NASA Technical Memorandum 82904

Large Space Systems/Propulsion Interactions

Material presented at a government-industry information exchange workshop held at NASA Lewis Research Center, Cleveland, Ohio, October 22-23, 1981

June 1982

NASA

PREFACE

With the emergence of the Shuttle, the space program has begun the transition from an era of demonstration to one of cost-effective utilization for commercial and defense applications. Future space missions will include large systems, with onboard propulsion requirements significantly different and more severe than those on present, relatively dense spacecraft. Therefore, a two-day workshop was held at the NASA Lewis Research Center to define propulsion requirements and identify technological issues which must be addressed in the design of large space systems.

The workshop provided experts from government and industry an opportunity to address the critical interactions between large space systems and their propulsion systems over the total mission life cycle - Shuttle launch to LEO; deployment and checkout; transfer to final orbit; and orbit maintenance for the remainder of the mission life. Presentations were made in three workshop panels dealing with the following major topics: Missions; system requirements and operations; and system design and integration. Summaries of the workshops were presented at a concluding plenary session. Workshops were kept informal, and that atmosphere provided the free and lively exchange of ideas and opinions which made this meeting highly successful.

This publication is a compilation of the material from the presentations and the conclusions of the three workshop panels.

Sol H. Gorland
NASA Lewis Research Center

Workshop Chairman

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CONTENTS

OVERVIEW OF LARGE SPACE SYSTEMS/PROPULSION INTERACTIONS

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LSST

ANTENNAS

LARC
- ANT. SYST. ANAL
- HOOP COLUMN
- WRAP RIB
- ELECTROSTATIC MEMBRANE

PLATFORMS

MSFC
- PLAT. SYST. ANAL.
- DEPLOYABLE PLAT.
- ASSEMBLY ANALYSIS
- SADE

JSC PLAT. ASSEMBLY TECH.

STATION

JSC, MSFC
- MAN PERMANENT OCCUPANCY IN SPACE

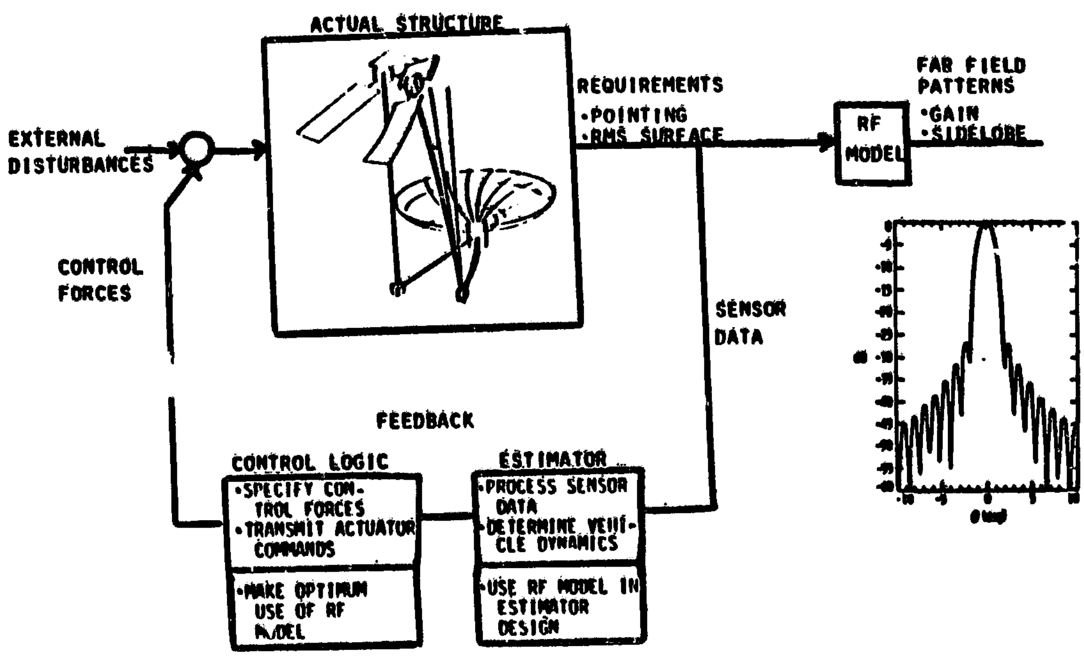
CONTROL & STAB.	LARC, MSFC, JPL
AUTOMATION	MSFC, JPL

PROPULSION

EARTH TO ORBIT (CHEM)	MSFC
ORBIT TRANSFER (CHEM)	LERC
ELECTRIC	LERC
PLANETARY (CHEM)	JPL
LOW THRUST (CHEM)	JPL
NUCLEAR ELECTRIC	JPL

- LARGE CONFIGURATION CHANGES
  - STOWED/DEPLOYED
  - ASSEMBLED
- COMPLEX STRUCTURES
  - VERY LOW FREQUENCY, DENSE MODES
  - DAMPING UNCERTAINTIES & NON-LINEARITIES
- ADVANCED CONTROL SYSTEMS
  - DISTRIBUTED
  - ADAPTIVE
- HIGHLY INTERACTIVE SUBSYSTEMS
  - STRUCTURE DYNAMICS
  - THERMAL
  - CONTROL

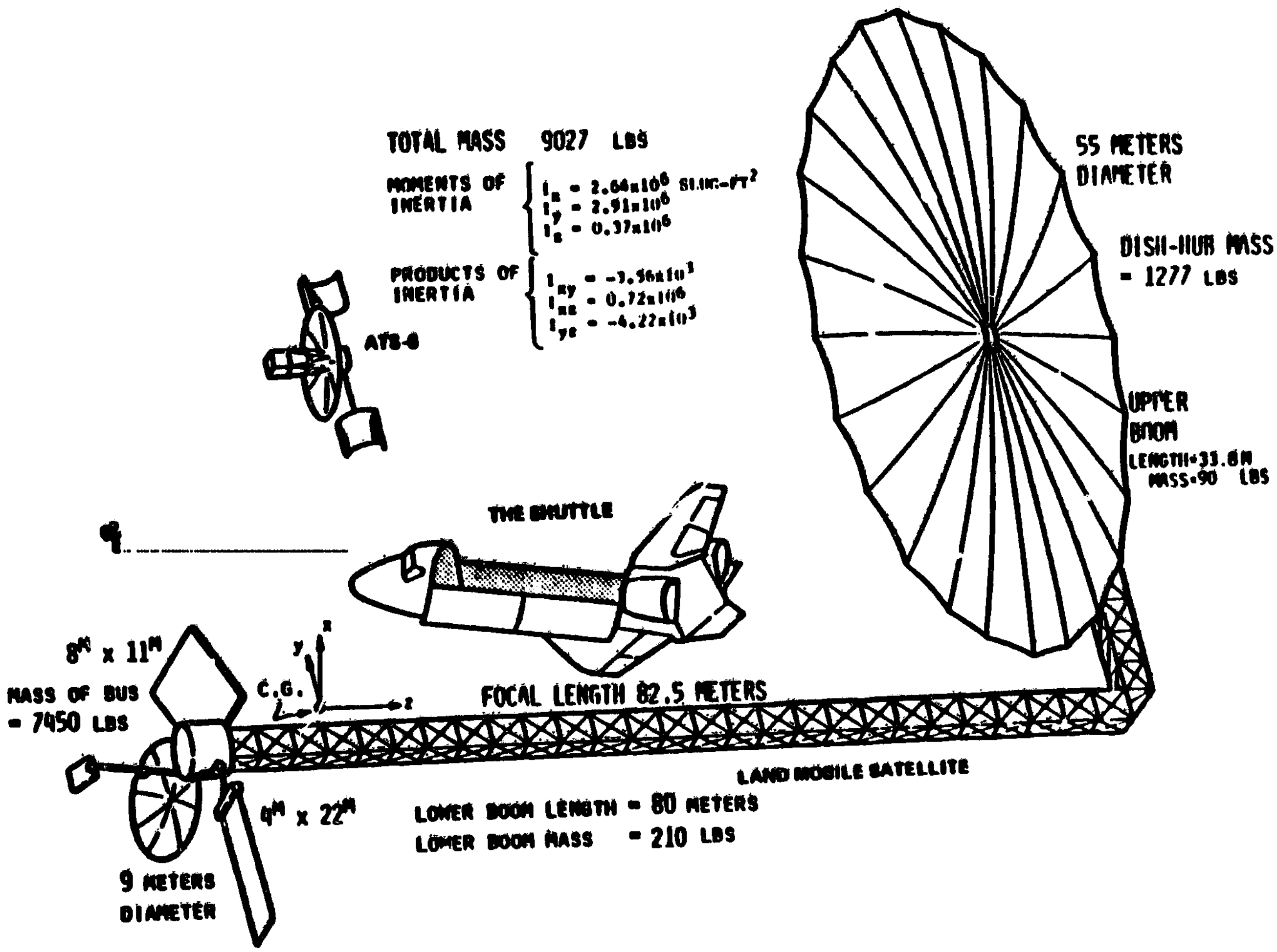
ANTENNA CONTROLLER DESIGNS BASED ON RF PERFORMANCE



NEW APPROACH

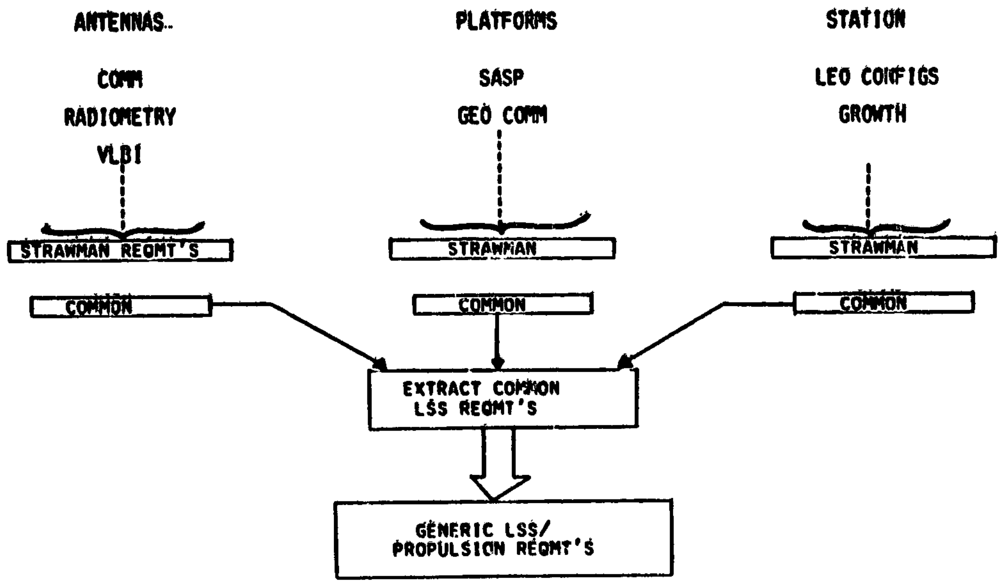
- RF MODEL USED TO OPTIMIZE FEEDBACK DESIGN PROCESS
- PERFORMANCE SPECIFIED IN TERMS OF RF PARAMETERS

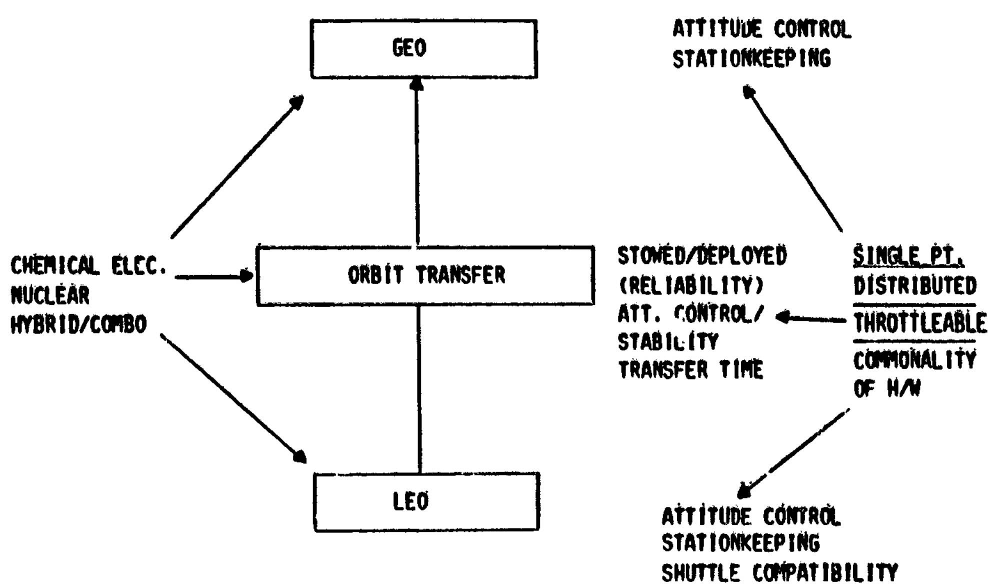
CONFIGURATION AND MASS PROPERTIES



THE SAME TYPE SYSTEM/MISSION ANALYSIS AND TRADES MUST BE PERFORMED WITH PARTICIPATION BY THE PROPULSION SYSTEM TO DEFINE PROPULSION SYSTEM DRIVERS, SUBSYSTEM INTERACTIONS AND REQUIREMENTS AND OPTIMUM CONFIGURATIONS.

SUGGESTED MISSION GROUPINGS ARE ON THE FOLLOWING CHART.





