polarization)

At 14/11 GHz: 8 Sets of Dual Pol Spots; 4 over North, Central, South America; 4 over Western Europe, Middle East, Africa

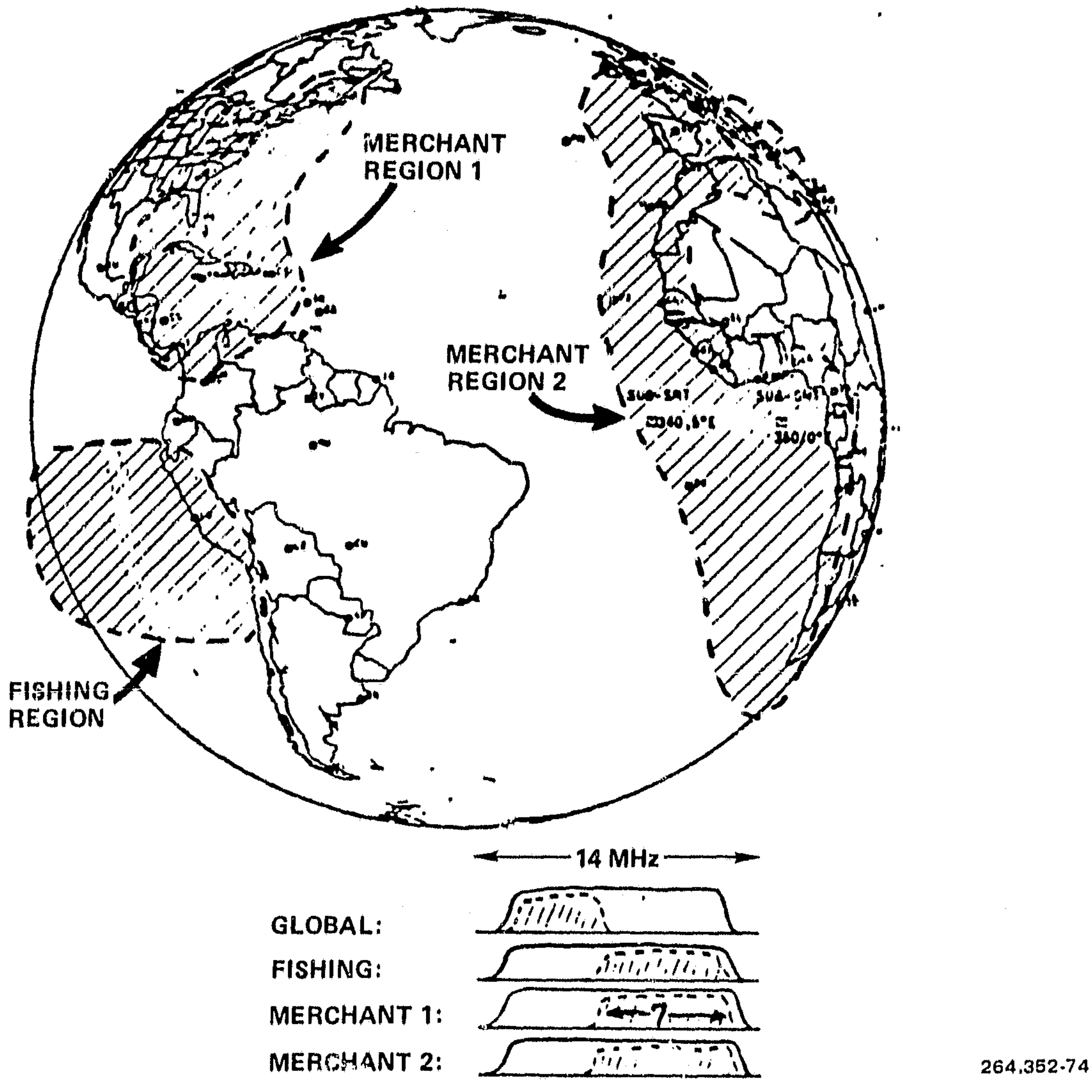

Figure 6-3. L-Band Sea Mobile Coverage - Examples of Shaped Beams Frequency Reuse

b. Alternative #2: (CONUS): high volume trunking (HVT) and customer premises service (CPS) includes Alternative #1 plus the following:

1. 25 spots (1°).
2. 25 by 25 dynamic switch.
3. SS/TDMA: 200 Mbps.
4. C/N_0 (down): 96 dB-Hz.
5. Link availability: 99.5 percent (no diversity).
6. Five 1-meter dishes (each producing 5 beams).

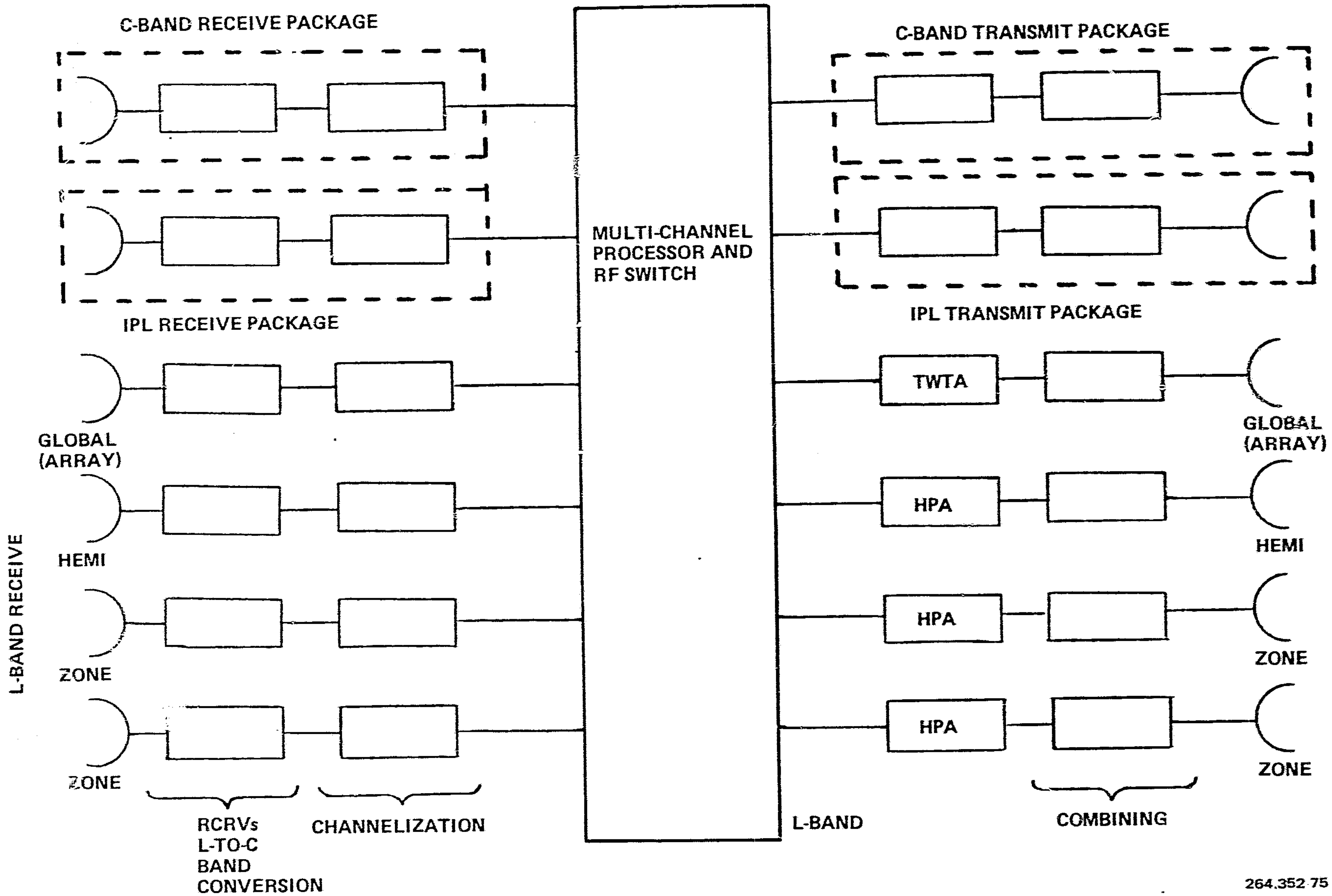

Figure 6-4. Sea Mobile Payload Concept

One of the more important tests to be performed using the 30/20 GHz system is that of beam isolation. For this test, the followi… …ould be included in the 10-beam trunk case:

a. One beam pointing at Boston.

b. One beam pointing at New York (opposite polarization).

c. One beam pointing at Washington (same polarization as Boston).

The test will determine/demonstrate how close tw… …pot beams can be placed, with a beam-to-beam interference level evaluatior

There are no definitive plans for demonstrating IPL systems between the experimental geostationary platform and other satellites now being planned or on the drawing board. Should the IPL experiment become a reality, system characteristics would typically be:

a. IPL system capability between the experimental platform and Satellite X, preferably with about 90° longitude separation.

b. 5-meter transmit/receive tracking antenna.

c. 23 GHz carrier frequency in one direction; 32 GHz carrier frequency in the other direction.

d. 1 GHz bandwidth each direction.

e. 10-watt TWTA.

f. Tests: antenna pointing, tracking, and transmit/receive operations.

A simplified schematic of the IPL system is shown in Figure 6-5, with an experimental platform capability for communicating with two other satellites.

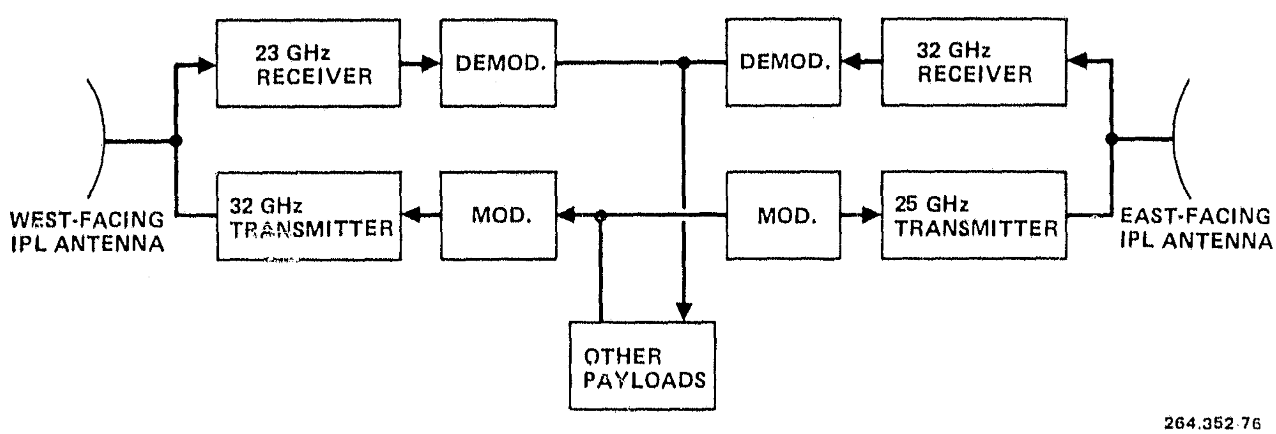

Figure 6-5. Interplatform Link Communications System

Table 6-6 summarizes the characteristics of the candidate communications systems for the experimental geostationary platform. They are not prioritized, and selection for allocation to an experimental platform will be governed by the number of Shuttle flights, OTV option performance, packaging constraints, payload weight, and communications systems test preference as yet to be determined.

6.6.2.2 Secondary Payloads. To select candidate secondary payloads for the experimental platform, the complete list of 84 candidate geostationary missions developed in Task 1 was first screened to eliminate those that were incompatible with the experimental platform philosophy in schedule, mission, or requirement for advanced technology. The remaining candidates were grouped according to sponsoring agency and/or subject, and prioritized. Eight were then selected for consideration as experimental platform payloads, representing the primary test and demonstration interests in DoD, OSTA, and OSS. These are shown in Table 6-7.

In developing the experimental platform concepts (as discussed in Section 6.7 to follow), payloads were selected from Table 6-7 based not only on priority and the desirability of providing a payload mix, but on the real limitations of weight, power requirement, and geometry. Some of the payloads are exceptionally heavy and high in power consumption, and if all were included on an experimental platform, the total could exceed the primary (communications) payload weight by 50 percent or more. Some require excessive volume and cannot be accommodated on the platform without severely reducing the accommodation available for other payloads, as shown by the geometry of the plasma wave injection experiment (Payload No. 80).

The payloads selected in this feasibility study are tentative and subject to change as the program progresses, to be compatible with the overall practicable platform weight, volume, and power requirements.

6.6.3 MISSION OPTIONS. To properly evaluate the feasibility of an experimental goestationary platform, a study must encompass options within a reasonable range of capability (and cost). For this study, the limit of investigation was set at configurations that could be placed in orbit with no more than two Shuttle flights, with a single Shuttle flight configuration as the minimum cost design goal. These configurations result in mission plan options and suboptions as summarized in Table 6-8.

6.6.3.1 Single Shuttle Flight Options. If sufficient experimental and demonstration capability can be installed on a platform restricted in weight and volume to less than half the Orbiter cargo bay capacity, mated to an orbital transfer vehicle that occupies the other half of the cargo bay, then the mission plan would be as described in Table 6-8. The mated platform/transfer vehicle would

Table 6-6. Candidate Communications Payload Characteristics - Experimental Geostationary Platform

Function	Frequency		Antennas				Beams	Transponder			Remarks
	Up GHz	Down GHz	No.	Size (m)	Beam-Width	Pointing Accuracy		BW MHz	EIRP dB-W	RF Watts	
Fixed-Service											FDM/FM/FDMA
C-Band		4.0	1	5	1.0°	0.1°					Global Not Included in Demo
	6.0		1	4							8 by 8 Switch
							Hemi (2) Zones (6)	30 80	32.0 32.0	16.00 4.00	Opposite Polarization
Ku-Band		11.0	1	4	0.5°	0.05-0.1°					SS - TDMA
											16 by 16 Switch
	14.0		1	3							
							Spot (16)	80	50.0	4-5	8 Sets Dual-Pol
Mobile	1.6	1.5	1	Array	(Global)	1°	Global (1)	7	35.0	100.00	Atlantic Ocean 4-Helix Array
L-Band	1.6	1.5	1	6	2.5°	0.25°	Hemi (1) Zones (2)	7 7	35.0 35.0	15.00 5.00	
30/20 GHz 10-Beam Trunk	30.0	20.0	2	4	0.3°	0.03°	Spots (10)	375	55.6	1.24	SS-TDMA 10 by 10 Switch
30/20 GHz 25 Beam CPS	30.0	20.0	5	1	1.1°	0.11°	Spots (25)	150	60.4	150.00	SS-TDMA 25 by 25 Switch
Interplatform Link	23.0	32.0	1	5	0.1°	0.01°	2	100	60.0	10.00	1 GHz BW in Each Direction

Table 6-7. Secondary (DoD and Science) Payload Candidates in Tentative Order of Priority

Payload No.	Weight, kg	Power, kW	Geometry
31 DMSP Data Relay (DoD)	150	0.1	18 in. by 45 in. by 24 in. Height
Tactical SATCOM Package (DoD)	320	1.0	62 in. by 83 in. by 50 in. Height
54 - EHF System			
55 - Aircraft Laser Relay			
60 - ECCM Processing TDMA			
61 - Space/Ground Lasercom			
17 Lightning Mapper (OSTA)	250	0.3	48 in. by 48 in. by 96 in. Height
Exposure Payload Group (DoD)	50	0.1	45 in. by 160 in. by 10 in. Height
33 - Materials Exposure			
43 - Magnetic Substorm Monitor			
56 - Fiber Optics Demonstration			
Large Space Structures Demo Group (DoD)	Incorporated in ACS and Platform Structure		
59 - Low CTE Structures			
34 - ACOSS Stabilization			
75 Imaging Spectrometer (OSS)	350	0.15	20 in. by 60 in. by 80 in. Height
79 Low Light Level TV (OSS)	300	1.0	40 in. Diameter by 100 in. Height
80 Plasma Wave Injection (OSS)	750	5.0	40 in. by 40 in. by 120 in., Plus 100 to 1000m Dipole Large Loop Antenna.

Table 6-8. Experimental Platform Mission Options

I - One Shuttle Flight

- Launch Mated Platform/Transfer Vehicle to LEO
- Orbital Transfer, LEO to 110°W GEO
- 1 Year Test and Demonstration
- (Option) Move to 15°W GEO for Atlantic Operation (Walking Orbit - 48 Days, 20 kg RCS Hydrazine)
- User Demonstration and Application 5 Years or More

II - Two Shuttle Flights

- Option A. Duplicate Platform/Transfer Vehicle Configurations; Different Payloads
  - DoD and Science Platform to 110°W GEO
  - Communications Platform to 15°W GEO
  - Western Hemisphere and Atlantic Test and Demonstration Operations for 1 Year
  - (Option) Move Western Hemisphere Platform to Atlantic to Demonstrate Rendezvous and Docking Technology; Continue User Demonstration and Application - 5 Years
- Option B. Platform in One Shuttle, Transfer Vehicle in Second
  - Platform to LEO; Deploy and Checkout
  - Transfer Vehicle to LEO and Mate
  - Orbital Transfer to 110°W GEO
  - 1 Year Test and Demonstration
  - Move to 15°W GEO for Atlantic Operation, 5 Years or More

be placed in LEO by the Orbiter. The platform would be deployed, checked out, and transferred to GEO at 110°W longitude. On-station tests, experiments, technology demonstrations, and verifications would be performed for a year or so. The platform could then either continue to serve a useful function over the Western Hemisphere (public service, etc.), or as a suboption, the platform could be moved to an Atlantic location for further experiments relevant to international communications, for example.

This option would be the lowest cost mission, and by reason of simplicity and a single Shuttle flight, would provide the earliest possible launch date for an experimental platform.

6.6.3.2 Two-Shuttle-Flight Options. With two Shuttle flights available to orbit an experimental platform, greater flexibility of platform configurations and mission planning is possible, as shown in Table 6-8.

In Option A, two basically identical Shuttle payloads are orbited independently, one to LEO and to GEO at 110°W longitude, the other to LEO and to GEO at 15°W longitude. Each Shuttle payload consists of a mated platform/transfer vehicle configuration similar to the single-Shuttle-flight option configuration, but with different platform payloads. This option provides twice the experimental payload capacity, greater payload selection, and better compatibility mix opportunities. The two small platforms also provide a capability for interplatform link demonstrations independent of other satellites, and an opportunity to demonstrate rendezvous and docking technology at GEO after moving the Western Hemisphere platform to the Atlantic platform location.

In Option B, one Shuttle flight is used to place a full-cargo-bay platform in LEO, where it is deployed and checked out. The second Shuttle flight delivers the orbital transfer vehicle to LEO, then monitors and controls the platform/transfer vehicle mating process, checkout, and transfer to GEO. The operations plan for this option is the same as that for the single-Shuttle-flight option - operations at 110°W longitude for a year or so, then transfer to the Atlantic location for further tests and experiments involving user applications.

Both Options A and B provide more payload and greater payload selection than does the single-Shuttle-flight mission. Option A provides the opportunity for interplatform link demonstration and rendezvous and docking technology demonstration, while Option B provides a better demonstration of economy of scale. Both options would have a cost somewhat more than double that of the single-Shuttle-flight mission, due to increased complexity.

6.6.3.3 Experimental Platform Area Coverage. Assuming a 5° elevation angle at the perimeter, area coverage for a Western Hemisphere experimental platform located at 110°W longitude is shown in Figure 6-6. It would cover the American continents completely except for the Alaskan northwest coast and land masses above 76°N latitude. The location is ideal for DoD and science experiments, and for communications experiments including sea mobile.

At 15°W longitude over the Atlantic, as shown in Figure 6-7, an experimental platform would cover all of Southern America, the eastern cost of North America (including the northeast corridor), all of Europe and all of Africa, involving international communications and sea mobile experiments.

6.6.3.4 Geostationary Orbital Arc Transfer. If we want to cover both the Western Hemisphere and Atlantic locations with a single experimental platform, it can be done by moving from one location to the other at any time desired during the on-station operational life of the platform. Employing the attitude

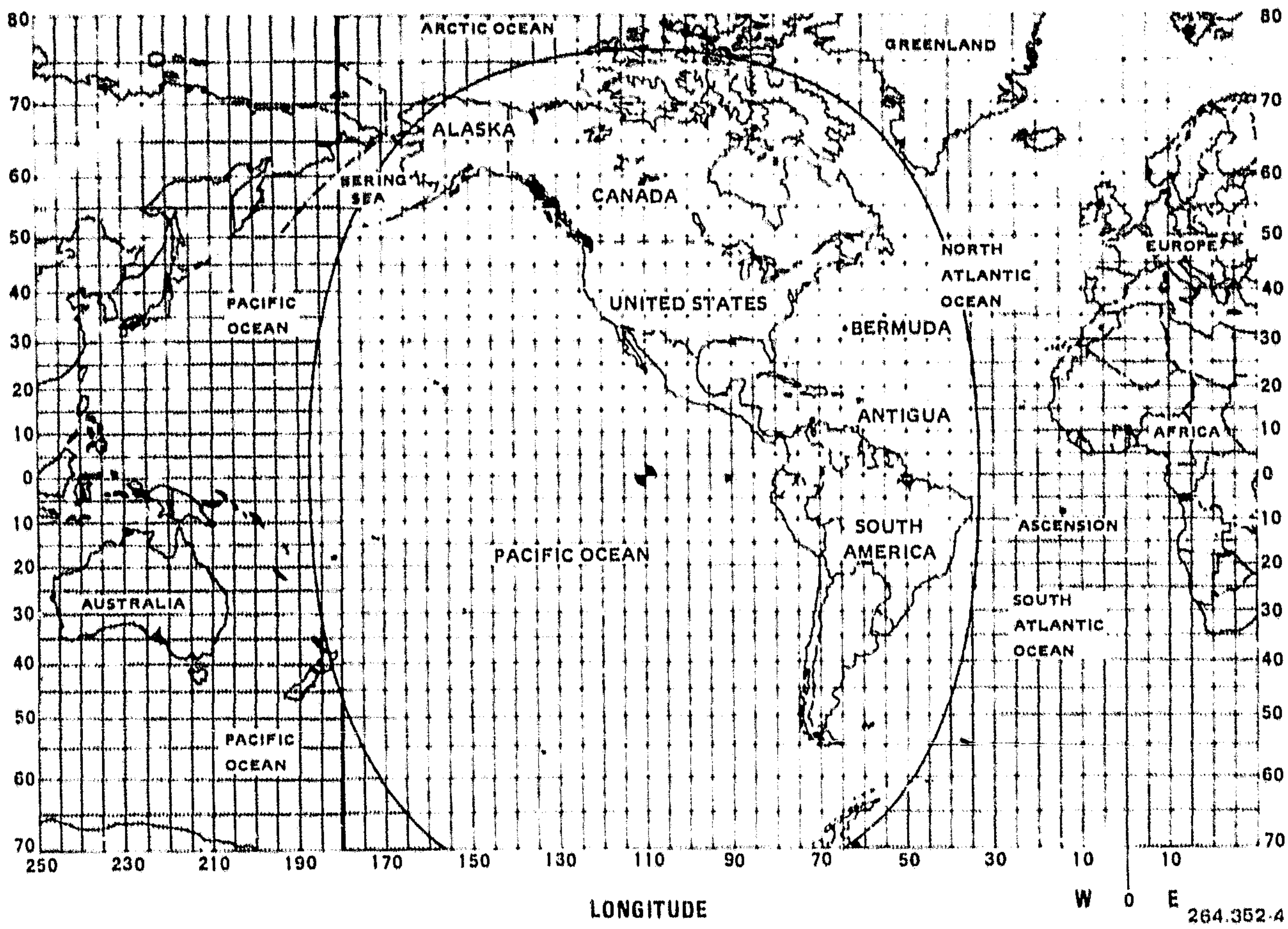

Figure 6-6. Experimental Platform at 110°W Longitude, 5° Elevation Angle

control system, the platform orbit is lowered slightly from geosynchronous with increased velocity, and moving east, returned to geostationary orbit at the desired second longitudinal location. The cost for the maneuver depends on the weight of the platform and the time allocated.

As shown in Figure 6-8, a 95° easterly shift in longitude for a 4000 kg platform with a hydrazine attitude control system could be accomplished with 40 kg of propellant in 24 days, or 20 kg of propellant in 48 days, etc. The advantages to be gained with respect to the increased scope of tests and experiments accomplished by the platform obviously far outweigh the modest investment in propellant weight and system capability.

6.6.4 STRUCTURAL CONCEPTS. Three generally accepted methods of creating large space structures in orbit were considered in developing concepts for an experimental platform:

a. Space fabrication.