PROPULSION/LSS INTERACTION
MMC (LOW THRUST CHEMICAL)

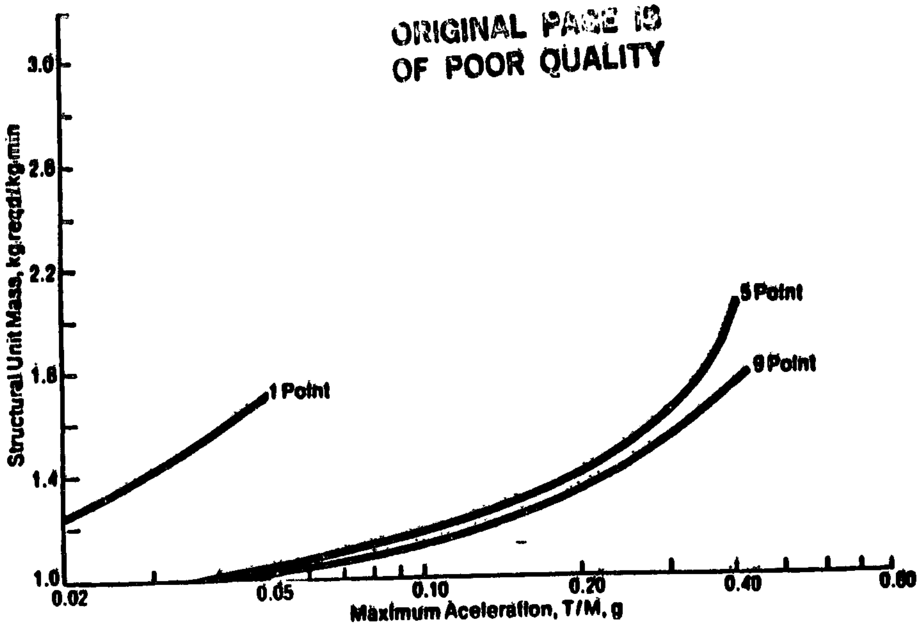
- LSS CONFIGURATIONS:
  - EXPANDABLE BOX TRUSS
  - WRAP RADIAL RIB
  - HOOP COLUMN
- MAXIMUM PAYLOAD MASS & LENGTH
- STRUCTURE MASS VS. THRUST/WEIGHT
  - CONFIGURATION SENSITIVITY ~ DIAMETER, DENSITY
  - RELATIVE CONFIGURATION TRADES
- THRUST TRANSIENTS (START-UP)
- DISTRIBUTED THRUSTING .

Average Structural Mass Impact for Start Transient Effects

Ramp Time T_R	Box Truss Average kg Reqd/kg Steady State	Radial Rib, Average kg Reqd/kg Steady State	Hoop/Column, Average kg Reqd/kg Steady State
Step	1.10	1.30	1.10
$1/3t_1$	1.03	1.10	1.01
$2/3t_1$	1.00	1.00	1.00
$3/3t_1$	1.00	1.00	1.00
$4/3t_1$	1.00	1.00	1.00
T_R Range for No Structural Impact	0.2 to 2 s	0.5 to 10 s	0.5 to 5 s

Note: Shutdown transients have negligible impact.

Effects



LSS/PROPULSION ISSUES

- SYSTEM OPTIMIZATION
  - PROPULSION/DISCIPLINE TRADES
  - CHEMICAL, ELECTRIC, NUCLEAR
- SYSTEM RELIABILITY/COST
  - PROPULSION SYSTEM TECHNOLOGY DRIVERS
  - SYSTEM COST DRIVERS
- SUBSYSTEM INTERACTION
  - PROPULSION/STRUCTURE/CONTROL
- PROPULSION/SYSTEM INTEGRATION
  - MODULAR GROWTH
  - ORBITAL ASSEMBLY OPERATIONS
- SUBSYSTEM COMMONALITY
  - SINGLE SYSTEM: ORBIT TRANSFER/CONTROL
  - CENTRAL VS. DISTRIBUTED (CONTROL)
- AUTOMATION
  - DEGREE WITHIN PROPULSION SUBSYSTEM
  - INTERACTION REQUIREMENTS

NASA SPACE SYSTEM TECHNOLOGY MODEL
FUTURE MISSION SYSTEM NEEDS

T. G. Reese

General Research Corporation

NASA SPACE SYSTEMS TECHNOLOGY MODEL
PURPOSE

PLANNING AID

- PROVIDES A REFERENCE FOR PLANNING TECHNOLOGY PROGRAMS AND OPTIONS
- IDENTIFIES CRITICAL TECHNOLOGIES NEEDED FOR FUTURE MISSIONS
- ESTABLISHES CRITERIA FOR EVALUATING ONGOING PROGRAMS

COMMUNICATION TOOL

- DISSEMINATES INFORMATION ON FUTURE CAPABILITIES OF SPACE TECHNOLOGY TO MISSION PLANNERS

NASA SPACE SYSTEMS TECHNOLOGY MODEL

- A CATALOG OF INFORMATION DETAILING PROPOSED FUTURE MISSION AND INSTRUMENT SYSTEMS ENDORSED BY NASA PROGRAM OFFICES
- A CHARACTERIZATION OF TRENDS AND FORECASTS OF CAPABILITIES IN THE MAJOR DISCIPLINES OF SPACE TECHNOLOGY
- A SET OF REAL AND GENERIC MISSIONS THAT ILLUSTRATE THE POTENTIAL OF SPACE TECHNOLOGY CURRENTLY BEING DEVELOPED
- REVISED AND UPDATED VERSION FOR 1981 NOW IN PUBLICATION

NASA SPACE SYSTEMS TECHNOLOGY MODEL CONTENTS

WITHIN TEN-YEAR HORIZON		DISTANT HORIZON
VOLUME I	VOLUME II	VOLUME III
System/Program Descriptions and Technology Needs	Space Technology Trends and Forecasts	Opportunity Systems/Programs and Technologies
• Part A: Mission Systems, Descriptions, and Parametric Tabulation of Needs • Part B: Instrument Systems, Descriptions, and Needs; OAST Experiments	• Transportation Systems • Spacecraft Systems • Information Systems • Chemical Propulsion • Electric Propulsion • Aerothermodynamics • Power • Materials and Structures • Automation, Guidance and Control • Sensors • Communications • Data Processing	• Part A: Program Office Opportunity Systems and Programs • Part B: Opportunity Systems Selected by OAST • Part C: Opportunity Technologies

1981 NASA MODEL - LARGE SPACE SYSTEMS

---- MISSION SYSTEMS EXTRACTED FROM THE MODEL ----

PROPOSED FUTURE LARGE SPACE SYSTEMS THAT ARE DRIVERS FOR PROPULSION TECHNOLOGY

PRIMARY PROPULSION: LOW THRUST-TO-WEIGHT RATIO ORBIT TRANSFER PROPULSION SYSTEMS

SECONDARY PROPULSION: ORBIT MAINTENANCE AND MOMENTUM MANAGEMENT SYSTEMS

PRIMARY PROPULSION DRIVER MISSIONS

EXPERIMENTAL GEOSTATIONARY PLATFORM

PROPOSED START DATE: 1986

PROPOSED LAUNCH DATE: 1990

COHERENT OPTICAL SYSTEM OF MODULAR IMAGING COLLECTORS (COSMIC)

PROPOSED START AND LAUNCH DATES: >1990

100 METER THINNED APERTURE TELESCOPE

PROPOSED START AND LAUNCH DATES: >1990

ORBITING DEEP SPACE RELAY STATION (ODSRS)

PROPOSED START AND LAUNCH DATES: >1990

EXPERIMENTAL GEOSTATIONARY PLATFORM

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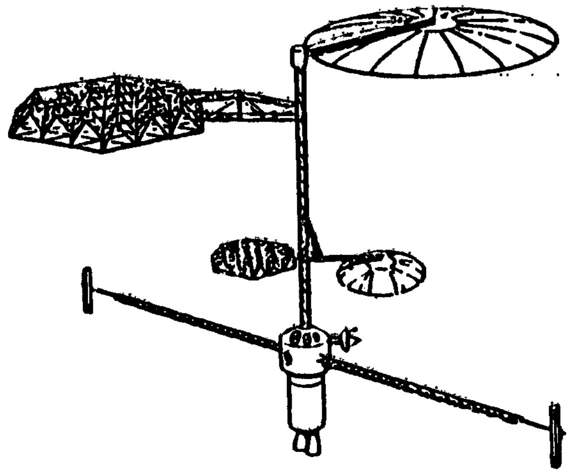
OF POOR QUALITY



STATUS:	PLANNED
LIFETIME:	8 YEARS
LAUNCH AND TRANSFER VEHICLES:	SHUTTLE/OTV
OPERATIONAL LOCATIONS:	GEO
TOTAL MASS AT OPERATIONAL LOCATIONS:	6,800-10,200
AVERAGE OPERATIONAL POWER:	25-50 kW

OBJECTIVE: DEMONSTRATE THE TECHNOLOGIES, SYSTEMS, AND POTENTIAL USES OF OPERATIONAL GEOSTATIONARY PLATFORMS. THE EXPERIMENTAL PLATFORM WILL PROVIDE AN OPPORTUNITY TO TEST NEW COMMUNICATIONS TECHNOLOGIES AND SERVICES AND ALSO PROVIDE A FREE-FLYING, LONG-LIFE TIME SPACECRAFT ON WHICH TO PERFORM SCIENCE AND APPLICATIONS EXPERIMENTS.

DESCRIPTION: A MATED TRANSFER VEHICLE/PACKAGED PLATFORM WILL BE FLOWN BY THE SHUTTLE INTO LOW EARTH ORBIT. AFTER PARTIAL DEPLOYMENT OF THE PLATFORM AND A PASSIVE DOCKING DEMONSTRATION WITH AN ADVANCED TMS, THE PAYLOAD WILL BE TRANSFERRED TO GEO. THE PLATFORM WILL BE PERIODICALLY VISITED BY AN ADVANCED TMS TO DEMONSTRATE RENDEZVOUS, DOCKING, AND SERVICING AT GEO. THE PLATFORM HAS FIVE DIFFERENT PAYLOADS: COMMUNICATIONS R&D PAYLOAD - COMMUNICATIONS TECHNOLOGY DEMONSTRATION, NEW SERVICE DEMONSTRATION; EARTH OBSERVATION R&D PAYLOAD - LIGHTING MAPPER, RADIOMETERS; OPERATIONAL ENVIRONMENTAL PAYLOAD - VAS, DATA COLLECTION; SCIENCE PAYLOAD - IMAGING SPECTRO OBSERVATORY, ATMOSPHERIC EMISSION IMAGER; DOD R&D PAYLOAD - NI/H_2 BATTERY, PASSIVELY DAMPED STRUCTURE, HARDENED OPTICAL COMMUNICATION LINKS, TACTICAL SATCOM.

COHERENT OPTICAL SYSTEM OF MODULAR IMAGING COLLECTORS (COSMIC)

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OF POOR QUALITY

STATUS: OPPORTUNITY

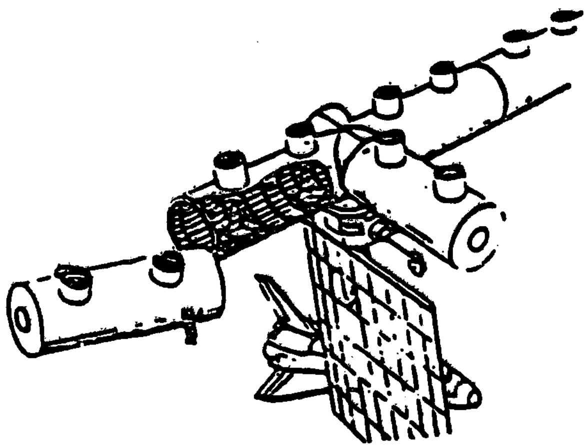
LIFETIME: 10 YEARS

LAUNCH AND TRANSFER VEHICLES: SHUTTLE, HLLV/OTV

OPERATIONAL LOCATIONS: INITIALLY 500-km ALTITUDE AT 28.5°; EVENTUALLY GEO-SYNCHRONOUS

TOTAL MASS AT OPERATIONAL LOCATIONS: APPROX. 67,000 kg

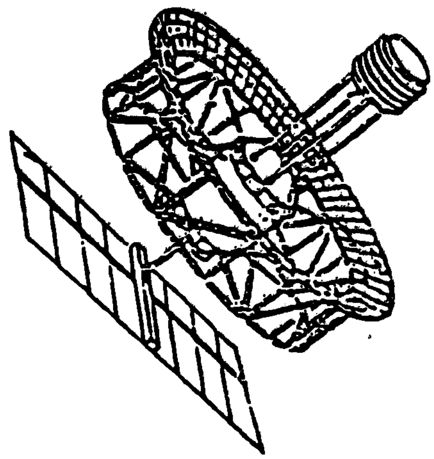
AVERAGE OPERATIONAL POWER: APPROX. 25 kW



OBJECTIVE: THE OVERALL OBJECTIVE OF THE COHERENT OPTICAL SYSTEM OF MODULAR IMAGING COLLECTORS (COSMIC) IS TO INCREASE THE CAPABILITIES OF UV/OPTICAL/IR ASTRONOMY BY SEVERAL ORDERS OF MAGNITUDE MORE THAN SPACE TELESCOPE. TYPICAL SCIENCE INVESTIGATIONS ARE CALIBRATION OF THE DISTANCE SCALE BEYOND VIRGO OUT TO THE COMA CLUSTER, HIGH-RESOLUTION STUDIES OF QUASARS, SEARCH FOR PLANETARY SYSTEMS.

DESCRIPTION: COSMIC COULD BE DEVELOPED BY NASA IN THE 1990s AS A LONG-LIVED INTERNATIONAL OBSERVATORY ANALOGOUS TO SPACE TELESCOPE. A LARGE COHERENT ARRAY OF OPTICAL COLLECTORS IS DEPLOYED IN ORBIT BY ASSEMBLY OF MODULES CARRIED INTO ORBIT INSIDE THE SHUTTLE ORBITER BAY. INITIALLY ONLY ONE MODULE CONSISTING OF A 10-m BASELINE ARRAY IS SUFFICIENT TO PROVE THE CONCEPT AND AT THE SAME TIME SIGNIFICANTLY INCREASE THE ANGULAR RESOLUTION CAPABILITY OVER SPACE TELESCOPE. SEVERAL ARRAY GEOMETRIES ARE UNDER CONSIDERATION WHICH MINIMIZE THE NUMBER OF ELEMENTS. AN EVOLUTIONARY DEVELOPMENT STARTING WITH A TWO-FOUR ELEMENT INTERFEROMETER AND EVOLVING TO A TWO-DIMENSIONAL ARRAY IS PROPOSED. THE ULTIMATE LIMIT OF SUCH AN APPROACH DEPENDS UPON THE ABILITY TO MANAGE THE BUILDUP OF TOLERANCES. THE 100-METER THINNED APERTURE TELESCOPE IS AN ALTERNATE METHOD FOR PROVIDING SIGNIFICANT ADVANCES FOR OPTICAL ASTRONOMY.

100 METER THINNED APERTURE TELESCOPE



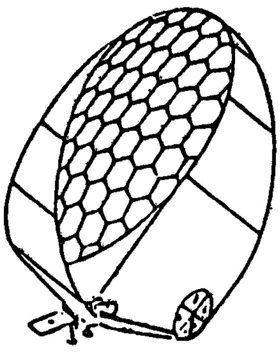
STATUS:	OPPORTUNITY
LIFETIME:	10 YEARS
LAUNCH AND TRANSFER VEHICLES:	SHUTTLE, HLV, OTV, IUS
OPERATIONAL LOCATIONS:	INITIALLY 500-km ORBIT AT 28.5°; EVENTUALLY GEO-SYNCHRONOUS
TOTAL MASS AT OPERATIONAL LOCATIONS:	APPROX. 85,000 kg
AVERAGE OPERATIONAL POWER:	APPROX. 25 kW

OBJECTIVE: THE 100-METER THINNED APERTURE TELESCOPE (TAT) HAS AS ITS BASIC OBJECTIVES A 30 FOLD INCREASE IN IMAGE RESOLUTION AND A 1000 FOLD INCREASE IN ASTROMETRIC PRECISION OVER THAT AFFORDED BY THE SPACE TELESCOPE. TYPICAL SCIENCE INVESTIGATIONS ARE CALIBRATION OF THE DISTANCE SCALE BEYOND VIRGO AND OUT TO THE COMA CLUSTER OF GALAXIES, HIGH-RESOLUTION STUDIES OF QUASARS, AND A SEARCH FOR PLANETARY SYSTEMS.

DESCRIPTION: THE TAT COULD BE DEVELOPED BY NASA IN THE 1990s AND MANAGED AS A LONG-LIVED INTERNATIONAL OBSERVATORY ANALOGOUS TO SPACE TELESCOPE. THE LARGE APERTURE TELESCOPE IS DEPLOYED IN LOW EARTH ORBIT USING ADVANCED ASSEMBLY TECHNIQUES. SEVERAL SHUTTLE FLIGHTS PROVIDE FOR ASSEMBLY ON THE INITIAL STRUCTURE INCLUDING FABRICATION OF STRUCTURAL COMPONENTS, ATTACHMENT OF THE EQUIPMENT AND INSTRUMENT SECTIONS AND THE SOLAR ARRAYS. ADDITION OF THE PRIMARY AND SECONDARY MIRROR SEGMENTS PROCEEDS IN AN INCREMENTAL FASHION TO PROVIDE AN EARLY INITIAL CAPABILITY TO OBTAIN HIGH RESOLUTION OBSERVATIONS OF BRIGHTER SOURCES. EVENTUAL FILLING IN OF SECTIONS OF ANNULAR MIRRORS WILL PROVIDE THE FULL CAPABILITY FOR FAINT OBJECT DETECTION.

TAT CONSTRUCTION WILL REQUIRE THE DEVELOPMENT OF EXTENSIVE ORBITAL CONSTRUCTION AND ASSEMBLY TECHNIQUES. THE BASIC STRUCTURE CAN BE CONSTRUCTED AND THEN INSTRUMENTED WITH RETRO-REFLECTORS TO IMPROVE DIMENSIONAL STABILITY USING LASER GAGE INTERFEROMETRY. THE INDIVIDUAL ELEMENTS ARE MOUNTED TO THIS STRUCTURE AND THEN CONTROLLED TO FORM A COHERENTLY PHASED ARRAY. INTERFEROMETRIC SENSORS IN THE FOCAL PLANE ARE USED TO DETECT WAVEFRONT ERRORS FOR EACH ELEMENT FOR IMAGE COMPENSATION AND FIGURE CONTROL.

ORBITING DEEP SPACE RELAY STATION (ODSRS)



STATUS:	OPPORTUNITY
LIFETIME:	10 YEARS
LAUNCH AND TRANSFER VEHICLES:	SHUTTLE/OTV
OPERATIONAL LOCATIONS:	GEO
TOTAL MASS AT OPERATIONAL LOCATIONS:	8500 kg
AVERAGE OPERATIONAL POWER:	5.5 kW

OBJECTIVE: TO PROVIDE DEEP SPACE TRACKING AND COMMUNICATIONS SUPPORT OF DEEP SPACE PROBES IN THE POST 1985 ERA.

DESCRIPTION: THE ODSRS WILL BE LAUNCHED INTO A LOW-EARTH ORBIT BY THE SPACE SHUTTLE. THIS ORBIT HAS AN INCLINATION OF ABOUT 28.5°. IT IS ANTICIPATED THAT THREE SHUTTLES WILL BE REQUIRED TO TRANSPORT ALL OF THE ODSRS HARDWARE INTO ORBIT, INCLUDING ONE SHUTTLE FOR THE ORBIT TRANSFER PROPULSION SYSTEM. THE ODSRS WILL BE ASSEMBLED, ALIGNED, AND TESTED IN THE LOW-EARTH ORBIT (LEO), AND THEN BOOSTED TO A GEOSYNCHRONOUS ORBIT (GEO). FINAL SYSTEM LEVEL PERFORMANCE AND ENVIRONMENTAL TESTS WILL BE PERFORMED IN LEO PRIOR TO A DECISION TO TRANSFER TO GEO.

THE ODSRS CONCEPTUAL DESIGN IS A 28 m, OFFSET FEED, TWO-REFLECTOR CASSEGRAIN ANTENNA. IT HAS A LIGHTWEIGHT DEPLOYABLE BACKUP STRUCTURE WITH PRECISION SURFACE PANELS ATTACHED IN LOW-EARTH ORBIT, USING THE SHUTTLE AS A WORK PLATFORM. SUPPORT SUBSYSTEMS FOR THE ODSRS ARE CONTAINED IN THE BOXLIKE BUS ATTACHED TO THE MAIN ANTENNA BACKUP STRUCTURE. THE ESTIMATED ODSRS MASS IS 8500 kg, AND ITS STOWED VOLUME IS APPROXIMATELY 2 SHUTTLE CARGO BAYS. ODSRS POWER CONSUMPTION IS ESTIMATED TO BE 5.5 kW, MOST OF WHICH WILL BE A CONTINUOUS LOAD FOR REFRIGERATORS FOR THE CRYOGENIC RECEIVERS.

SECONDARY PROPULSION DRIVER MISSIONS

SPACE PLATFORM ALPHA

START DATE: 1984

LAUNCH DATE: 1987

SPACE STATION

START DATE: 1986

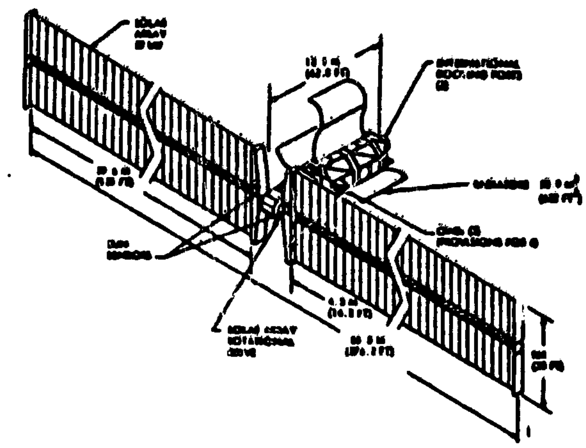
LAUNCH DATE: 1990-91

AUTOMATED PLANETARY STATION

START AND LAUNCH DATES: >1990

SPACE PLATFORM ALPHA

STATUS:	PLANNED
LIFETIME:	5 YEARS WITH ON-ORBIT MAINTENANCE
LAUNCH AND TRANSFER VEHICLES:	SHUTTLE DEPLOYED AND SERVICED
OPERATIONAL LOCATIONS:	LEO
TOTAL MASS AT OPERATIONAL LOCATIONS:	12,500 kg
AVERAGE OPERATIONAL POWER:	11-12 kW AVAILABLE FOR PAYLOADS



OBJECTIVE: THE SPACE PLATFORM ALPHA IS A SHUTTLE-DEPLOYED AND SHUTTLE-TENDED FACILITY PLACED IN LOW EARTH ORBIT FOR AN INDEFINITE TIME. IT WILL PROVIDE STABILITY, POINTING, COMMUNICATIONS, POWER, AND THERMAL DISSIPATION SERVICES TO A VARIETY OF TEMPORARILY EMPLACED PAYLOADS. THIS PROGRAM IS INTENDED TO PROVIDE THE CAPABILITY TO OPERATE THOSE PAYLOADS THAT ARE RESTRICTED BY THE LIMITED TIME AND POWER AVAILABLE DURING A SHUTTLE SORTIE MISSION.

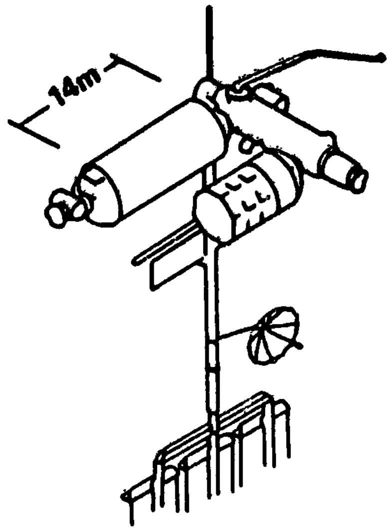
DESCRIPTION: THE PROGRAM IS A SYNTHESIS OF SEVERAL ACTIVITIES WITHIN THE OFFICE OF SPACE TRANSPORTATION. STUDIES OF THE 25 kW POWER SYSTEM, SCIENCE AND APPLICATIONS SPACE PLATFORM AND THE MATERIALS EXPERIMENT CARRIER FORM A BASE FOR THIS CONCEPT.

THE BASELINE DESIGN FOR THE SPACE PLATFORM IS AN 11-12 kW SYSTEM WITH FACILITIES TO BERTH WITH AND OPERATE AT LEAST THREE PAYLOAD COMPLEMENTS.

THE OPERATIONAL MODES WILL INCLUDE THE FREE-FLYING PLATFORM MODE (PRIMARY) AND THE ATTACHED OR SORTIE MODE. THE SHUTTLE-ATTACHED MODE WOULD BE USED TO: (1) EXCHANGE PAYLOADS; (2) PERFORM PLATFORM MAINTENANCE AND REPAIR; (3) AUGMENT PLATFORM CAPABILITIES; (4) PERMIT EXTENSION OF THE ORBITER ON-ORBIT STAY TIME.

THE PROGRAM IS CURRENTLY IN THE SYSTEM DEFINITION AND PRELIMINARY DESIGN PHASE. TWO STUDY CONTRACTORS (TRW AND MDAC) ARE DEVELOPING COMPETITIVE ALTERNATE SYSTEM CONCEPTS.

SPACE STATION



STATUS:	CANDIDATE
LIFETIME:	>10 YEARS
LAUNCH AND TRANSFER VEHICLES:	SHUTTLE
OPERATIONAL LOCATIONS:	LOW EARTH ORBIT
TOTAL MASS AT OPERATIONAL LOCATIONS:	59,000 kg FOR INITIAL SOC
AVERAGE OPERATIONAL POWER:	50 kW SUNLIT

OBJECTIVE: THE SPACE STATION WILL PROVIDE A MANNED OPERATIONAL BASE IN LOW EARTH ORBIT TO PERFORM MISSIONS REQUIRING EXTENDED ORBITAL STAY TIMES WITH FREQUENT OR CONTINUOUS CREW INVOLVEMENT.

DESCRIPTION: THE SPACE STATION CONCEPT DESCRIBED HERE IS THE SPACE OPERATIONS CENTER NOW BEING ANALYZED BY BOEING. THE SOC IS SPACE ASSEMBLED FROM MODULES PLACED IN ORBIT BY MULTIPLE SHUTTLE LAUNCHES. THE INITIAL VERSION INCLUDES ONE HABITAT MODULE, ONE SERVICE MODULE, ONE LOGISTICS MODULE AND AN AIRLOCK. IT CAN BE CONTINUOUSLY MANNED BY A CREW OF FOUR (90 DAY NORMAL RESUPPLY), CAN PERFORM SATELLITE SERVICING, UPPER-STAGE TO PAYLOAD MATING, AND CAN ACCOMMODATE A VARIETY OF SCIENCE AND APPLICATIONS PAYLOADS AND EXPERIMENTS. AN AUGMENTED VERSION, THE OPERATIONAL SOC, INCLUDES AN ADDITIONAL HABITAT MODULE, LOGISTICS MODULE AND SERVICE MODULE. A DOCKING MODULE AND FINGER PIERS WOULD ALSO BE ADDED. THE OPERATIONAL SOC PROVIDES FOR S/C DEPLOYMENT AND ASSEMBLY, LIQUID UPPERSTAGE STORAGE, AND UNASSISTED MATING OF UPPER STAGES TO SPACECRAFT. THE CREW COMPLEMENT CAN BE INCREASED TO EIGHT. FUTURE VERSIONS OF SOC ARE CONTEMPLATED. THEY WOULD PERMIT ON-ORBIT PROPELLANT STORAGE, OTV BASING AND FACILITIES FOR ASSEMBLY OR CONSTRUCTION OF LARGE SPACECRAFT.