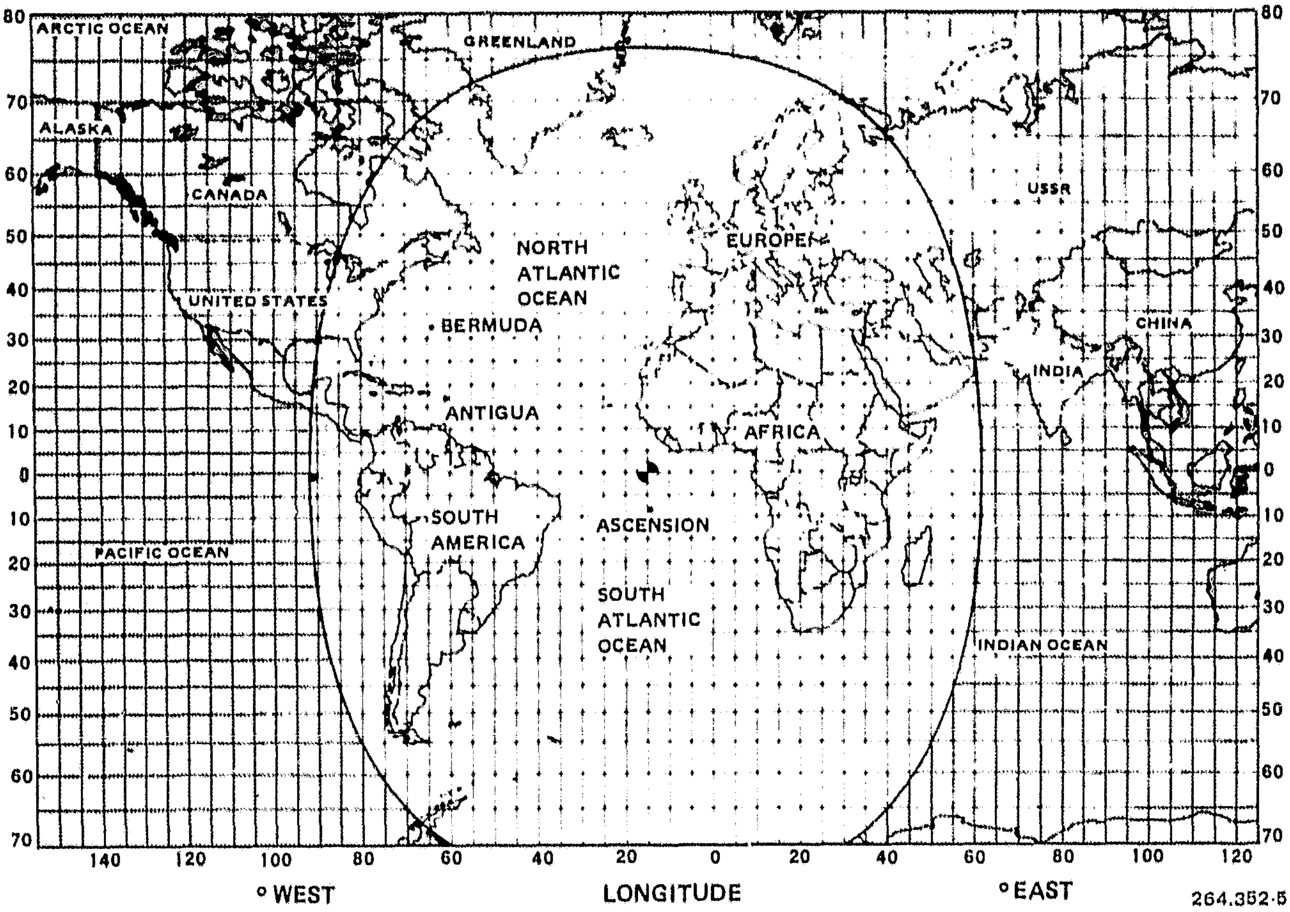

Figure 6-7. Experimental Platform at 5°W Longitude, 5° Elevation Angle

b. Assembly.

c. Deployment.

Of these, deployment offers the minimum in technology risk, development costs and schedule, time in LEO, and EVA requirements. Deployment may not be the optimum choice for eventual large-area space structures, but for a modestly-sized structure, its advantages place it in the most-favored category for near-term programs. The deployment method, with its inherent minimum need for stay-time in LEO, also favors use of high-energy cryogenic propellant transfer vehicles (minimum boiloff losses), and minimum support requirements from the Shuttle in terms of propellants, life support systems, etc., during the Orbiter stay in LEO.

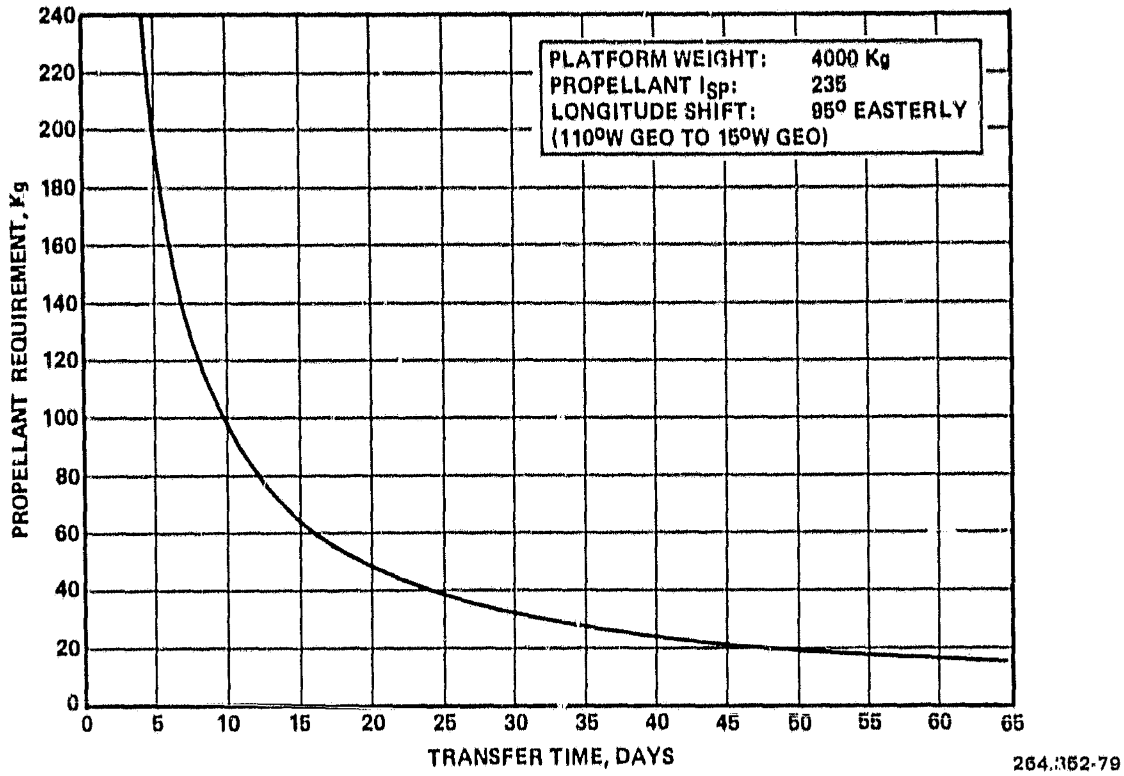

Figure 6-8. Walking Orbit Propellant Requirement for 95° Geosynchronous Orbit Shift

The basic deployment system selected for experimental platform concepts is shown in Figure 6-9. It was developed during the Air Force On-Orbit-Assembly study (Contract F04701-77-C-0178), and was selected primarily for its:

a. Packagability of structure, subsystems, and payloads.

b. Flexibility with respect to payload mounting locations.

c. Centralization of subsystems in a common core.

d. Adaptability to servicing, docking, and modular growth for test and demonstration in LEO or GEO.

e. Geometry of interface with a transfer vehicle.

f. Symmetry of design and loading for orbital transfer.

g. Structural reliability.

h. Low weight.

The structure consists basically of a central core for subsystems, with interfaces for deployable payload mounting arms, for direct-mounted payloads, and for an orbital transfer vehicle. The system is shown stripped of everything but

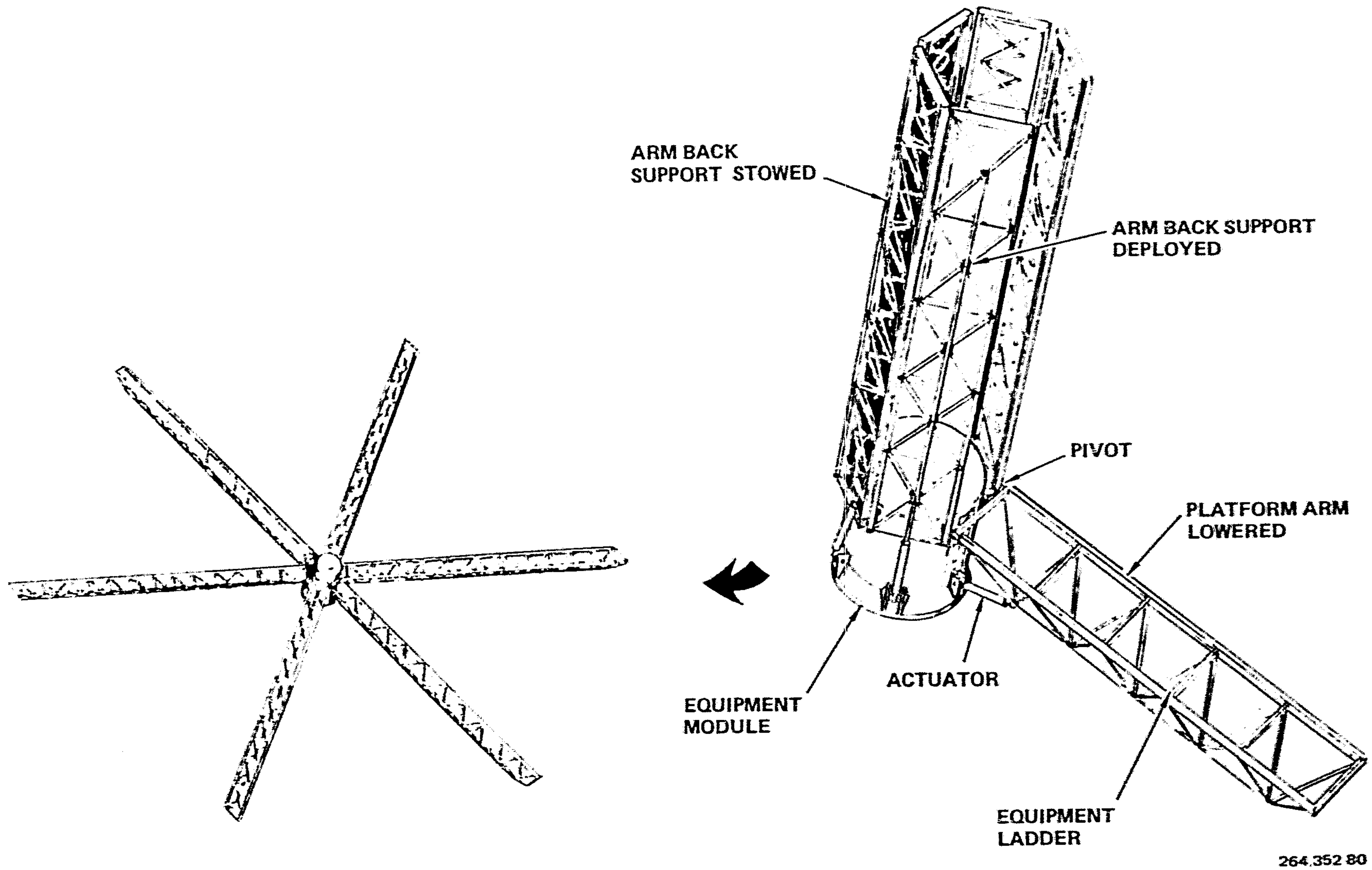

Figure 6-9. Experimental Platform Deployable Structural Support Concept

core and structure, for clarity. In the generic form shown, it consists of six semideployable arms attached to the core or hub. Four arms are shown completely folded and packaged; a fifth has been expanded with its back support deployed; and the sixth is shown fully deployed and extended to its operational position.

The basic technologies involved include the packaging system, the deployment/adjustment system, and the structural element configuration and material.

6.6.4.1 Packaging and Deployment Systems. Two concepts have been developed in General Dyanmics Convair's IRAD programs, for application to deployable space structures.

Semideployable Arm Concept. For relatively small structures such as the experimental platform, semideployable (fixed length, expandable triangular cross-section) arms can be used, Figure 6-10. The arm shown is 5 feet wide, 4.3 feet deep, 17.5 feet long for a half-cargo-bay experimental platform, and 45 feet long for a full-cargo-bay experimental platform. The arm structure is fixed in the direction of its major axis, and need be deployed only in cross-section for stiffness, using hinged-strut technology.

Each of the bays in the arm is approximately 6 feet long, and is ideal for packaging of experimental and demonstration payloads. Illustrated in Figure 6-11 is a female docking port, stowed within the protective frame of the arm bay, deployed to a position away from the earth-facing payload surface of the structure, where it can be approached by a teleoperator or other servicing/docking vehicle to demonstrate docking technology. Adjacent bays or bays on other arms can be used for component and fuel servicing technology demonstrations.

A one third scale model of two bays of an experimental platform arm is shown in the stowed position in Figure 6-11, and fully expanded in Figure 6-12. This is a demonstration model and has not been optimized for low cost and weight. In its final configuration, the arm will use low CTE composites with tube thicknesses approaching 0.009 in., tension cord diagonals or corner gussets for space availability within bays, "Z" sections instead of channel, and simple strut pin-joints with dog-leg end fittings rather than universals. If retraction as well as deployment proves to be a sufficient reliability/safety advantage, the carpenter-tape hinges shown could be replaced with tension or torsion spring type hinges, with an open locking device and a motor or spring drive. Full scale arm weight is estimated at approximately 1.5 lb/ft, when optimized.

Advantages of the semideployable arm include:

a. Internal volume of the arm available for equipment installation.

b. High volumetric efficiency; the arm takes only 2.5 percent of the available cargo bay volume.

c. Simplicity - deployment for stiffening only.

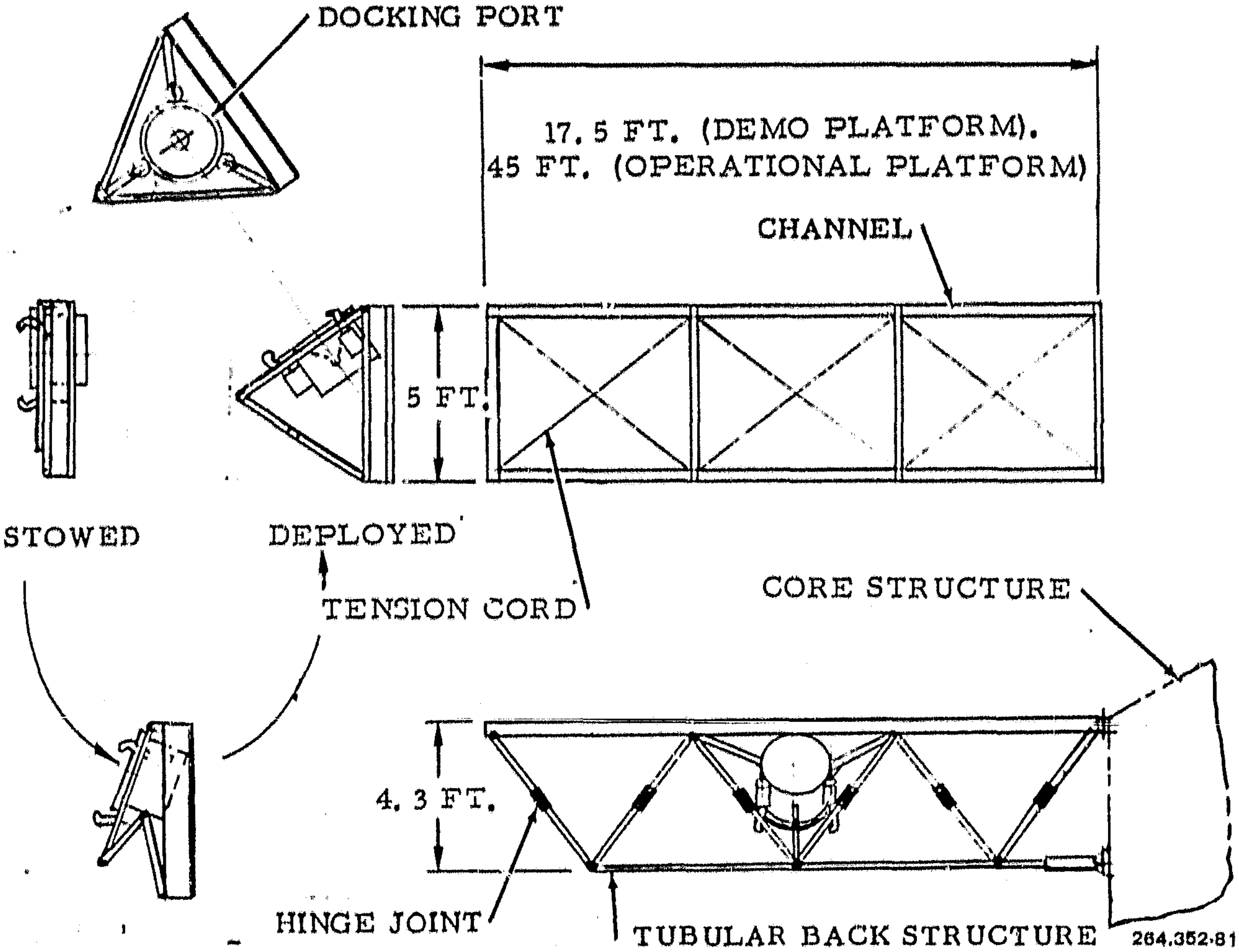

Figure 6-10. Semideployable Arm Concept

d. Length - accommodated by the cargo bay.

e. Protection - all payloads surrounded by structure when packaged, desirable for transportation and handling.

f. Divisibility - multination/multicontractor sharing of tasks is made possible: each arm, like a pallet, can be provided as a basic subsystem or experiment mount.

g. Adaptability - each arm is separable and can be replaced with optional arrangements during final assembly.

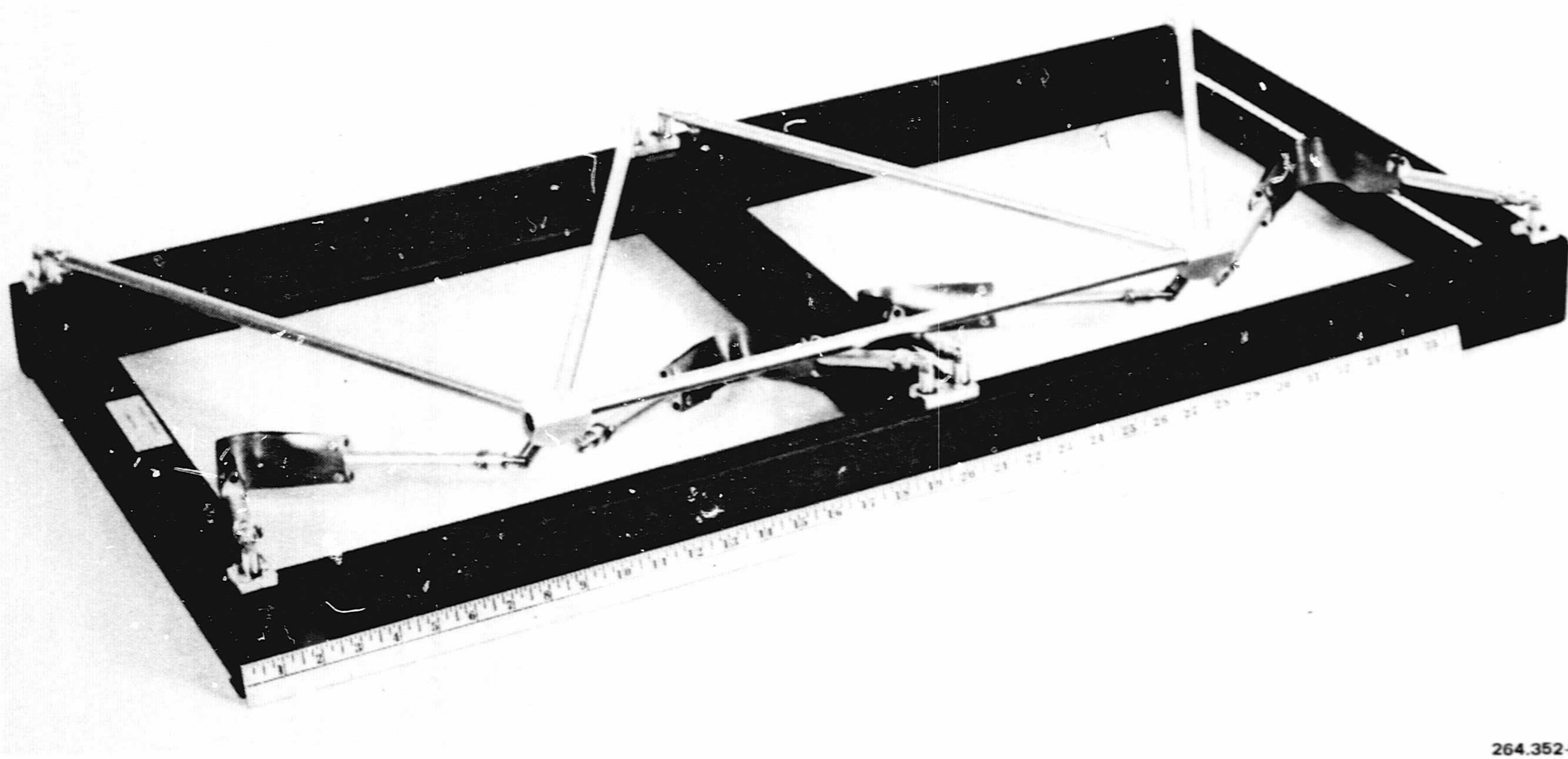

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Figure 6-11. Semideployable Arm Concept - Two Bay, 1/3 Scale Model, Folded

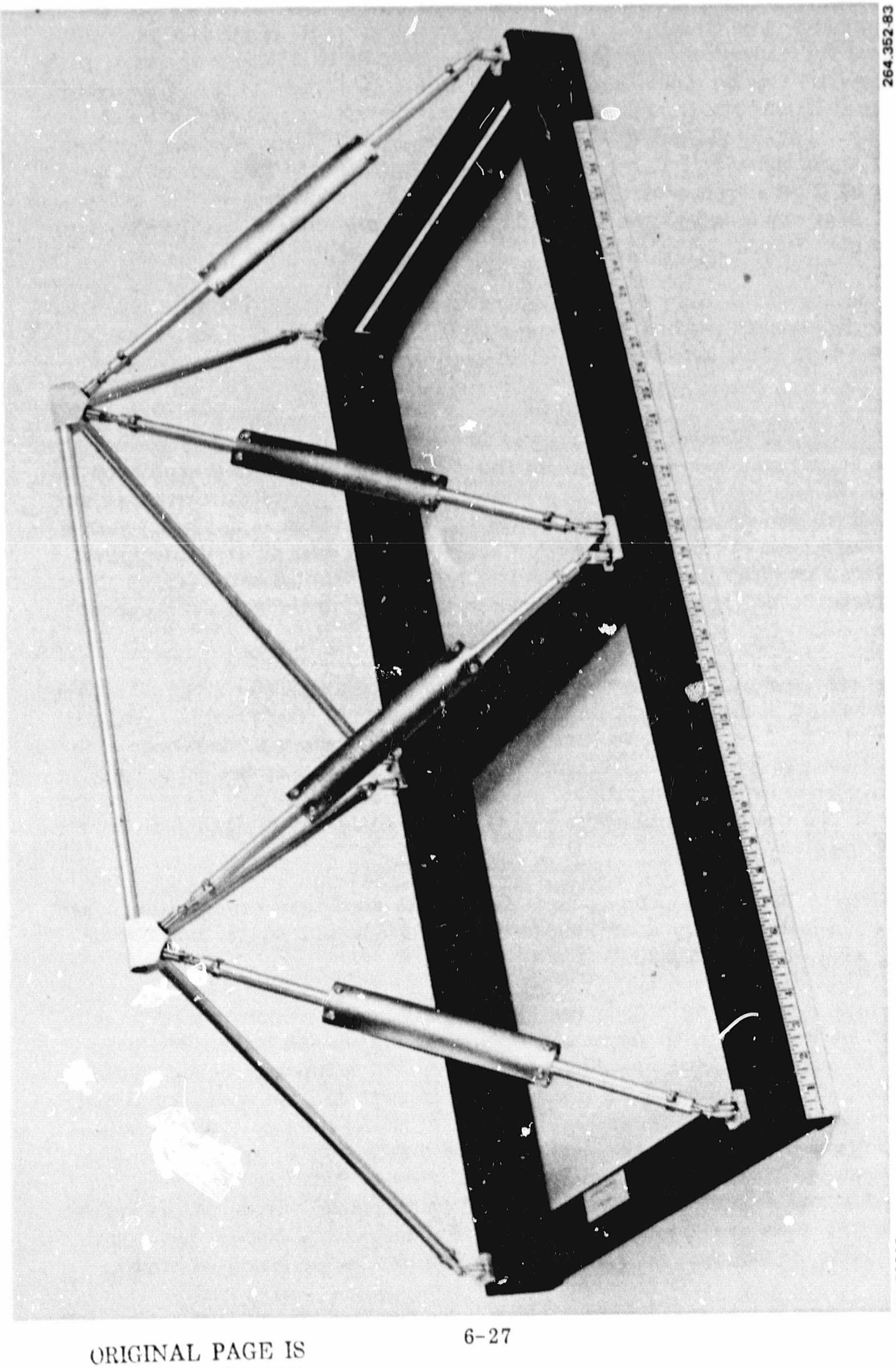

Figure 6-12. Semideployable Arm Concept - Two Bay, 1/3 Scale Model, Deployed

Fully-Deployable Arm Concept. For large structures such as those that could be required for operational platforms with arm lengths of 250 feet or more, fully deployable arms can be used, such as those shown in Figure 6-13. This arm has a double-triangular (diamond) cross section for structural rigidity and redundancy. Packaged, a 5 ft wide, 7.5 ft deep, 280 ft long arm has a cross section of approximately 1 ft by 5 ft and a length of 31 ft. The arm is shown deploying as it would be controlled from the Orbiter. The rectangular cross section is first expanded to the diamond shape, then extended bay-by-bay to its full length.

A 5-bay section of the fully deployable arm has been built at GDC and the controlled deployment mechanism is being added. The truss is a full-scale, extensible, deployable arm for demonstration, vibration, thermal, and load testing.

6.6.4.2 Structural Element Technology. The basic strut technology for expendable truss structures was developed during the on-orbit-assembly study for the Air Force. Struts are fabricated of graphite-epoxy composite with titanium end-fittings and midcenter hinges where required. The material is designed with a negative coefficient of thermal expansion to provide an overall strut designed CTE of zero. Manufacturing tolerances preclude an absolute zero CTE in the actual structure, but analysis has shown the deviation to be well within acceptable limits.

The carpenter tape hinges shown in Figure 6-13 are used in the diagonal struts for lightness and simplicity, where tension and moderate compressive strength is required. The over-center locking hinge is used on the main longeron struts, where high compressive strength is needed. The over-center self-locking hinge has recently been modified to provide an unlocking capability, if retraction and restowing in the Orbiter should prove to be sufficiently advantageous to warrant the added complexity.