AUTOMATED PLANETARY STATION

STATUS:	OPPORTUNITY	OPERATIONAL LOCATION:	LOW EARTH ORBIT
LIFETIME:	5-10 YEARS	TOTAL MASS AT OPERATIONAL LOCATION:	25,000 kg
LAUNCH AND TRANSFER VEHICLES:	SHUTTLE	AVERAGE OPERATIONAL POWER:	25 kW

OBJECTIVE: PLANETARY OBSERVATIONS ARE TO BE MADE FROM AN ORBITING PLATFORM. CONTINUOUS OBSERVATIONS OF DYNAMIC PHENOMENA (e.g., ATMOSPHERES) ARE POSSIBLE. EMPHASIS WILL BE ON WAVELENGTHS THAT CANNOT BE OBSERVED WITH EARTH-BASED TELESCOPES, ON THE USE OF INTERFEROMETERS WITH 0.01-ARC SEC ANGULAR RESOLUTIONS THAT CANNOT BE ATTAINED WHEN VIEWING THROUGH THE EARTH'S ATMOSPHERE, AND ON NEW MEASUREMENT TECHNIQUES FOR DEEP SPACE MISSIONS.

DESCRIPTION: A 25-kW POWER MODULE PROVIDES THE BASIC SUPPORT FUNCTIONS FOR THIS MISSION. THE POWER MODULE PROVIDES POWER, ATTITUDE CONTROL, AND COMMUNICATIONS. INSTRUMENTS ARE PLACED ON PALLETS (SIMILAR TO SPACELAB) AND MAY BE EITHER RELATIVELY SMALL PRINCIPAL INVESTIGATOR EXPERIMENTS OR LARGER MULTI USER FACILITIES. SHUTTLE REVISITS COULD BE UTILIZED FOR REPLENISHING CONSUMABLES (e.g., FILM, CRYOGENIC LIQUIDS), MAINTENANCE, AND FOR RECOVERY OF THE PALLET AND ITS INSTRUMENTATION. EARLIEST POSSIBLE LAUNCH DATE IS 1990. CONSEQUENTLY, LITTLE EFFORT HAS BEEN EXPENDED STUDYING THIS OPPORTUNITY.

FUTURE LSS/PROPULSION SYSTEM DRIVERS

LEO TO GEO ORBIT TRANSFER

LOW THRUST TO WEIGHT PROPULSION FOR LARGE SPACE STRUCTURES WITH MASS OF 6800 to 85,000 kg

ATTITUDE AND MOMENTUM CONTROL

LONG LIFE (>10 YEARS), RELIABLE AUXILIARY PROPULSION SYSTEMS

PROPULSION SYSTEMS FOR VERY LARGE (100 METER) STRUCTURES

PROPULSION SYSTEMS FOR SPACE PLATFORMS WITH CHANGING MASS AND INERTIAL PROPERTIES

N82 27361

POTENTIAL LARGE SPACE SYSTEMS MISSION OPPORTUNITIES FOR THE POST 1990s

R. L. CHASE

Analytic Services Inc. (ANSER)
400 Army-Navy Drive
Arlington, Virginia 22202

100 STRATEGIC WARFARE
- 121 BALLISTIC MISSILE DEFENSE
- 122 STRATEGIC AIR DEFENSE
- 123 SPACE DEFENSE
- 131 STRATEGIC COMMAND AND CONTROL
- 132 STRATEGIC SURVEILLANCE AND WARNING
- 133 STRATEGIC COMMUNICATIONS
- 121 STRATEGIC SUPPORT

200 TACTICAL WARFARE
- 212 FIRE SUPPORT
- 213 MINE WARFARE
- 214 LAND COMBAT SUPPORT
- 221 COUNTERAIR
- 222 CLOSE AIR SUPPORT/INTERDICTION
- 223 DEFENSE SUPPRESSION
- 224 AIR WARFARE SUPPORT
- 231 ANTI-AIR WARFARE
- 232 ANTI-SURFACE WARFARE
- 233 ANTI-SUBMARINE WARFARE
- 235 AMPHIBIOUS WARFARE
- 236 NAVAL WARFARE SUPPORT
- 241 BATTLEFIELD TNW
- 242 THEATER-WIDE TNW
- 243 DEFENSIVE TNW
- 244 SEA CONTROL TNW

- 261 THEATER COMMAND AND CONTROL
- 262 THEATER SURVEILLANCE AND RECONNAISSANCE
- 263 THEATER AND COMMON USER COMMUNICATIONS
- 265 TACTICAL COMMAND AND CONTROL
- 266 TACTICAL SURVEILLANCE AND RECONNAISSANCE
- 267 TACTICAL COMMUNICATIONS
- 268 ELECTRONIC WARFARE AND COUNTER C^3
- 264 REFUELING

300 DEFENSE-WIDE C^3I
- 312 GENERAL DEFENSE INTELLIGENCE PROGRAMS
- 313 CLASSIFIED PROGRAMS
- 314 OTHER INTELLIGENCE PROGRAMS
- 321 NAVIGATION AND POSITION FINDING
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- 323 OTHER SUPPORT PROGRAMS
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400 DEFENSE-WIDE MISSION SUPPORT
- 411 ORBITAL SUPPORT, GROUND
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MISSION DEFINITION

- MISSILE DEFENSE
- THEATER TNW
- SPACE DEFENSE
- COMMAND, CONTROL, AND COMMUNICATIONS
- SURVEILLANCE AND WARNING
- DEFENSE SUPPRESSION
- FORCE SUPPORT
- FIRE SUPPORT/INTERDICTION
- MINE WARFARE
- SPACE TRANSPORTATION
- ORBITAL SUPPORT
- ENVIRONMENTAL SUPPORT

POTENTIAL MISSIONS REQUIRING LARGE SPACE SYSTEMS

- MISSILE DEFENSE
- THEATER TNW
- SPACE DEFENSE
- COMMAND, CONTROL, AND COMMUNICATIONS
- SURVEILLANCE AND WARNING
- DEFENSE SUPPRESSION
- FORCE SUPPORT
- FIRE SUPPORT/INTERDICTION
- MINE WARFARE
- SPACE TRANSPORTATION
- ORBITAL SUPPORT
- ENVIRONMENTAL SUPPORT