6.6.4.3 Growth Potential. The generic deployable structure concept discussed in this section is inherently adaptable to growth either as a single module, or by joining with other modules.

Single Module. As already noted, the platform structure as configured in this section can be designed with semideployable arms for a small half-cargo-bay platform (17.5 ft arms); with semideployable arms for a full-cargo-bay platform (45 ft arms); or with fully-deployable arms for growth into an operational platform (280 ft arms). A full-cargo-bay platform of moderate size, for example, is shown in Figure 6-14. Here, the deployed platform will be approximately 100 ft in diameter, with semideployable rigid arms. Communications payloads can be mounted at the arm tips, on the arms, and on the core. Solar panels can be mounted at the ends of the arms, as shown. Subreflectors, horns, and other components can be mounted on a central fixed or extensible mast, as shown.

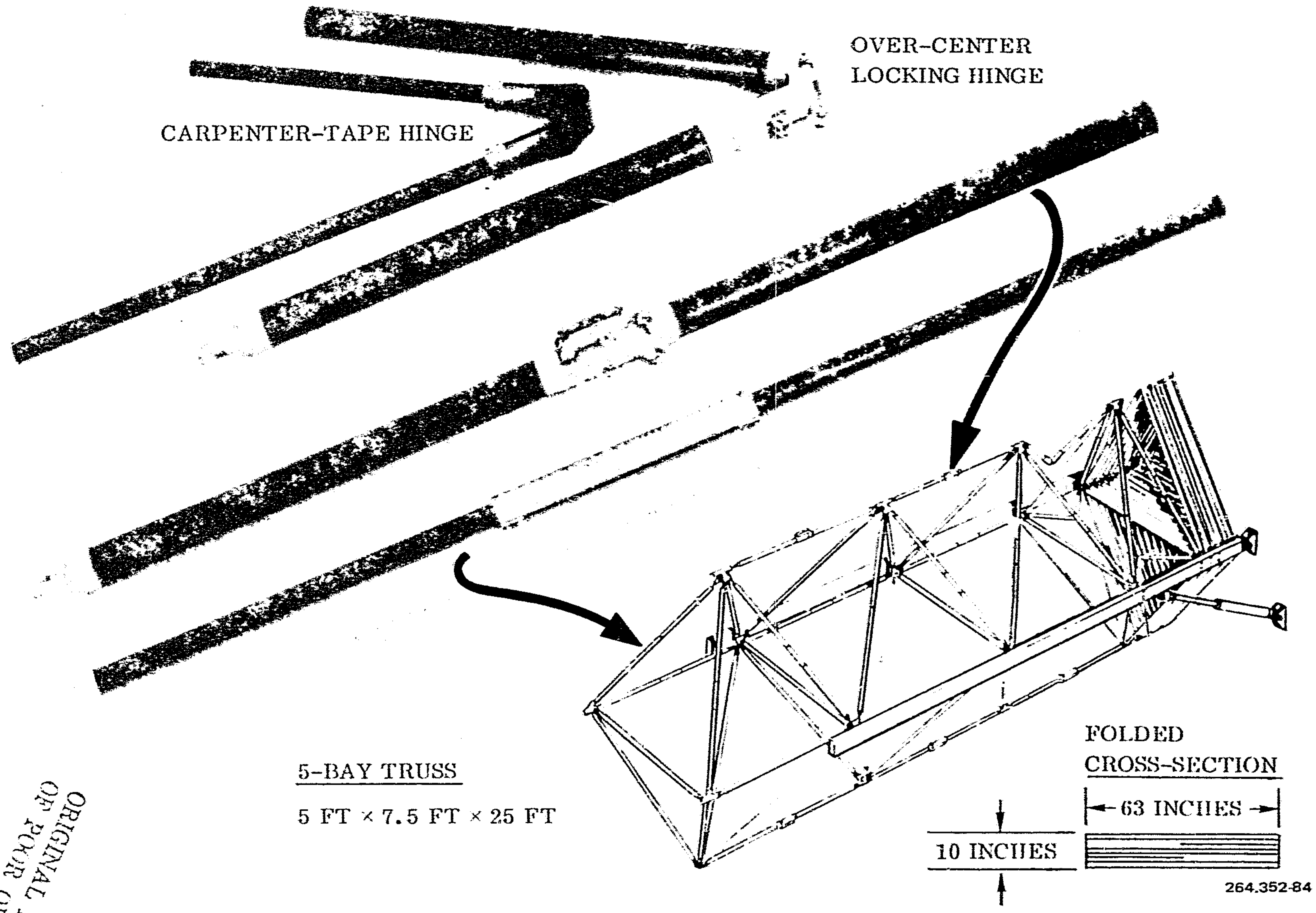

Figure 6-13. Fully Deployable Arm Concept

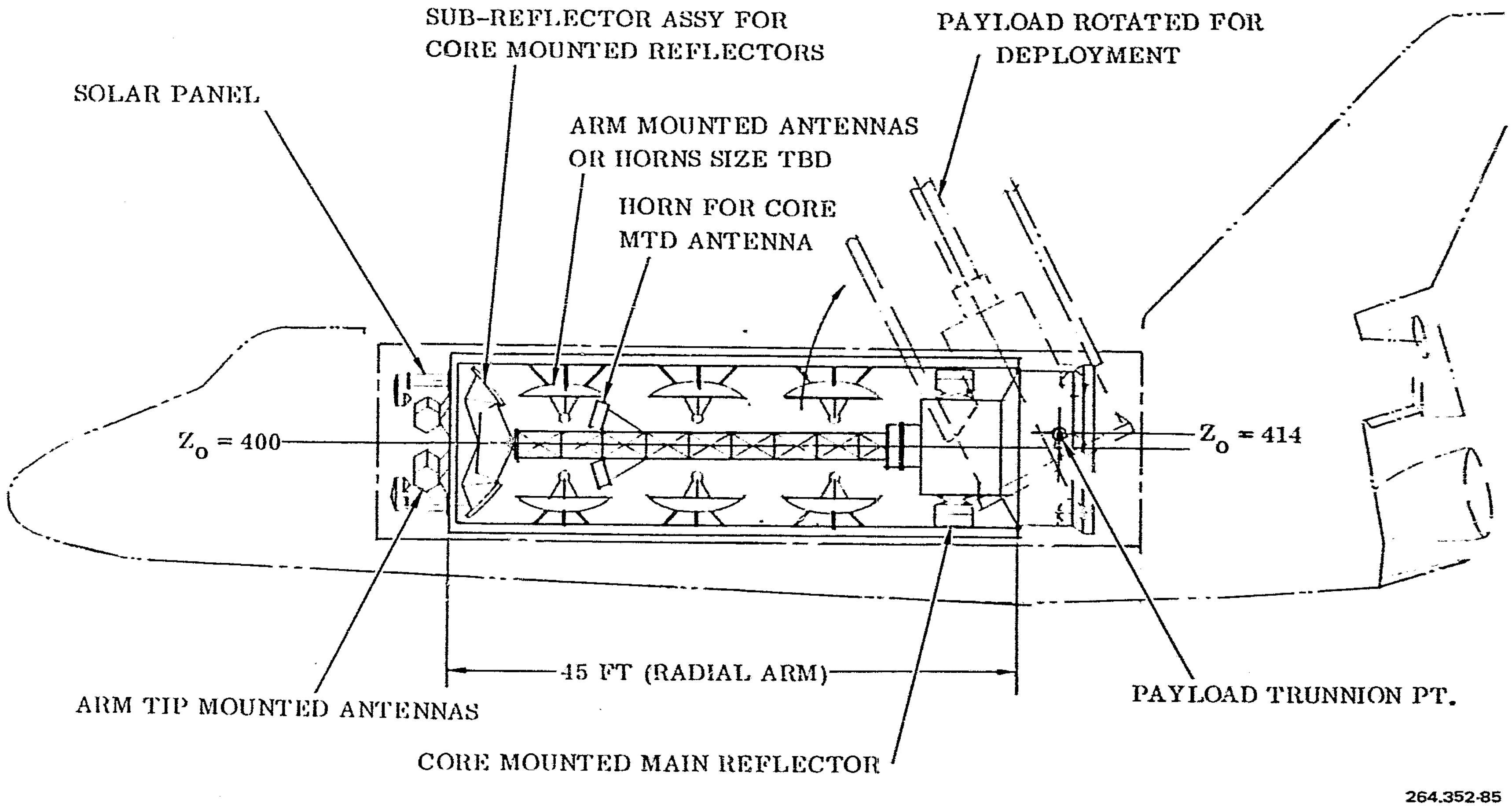

Figure 6-14. Growth Potential - Full Cargo Bay, Packaged

Multiple Docked-Module Platforms. The deployable radial-arm concept is also adaptable to docked-module configurations, with either the semideployable or fully deployable arms. Docking hardware, as will be discussed in a following section on subsystems, is mounted at the arm tips. Docking will be accomplished at two points between modules, at the ends of arm-pairs between the solar array arms, using a soft-docking technique.

An example of growth by module linear expansion is shown in Figure 6-15. Here, three operational platform modules are docked together in an east-west direction, with solar panels extended in the north-south direction at different distances from the central cores to eliminate or minimize solar array shadowing.

Figure 6-16 shows operational platform modular growth by lateral addition. Here, the growth concept centralizes the platform area somewhat, creating an alternating ladder addition that interlocks modules at five of the six arms of each module, with solar arrays on the sixth. Five platform modules are shown, providing a fully operational platform area approximately 300 ft by 300 ft. The concept illustrates the varying distance of the solar panels from the cores to minimize shadowing, and also illustrates the two- and three-point docking techniques developed in the on-orbit-assembly study.

6.6.5 SUBSYSTEM REQUIREMENTS. Subsystems for an experimental platform have been identified in sufficient depth to assist in determining the feasibility of the experimental platform concept with a reasonable degree of credibility. No attempt has been made to design recommended systems. Rather, system requirements and existing, acceptable systems have been looked at to determine their overall acceptability and application to a feasible platform concept.

6.6.5.1 Docking. Shown in Figures 6-17 and 6-18 is the soft-docking technology developed at GDC for large, flexible space structures involving appreciable mass, as exemplified by the geostationary platform concept. The hardware consists essentially of male/female soft docking units, sensors, umbilical receptacles, and associated electronics/ACS.

In the two-point docking concept shown, the passive platform is on station in geostationary orbit; the active platform approaches following a preprogrammed rendezvous trajectory involving initial search and acquisition, approach with incremental velocity decreases, braking, positioning, and stationkeeping. Typical rendezvous hardware for the system would be microwave interferometers, such as the 9.7 GHz Cubic ELF III using both angle-measuring and distance-measuring sensors with pairs of both coarse and fine antennas.

In the docking phase of the operation, the platforms are in position within 5 feet of each other. The active module extends an extensible, steerable probe (Figure 6-18) to full length. Relative position is determined by scanning laser radar (SLR) sensors capable, with the support systems, of maintaining relative module positioning within plus or minus 1.3 cm, and relative velocity less than 1.3 cm/sec. This equipment has been developed by ITT/Gilfillen.

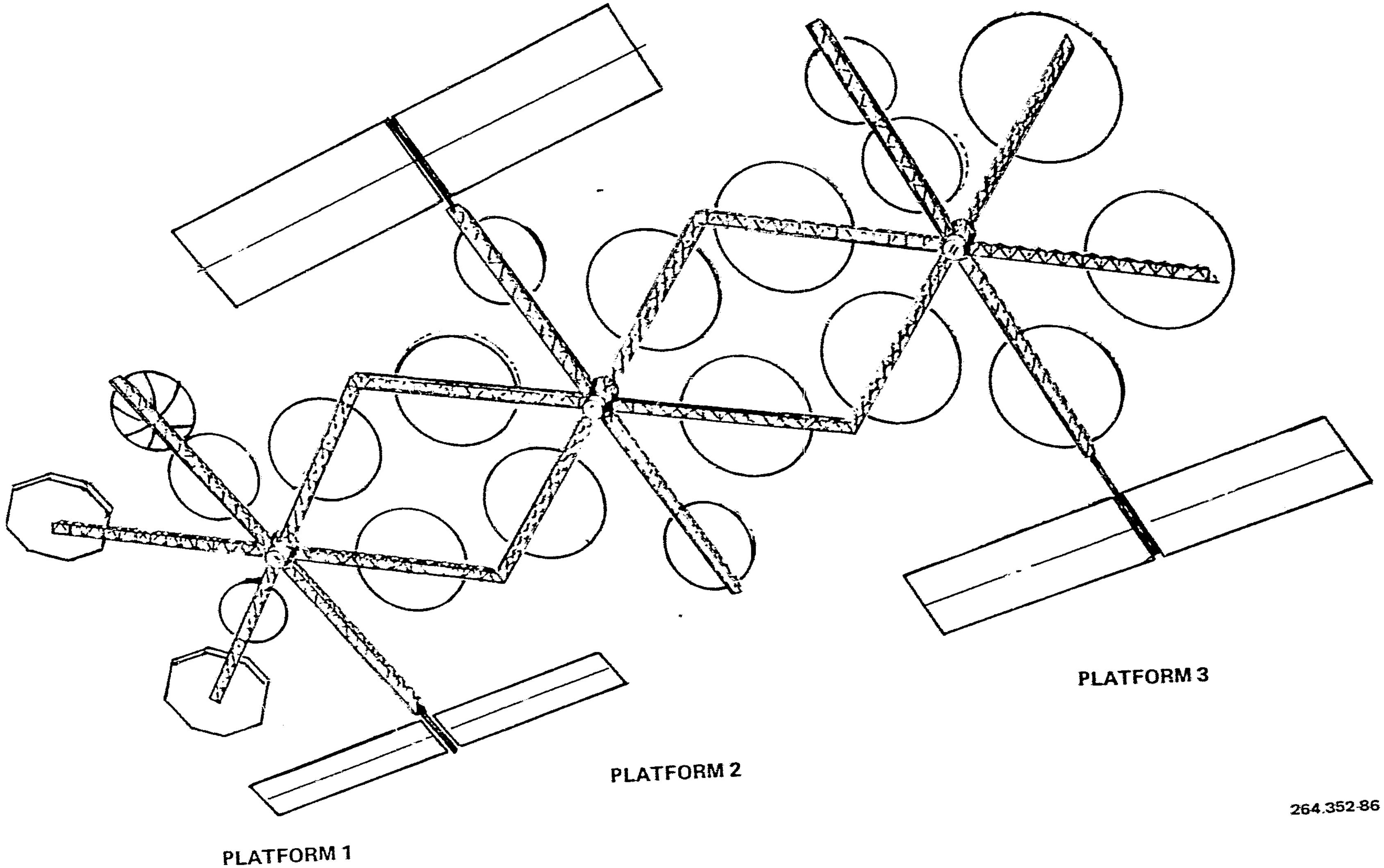

Figure 6-15. Operational Geostationary Platform Growth - Linear Expansion

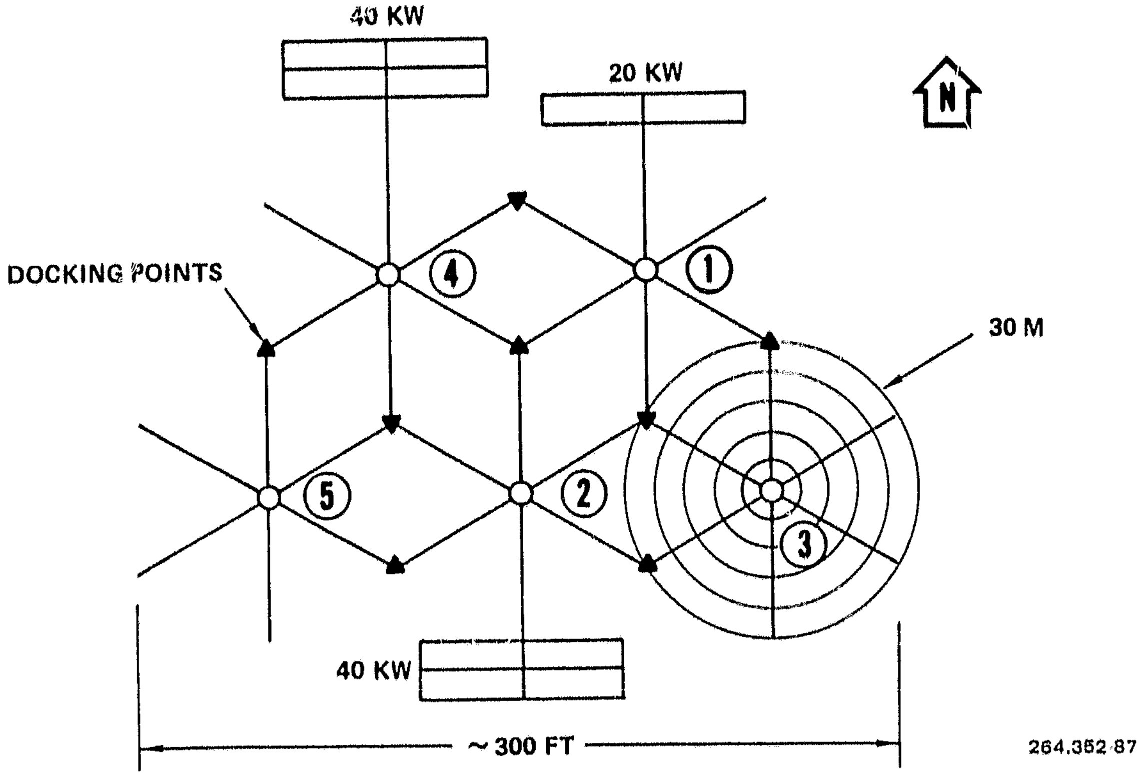

Figure 6-16. Operational Geostationary Platform Growth - Lateral Expansion

Steered by the SLR, the probe enters the docking cone, the tip engages and is captured, and the probe draws the two structures together. The mating docking cones provide final alignment and the latching pawls lock the two structures together, structurally stronger than the radial arms themselves.

The mechanical components of the system have not been manufactured, but have been preliminarily designed, and the required mechanisms and components are available.

Docking of large flexible space structures of appreciable mass is practical, feasible, and with the system shown, produces structural loads that are insignificant in comparison with the other operational loads encountered.

6.6.5.2 Attitude Control. To obtain an estimate of attitude control system requirements for an experimental platform, a typical ACS was set up as shown in Figure 6-19, in block diagram form. Earth and sun sensors are used as input to the attitude reference unit. The control processor interprets input data and commands the reaction/momentum wheels and the thrusters. Hydrazine is assumed as the propellant.

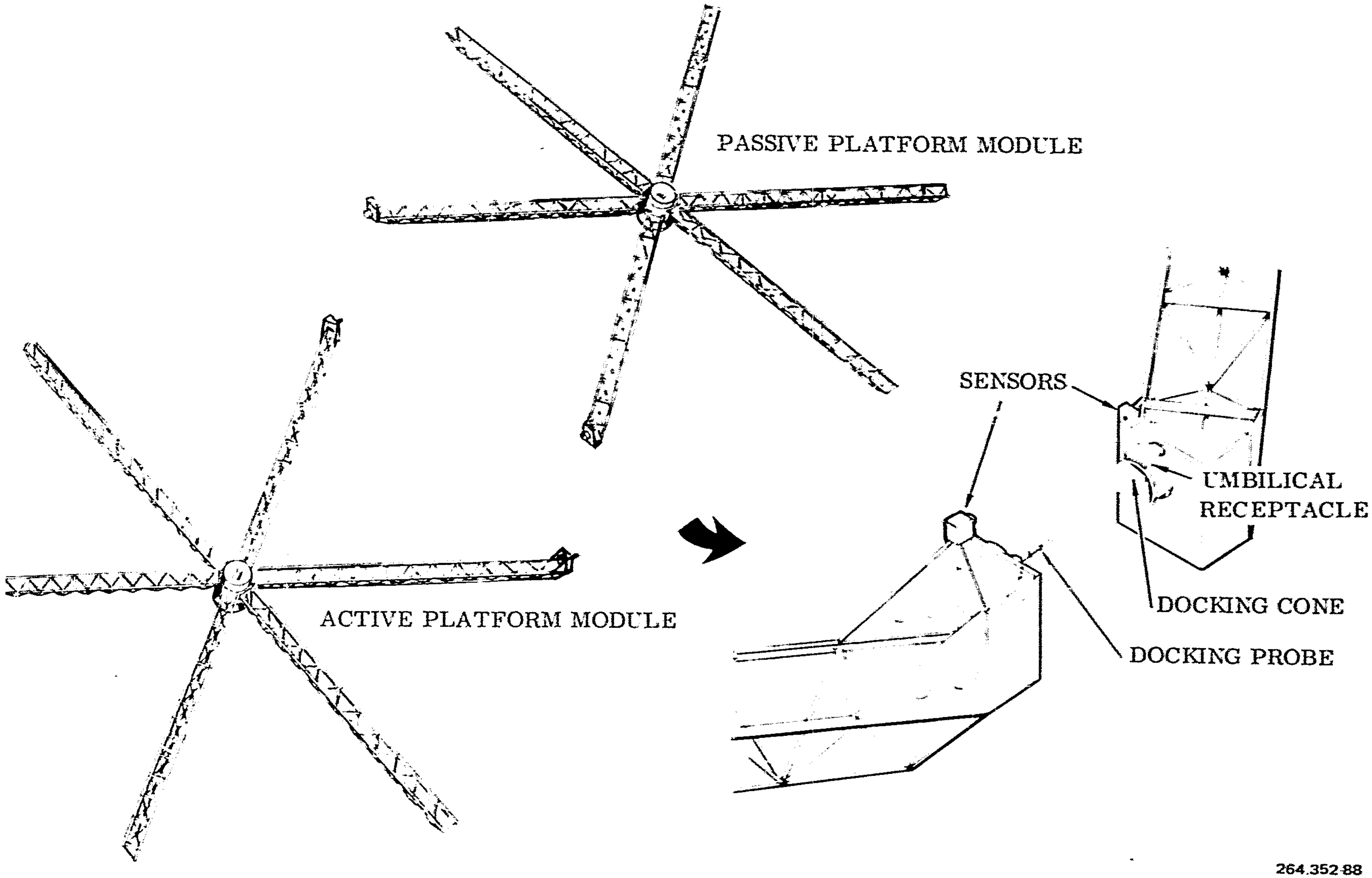

Figure 6-17. Operational Geostationary Platform Soft Docking Concept

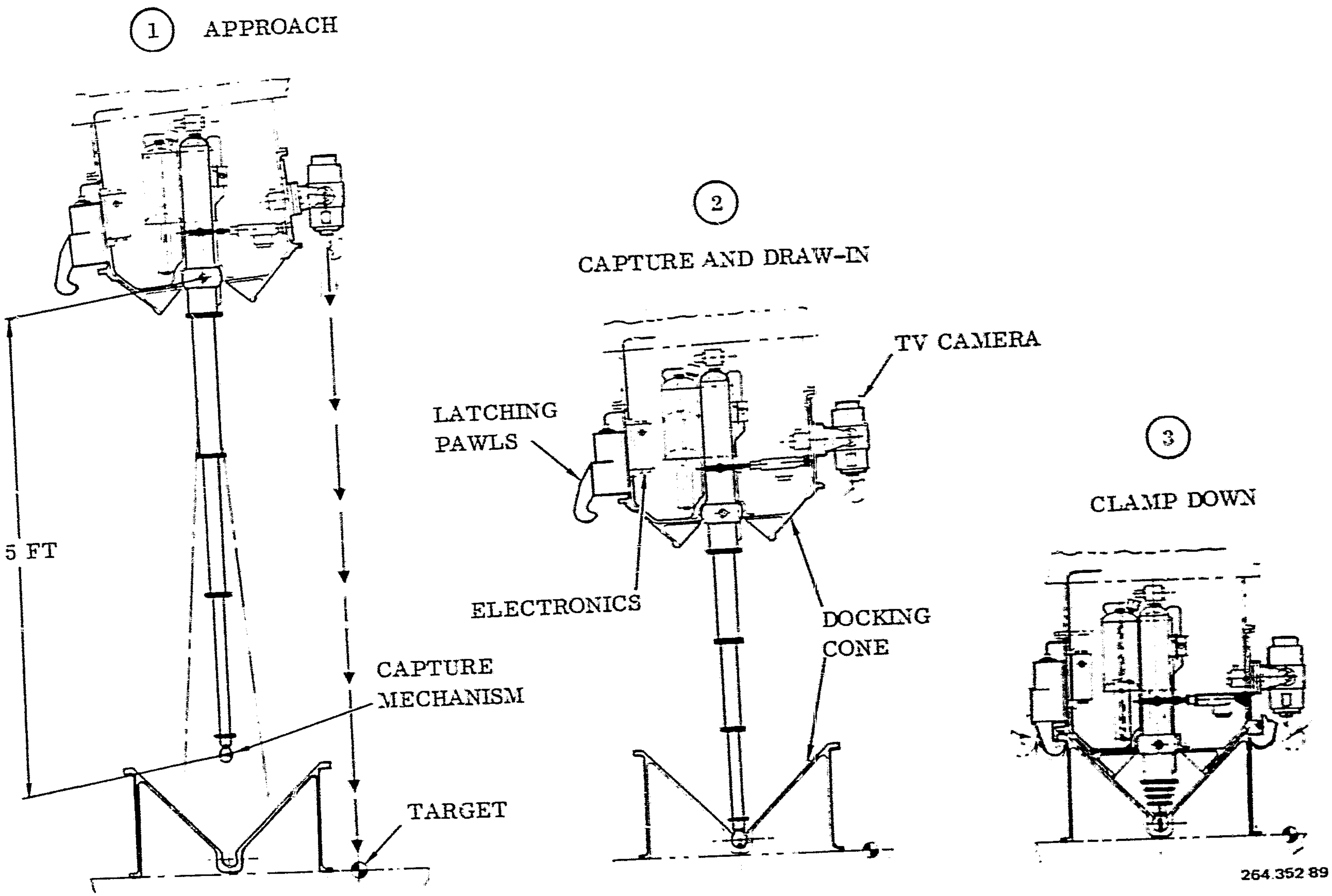

Figure 6-18. Soft-Docking System Hardware

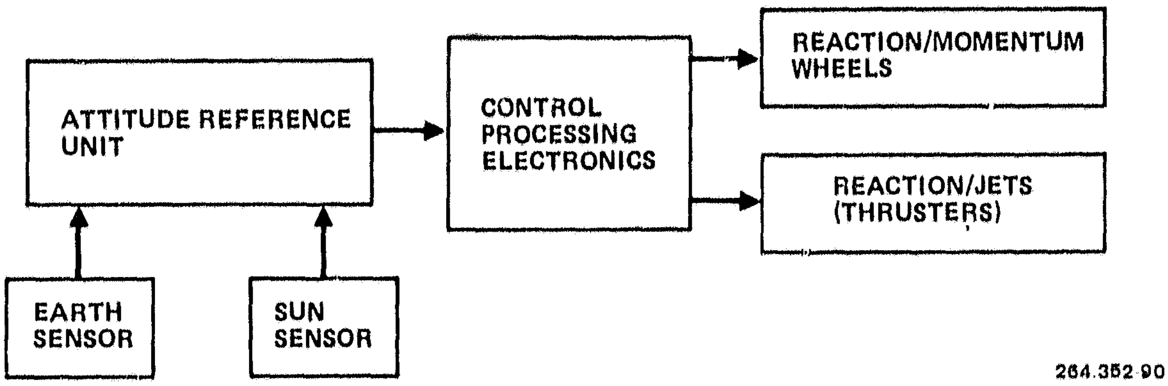

Figure 6-19. Experimental Platform Attitude Control System

A preliminary platform concept, Concept 1 in Section 6.7.2, was used as the basis for analysis. As expected, the maximum moment of inertia for the platform occurs in yaw, about the local vertical, with an estimated 105,000 kg-m^2. Pitch about the north-south axis and roll about the east-west velocity vector show only about half the yaw moment of inertia, again as expected from the flat platform configuration.

Gravity gradient analysis shows the platform to be stable in yaw, but unstable in both pitch and roll. Active control is required for stability, primarily for the pitch instability. Roll can be stabilized with approximately 15 N m-sec. of bias momentum.

Estimated propellant requirements for the platform are approximately 1/40th of the platform weight, or about 100 kg for a 4000 kg platform, per year. Of this, 99 percent is used for stationkeeping. The remainder, or about 1 kg N_2H_4 per year, is used for wheel unloading, assuming 10 percent solar panel unbalance and 1.0° principal axis offset in pitch.

Pointing accuracy for the ACS and platform is taken as plus or minus 0.1° in all axes, and includes an attitude reference error of 0.05°. More sensitive pointing accuracy requirements associated with large operational platforms, large diameter reflectors, and very narrow beams are not required for the experimental platform. If an experiment requiring pointing accuracy better than 0.1° is to be used, it will be provided by using a gimbal mount.

Components for the ACS will include proven components such as:

a. Magnetic bearing momentum, reaction wheels.

b. Solid state earth sensors with no moving parts.

c. Rate integrating gyros with update sensors and filtering to provide attitude reference.

6.6.5.3 Active Stabilization. Control is recognized as a significant concern in the development of large space structures, particularly if the structures are essentially flat and flexible, as platforms will be, with the characteristics shown in Figure 6-20.

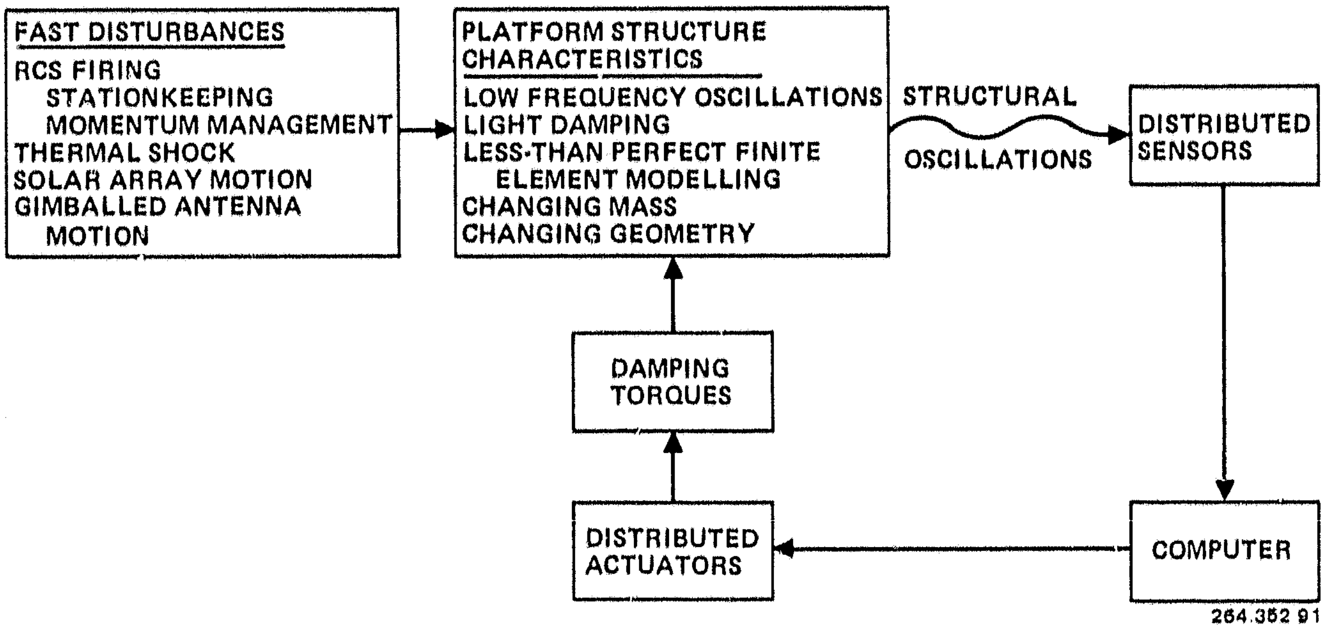

Figure 6-20. Active Stabilization System (Modern Control Theory)

So called fast disturbances, as shown in the figure, produce low-frequency oscillations in the structure, essentially giving false indications of platform attitude as detected by the sensors, and establishing the possibility of control-system-induced reinforcement of the oscillations and attendant structural damage or loss of control. A control system that neutralizes this problem has been developed at GDC, using rigid body state estimators in the ACS electronics, which in effect filter out the false oscillatory signals going to the computer, letting it see the attitude of the basic rigid structural body. This is the system that will be used on the platform.

An advancement in the state-of-the-art being studied is active stabilization of space structures (ACOSS) which, in effect, distinguishes between the rigid body attitude and the oscillatory signals through distributed sensors; through distributed actuators it damps out the oscillatory vibrations and gives us, essentially, an actual rigid body. The system can be integrated with the ACS on the experimental platform. for experiment and demonstration as required.

6.6.5.4 Avionics. The avionics system for platform monitoring and control, Figure 6-21, is centralized in the platform core, and is comprised essentially of the:

a. Control and data processor.

b. Processor interface unit.

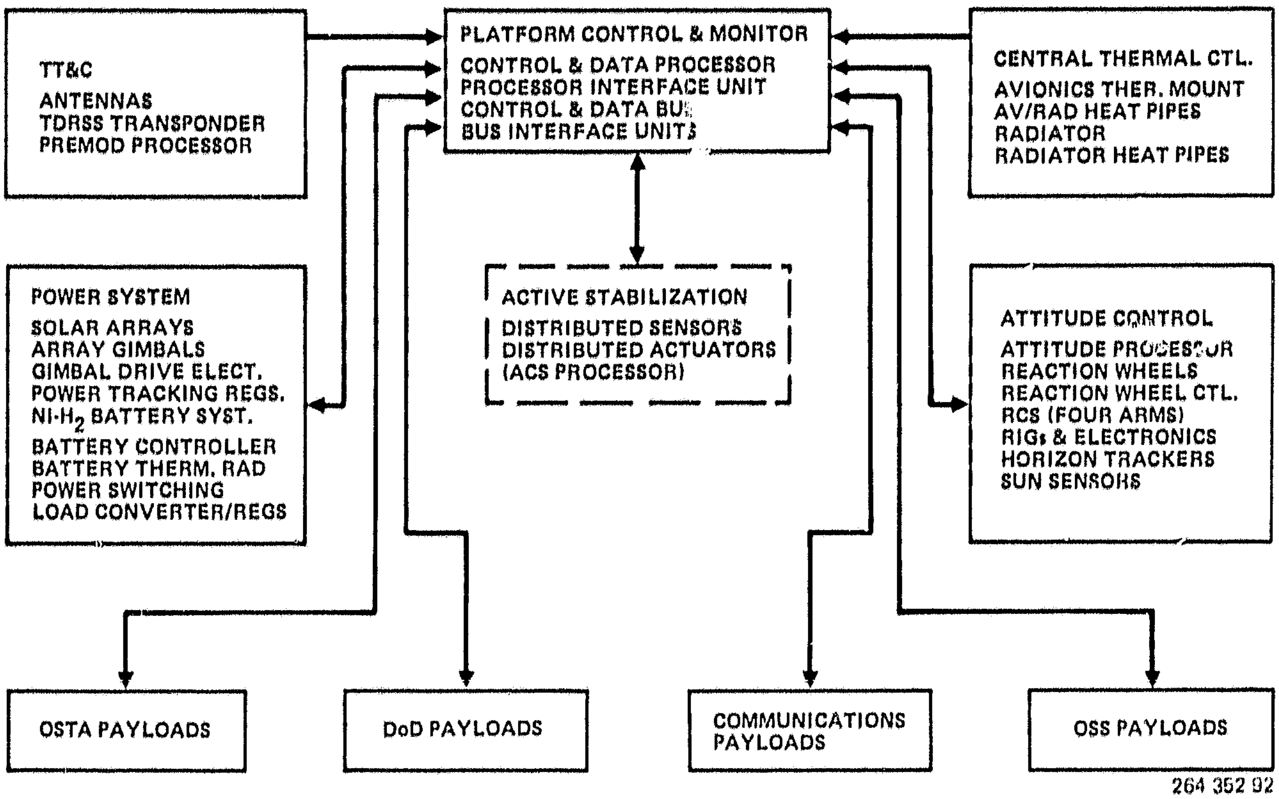

Figure 6-21. Experimental Platform Avionics Subsystem

c. Control and data bus.

d. Bus interface units.

The four platform subsystems shown, plus the payloads, represent the total probable interfaces for experimental platform subsystems with the centralized avionics system. All platform and payload control functions are integrated through the platform control and monitor unit for on-station operations, including stationkeeping, thermal control, power management, and command.

ACOSS has been added as a demonstration payload, to be activated and controlled as required during the platform's experimental operations phase.

6.6.5.5 Platform Communications. The interfaces between the experimental platform links, and the central communications controller are shown in Figure 6-22.

The central communications controller provides the external control for each individual platform subsystem. Each subsystem operates semiautonomously between revisions of the external control functions. The central communications controller also controls the interplatform communications subsystem by establishing the matrix switch controller operating modes, coordinating the communications subsystem being crosslinked, maintaining synchronization, and doing any formatting required.

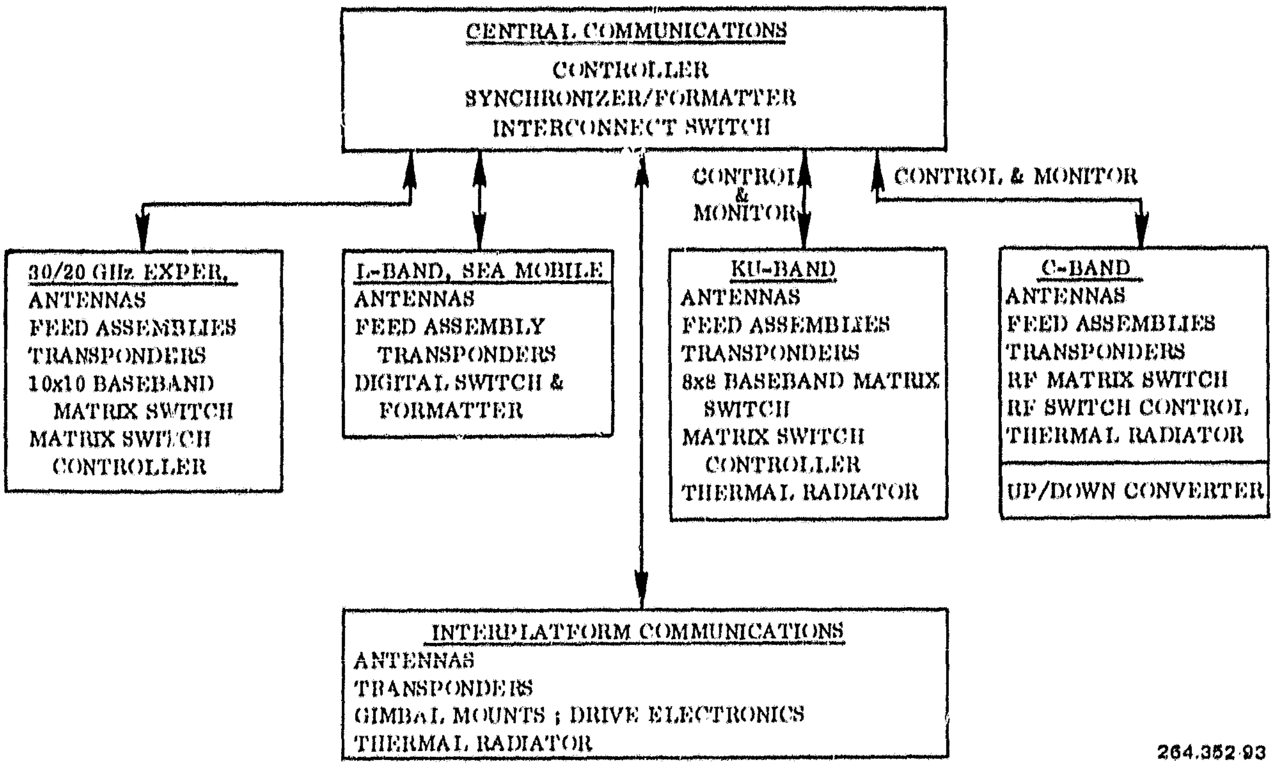

Figure 6-22. Experimental Platform Communications Subsystem

6.6.5.6 Power. The power system envisioned for the experimental platform consists of solar arrays, batteries, and DC power distribution, and management subsystems. System capacity and weight will vary with platform size, number of payloads, and payload requirements. To determine typical power requirements for system sizing and feasibility analysis, the six experimental platform concepts (to be discussed in Section 6.7) were analyzed.

Typical power requirement for a half-cargo-bay size platform proved to be between 6 and 8 kW. For a full-cargo-bay platform, the requirement fell between 10 and 13 kW. The power requirement for experimental Concept 2, Table 6-9 is typical.

A preliminary weight estimate of the power subsystem for each platform concept was also made. A typical example (Concept 2) is shown in Table 6-10. For the six concepts, the power subsystem weight ranged from 373 kg to 756 kg.

6.7 EXPERIMENTAL PLATFORM CONCEPTS

In developing experimental platform configurations to determine feasibility of the concepts, the intent has been to investigate a range of alternatives and options:

a. Various combinations of primary and secondary payloads.

b. Minimum and maximum power requirements.

Table 6-9. Power Requirements, Experimental Platform Concept 2

Power Load		Watts
Communications Payloads		3082
OSS Payloads		150
OSTA Payload		300
TT&C		297
Central Communications Control		198
Attitude Control System		209
Active Stabilization		90
Load Contingency		216
	Total Power Load	4542
Power Losses		
Distribution		111
Battery Charge		465
Conditioning/Regulating		402
	Total Power Losses	978
Array Contingencies		
Worst-Case Solar Incidence		458
5 Percent Design Margin		299
	Total Contingencies	757
	Total Array EOL Requirement	6277
	Total Array BOL Requirement	7798

c. Number of Shuttle flights (1 or 2).

d. Capabilities with respect to Shuttle flight constraints.

e. Transfer vehicle options.

f. Types of antennas.

g. Structural configuration options.

Input data used in developing the concepts was derived from the results of Tasks 2, 3, 4, and 5; from reference documents listed previously; and from the analyses in Section 6.6:

a. Identification of candidate technologies.

b. Identification of candidate payloads.

Table 6-10. Power Subsystem Weight Estimate, Experimental Platform Concept 2

		Weight lb	Weight kg
Solar Array			
BOL Power, watts	7,798		
Cell Area (15,868 W/sq ft), sq ft	477		
Panel Area (sq ft × 1.09), sq ft	520		
Number of Solar Cells	57,020		
Weights			
Cells, Cover Glass, Adhesive		58	
Blanket and Substructure		26	
Hinge and Latch, Array Harness, Mount, Clampdowns		45	
Gimbal Electronics		2	
Total Array Weight		131	59
Battery Subsystem			
Capacity, watt-hours	5,902		
Weights			
Batteries		640	
Harness		10	
Controller		4	
Peak Tracking Regulator		31	
Load Converter/Regulator		62	
Electronic Assembly		4	
Power Switches		7	
Power Transfer Relay		4	
Power Conductors		22	
Distribution Harness		150	
Total Battery Subsystem Weight		934	424
Total Power Subsystem Weight			483

c. Analysis of mission options.

d. Structural element concepts.

e. Structural growth potential.

f. Analysis of platform subsystem requirements.

6.7.1 CANDIDATE ANTENNA CONFIGURATIONS. A significant lesson was learned from Task 3. In developing conceptual designs for platforms and their communications payloads, packaging proved to be the major design driver in most cases, particularly with respect to dish antenna types and deployment concepts. Six reflector configurations were considered for application on the experimental platforms:

a. Offset parabola.

b. Offset parabola with frequency selective subreflector.

c. Confocal.

d. Offset Cassegrain.

e. Gregorian.

f. Triple reflector.

Of these, the offset Cassegrain proved to be the most compatible with packaging and performance design requirements, with good magnification and scan performance, and good F/D ratios. The offset parabolic configuration generally required too large an F/D for effective packaging. Using the same basic reflector, the effective F/D for the Cassegrain is double that of the offset parabola alone, decreasing scan gain loss (e.g., 3 dB and 0.5 dB) and minimizing sidelobes. The confocal and triple reflector configurations are too complex, and the Gregorian, while offering less offset, requires too long a path. Dielectric lenses and phased arrays show even better characteristics than the offset Cassegrain, but the complexity and other problems associated with them are not worth the little advantage to be gained from the slightly reduced gain loss.

No single deployable antenna concept can satisfy all the desirable characteristics for an experimental platform application. Shown in Figure 6-23 and in Table 6-11 are the three most promising antenna concepets for existing, deployable, offset antennas.

The parabolic expandable truss antenna, or PETA, has an advantage of high density packaging in a short cylindrical volume, and can be side-mounted with no support to the central axis.

The wrapped rib antenna is lightweight, and the proposed design shown here has an extremely small packaged volume in the shape of a flywheel, an advantage offset in some packaging concepts by the necessity for a mast or boom extending out to the center axis.

The sunflower solid-surface antenna is best suited for high-frequency applications, but suffers from a larger packaging volume.

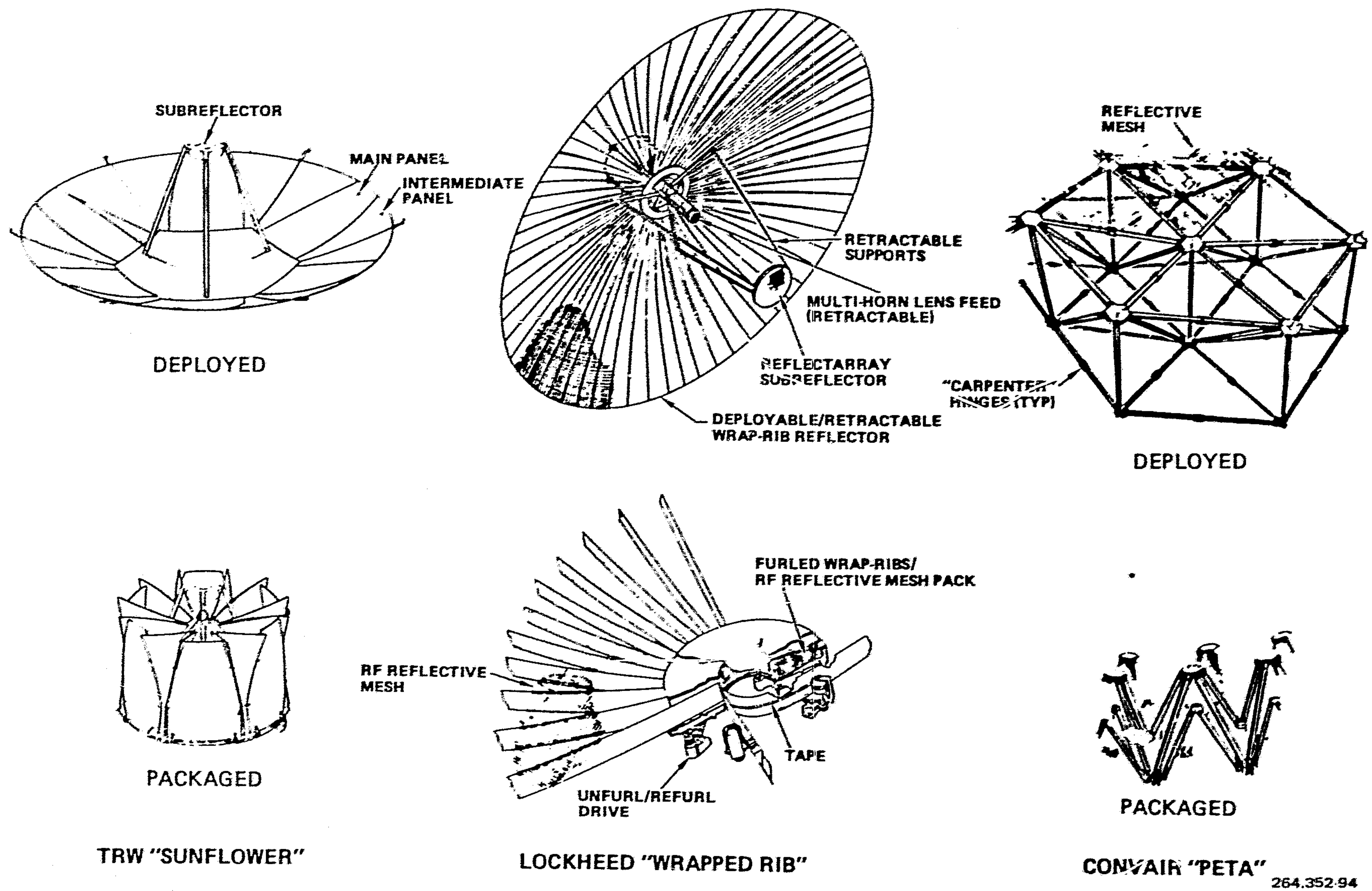

Figure 6-23. Candidate Antenna Concepts

Table 6-11. Existing, Deployable Antenna Concepts for the Experimental Geostationary Platform

Deployed Diameter (ft)	Packaged Dimensions, ft (diameter by length) Convair "PETA" (Type A)	Lockheed "Wrapped Rib"[1]	TRW "Sunflower" Solid Surface (12 Panel)
5	0.75 × 0.49	1.5 × 1.0	1.96 × 1.78
10	0.50 × 0.98	1.7 × 1.5	3.60 × 3.90
26	3.73 × 2.45	2.5 × 2.5	9.80 × 8.90
50	7.50 × 5.00	3.7 × 4.0	17.90 × 19.60
75	11.20 × 7.35	5.1 × 6.0	29.40 × 26.80
100	14.90 × 9.80	5.9 × 8.0	35.70 × 39.20

[1] Improved ATS Design.

As experimental platform concepts were developed, the advantages and disadvantages of candidate antenna types were evaluated to provide the most compatible configurations. Antenna types must continue to be evaluated as the experimental platform program progresses, to ensure the best possible selection compatible with technology requirements and demonstration, performance, weight, and packaging constraints.

Wave guide considerations, servicing compatibility, mast length, and feed horn and subreflector sizes have also proven to be as important to platform design as antenna configurations, and will undoubtedly influence follow-on preliminary experimental platform designs.

6.7.2 CANDIDATE PLATFORM CONFIGURATIONS. To encompass the range of alternatives and options needed to properly evaluate feasibility of the experimental platform concept, six platform configurations were developed. As a starting point, dish antennas were assumed to be the primary design challenge, and primary payload dish antennas were accommodated for all bandwidths on Concepts 1 and 2, as shown in Table 6-12. Concept 1 had no DoD or science payloads aboard. By rearranging payloads and adding an additional horn/subreflector mast for Concept 2, three science payloads were added to the same basic 6-arm structure, with an increase in platform weight that changed it from a low-thrust Centaur class payload to a low-thrust offloaded IOTV class payload, as shown in the table.

Table 6-12. Experimental Platform Concepts - Payload Allocation and Platform Weight Summary

Concept	No. of Arms	No. of Shuttle Flights	Payloads						Platform Location		Total Platform Weight, kg
			Communications					DoD-Science			
			C	Ka	L	Ka	IPL	Payload No.	WH	ATL	
1	6	1	X	X	X	(1)	X	None	X†	X†	4718*
2	6	1	X	X	X	(1)	X	17, 75, 79	X†	X†	5278**
3	4	1	X	X	X		X	None		X	4261*
4	5	1	X			(1)		33, 43, 56	X		4758*
5	4	1				(1)	X	31, 33, 43, 56, 54, 55, 60, 61	X		4179*
6a	5	2	X			(1)	X	31, 17, 33, 43, 56, 54, 55, 60, 61, 75, 79	X		9021***
6b	5	2	X			(2)	X	31, 17, 33, 43, 56, 54, 55, 60, 61	X		7992***

*Centaur capability, low thrust: 4772.

**IOTV capability, low thrust, offloaded: 5670.

***IOTV capability, low thrust: 9190

†Optional transfer from Western Hemisphere to Atlantic location after initial demonstration phase.