MISSILE DEFENSE

SPACE DEFENSE

COMMAND, CONTROL, AND COMMUNICATIONS

SURVEILLANCE AND WARNING

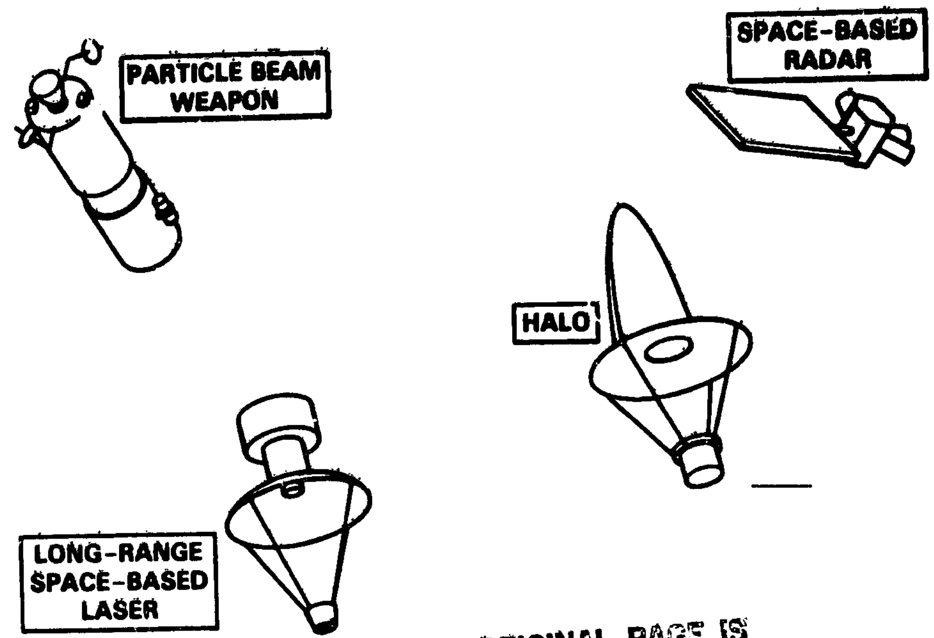
DEFENSE SUPPRESSION

FORCE SUPPORT

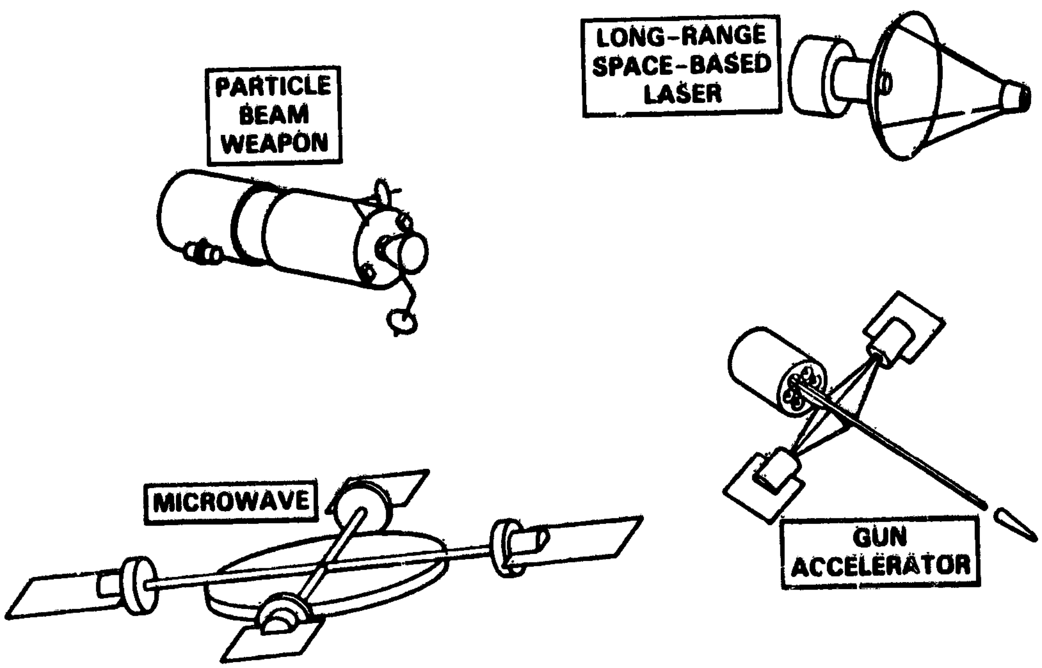
ORBITAL SUPPORT

SPACE TRANSPORTATION

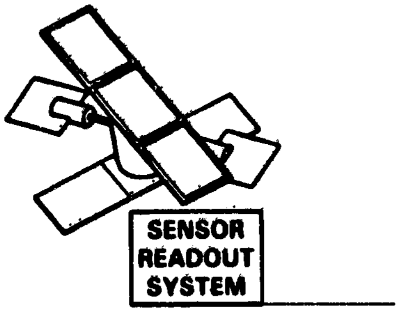
MISSILE DEFENSE

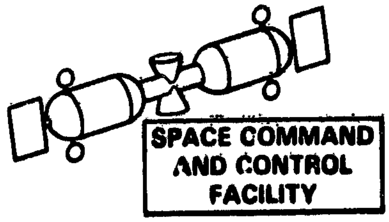


SPACE DEFENSE

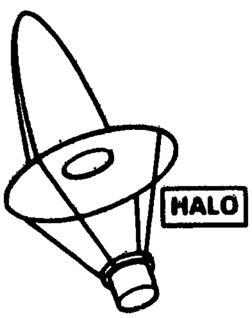


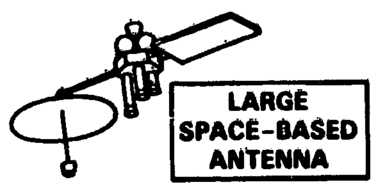
COMMAND, CONTROL, AND COMMUNICATIONS

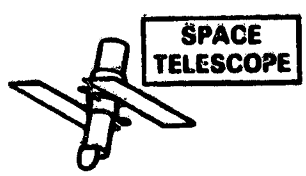




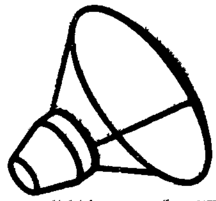
SURVEILLANCE AND WARNING





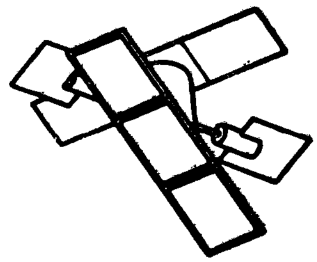


DEFENSE SUPPRESSION



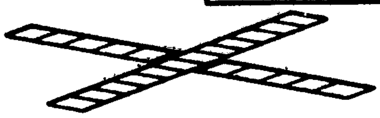
MULTIPURPOSE ORBITAL JAMMER

FORCE SUPPORT

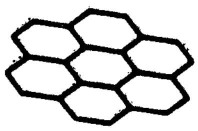


GLOBAL LOGISTIC INFORMATION SYSTEM

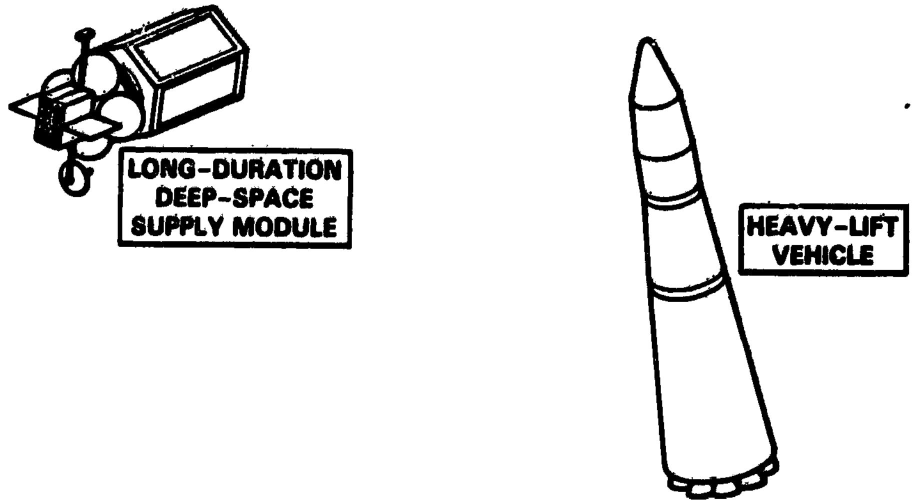
PERSONAL NAVIGATION SYSTEM



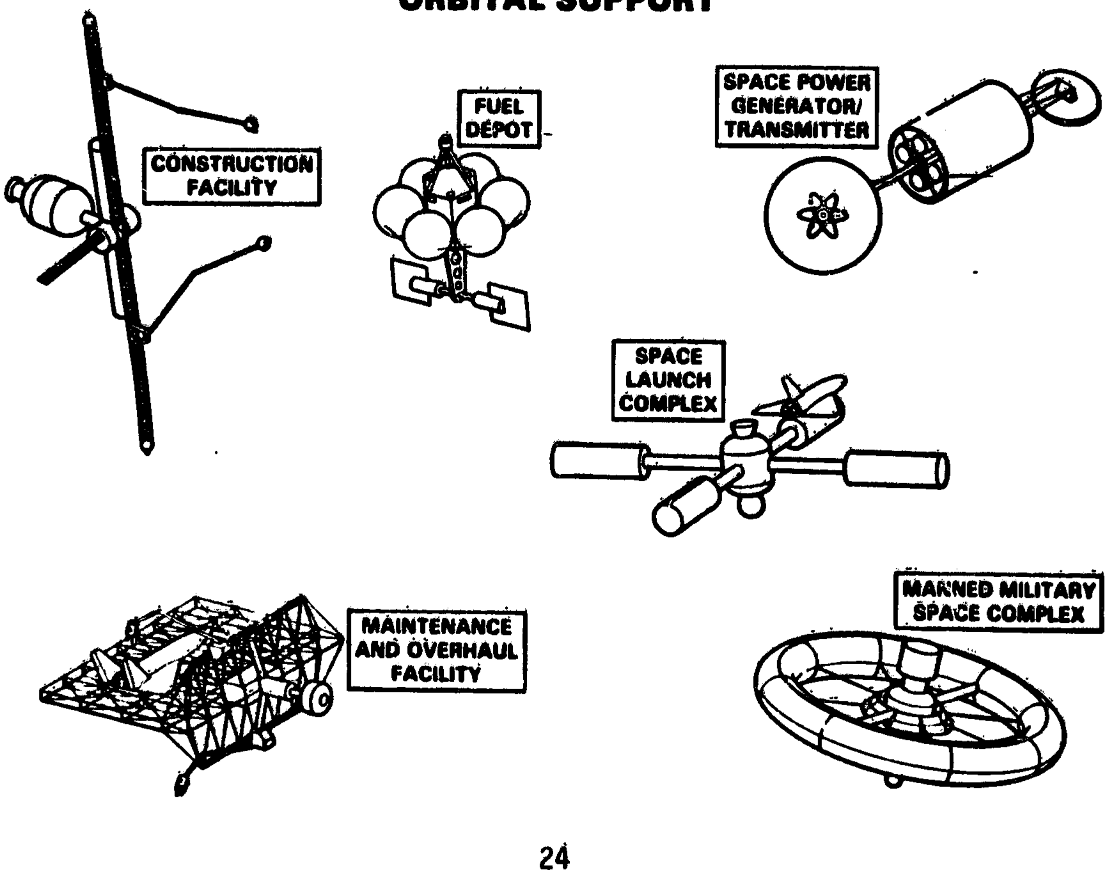
BATTLEFIELD ILLUMINATOR



SPACE TRANSPORTATION



ORBITAL SUPPORT



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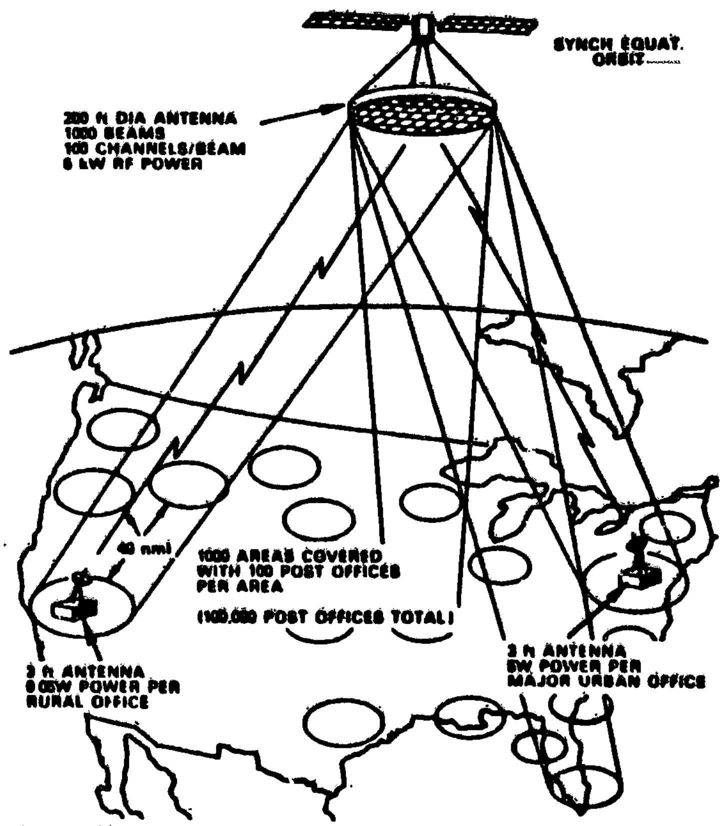
ADVANCED SPACE SYSTEM CONCEPTS AND THEIR ORBITAL SUPPORT NEEDS (1980-2000)

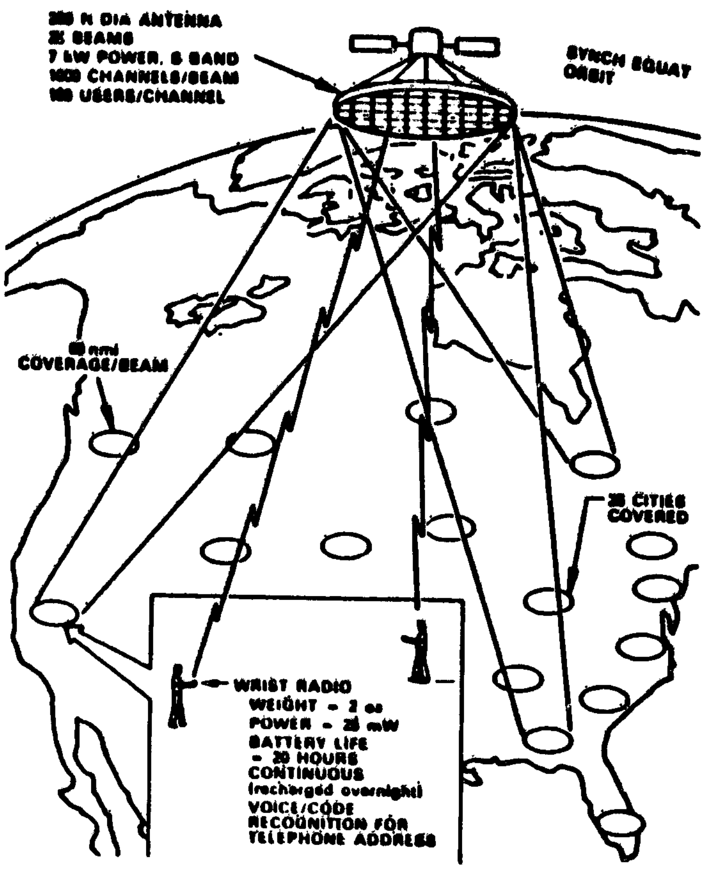
J. Butts

THE AEROSPACE CORPORATION
El Segundo, California 90245

ELECTRONIC MAIL TRANSMISSION



- **PURPOSE**
  To speed up delivery and lower costs of most mail.
  To service thinly populated areas.
- **RATIONALE**
  Delivery of physical letters is slow and needless in most cases when locally reproduced facsimile could do.
- **CONCEPT DESCRIPTION**
  Page readers and facsimile printers at each post office read, transmit, receive, and reproduce mail. Satellite acts as multi-channel repeater.
- **CHARACTERISTICS**
  - WEIGHT: 20,000 lb
  - SIZE: 200-ft dia antenna
  - RAW POWER: 15 kW
  - ORBIT: Synch. Equat.
  - CONSTELLATION SIZE: 1
  - RISK CATEGORY: I (Low)
  - TIME FRAME: 1990
  - IOC COST (Space only): 430 M
- **PERFORMANCE**
  Transmits facsimile at 10 pages (8 1/2 x 11") per second per post office. Up to 100,000 post offices serviced in up to 50% of area of U. S. A. Total service = 100 billion pages/day.
- **BUILDING BLOCK REQUIREMENTS**
  - TRANSPORTATION: Shuttle and large tug or SEPS
  - ON-ORBIT OPERATIONS: Automated or manual servicing unit; assembly on orbit
  - SUBSYSTEMS: Attitude control; antenna; processor
  - TECHNOLOGY: Large multibeam antenna; multi-channel transponder; LSI processor; multiple-access techniques
  - OTHER: None



- **PURPOSE**
  To allow citizens to communicate through exchanges by voice, from anywhere.
- **RATIONALE**
  Mobile telephones are desirable, but should be wrist worn. Uses include emergency, recreation, business, rescue, etc.
- **CONCEPT DESCRIPTION**
  Multichannel switching satellite and wrist transmitter-receivers connect people anywhere to each other directly or to telephone networks. Analog or vocoded voice used.
- **CHARACTERISTICS**

• WEIGHT	16,000 lb
• SIZE	200 ft dia antenna
• RAW POWER	21 kW
• ORBIT	Synch. Equat.
• CONSTELLATION SIZE	1
• RISK CATEGORY	I (Low)
• TIME FRAME	1990
• IOC COST (SPACE ONLY)	300M

- **PERFORMANCE**
  25,000 simultaneous voice channels, each shared by up to 100 users: 2.5 million people communicate by normal voice.
- **BUILDING BLOCK REQUIREMENTS**

• TRANSPORTATION	Shuttle and large/tandem tug or SEPS
• ON-ORBIT OPERATIONS	Automated or manual servicing unit; assembly on orbit
• SUBSYSTEMS	Attitude control; antenna; processor; repeater
• TECHNOLOGY	Large multibeam antenna; multi-channel repeater; LSI processor, multiple-access techniques
• OTHER	Wrist transceiver, LSI technology

VEHICLE / PACKAGE LOCATOR

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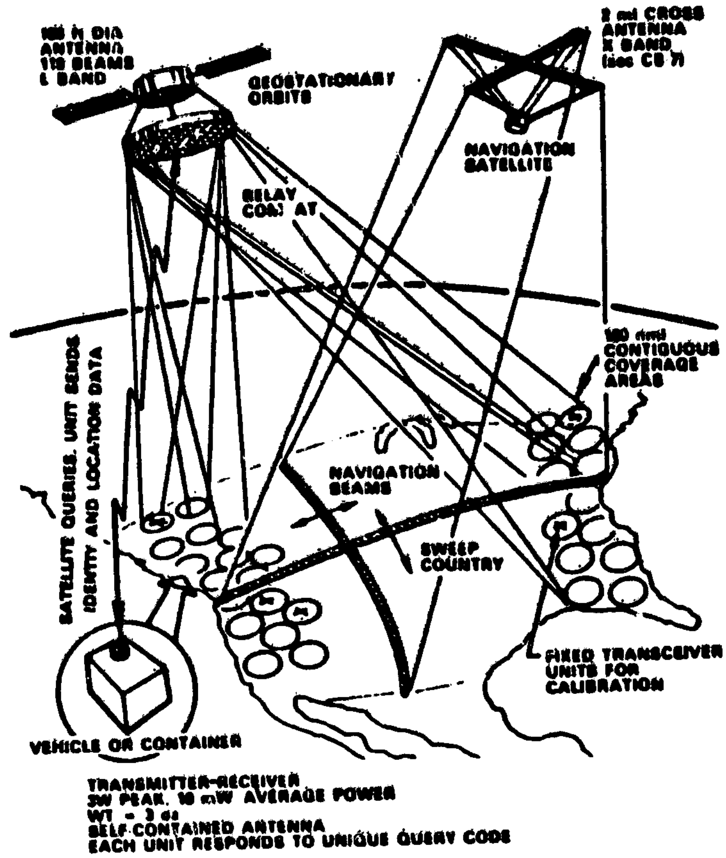


- **PURPOSE**
  To locate vehicles or articles in shipment continuously anywhere in U. S. A.