Active elements such as feed horns, when mast-mounted, create design problems in packaging cables and wave guides in deployable masts. To alleviate this problem, core-mounted feed assemblies were tried on Concept 3 and on. This solves the waveguide packaging problem, but results in additional packaging problems with the larger feed assemblies and subreflectors required. The next-step solution will be to eliminate the subreflectors, core-mount the resulting smaller feed assemblies, and mast-mount the dish antennas. Again, this solves the waveguide, subreflector, and feed assembly size problems, but results in more complex antenna packaging/deployment and mast stiffness requirements. This concept will be evaluated in the follow-on studies.

Concepts 3, 4, and 5 were developed for single Shuttle-flight missions stressing DoD and science payloads, and a change in basic structure. These concepts all fell in the Centaur-class transfer vehicle capability range, as shown in Table 6-12.

To investigate the feasibility of a two-Shuttle mission (platform in one, orbital transfer vehicle in the other), a 5-arm structure was tried with a 10-meter C-band antenna, a 30/20 experiment, and most of the candidate secondary payloads as listed in Table 6-7. Two payload combinations were considered. Alternative #1 included two OSS payloads; Alternative #2 replaced these OSS payloads with five additional one-meter Ka-band antennas and a 25 by 25 switch. As shown in Table 6-12, both alternatives fell within the low-thrust, fully-loaded IOTV capability.

Details of the six experimental platform concepts are shown in Figures 6-24 through 6-44 and Tables 6-13 through 6-30, as shown:

Concept 1	Figures 6-24 through 6-26 and Tables 6-13 through 6-15
Concept 2	Figures 6-27 through 6-29 and Tables 6-16 through 6-18
Concept 3	Figures 6-30 through 6-32 and Tables 6-19 through 6-21
Concept 4	Figures 6-33 through 6-35 and Tables 6-22 through 6-24
Concept 5	Figures 6-36 through 6-39 and Tables 6-25 through 6-27
Concept 6	Figures 6-40 through 6-44 and Tables 6-28 through 6-31

The figures include deployed and plan elevation views, section views, and a packaged view of each platform concept, payloads and technologies accommodated, antenna parameters, and weight estimates.

In general, all of the concepts display the following characteristics:

a. A central core or hub enclosing avionics, power, ACS, and communications support subsystems.

b. Four to six semideployable radial arm trusses.

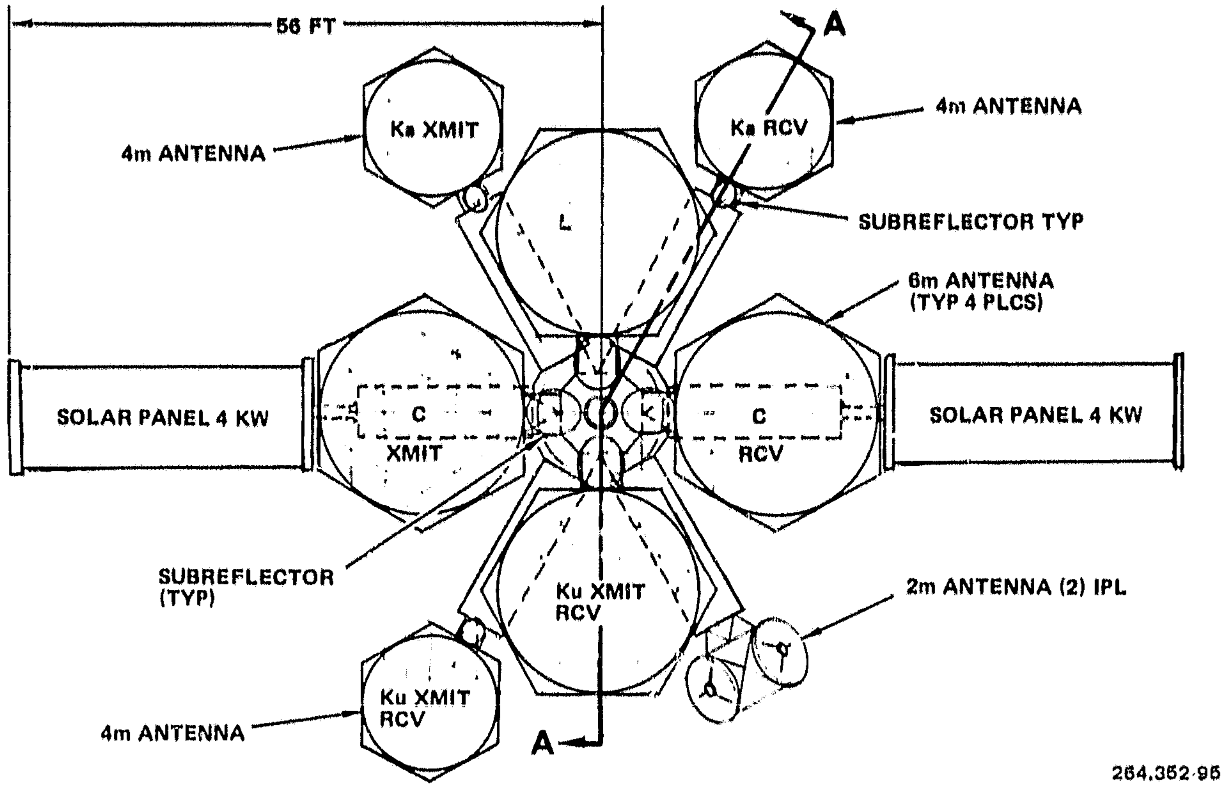

Figure 6-24. Experimental Platform Concept 1, Deployed - Plan View

c. A central core adapter designed to interface with the orbital transfer vehicle or with a service vehicle.

d. Telescoping or deployable masts.

e. Low-CTE structure.

f. Two deployable solar arrays.

g. Fixed antennas, less than 2 meters in diameter.

h. Deployable antennas, over 2 meters in diameter.

i. Orbiter-controlled deployment and checkout with no EVA except for contingencies.

j. A 15 percent weight design contingency factor.

k. Centaur and IOTV-class platform weights.

(Continued on Page 6-83)

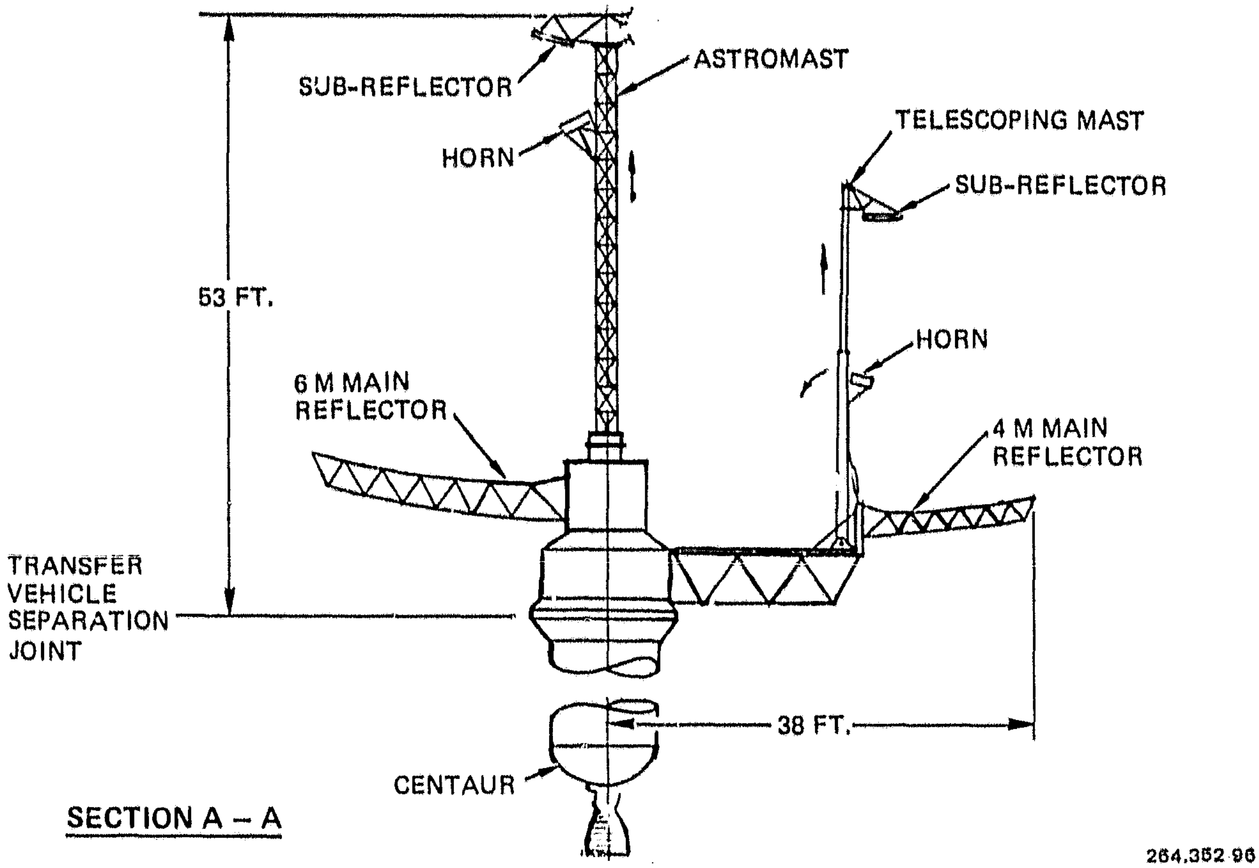

Figure 6-25. Experimental Platform Concept 1, Deployed - Side View

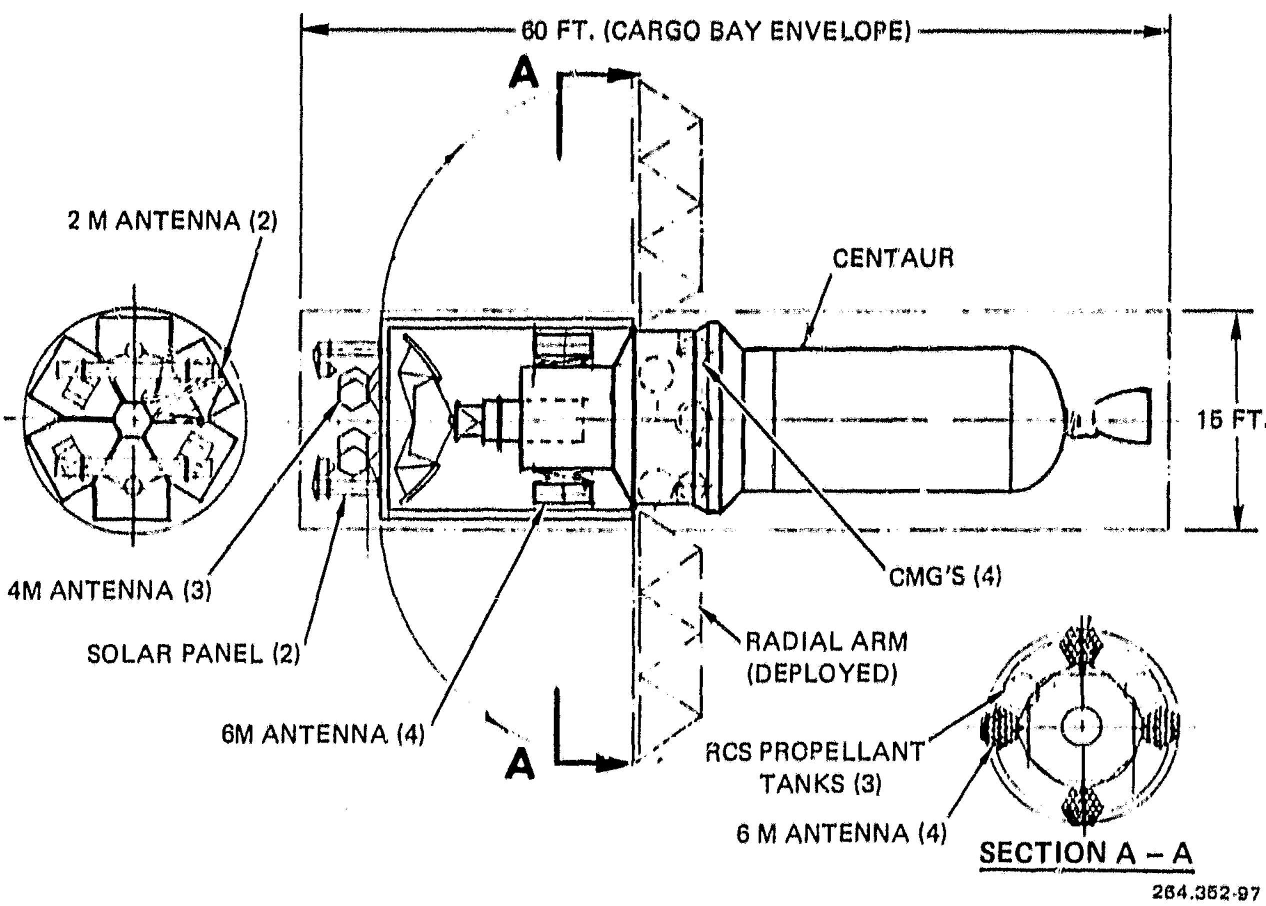

Figure 6-26. Experimental Platform Concept 1, Packaged

Table 6-13. Experimental Platform Concept 1, Payloads and Technologies

Payloads		
C Band	–	Two 6-meter antennas; 10 × 10 RF switch
Ku Band	–	One 6-meter, one 4-meter antenna
L Band	–	One 6-meter antenna
Ka Band	–	Two 4-meter antennas
IPL	–	Two 2-meter antennas
Technology Demonstrations		
Advanced communications technology - 4 separate bands		
C Band beam shaping		
IPL technology		
Deployable antennas		
All platform technologies		

Table 6-14. Experimental Platform Concept 1, Antenna Characteristics

Freq Desig	Freq. GHz	Antenna Dia. Type	Eff F/D	Function	Beam Width	Point. Acc.	Coverage	Beam Type	Scanned Angle	Notes
C-BAND	4	6M, O/CASS, MBFR	0.7	XMIT	0.6°	0.06°	SECTOR	RECONFIG.	3 BW	SMALL FEED
	6	6M, O/CASS, MBFR	0.7	RCV	0.9°	0.10°	SECTOR	RECONFIG.	3 BW	SMALL FEED
	4	0.25M HORN	—	XMIT	18°	1.0°	EARTH	FIXED	—	
	6	0.16M HORN	—	RCV	18°	1.0°	EARTH	FIXED	—	
Ku-BAND	14/11	6M, O/CASS, MBFR	0.8	T/R	0.28°	0.03°	CONUS	SWITCHED	6 BW	W/FSS
	14/11	4M, O/CASS, MBFR	0.8	T/R	0.42°	0.04°	EUROPE	SWITCHED	5 BW	W/FSS
Ka-BAND	20	4M, O/CASS, MBFR	1.0	XMIT	0.3°	0.03°	CON/EUR	FIX/SCAN'G	10 BW	
	30	4M, O/CASS, MBFR	1.0	RCV	0.2°	0.02°	CON/EUR	FIX/SCAN'G	15 BW	
L-BAND	1.6/1.5	6M, O/CONFOCAL	0.5	T/R	2.3°	0.2°	1 AREA/2 SPOT	FIXED	4 BW	
	1.6/1.5	0.4M, HELIX ARR.	—	T/R	18°	1.0°	EARTH	FIXED	—	
	6/5	0.2M, HORN	—	T/R	18°	1.0°	EARTH	FIXED		
IPL	32	2M, CASS	0.4	T OR R	0.3°	0.03°	GIMBAL	PENCIL	—	
	25	2M, CASS	0.4	T OR R	0.4°	0.04°	GIMBAL	PENCIL	—	

Table 6-15. Experimental Platform Concept 1, Weight Estimate

	Estimated Weight, kg		Contingency, 15%, kg		Total, kg
<u>PLATFORM</u>					
STRUCTURE	1,167				
THERMAL CONTROL	177				
ATTITUDE CONTROL	880				
ELECTRICAL POWER (7.5 KW)	473				
AVIONICS	200				
	2,897	+	434	=	3,331
<u>PAYLOADS</u>					
C-BAND	299				
Ku-BAND	256				
L-BAND	131				
Ka-BAND	392				
IPL	78				
SECONDARY	50				
	1,206	+	181	=	1,387
TOTAL PLATFORM WEIGHT, WITH 15% CONTINGENCY:					4,718
<u>TRANSFER VEHICLE</u> – CENTAUR STS (4,763 KG CAPABILITY)					16,083
<u>ASE</u>					4,424
ORBITER LAUNCH WEIGHT: (29,484 KG CAPABILITY)					25,225

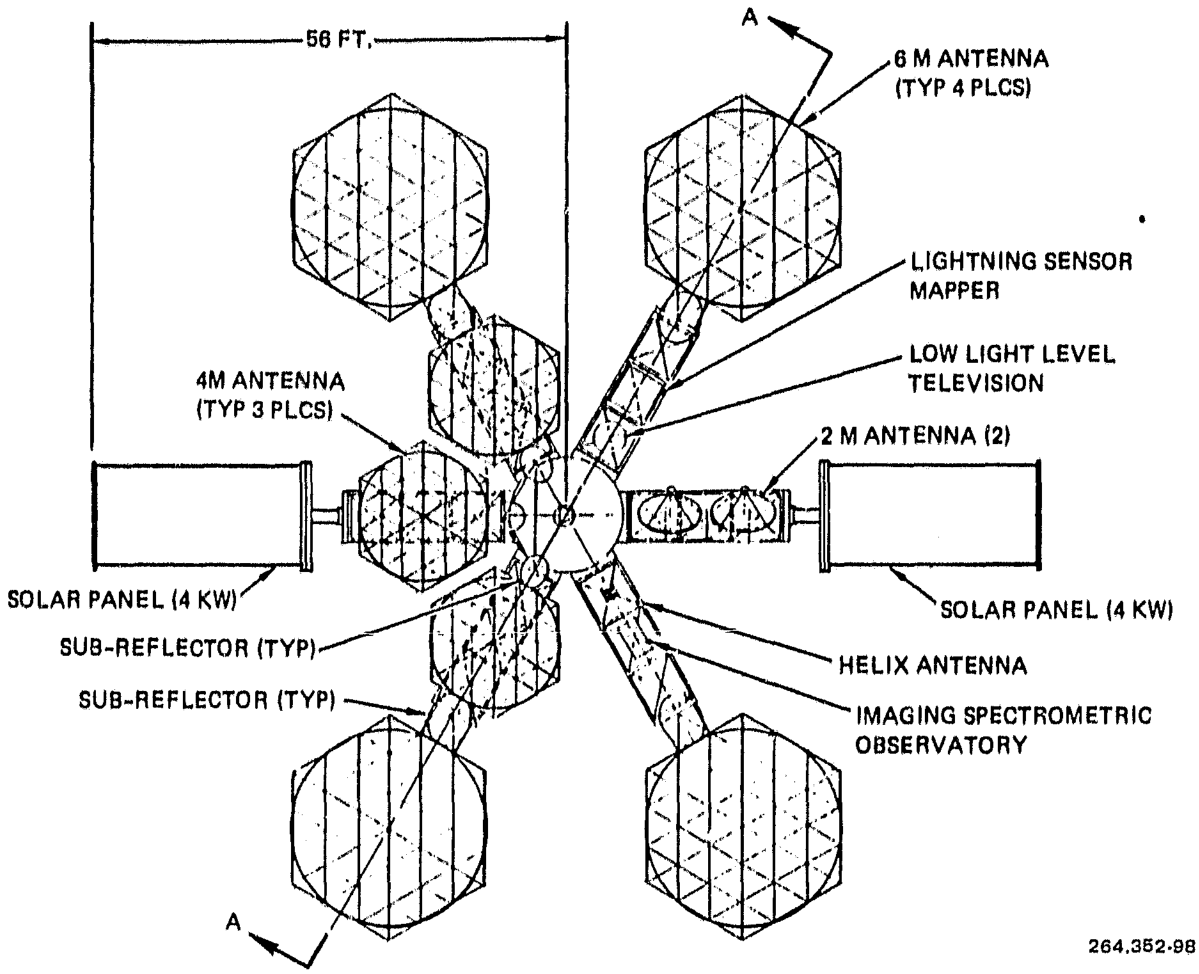

Figure 6-27. Experimental Platform Concept 2, Deployed - Plan View

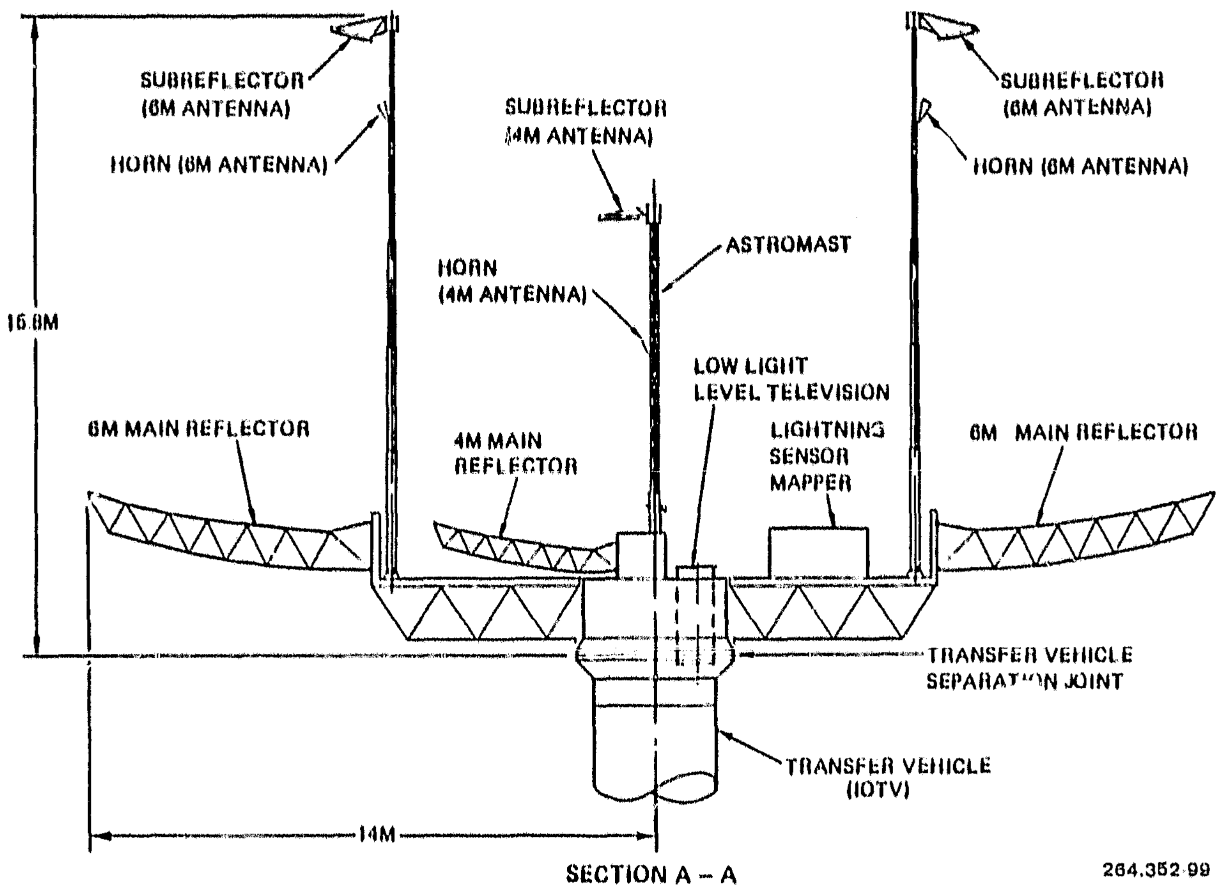

Figure 6-28. Experimental Platform Concept 2, Deployed - Side View

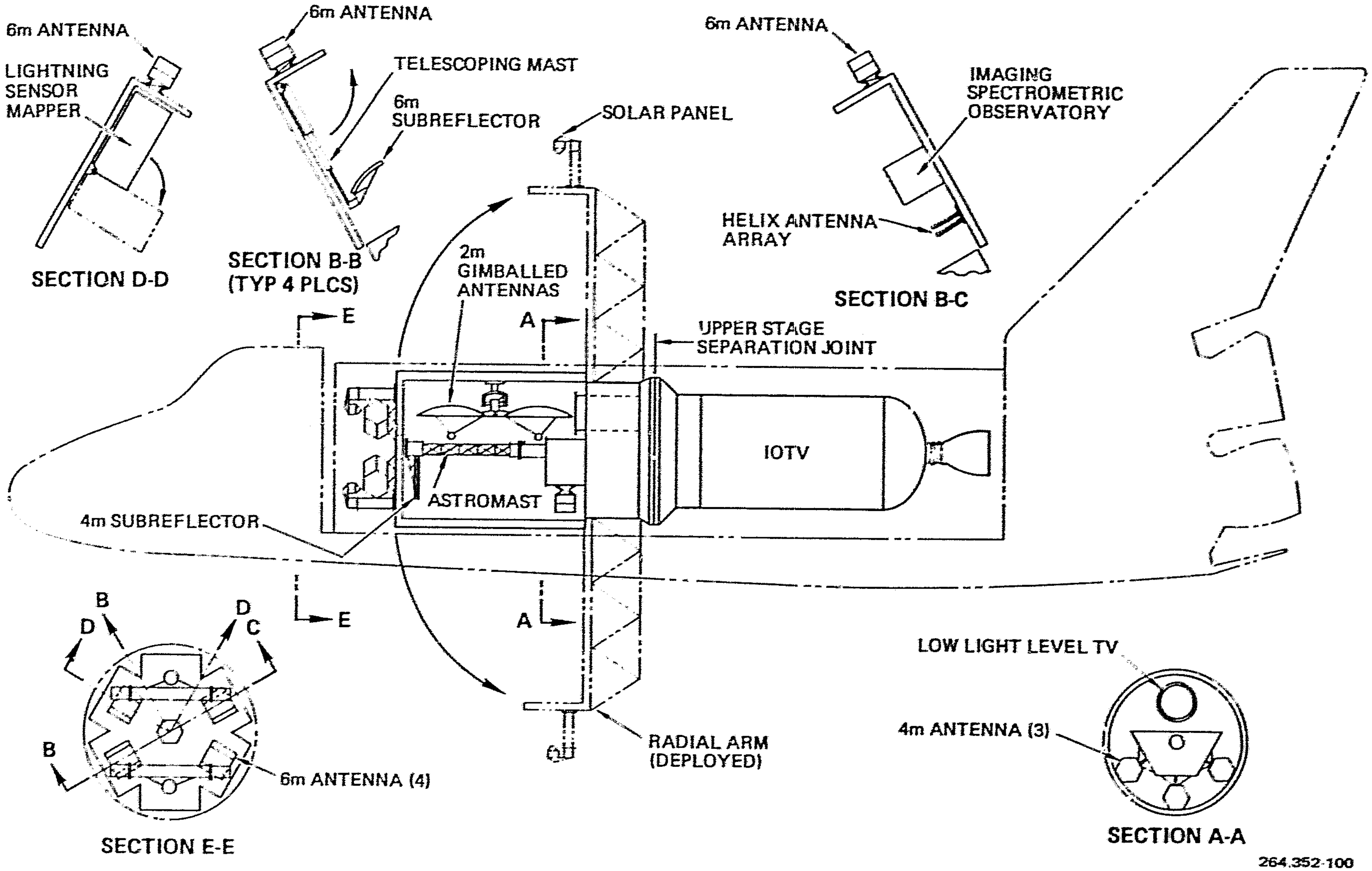

Figure 6-29. Experimental Concept 2, Packaged

Table 6-16. Experimental Platform Concept 2, Payloads and Technologies

Payloads

C Band	–	Two 6-meter antennas; 10 × 10 RF switch
Ku Band	–	One 6-meter, one 4-meter antenna
L Band	–	One 6-meter antenna
Ka Band	–	Two 4-meter antennas
IPL	–	Two 2-meter antennas
OSS #75	–	Imaging spectrometer observatory
OSS #79	–	Low light level TV
OSTA #17	–	Lightning mapper