- **RATIONALE**
  To aid in prevention of theft or hijacking, increase efficiency, and minimize error in shipments

- **CONCEPT DESCRIPTION**
  A small transceiver is attached to (or enclosed in) each unit to be tracked. The unit determines its location using crossed antenna NAVSAT, and relays the data to a control center via a special Comsat when queried.

- **CHARACTERISTICS**

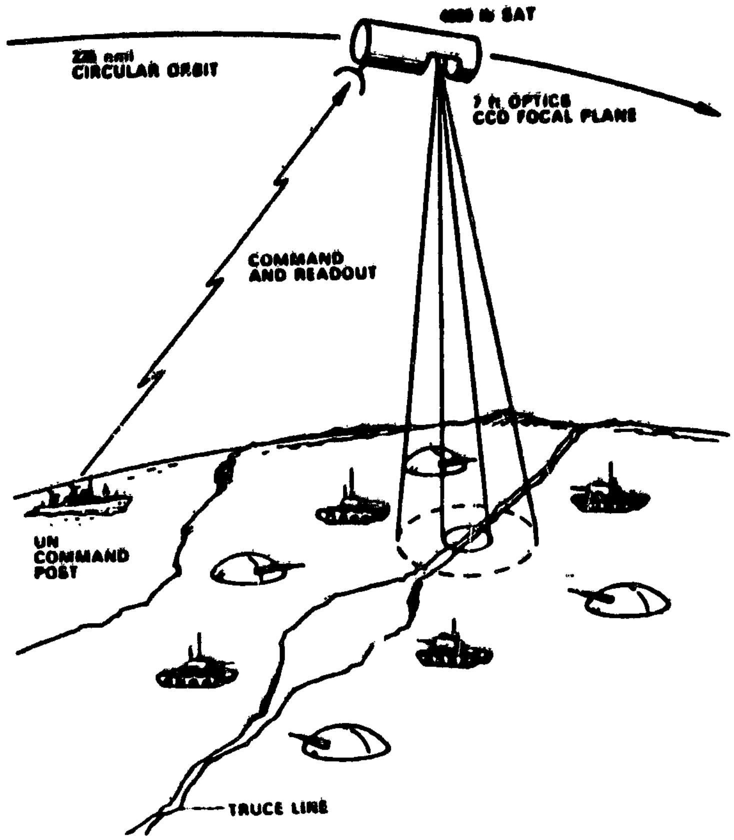
• WEIGHT	20,000 lb (Total)
• SIZE	2-mi antenna
• RAW POWER	23 kW
• ORBIT	Geostationary
• CONSTELLATION SIZE	2
• RISK CATEGORY	II (Medium)
• TIME FRAME	1990
• IOC COST (Space only)	400 M

- **PERFORMANCE**
  Up to one billion vehicles or containers can be located ± 300 ft every hour anywhere in U. S. A. Location package could cost less than $10, weigh 3 ounces.

- **BUILDING BLOCK REQUIREMENTS**

• TRANSPORTATION	Shuttle and large/tandem tug or SEPS
• ON-ORBIT OPERATIONS	Automated or manned assembly and servicing
• SUBSYSTEMS	Antenna attitude control, laser radar, channelizer/processor, stationkept antenna
• TECHNOLOGY	Phase control, LSI processor, multiple access technique, stationkept sub-units
• OTHER	Cheap - LSI - container - transponder

U.N. TRUCE OBSERVATION SATELLITE

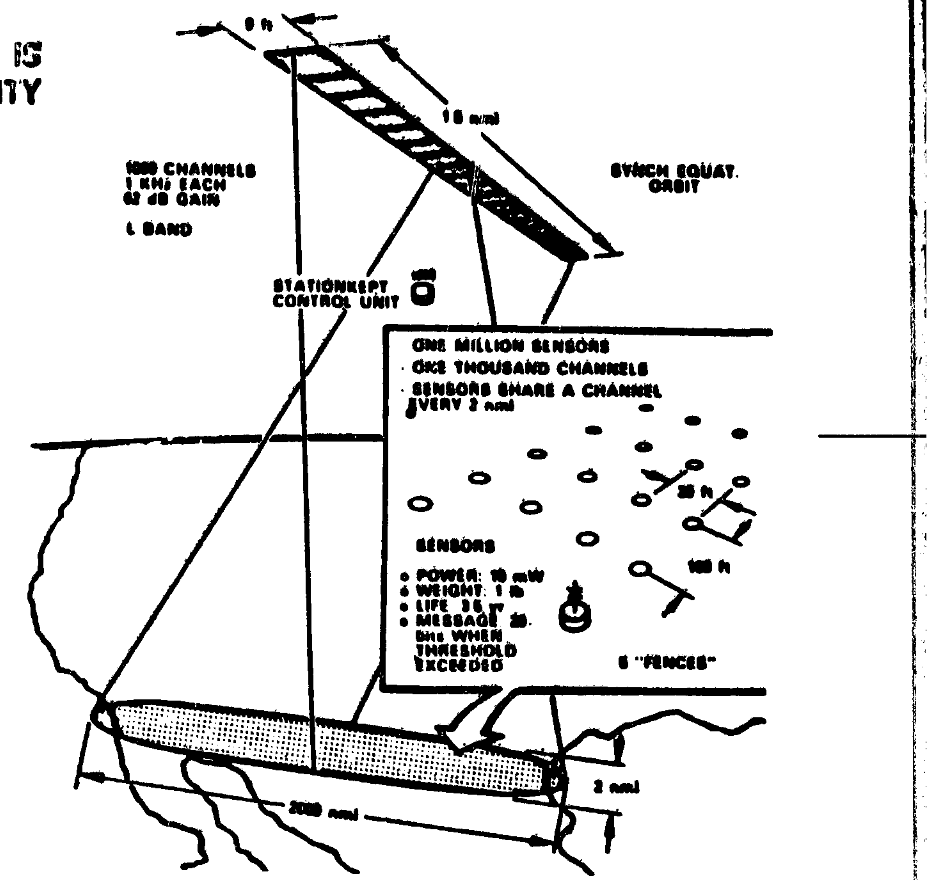


- **PURPOSE**
  Aid U.N. teams to monitor truce agreements, particularly border zones, and weapon system dispositions such as missile launchers.
- **RATIONALE**
  U.N. will have responsibility for truce monitoring, but will be denied on-site capability in some cases. Space systems are free from local control or interference.
- **CONCEPT DESCRIPTION**
  One low altitude satellite with visible light optics for daytime monitoring and infrared optics for night-time operation.
- **CHARACTERISTICS**

• WEIGHT	4,000 lb
• SIZE	15 x 60 ft
• RAW POWER	3 kW
• ORBIT	225 nmi near-polar
• CONSTELLATION SIZE	1
• RISK CATEGORY	I (Low)
• TIME FRAME	1985
• K/C COST (Space only)	90 M

- **PERFORMANCE**
  Ground resolution, < 6 ft. (Visible) 120-ft I.R. Location accuracy, 300 ft. Truce area covered twice a day.
- **BUILDING BLOCK REQUIREMENTS**

• TRANSPORTATION	Shuttle
• ON-ORBIT OPERATIONS	Shuttle attached manipulator
• SUBSYSTEMS	Focal plane
• TECHNOLOGY	Similar to weather satellites and ERTS; CCD focal plane
• OTHER	



- **PURPOSE**
  To detect overt or covert attempts at crossing a border.
- **RATIONALE**
  Flow of illegal aliens and drug traffickers is a major problem. Detection is difficult along long, unpatrolled borders.
- **CONCEPT DESCRIPTION**
  Very many, very small seismic sensors are read out by a satellite with very large antenna. Penetration causes vibrations which are picked up and correlated at a central site.
- **CHARACTERISTICS**

• WEIGHT	8000 lb
• SIZE	9000 ft x 9 ft
• RAW POWER	20 kW
• ORBIT	Synch. Equat.
• CONSTELLATION SIZE	1
• RISK CATEGORY	II (Medium)
• TIME FRAME	1990
• IOC COST (Space only)	170 M

- **PERFORMANCE**
  Virtually all moving objects detected. False alarms sorted by correlation between sensors and fences. Sensor life 3.5 years at one penetration attempt per sensor per month.
- **BUILDING BLOCK REQUIREMENTS**

• TRANSPORTATION	Shuttle and tug
• ON-ORBIT OPERATIONS	Automated or manual assembly and servicing unit
• SUBSYSTEMS	Structure; attitude control; antenna
• TECHNOLOGY	Large passive microwave antenna - stationkeeping subsatellites; laser master measuring and control unit
• OTHER	Small, light, long-lived sensor units which are very cheap in mass production.

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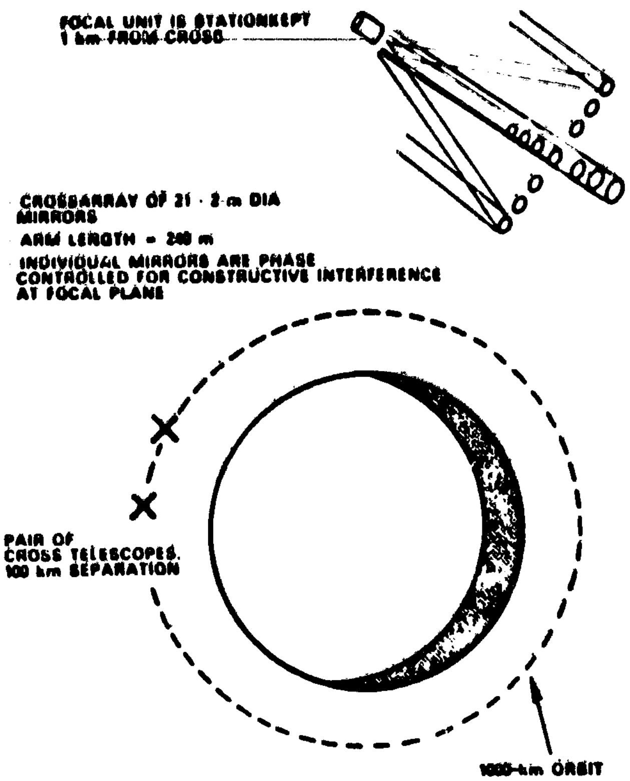
ASTRONOMICAL SUPER TELESCOPE



- **PURPOSE**
  To extend knowledge of universe by examination of most distant objects.
- **RATIONALE**
  Largest earth telescopes have insufficient resolution. Need even more than LST will provide.
- **CONCEPT DESCRIPTION**
  A cross-array of visible light and 100 μm mirrors is phase controlled at mirrors or near focal plane for constructive interference. Laser link to other cross-array.
- **CHARACTERISTICS**

• WEIGHT	40,000 lb
• SIZE	800 ft cross
• RAW POWER	10 kW
• ORBIT	300 nmi circular
• CONSTELLATION SIZE	2-100 km apart
• RISK CATEGORY	IV (High)
• TIME FRAME	2000
• IOC COST (Space only)	450 M.

- **PERFORMANCE**
  Direct parallax measurements to 6500 light years with one cross. Resolution of one cross = 3×10^{-9} radians. Resolution of 2 crosses = 10^{-11} radians.
- **BUILDING BLOCK REQUIREMENTS**

• TRANSPORTATION	Shuttle
• ON-ORBIT OPERATIONS	Automated or manual service unit, manned assembly
• SUBSYSTEMS	Mirrors, stationkeeping, structure, sensor, phase control mechanism
• TECHNOLOGY	Adaptive focal plane, mirrors, stationkeeping sensors
• OTHER	

HIGH RESOLUTION EARTH MAPPING RADAR

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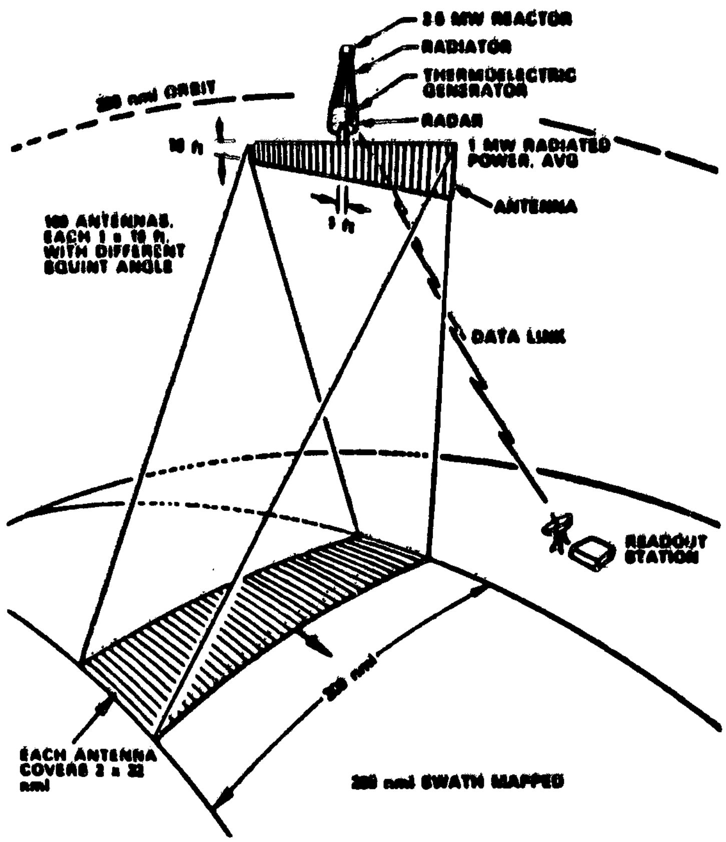