Technology Demonstrations

Advanced communications technology - 4 separate bands

C Band beam shaping

IPL technology

Deployable antennas

All platform technologies

Table 6-17. Experimental Platform Concept 2, Antenna Characteristics

Freq Desig	Freq. GHz	Antenna Dia. Type	Eff F/D	Function	Beam Width	Point. Acc.	Coverage	Beam Type	Scanned Angle	Notes
C-BAND	4	6M, O/CASS, MBFR	0.7	XMIT	0.6°	0.06°	SECTOR	RECONFIG.	3 BW	SMALL FEED
	6	6M, O/CASS, MBFR	0.7	RCV	0.9°	0.10°	SECTOR	RECONFIG.	3 BW	SMALL FEED
	4	0.25M HORN	—	XMIT	18°	1.0°	EARTH	FIXED	—	
	6	0.16M HORN	—	RCV	18°	1.0°	EARTH	FIXED	—	
Ku-BAND	14/11	6M, O/CASS, MBFR	0.8	T/R	0.28°	0.03°	CONUS	SWITCHED	6 BW	W/FSS
	14/11	4M, O/CASS, MBFR	0.8	T/R	0.42°	0.04°	EUROPE	SWITCHED	5 BW	W/FSS
Ka-BAND	20	4M, O/CASS, MBFR	1.0	XMIT	0.3°	0.03°	CON/EUR	FIX/SCAN'G	10 BW	
	30	4M, O/CASS, MBFR	1.0	RCV	0.2°	0.02°	CON/EUR	FIX/SCAN'G	15 BW	
L-BAND	1.6/1.5	6M, O/CONFOCAL	0.5	T/R	2.3°	0.2°	1 AREA/2 SPOT	FIXED	4 BW	
	1.6/1.5	0.4M, HELIX ARR.	—	T/R	18°	1.0°	EARTH	FIXED	—	
	6/5	0.2M, HORN	—	T/R	18°	1.0°	EARTH	FIXED		
IPL	32	2M, CASS	0.4	T OR R	0.3°	0.03°	GIMBAL	PENCIL	—	
	25	2M, CASS	0.4	T OR R	0.4°	0.04°	GIMBAL	PENCIL	—	

Table 6-18. Experimental Platform Concept 2, Weight Estimate

	Estimated Weight, kg		Contingency, 15%, kg		Total, kg
PLATFORM					
STRUCTURE	1,167				
THERMAL CONTROL	182				
ATTITUDE CONTROL	1,002				
ELECTRICAL POWER (7.8 KW)	483				
AVIONICS	200				
	3,034	+	455	=	3,489
PAYLOADS					
C-BAND	299				
Ku-BAND	256				
L-BAND	131				
Ka-BAND	392				
IPL	78				
SECONDARY	400				
	1,556	+	233	=	1,789
TOTAL PLATFORM WEIGHT, WITH 15% CONTINGENCY:					5,278
TRANSFER VEHICLE — IOTV (5,670 KG CAPABILITY)					19,090
ASE					2,566
ORBITER LAUNCH WEIGHT: (29,484 KG CAPABILITY)					26,934

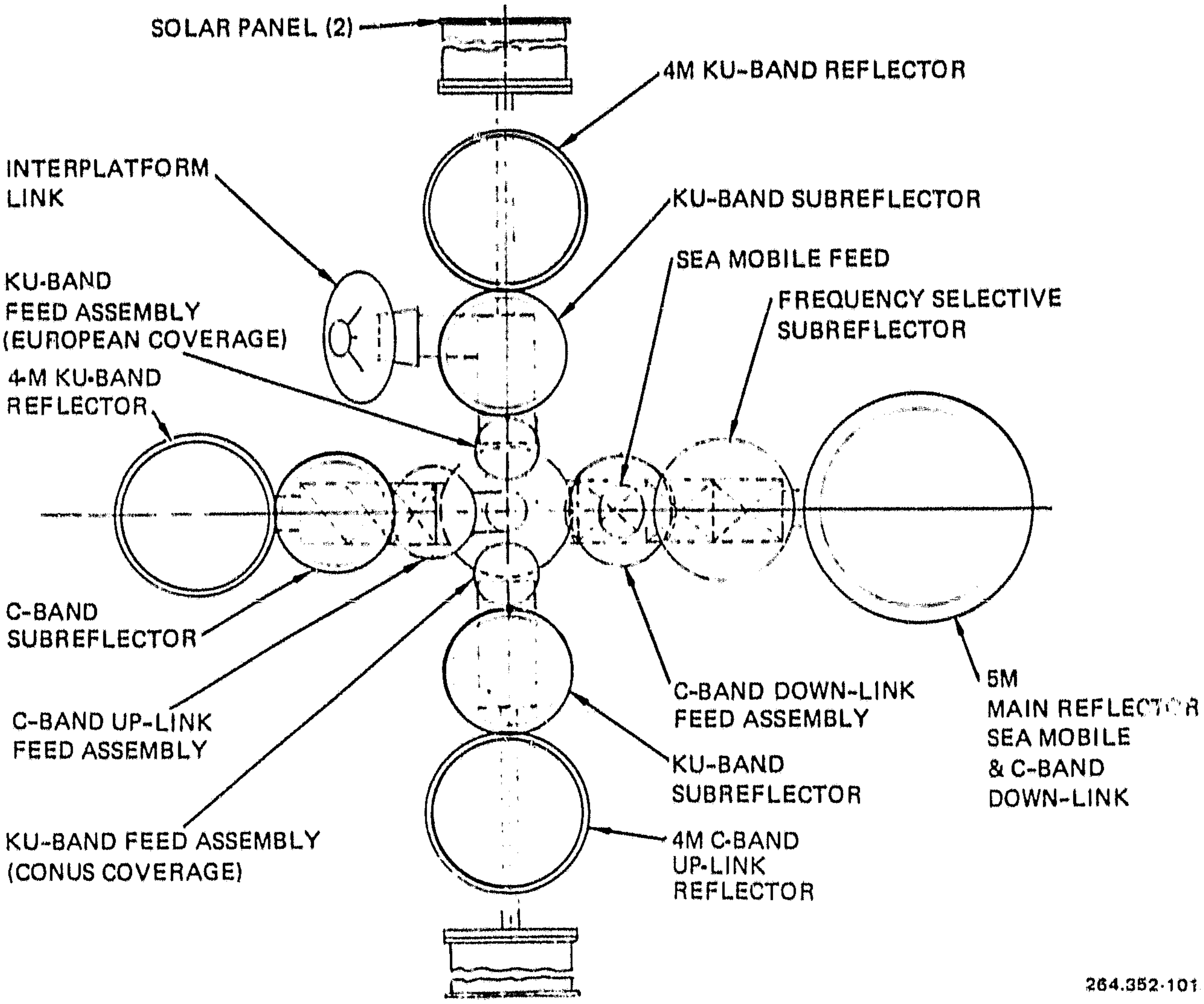

Figure 6-30. Experimental Platform Concept 3, Deployed - Plan View

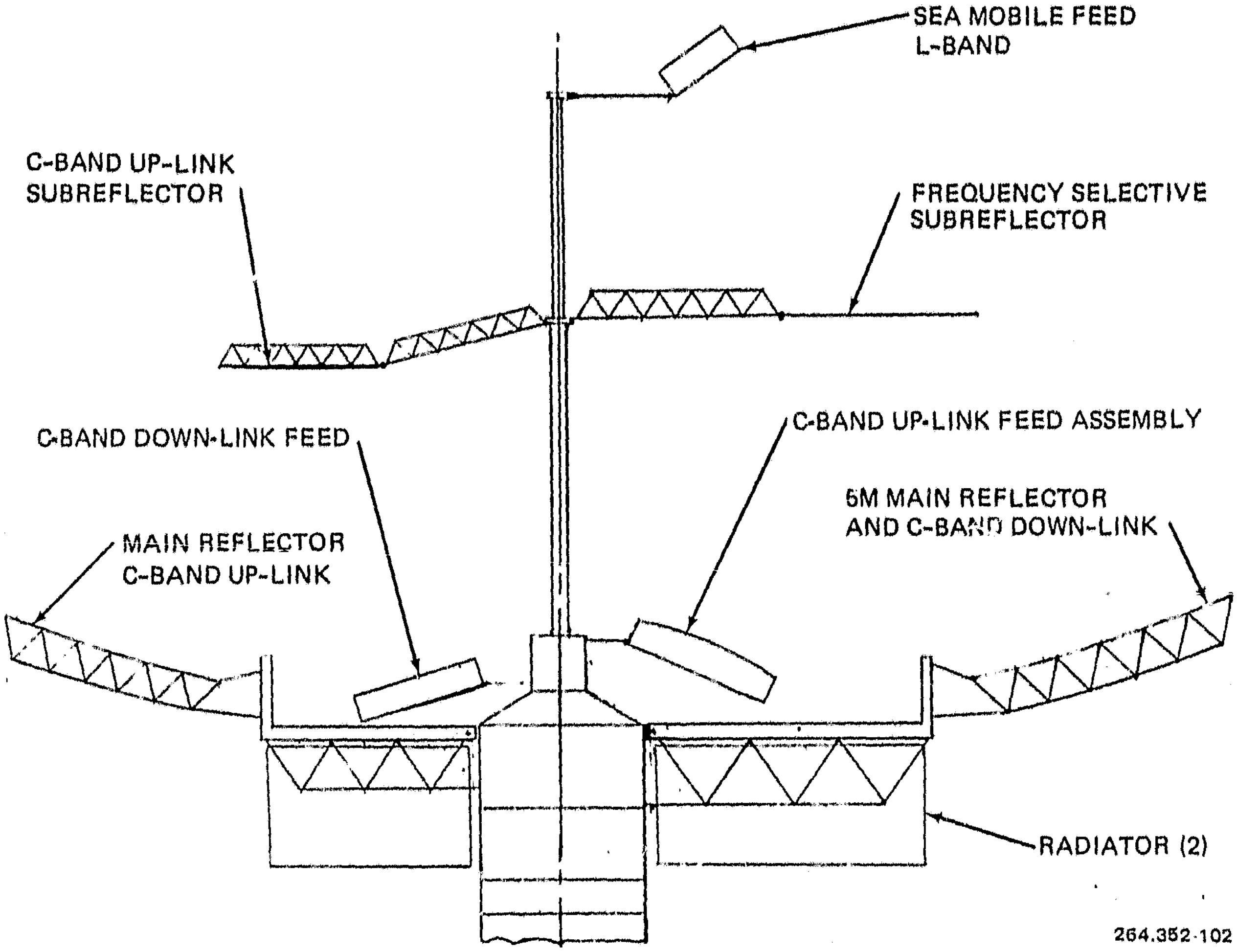

Figure 6-31. Experimental Platform Concept 3, Deployed - Side View

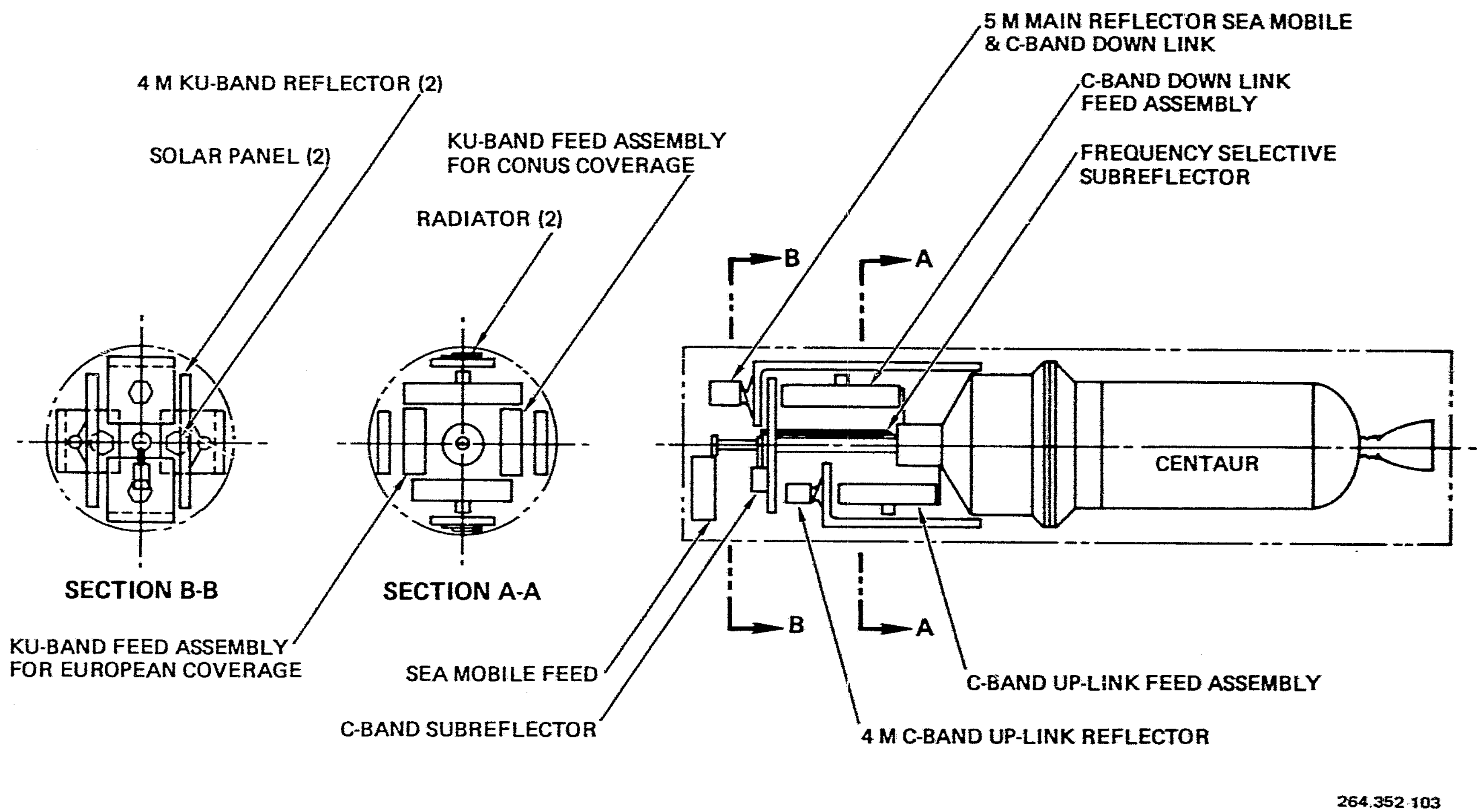

Figure 6-32. Experimental Platform Concept 3, Packaged

Table 6-19. Experimental Platform Concept 3, Payloads and Technologies

Payload

C-Band	–	One 5-meter, one 4-meter antenna; 10 × 10 switch; baseband processor.
Ku Band	–	Two 4-meter antennas; satellite switch.
L Band	–	One 5-meter antenna (shared with 6 GHz C-Band).
IPL	–	One 2.4-meter antenna.

Technology Demonstrations

Advanced communications technology - four separate bands.

C-Band beam shaping/reconfigurability.

C-Band direct-to-user.

Ku Band switch beam shaping.

Frequency selective subreflector surface - C-Band and L-Band.

IPL technology.

Deployable antennas.

All platform technologies.

Table 6-20. Experimental Platform Concept 3, Antenna Characteristics

Freq Desig	Freq. GHz	Antenna Dia. Type	Eff F/D	Function	Beam Width	Point. Acc.	Coverage	Beam Type	Scanned Angle	Notes
C-BAND	4	5M, O/CASS, MBFR	0.6	XMIT	1.0°	0.1°	EARTH	RECONFIG.	9 BW	W/L-FSS
	6	4M, O/CASS, MBFR	0.6	RCV	0.9°	0.1°	EARTH	RECONFIG.	10 BW	
	4	0.25M HORN	—	XMIT	18°	1.0°	EARTH	FIXED	—	
	6	0.16M HORN	—	RCV	18°	1.0°	EARTH	FIXED	—	
Ku-BAND	14/11	4M, O/CASS, MBFR	0.7	T/R	0.4/0.5°	0.04°	CONUS	SWITCHED	11/9 BW	W/FSS
	14/11	4M, O/CASS, MBFR	0.7	T/R	0.4/0.5°	0.04°	EUROPE	SWITCHED	11/9 BW	W/FSS
L-BAND	1.6/1.5	5M, O/PARA, MBFR	0.6	T/R	2.2°	0.2°	1 AREA/2 SPOT	FIXED	4 BW	W/C-FSS
	1.6/1.5	0.4M, HELIX ARR.	—	T/R	18°	1.0°	EARTH	FIXED	—	
	6/5	0.2M, HORN	—	T/R	18°	1.0°	EARTH	FIXED	—	
IPL	32/25	2.4M, CASS	0.4	T/R	0.3/0.4°	0.03°	GIMBAL	PENCIL	—	W/FSS

Table 6-21. Experimental Platform Concept 3, Weight Estimate

	Estimated Weight, kg		Contingency, 15%, kg		Total, kg
PLATFORM					
STRUCTURE	1,207				
THERMAL CONTROL	165				
ATTITUDE CONTROL	880				
ELECTRICAL POWER (7.0 KW)	427				
AVIONICS	200				
	2,879	+	432	=	3,311
PAYLOADS					
C-BAND	366				
Ku-BAND	270				
L-BAND	120				
Ka-BAND	—				
IPL	70				
SECONDARY	—				
	826	+	124	=	950
TOTAL PLATFORM WEIGHT, WITH 15% CONTINGENCY:					4,261
TRANSFER VEHICLE – CENTAUR STS (4,763 KG CAPABILITY)					16,083
ASE					4,424
ORBITER LAUNCH WEIGHT: (29,484 KG CAPABILITY)					24,768

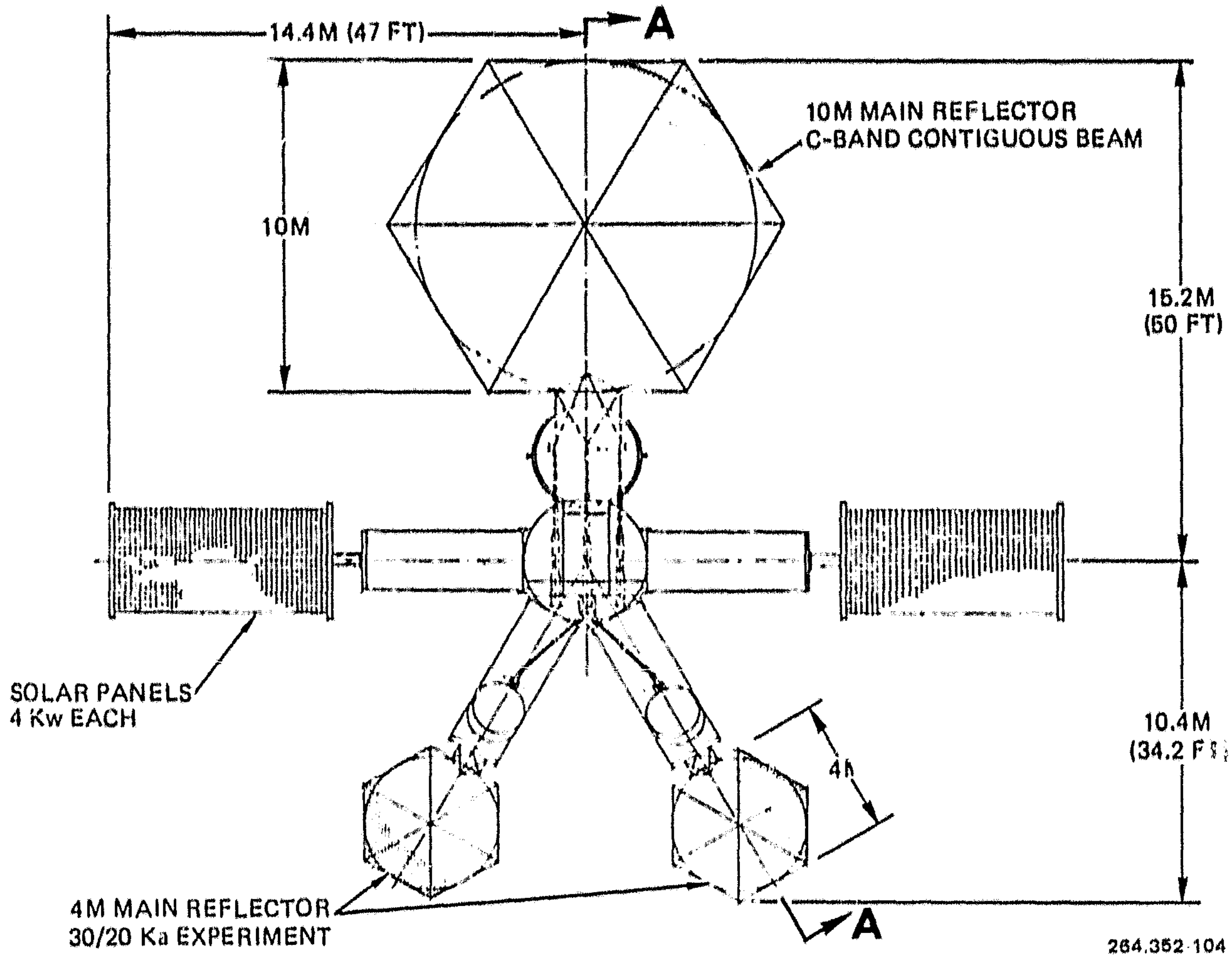

Figure 6-33. Experimental Platform Concept 4, Deployed - Plan View

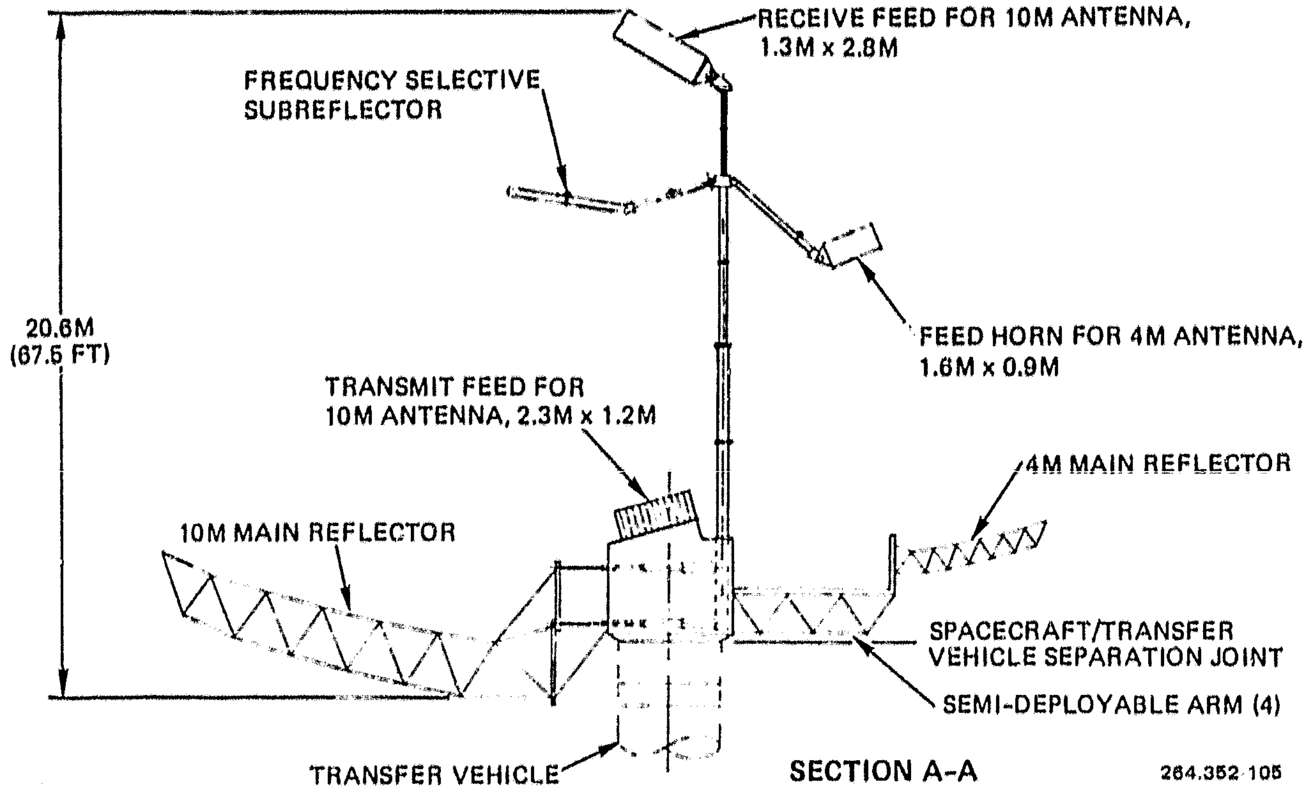

Figure 6-34. Experimental Platform Concept 4, Deployed - Side View

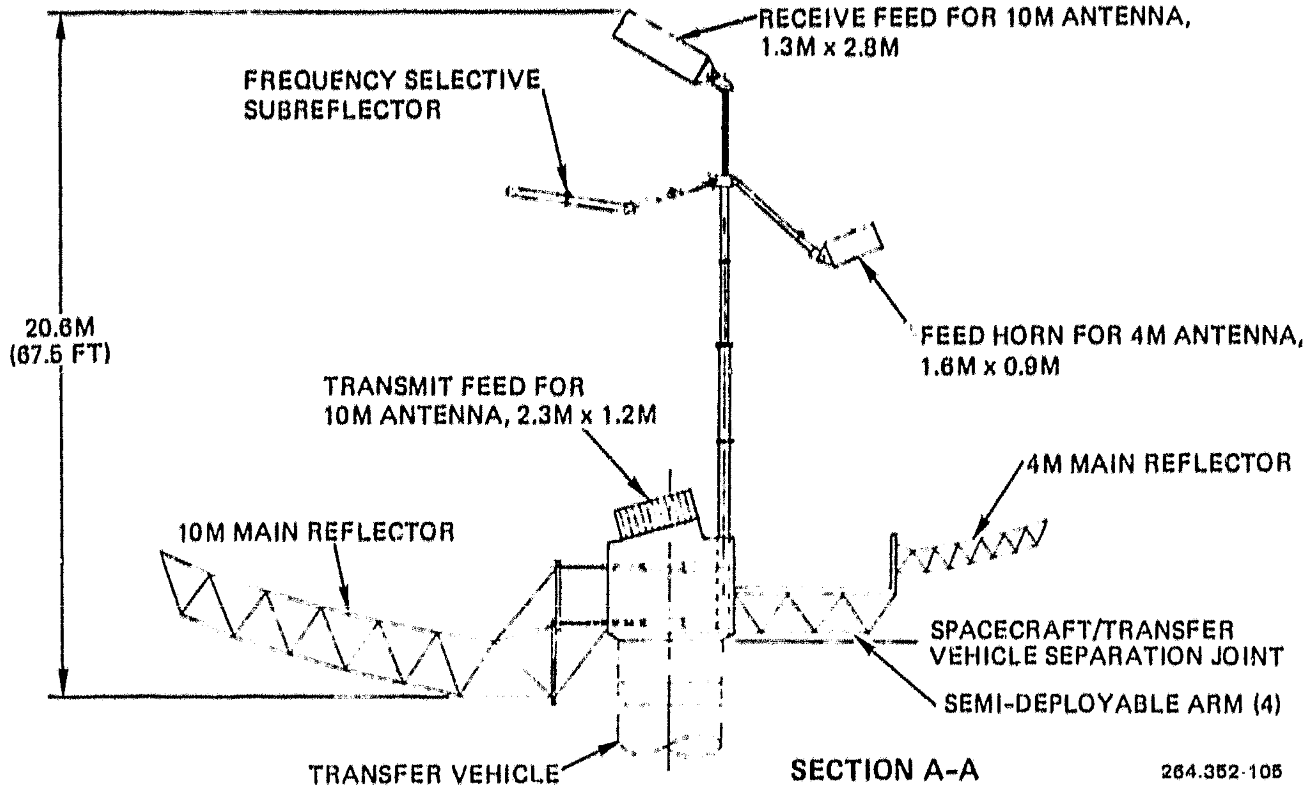

Figure 6-34. Experimental Platform Concept 4, Deployed - Side View

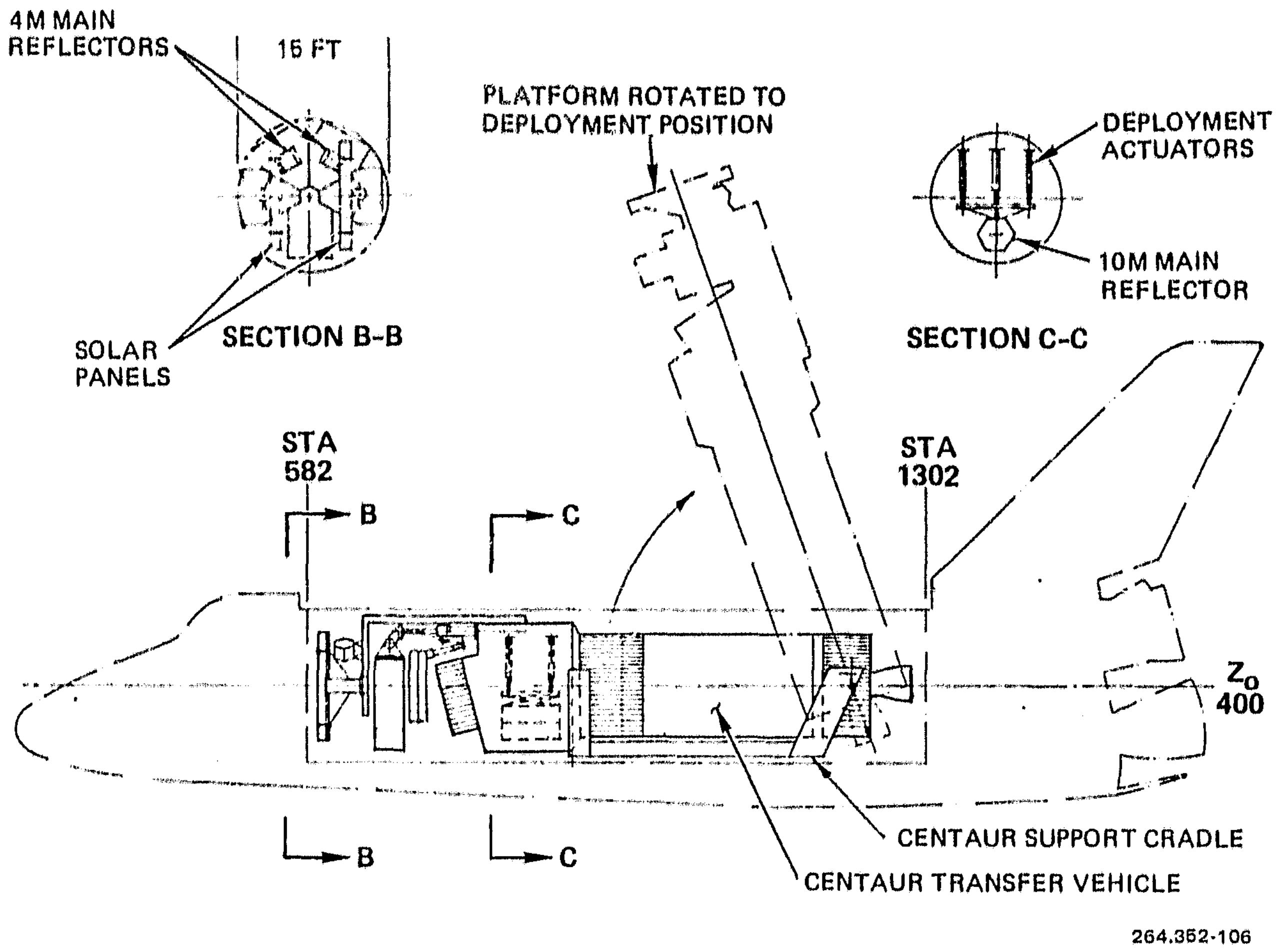

Figure 6-35. Experimental Platform Concept 4, Packaged

Table 6-22. Experimental Platform Concept 4, Payload and Technologies

Payloads		
C Band	–	One 10-meter antenna; 100 × 100 switch; baseband processor
Ka Band	–	Two 4-meter antennas; 10 × 10 switch; baseband processor
DOD #33	–	Materials exposure
DOD #43	–	Magnetic Substorm monitor
DOD #56	–	Fiber optics demonstrator

Technology Demonstrations

- Advanced communications technology – C and Ka Bands
- C Band beam shaping/reconfigurability
- C Band direct-to-user
- C Band 100 reuse satellite switch
- Ka Band beam scanning
- Frequency selective subreflector surface – C Band
- Large deployable antennas
- All platform technologies

Table 6-23. Experimental Platform Concept 4, Antenna Characteristics

Freq Desig	Freq. GHz	Antenna Dia. Type	Eff F/D	Function	Beam Width	Point. Acc.	Coverage	Beam Type	Scanned Angle	Notes
C-BAND	6/4	10M, O/CASS, MBFR	0.6	T/R	0.35/0.5°	0.03°	CONUS	RECONFIG.	12/8 BW	W/FSS
	4	0.25M HORN	—	XMIT	18°	1.0°	EARTH	FIXED	—	
	6	0.16M HORN	—	RCV	18°	1.0°	EARTH	FIXED	—	
Ka-BAND	20	4M, O/CASS, MBFR	1.0	XMIT	0.3°	0.03°	CONUS	FIX/SCAN'G	13 BW	
	30	4M, O/CASS, MBFR	1.0	RCV	0.3°	0.03°	CONUS	FIX/SCAN'G	13 BW	

Table 6-24. Experimental Platform Concept 4, Weight Estimate

	Estimated Weight, kg		Contingency, 15%, kg		Total, kg
PLATFORM					
STRUCTURE	1,450				
THERMAL CONTROL	174				
ATTITUDE CONTROL	880				
ELECTRICAL POWER (7.4 KW)	454				
AVIONICS	200				
	3,158	+	474	=	3,632
PAYLOADS					
C-BAND	453				
Ku-BAND	—				
L-BAND	—				
Ka-BAND	476				
IPL	—				
SECONDARY	50				
	979	+	147	=	1,126
TOTAL PLATFORM WEIGHT, WITH 15% CONTINGENCY:					4,758
TRANSFER VEHICLE — CENTAUR STS (4,763 KG CAPABILITY)					16,083
ASE					4,424
ORBITER LAUNCH WEIGHT: (29,484 KG CAPABILITY)					25,265

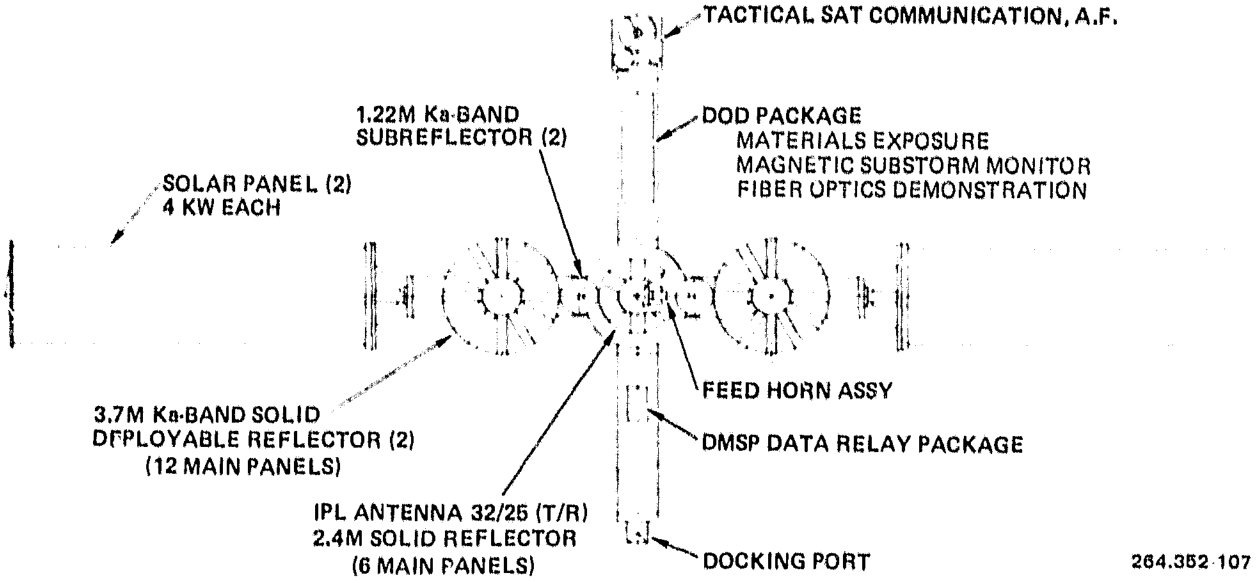

Figure 6-36. Experimental Platform Concept 5, Deployed - Plan View

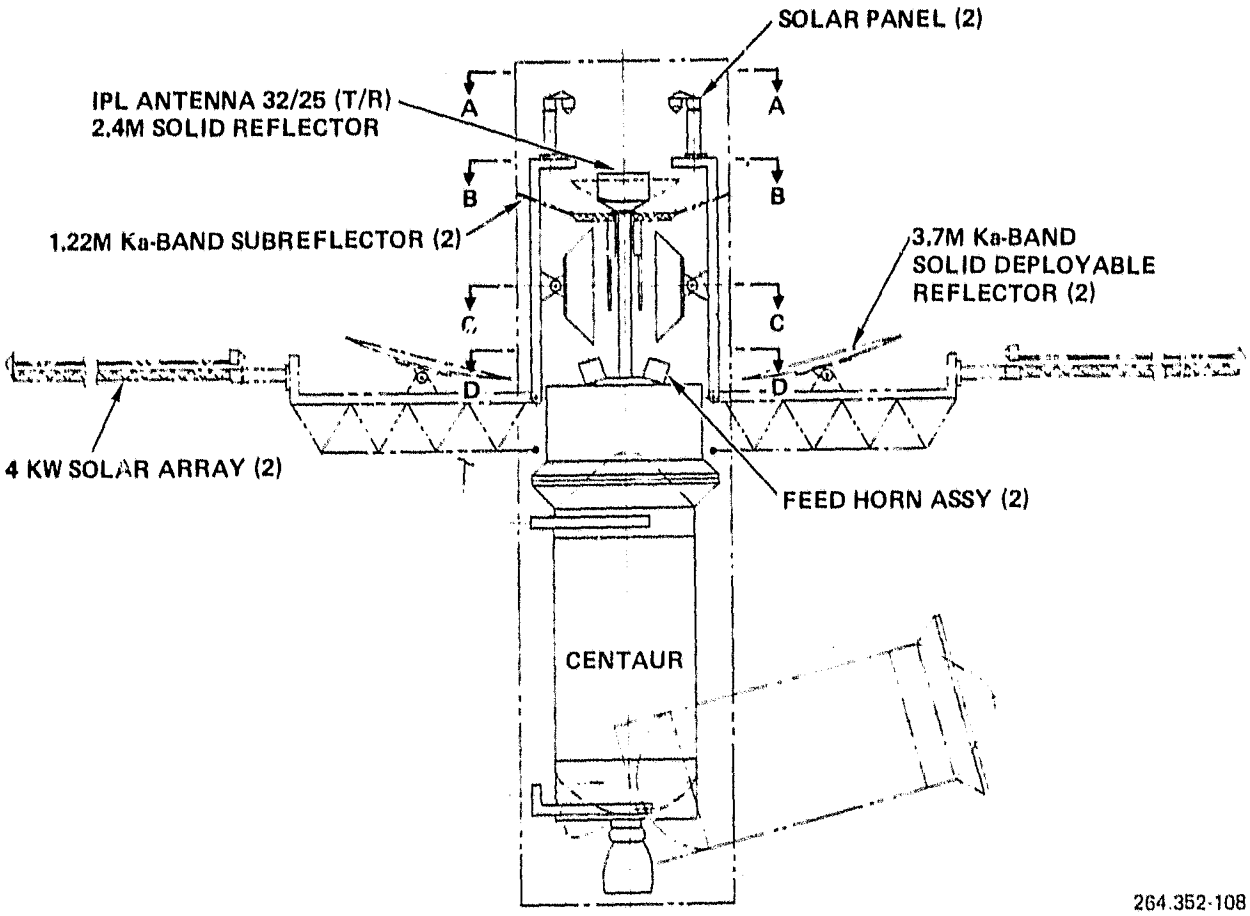

Figure 6-37. Experimental Platform Concept 5, Packaged - North-to-South Side View

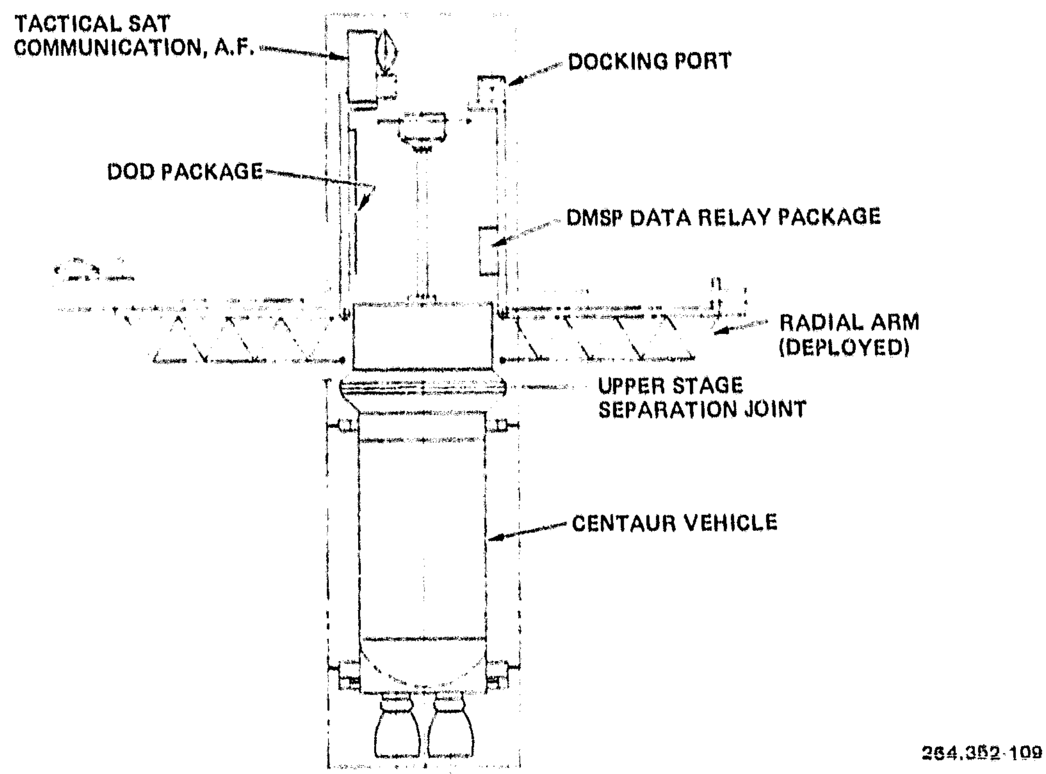

Figure 6-38. Experimental Platform Concept 5, Packaged - East-to-West Side View

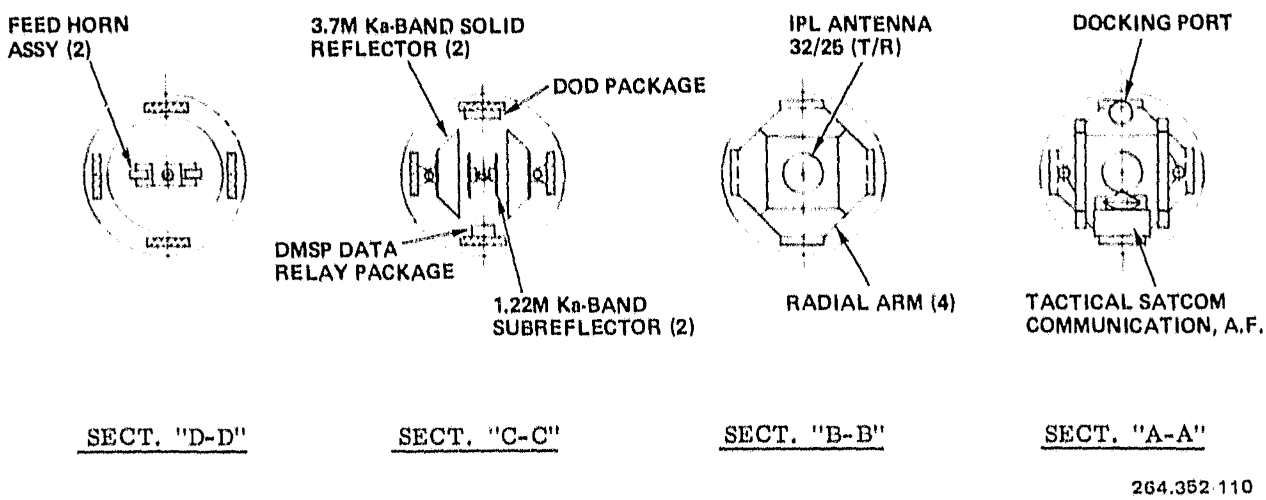

Figure 6-39. Experimental Platform Concept 5, Packaged - Cross Sections

Table 6-25. Experimental Platform Concept 5, Payloads and Technologies

Payloads

Ka Band	–	One 3.7-meter solid surface antenna; 25 × 25 switch; baseband processor
IPL	–	One 2.4-meter solid surface antenna
DOD #31	–	DMSP data relay
Tactical AF satellite communications		
DOD #33	–	Materials exposure
DOD #43	–	Magnetic substorm monitor
DOD #56	–	Fiber optics demonstrator

Technology Demonstrations

Ka Band advanced communications technology

Ka Band beam scanning

High frequency deployable solid surface antennas

All platform technologies

Table 6-26. Experimental Platform Concept 5, Antenna Characteristics

Freq Desig	Freq. GHz	Antenna Dia. Type	Eff F/D	Function	Beam Width	Point. Acc.	Coverage	Beam Type	Scanned Angle	Notes
Ka-BAND	20	3.7M, O/CASS, MBFR	0.8	XMIT	0.3°	0.03°	CONUS	FIX/SCAN'G	13 BW	
	30	3.7M, O/CASS, MBFR	0.8	RCV	0.3°	0.03°	CONUS	FIX/SCAN'G	13 BW	
IPL	32/25	2.4M, CASS	0.4	T/R	0.3/0.4°	0.03°	GIMBAL	PENCIL	—	W/FSS

Table 6-27. Experimental Platform Concept 5, Weight Estimate

	Estimated Weight, kg		Contingency, 15%, kg		Total, kg
PLATFORM					
STRUCTURE	1,074				
THERMAL CONTROL	135				
ATTITUDE CONTROL	820				
ELECTRICAL POWER (5.8 KW)	373				
AVIONICS	200				
	2,602	+	390	=	2,992
PAYLOADS					
C-BAND	—				
Ku-BAND	—				
L-BAND	—				
Ka-BAND	442				
IPL	70				
SECONDARY	520				
	1,032	+	155	=	1,187
TOTAL PLATFORM WEIGHT, WITH 15% CONTINGENCY:					4,179
TRANSFER VEHICLE — CENTAUR STS (4,763 KG CAPABILITY)					16,083
ASE					4,424
ORBITER LAUNCH WEIGHT: (29,484 KG CAPABILITY)					24,686