- **PURPOSE**
  To provide maps of the surface with high resolution through cloud cover.
- **RATIONALE**
  Resources, pollution, crop, water, and other observations may be aided by high resolution and frequent coverage regardless of weather.
- **CONCEPT DESCRIPTION**
  Synthetic array radar of very high power provides high resolution. On-board image processing allows microwave data link for all weather capability.
- **CHARACTERISTICS**

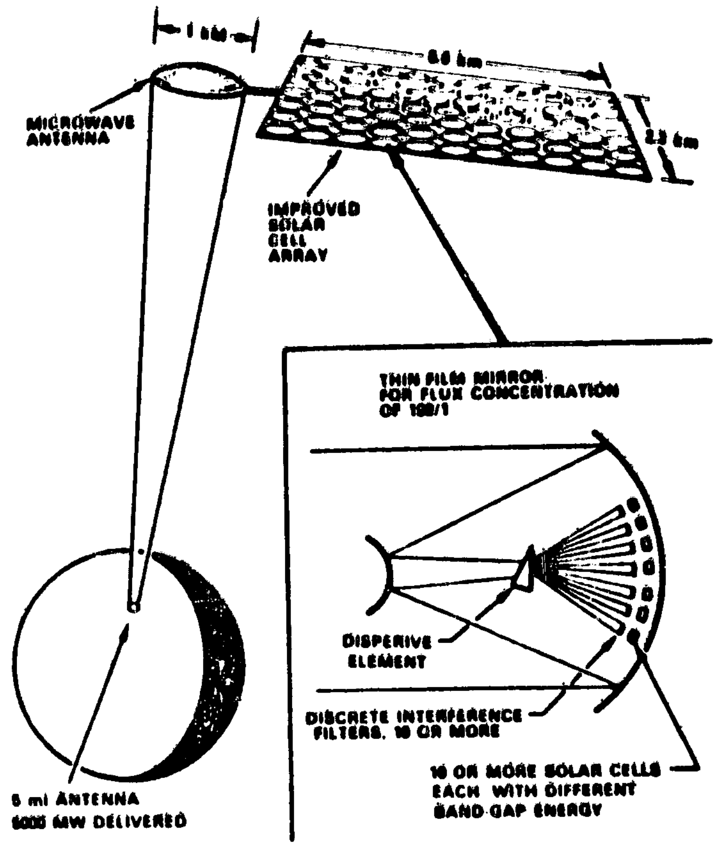
• WEIGHT	110,000 lb
• SIZE	16 x 100 ft
• RAW POWER	2.5 MW
• ORBIT	200 nmi polar
• CONSTELLATION SIZE	1
• RISK CATEGORY	II (Medium)
• TIME FRAME	1990
• IOC COST (Space only)	500 M —

- **PERFORMANCE**
  200 nmi ground swath mapped to less than a few feet resolution once a day. U. S. covered every six days.

- **BUILDING BLOCK REQUIREMENTS**

• TRANSPORTATION	Shuttle
• ON-ORBIT OPERATIONS	Shuttle manipulator; servicing
• SUBSYSTEMS	Thermal, nuclear, power generator, radar
• TECHNOLOGY	High power transmitter; automated image processor, reactor, shielding
• OTHER	None

HIGH EFFICIENCY SOLAR ENERGY GENERATION

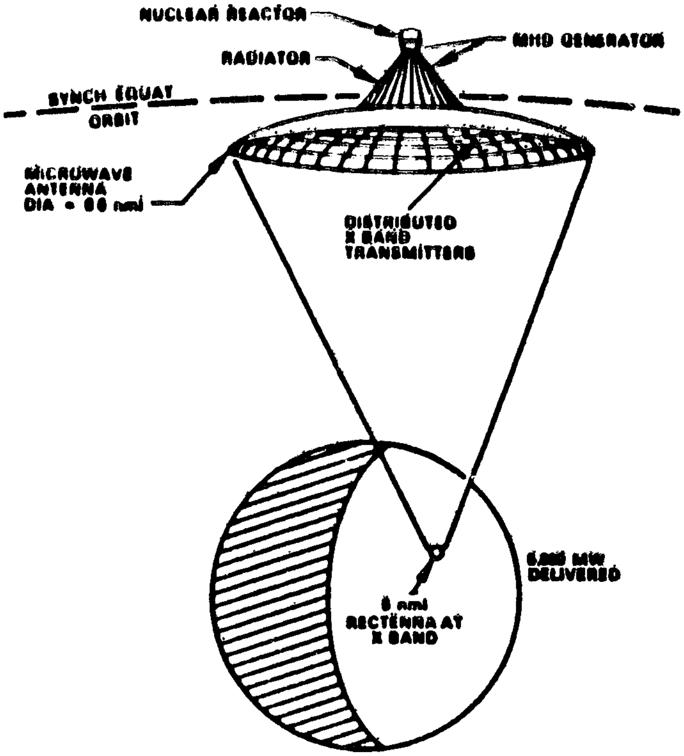


- **PURPOSE**
  To increase the efficiency and decrease the cost of solar power delivery from space.
- **RATIONALE**
  Solar power satellites will be large, heavy, and expensive.
- **CONCEPT DESCRIPTION**
  Efficiency and cost of solar-voltaic conversion can be greatly increased by using multiple cells, each tailored to the photon energy in a restricted spectrum; and by using high ratio solar flux concentration.
- **CHARACTERISTICS**

• WEIGHT	8,000,000 lb
• SIZE	5.6 x 2.3 km
• RAW POWER	10,000 MW
• ORBIT	Synch. Equat.
• CONSTELLATION SIZE	1
• RISK CATEGORY	IV (High)
• TIME FRAME	2000
• IOC COST	TBD

- **PERFORMANCE**
  Same power delivered with one fifth the weight in orbit compared to the current solar power satellite concept (CS-1) and probable but undetermined cost reduction.
- **BUILDING BLOCK REQUIREMENTS**

• TRANSPORTATION	LLV and large tug and large SEPS
• ON-ORBIT OPERATIONS	Manned servicing unit; assemble in orbit
• SUBSYSTEMS	Attitude control; structures; power antenna
• TECHNOLOGY	Large economical solar arrays; large active microwave antenna; high power tubes; lightweight concentrator; thermal design
• OTHER	Rectenna on ground



- **PURPOSE**

  To generate and deliver electrical energy without pollution or hazard.

- **RATIONALE**

  Power is needed which requires no radioactive material on earth, produces no atmospheric heating, and no resource consumption.

- **CONCEPT DESCRIPTION**

  A breeder reactor, MHD power generator, microwave transmitter, and microwave antenna are used to beam energy to a ground receiver. Fuel breeding supplies fuel.

- **CHARACTERISTICS**

• WEIGHT	TBD
• SIZE	3,600-ft dia
• RAW POWER	10,000 MW
• ORBIT	Synch. Equat.
• CONSTELLATION SIZE	1
• RISK CATEGORY	IV (High)
• TIME FRAME	2000
• IOC COST (Space only)	TBD

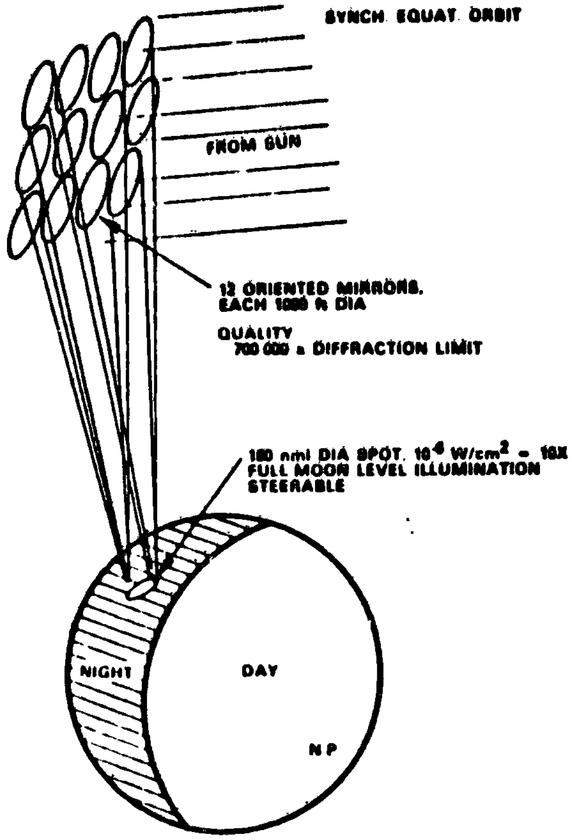
- **PERFORMANCE**

  5,000 Megawatts delivered power continuously - with sufficient fuel breeding for a life of at least 1000 years.

- **BUILDING BLOCK REQUIREMENTS**

• TRANSPORTATION	LLV and large tug and large SEPS
• ON-ORBIT OPERATIONS	Manned service unit, automated servicing unit; assemble in orbit
• SUBSYSTEMS	Structure; attitude control; antenna; reactor; power unit
• TECHNOLOGY	Large active microwave antenna; large reactor; heat radiator; MHD power generator; pointing and tracking sensor
• OTHER	Rectenna on ground; safety

NIGHT ILLUMINATOR



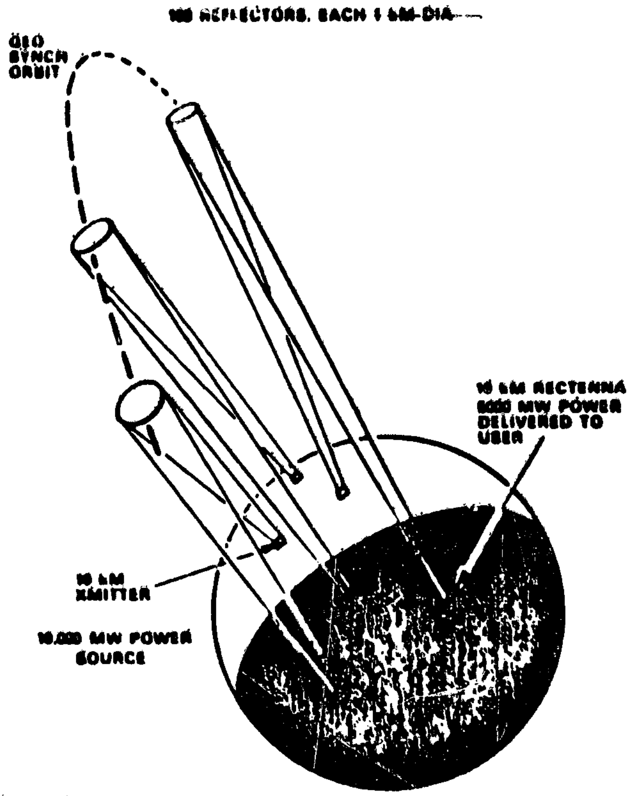
- **PURPOSE**
  To provide night lighting without earth-based energy, pollution, street lights, cables, trenches, etc.
- **RATIONALE**
  Alternative energy sources are needed.
- **CONCEPT DESCRIPTION**
  Large area reflectors in space reflect the image of the sun onto the earth. Multiple satellites used to minimize construction difficulties.
- **CHARACTERISTICS**

• WEIGHT	100,000 lb
• SIZE	12 mirrors each 1,000-ft dia
• RAW POWER	1.2 kW
• ORBIT	Synch. Equat.
• CONSTELLATION SIZE	1
• RISK CATEGORY	II (Medium)
• TIME FRAME	1990
• IOC COST (Space only)	160 M

- **PERFORMANCE**
  Ten times full-moon level illumination at night provided to area 180 nmi dia (no clouds). Full moon level provided through moderate clouds.
- **BUILDING BLOCK REQUIREMENTS**

• TRANSPORTATION	Shuttle and large tug and/or SEPS
• ON-ORBIT OPERATIONS	Automated or manual servicing unit
• SUBSYSTEMS	Attitude control; mirrors, structure
• TECHNOLOGY	Large reflector; pointing; stationkeeping master control
• OTHER	None

POWER RELAY SATELLITE



- **PURPOSE**
  To provide for transmission of electrical power from remote regions, minimizing environmental impact.
- **RATIONALE**
  Power should be generated in remote regions.
  Sunny side of Earth can supply power to night side.
- **CONCEPT DESCRIPTION**
  Source power is converted to a microwave beam, bounced off an orbiting reflector, and reconverted to DC at receiving antenna on ground.
- **CHARACTERISTICS**

• WEIGHT	600,000 lb
• SIZE	0.5-nml dia
• RAW POWER	---
• ORBIT	Synch. Equat.
• CONSTELLATION SIZE	100
• RISK CATEGORY	IV (High)
• TIME FRAME	1995
• IOC COST (Space only)	36 B

- **PERFORMANCE**
  5,000 megawatts delivered to each of 100 user areas. 53 percent overall DC-DC efficiency attained. Total energy is about 10 percent of U. S. consumption.
- **BUILDING BLOCK REQUIREMENTS**

• TRANSPORTATION	LLV and large tug or large SEPS
• ON-ORBIT OPERATIONS	Manned/automated servicing, assemble in orbit
• SUBSYSTEMS	Attitude control; structures, phase front control
• TECHNOLOGY	High efficiency, large, passive steerable phase front antenna; ion thrusters
• OTHER	Ground-based elements

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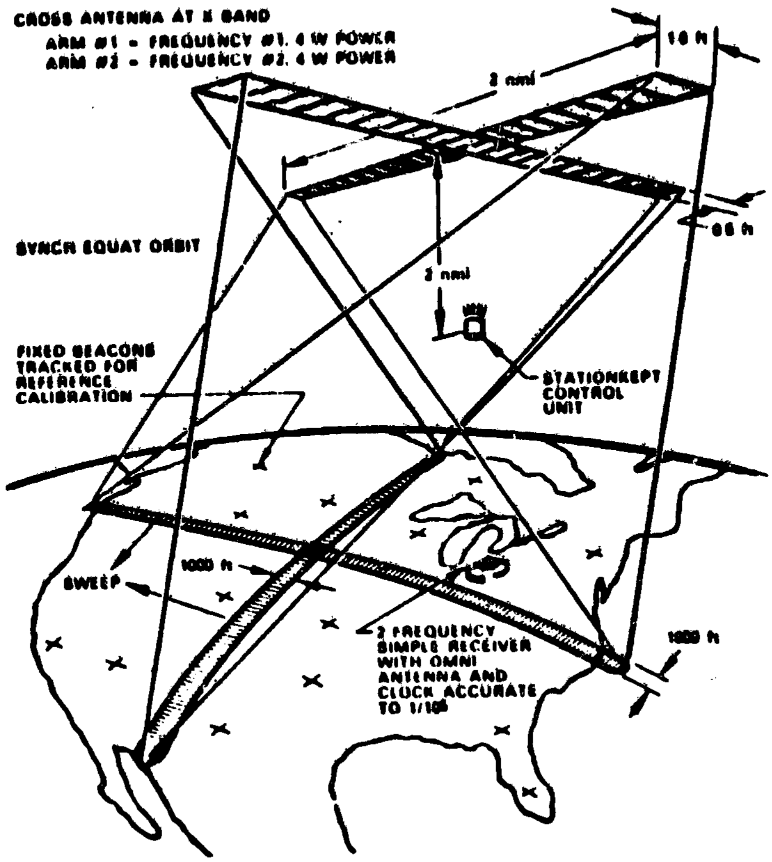
PERSONAL NAVIGATION WRIST SET



- **PURPOSE**
  To provide accurate relative position location with very inexpensive user equipment.