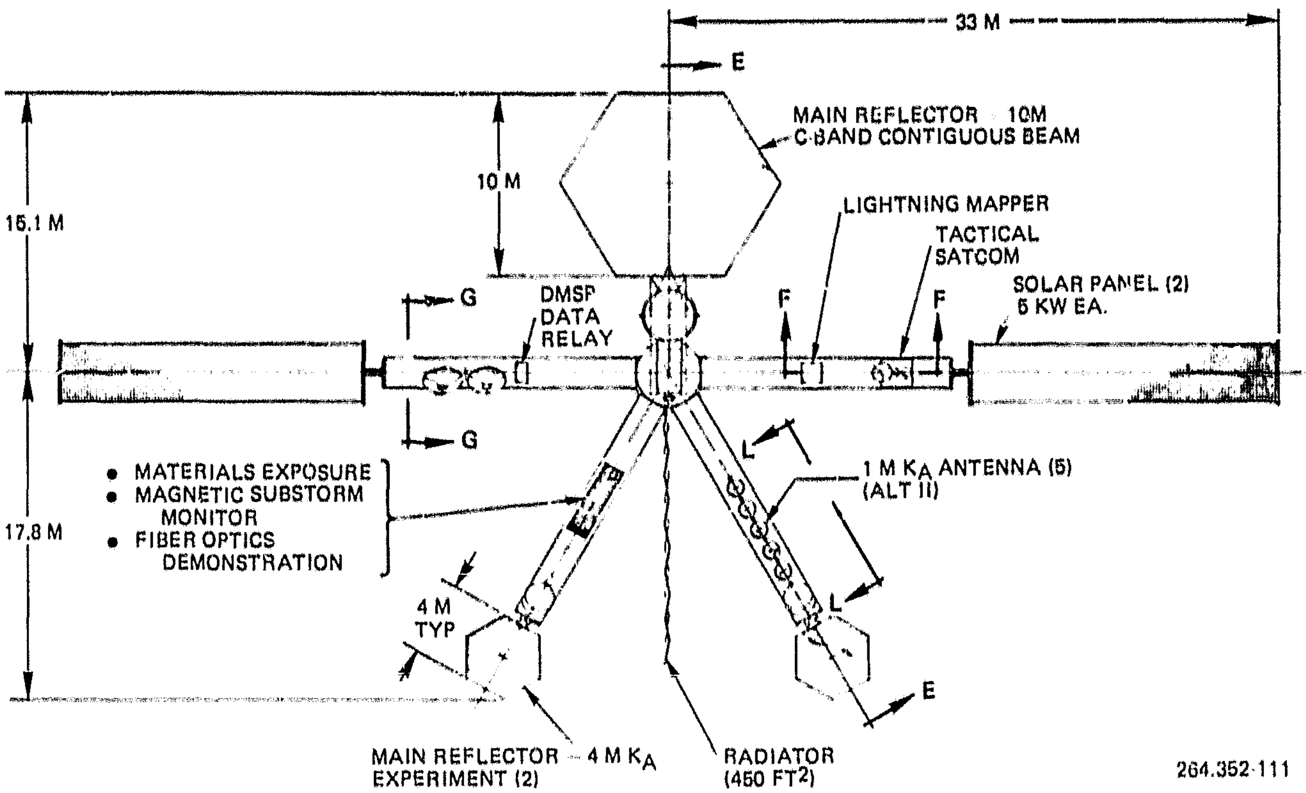

Figure 6-40. Experimental Platform Concept 6, Deployed - Plan View

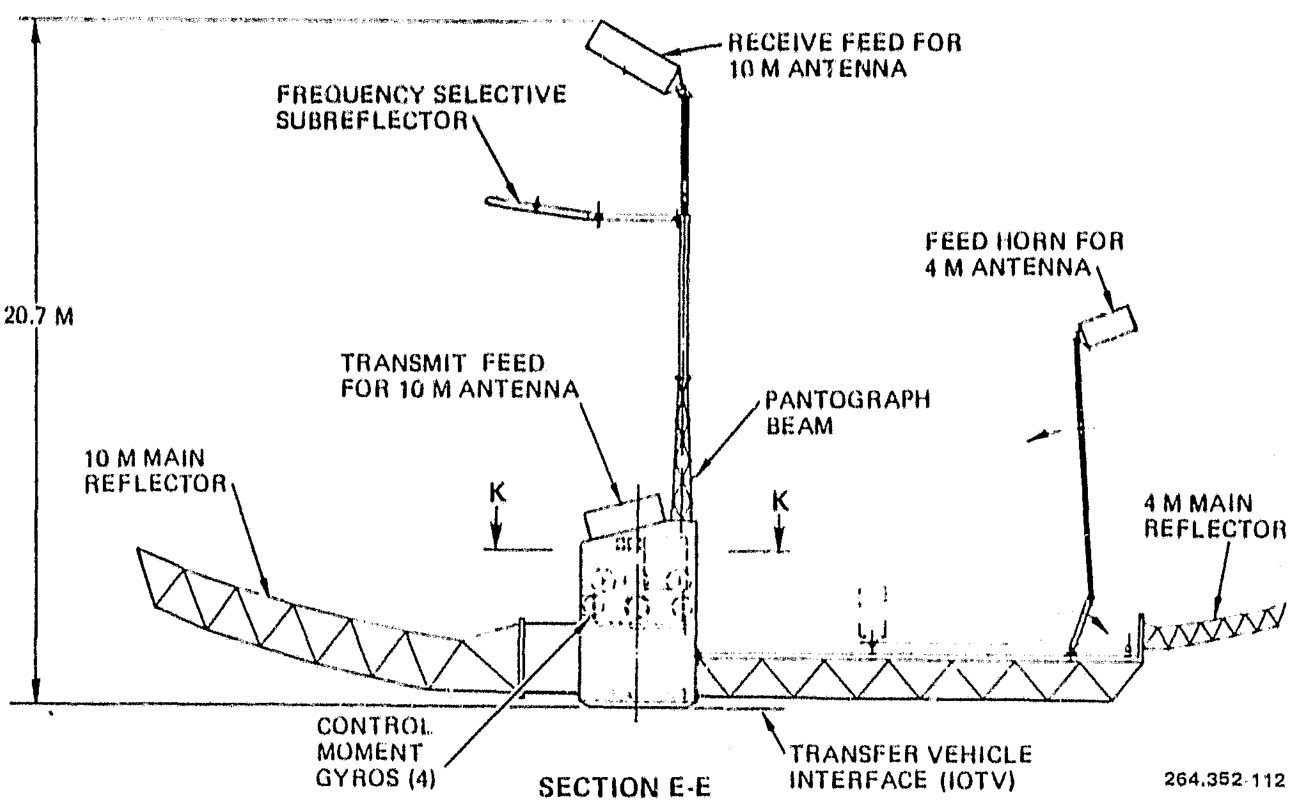

Figure 6-41. Experimental Platform Concept 6, Deployed - Side View

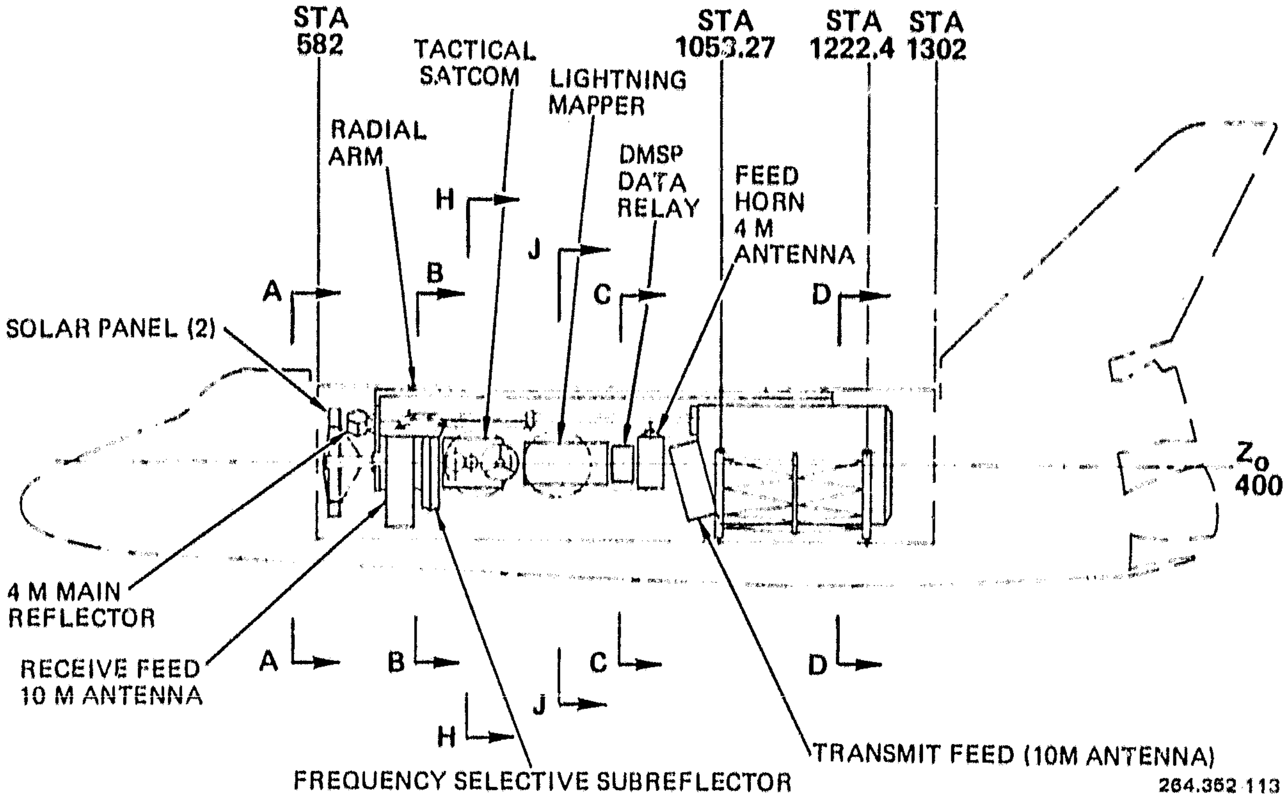

Figure 6-42. Experimental Platform Concept 6, Packaged

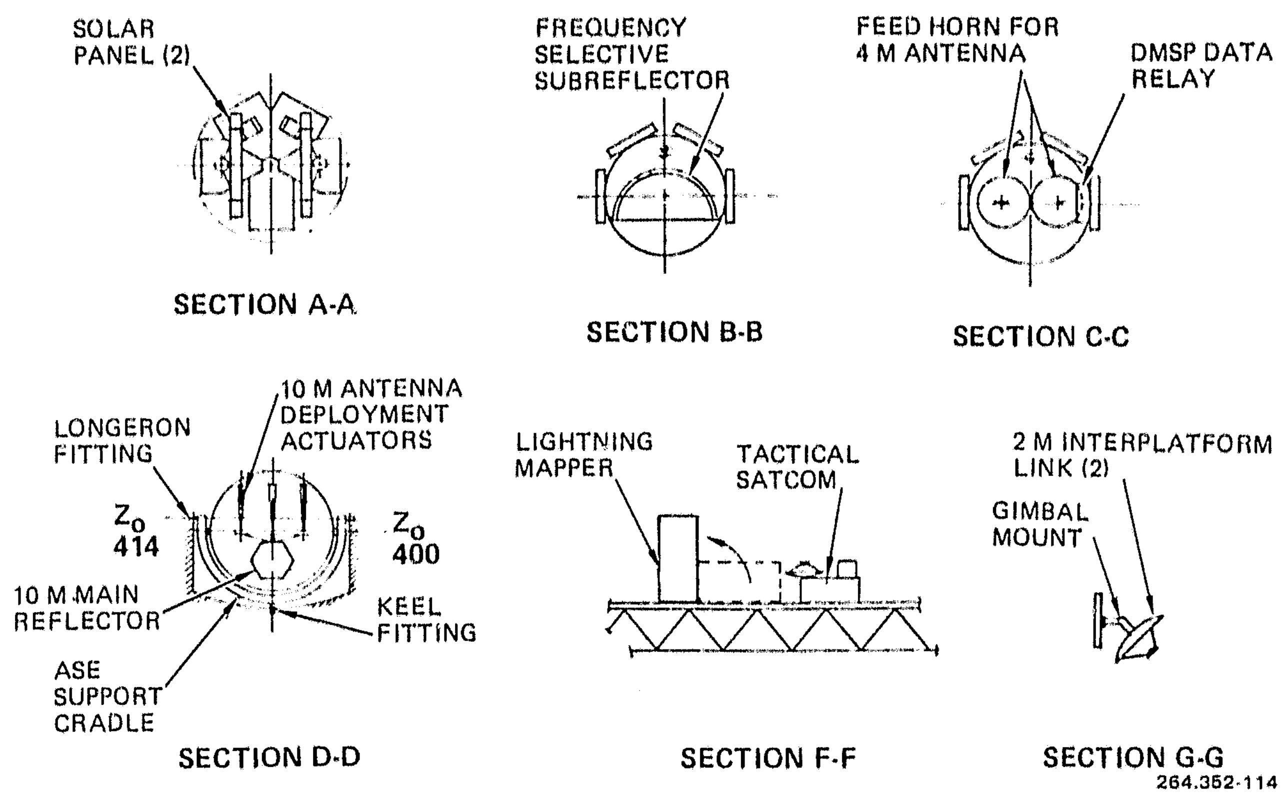

Figure 6-43. Experimental Platform Concept 6, Packaged - Cross Sections

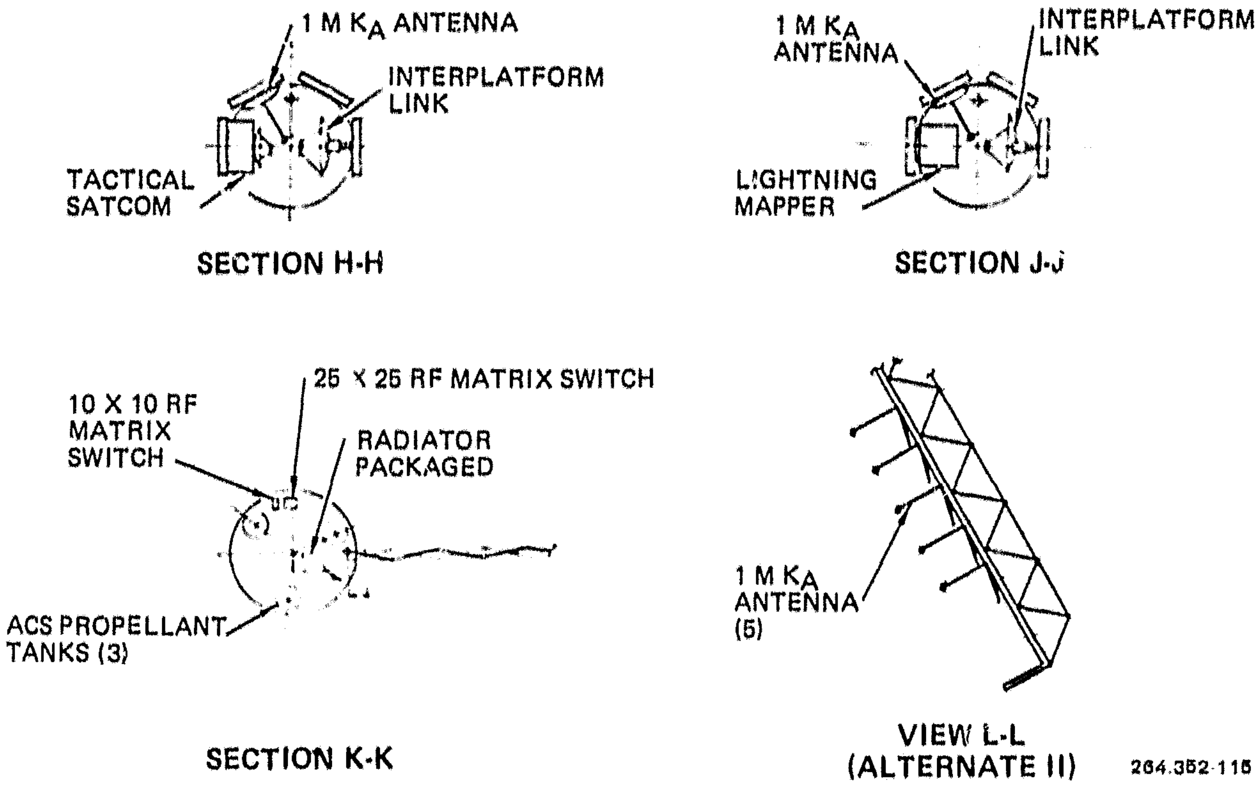

Figure 6-44. Experimental Platform Concept 6, Packaged - Cross Sections

Table 6-28. Experimental Platform Concept 6, Payloads and Technologies

Payloads		
C Band	–	One 10-meter antenna; 100 reuse switch; baseband processor
IPL	–	Two 2-meter antennas
Ka Band	–	Two 4-meter antennas; 10 × 10 switch; baseband processor
DOD #31	–	DMSP data relay
DOD tactical AF satellite communications		
DOD #33	–	materials exposure
DOD #43	–	magnetic substorm monitor
DOD #56	–	fiber optics demonstrator
OSTA #17	–	lightning mapper
		PLUS
Alternative 1		
OSS #75	–	imaging spectrometer
OSS #79	–	low light level TV
Alternative 2		
Ka Band	–	Five 1-meter antnnas; 25 × 25 switch
Technology Demonstrations		
Advanced communications technology – C and Ka Bands		
C Band beam shaping/reconfigurability		
C and Ka Band direct-to-user		
Frequency selective subreflector surface – C Band, 4 and 6 GHz		
Large deployable antenna – C Band		
Ka Band beam scanning		
High frequency deployable solid surface antennas		
IPL technology		
All platform technologies		

Table 6-29. Experimental Platform Concept 6, Antenna Characteristics

Freq Desig	Freq. GHz	Antenna Dia. Type	Eff F/D	Function	Beam Width	Point. Acc.	Coverage	Beam Type	Scanned Angle	Notes
C-BAND	6/4	10M, O/CASS, MBFR	0.6	T/R	0.35/0.5°	0.03°	CONUS	RECONFIG.	12/8 BW	W/FSS
	4	0.25M HORN	—	XMIT	18°	1.0°	EARTH	FIXED	—	
	6	0.16M HORN	—	RCV	18°	1.0°	EARTH	FIXED	—	
Ka-BAND	20	4M, O/CASS, MBFR	1.0	XMIT	0.3°	0.03°	CONUS	FIX/SCAN'G	13 BW	
	30	4M, O/CASS, MBFR	1.0	RCV	0.3°	0.03°	CONUS	FIX/SCAN'G	13 BW	
ALT 2	30/20	1M, O/PARA, MBFR	0.5	T/R	1.0°	0.1°	CONUS	FIXED, 25	4 BW	5 REFLEC. W/FSS

Table 6-30. Experimental Platform Concept 6, Alternate #1, Weight Estimate

	Estimated Weight, kg		Contingency, 15%, kg		Total, kg
PLATFORM					
STRUCTURE	2,582				
THERMAL CONTROL	301				
ATTITUDE CONTROL	1,460				
ELECTRICAL POWER (12.8 KW)	756				
AVIONICS	243				
	5,342	+	801	=	6,143
PAYLOADS					
C-BAND	453				
Ku-BAND	—				
L-BAND	—				
Ka-BAND	452				
IPL	78				
SECONDARY	1,520				
	2,503	+	375	=	2,878
TOTAL PLATFORM WEIGHT, WITH 15% CONTINGENCY:					9,021
TRANSFER VEHICLE — IOTV (LOW THRUST, LAUNCHED SEPARATELY 9190 KG CAPABILITY)					—
ASE					1,900
ORBITER LAUNCH WEIGHT: (29,484 KG CAPABILITY)					10,921

Table 6-31. Experimental Platform Concept 6, Alternative #2, Weight Estimate

	Estimated Weight, kg		Contingency, 15%, kg		Total, kg
PLATFORM					
STRUCTURE	2,571				
THERMAL CONTROL	253				
ATTITUDE CONTROL	1,320				
ELECTRICAL POWER (10.7 KW)	662				
AVIONICS	243				
	5,049	+	757	=	5,806
PAYLOADS					
C-BAND	453				
Ku-BAND	—				
L-BAND	—				
Ka-BAND	500				
IPL	78				
SECONDARY	870				
	1,901	+	285	=	2,186
TOTAL PLATFORM WEIGHT, WITH 15% CONTINGENCY:					7,992
TRANSFER VEHICLE — IOTV (LOW THRUST, LAUNCHED SEPARATELY 9190 KG CAPABILITY)					—
ASE					1,900
ORBITER LAUNCH WEIGHT: (29,484 KG CAPABILITY)					9,892

6.8 TRANSFER VEHICLE OPTIONS

Per study ground rule, feasibility of the experimental platform mission was to be evaluated considering the Centaur, IUS, and IOTV as candidate transfer vehicles for platform transfer from LEO to GEO.

Transfer vehicle performance was taken from "STS Upper Stage Geosynchronous Payload Capability", NASA/MSFC memo PD01-79-70, and subsequent updates.

The matrix of candidate transfer vehicles versus number of Shuttle flights for the mission (1 or 2), is summarized in Figure 6-45. Transfer vehicle performance capability is given in the righthand column, in kilograms. As shown previously in Table 6-12, both Centaur and IOTV are extremely well-suited to the requirements of a mated platform/transfer vehicle mission using one Shuttle flight. Payload weights for platform Concepts 1 through 5 range from 4179 kg for Concept 5 to 5278 kg for Concept 2; Centaur performance capability to GEO is 4772 kg, the offloaded IOTV capability is 5670 kg. The performance capability of the IUS at 2636 kg, however, could only accommodate a very small platform with limited payloads, filling only about half of the available payload space in the cargo bay, and is therefore unsuited for the experimental platform mission.

STS REQUIREMENT	MISSION				OPERATIONS					MAX. P/L, KG
	OPTION NO.	FLIGHT NO.	OTV	CARGO	TO LEO	DEPLOY	MATE	XFER TO GEO	MATE	
SINGLE SHUTTLE FLIGHT	I A	1	CENT	28'	←	X		←		4772
	I B	1	IUS (3 STG)	32.6'	←	X		←		2636
	I C	1	IOTV	24'	←	X		←		5670
TWO SHUTTLE FLIGHTS	II A	1	CENT	28'	←	X		←	√	9544
		2	CENT	28'	←	X		←		
	II B	1	IUS (3 STG)	32.6'	←	X		←	√	5272
		2	IUS (3 STG)	32.6'	←	X		←		
	II C	1	IOTV	24'	←	X		←	√	11340
		2	IOTV	24'	←	X		←		
	II D	1		56'	←	X	√	←		9100
		2	IOTV		←					

264.352-116

Figure 6-45. Transfer Vehicle Options

For the two-Shuttle flight mission, either Centaur or the offloaded IOTV can accommodate the twin-configuration option, II-A and II-C, where each Shuttle carries a nearly identical mated platform/transfer vehicle cargo differing only in the payloads carried on each platform. These two options exceed the performance capability of Option II-D, where a fully-loaded IOTV is carried in one Shuttle, and a full-cargo-bay length platform such as Concept 6A or 6B is carried in the other. Performance capability for Option II-A and II-C is 9544 kg and 11,340 kg respectively; Option II-D is limited to 9190 by IOTV performance.

Option II-B, using the twin mated platform/transfer vehicle concept and three-stage IUS transfer vehicles, is limited to 5272 kg, less than the single Shuttle flight performance with the IOTV (Option I-C), and only slightly better than the single Shuttle flight performance with the Centaur (Option I-A).

6.9 EVALUATION

The results of this preliminary conceptual feasibility study are in no way intended to imply any limitations on the possible configurations that could satisfy the experimental platform concept requirements, nor are they intended as recommendations. The intent was to determine feasibility of the concept, and the study to date has certainly shown the concept to be both feasible and practicable.

The variety of structural configurations, payload combinations, concepts and options for an experimental geostationary platform are almost limitless. The ones shown in this study are only samples of what the experimental platform could be. What we have learned from the study is significant, however:

a. Single-flight mission platforms show:

   1. Lowest program costs.

   2. Total platform weights from 4000 to 5300 kg, compatible with low-thrust Centaur and low-thrust offloaded IOTV performance capability, but far exceeding the three-stage IUS capability of 2700 kg.

   3. Lower density Shuttle cargoes for predominantly communications payloads with deployable antennas and subreflectors, with weights from 4100 to 4700 kg, in the low-thrust Centaur capability.

   4. Higher density Shuttle cargoes for predominantly DoD and science experiments, with weights from 4700 to 5700 kg, in the low-thrust offloaded IOTV capability.

b. Two-flight missions, with the platform in one Orbiter and transfer vehicle in the other, show:

   1. Total platform weights to 9190 kg.
   2. Best demonstration of economy of scale.
   3. A possible advantage in operations when a cryo transfer vehicle is used. The platform can be deployed, checked out, any faults rectified, and prepared for transfer before the cryo transfer vehicle is boosted to LEO, minimizing any boiloff losses.

c. Two-flight missions, with nearly identical cargoes of mated platform/transfer vehicle, differing only in the platform payloads, show:

   1. Total platform weights to 11,340 kg.
   2. Maximum payload weight and payload selection options.
   3. Maximum mission flexibility. The two platforms can be placed at different longitudes in the geostationary orbital arc, serving a greater variety of payload objectives.
   4. Opportunity to test and demonstrate interplatform link technology with no dependence on other satellites or programs.
   5. Opportunity to test and demonstrate rendezvous and docking technology, by moving one platform in a walking orbit to the longitudinal location of the other.

In summary, the study has revealed no insurmountable design problems; it has significant test, experimental, and user application capability; it can be accomplished in either a single Shuttle mission, or expanded in capability to a two-Shuttle mission; it reflects highly reliable, current state of the art technology for the most part, but demonstrates sufficient advancement in communications and platform technology to warrant outside support and the use of public funds by the government.

Selection of a platform configuration for further study and definition will be dependent on priorities and concurrent program constraints. Funding priorities will dictate the one or two-Shuttle flight mission for the experimental platform; communications, DoD and science payload priorities will influence the platform design; and finally, transfer stage availability (dependent on transfer vehicle program priorities) will determine the design constraint with respect to weight and volume.

In any case, or combination of influencing factors, the experimental platform concept is viable, and the configuration can be easily tailored to fit the current priorities and constraints.

SECTION 7

FUTURE WORK

The thrust of this initial study was toward the characterization of operational geostationary platforms of the 1990s.

In the course of the study, it was recognized that a NASA experimental geostationary platform was required to demonstrate critical technologies.

NASA's primary interest in the follow-on study is to lay the basis for a Phase B study of the experimental geostationary platform. To do this, it is necessary to further characterize the most probable concepts for operational platforms, so that the proper technologies can be demonstrated on the experimental platform.

In some cases, this requires expanding the range of options considered in the initial study. For example, a range of multislot alternate architectures will be examined in addition to the single-slot initial communications system architecture, which was the basis for the initial study. This will enable NASA to determine the economic and system impact of using less ambitious communications technology than was postulated in Task 1 and characterized in Task 4.

In other cases, the range of options will be narrowed. For example, single-Shuttle launch concepts will be emphasized, since the economic return of going to the more complex concepts involving assembly at LEO appears to be marginal. An attempt will be made to determine with greater precision the relative merits of the constellation and docked module concepts represented by Alternatives #1 and #2 identified in Task 2. This will enable NASA to prioritize the requirements for demonstration of the very different technologies associated with those concepts.

In order to give both NASA and the communications industry a better feel for platform economics, a return on investment (ROI) analysis of Alternatives #1 and #2 will be performed considering only communications payloads (Payload numbers 1 through 11).

Continuing interaction with potential commercial and governmental user agencies and payload suppliers will result in a refined (and hopefully prioritized) list of candidate payloads for the experimental platform.

An updated set of candidate upper stages for possible use with the experimental platform will be considered and the weight, volume, g-level, and other constraints on the platform determined.

Finally, a few experimental platform concepts will be developed that relate the candidate upper stages to appropriate subsets of the candidate payloads. Program costs for these concepts will be estimated to provide NASA with a basis for selecting the experimental platform concept(s) that satisfy critical mission requirements at an affordable price. These may include:

a. A modest experimental program using a solid upper stage and small platform deployed at GEO with a very limited number of payloads.

b. More realistic demonstrations with low thrust liquid upper stages that allow deployment, checkout, and initial experimentation at LEO before transfer.

c. A quasi-operational prototype platform employing significant user payloads.

The degree of user interest and participation may well influence the nature of the experimental platform program.