- **RATIONALE**
  Navigation system costs are dominated by user equipment costs.

- **CONCEPT DESCRIPTION**
  Narrow beams are swept over the U. S. by large phased arrays in space. Very simple receivers measure time elapsed between pulses received and display distances (N-S, E-W) to fixed point.

- **CHARACTERISTICS**

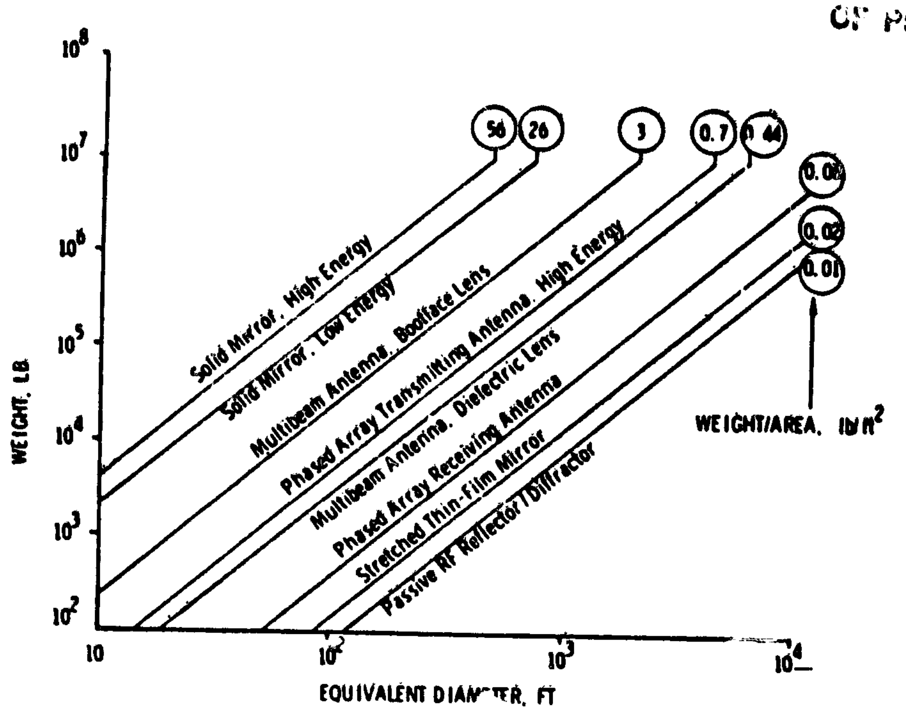
• WEIGHT	3000 lb
• SIZE	2 nmi cross
• RAW POWER	2 kW
• ORBIT	Sync. Equat.
• CONSTELLATION SIZE	1
• RISK CATEGORY	II (Medium)
• TIME FRAME	1990
• IOC COST (SPACE ONLY)	100 M

- **PERFORMANCE**
  - User position located to 300 ft every 10 sec relative to a fixed location < 100 nmi away.
  - User receiver can cost less than $10 in mass production.

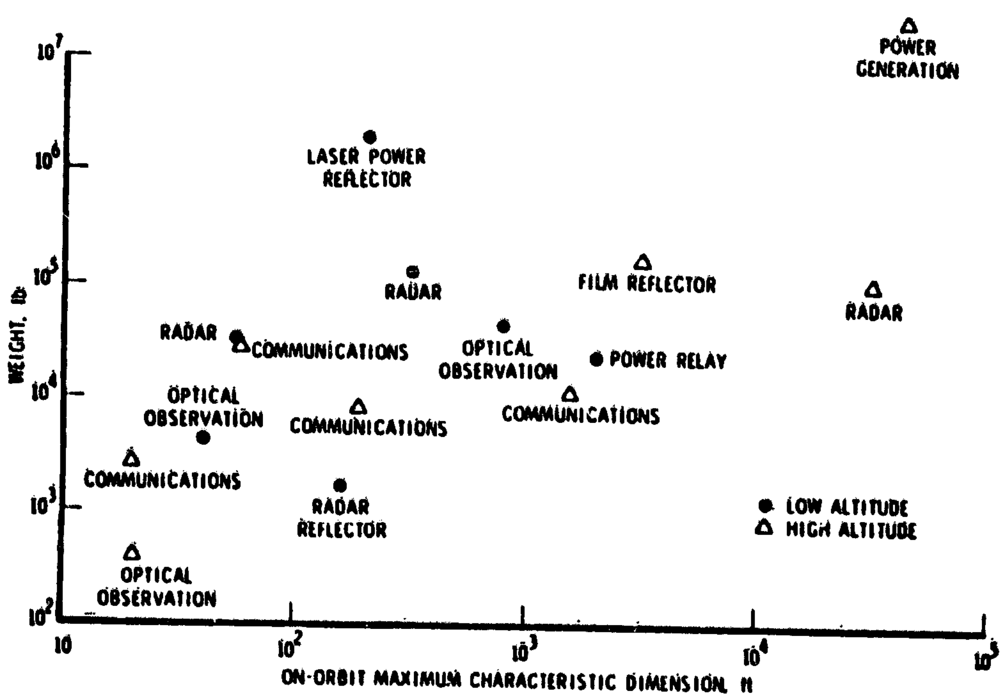
- **BUILDING BLOCK REQUIREMENTS**

• TRANSPORTATION	Shuttle and Tug
• ON-ORBIT OPERATIONS	Manned or automated assembly and servicing units
• SUBSYSTEMS	Antenna with independently stationkept subunits.
• TECHNOLOGY	Ion thruster, adaptive RF phase control, laser master measuring unit
• OTHER	LSI receivers

LARGE STRUCTURES WEIGHT ESTIMATION



MAJOR PARAMETERS OF REPRESENTATIVE CONCEPTS



"MISSIONS" PANEL WORKSHOP SUMMARY

R. L. Chase

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400 Army-Navy Drive
Arlington, Virginia 22202

The objective of the "Missions" panel was to define a mission set that would require the potential use of very large, advanced space systems; to rank the mission on the bases of need from both a civilian and a military perspective; to evaluate the advanced space systems with respect to propulsion needs, both primary and secondary; and to summarize as concisely as possible the high-priority mission forces which require advanced propulsion technology. In addition, the panel was to provide specific propulsion-technology guidance for use by NASA propulsion-technology planners. To accomplish these objectives, this panel was made up of a representative cross section of advanced civilian and military mission planners and advanced systems planners.

To allow the panel maximum time during the two-day workshop for deliberation and formulation of a consensus on each of the stated objectives, the approach outlined in figure 1 was followed. On the first day, three presentations summarizing future mission requirements were made to the panel. The first presentation was based on a technology mission model study by General Research Corporation on the potential need for large space systems for future civilian missions. The second was based on an advanced military space architecture study by Analytical Services on the potential need for large space systems for future military missions. The third was based on an advanced space system concepts study by Aerospace Corporation on the civilian and military needs for large space systems.

The initial effort by the panel was to select a set of missions and advanced system concepts representative of the broad spectrum of opportunities described in the presentations. This effort was followed by the establishment of assessment criteria that enabled the ranking of each mission. After mission ranking, a representative set of propulsion-system evaluation criteria was prepared. The panel then ranked mission/system propulsion needs with respect to the evaluation criteria. The panel activity was concluded by the formulation of a concise overall mission/system propulsion technology guidance statement for use by NASA propulsion technology planners.

Figure 2 presents a summary of the mission set and the large space systems associated with each mission. The seven missions shown on this figure are not a unique set but represent all the generic large space systems considered by the panel. The seven missions are a subset of over 50 civilian and military missions.

Having selected a representative subset of missions and systems, the panel ranked the mission set on the basis of perceived national need, both civilian and military. A delphi repetitive voting process was used to achieve a consensus. Figure 3 presents the results of the voting algorithm used. The missions and rankings clearly favor military over civilian needs, but they are intended to be representative of national needs. Also, they acknowledge the growing importance of communication satellites over a broad spectrum of civilian and military uses.

To assess the propulsion technology needs and risks, the panel established the following set of evaluation factors:

Propulsion design need factors:

- Satellite weight
- Satellite orbit
- On-orbit maneuvering
- Station keeping
- Pointing
- Acceleration limit
- Figure control
- Contamination
- Fueling/servicing

Technical risk factors:

- Technical risk
- Time period
- Priority

The results of the panel assessment are summarized in figure 4 and show an emphasis on large-antenna-pointing propulsion technology.

The panel concluded that the future utilization of large structures will be mainly for surveillance, communications, and defense missions. The systems associated with these missions are large antennas and platforms. The propulsion needs associated with these systems are primary propulsion for placement and maneuvering and secondary propulsion for pointing. It is also recommended that future propulsion technology include a concern about contamination and provide for orbital servicing, including refueling.

FIGURE 1
APPROACH

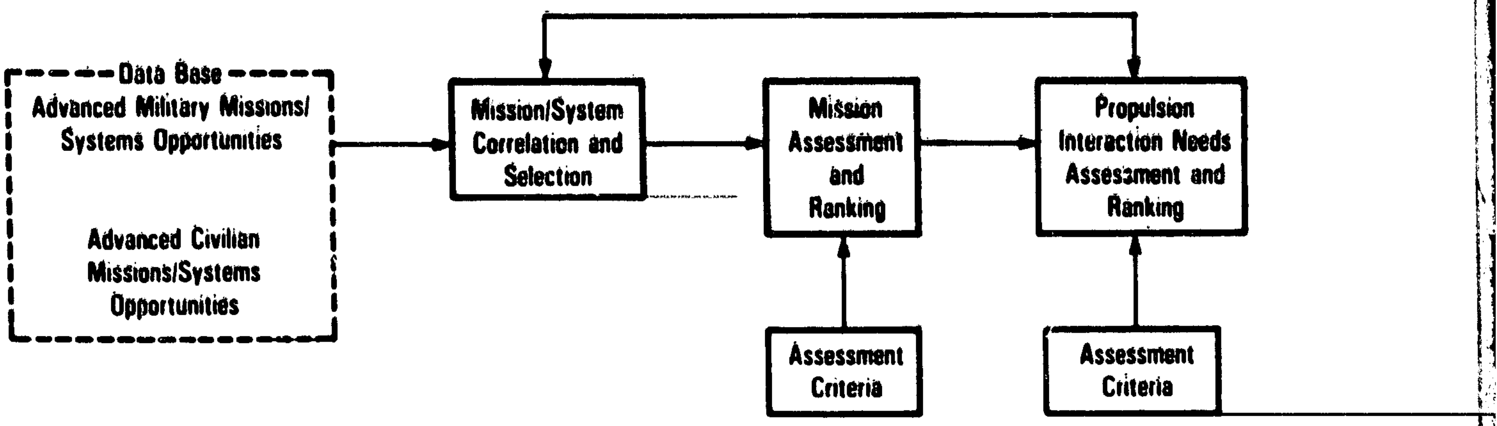

FIGURE 2
MISSION/SYSTEM CORRELATION SUMMARY

MISSIONS	SYSTEMS
O SPACE SCIENCE	LARGE ANTENNA, OPTICAL SYSTEMS, PLANETARY STATION
O COMMUNICATIONS	LARGE MULTIPLE ANTENNA
O COMMAND AND CONTROL	COMMAND POST
O SURVEILLANCE	RADAR, IR
O TERRESTRIAL SUPPORT	LARGE MULTIPLE ANTENNA
O ORBITAL SUPPORT	POWER STATION, SPACE STATION*, PLATFORM†
O DEFENSE	LASER (LOW POWER), LASER (HIGH POWER)

* SPACE STATION: MANNED, INCLUDING FUEL DEPOT AND MAINTENANCE FACILITIES
† PLATFORM: UNMANNED, SENSORS, COMMUNICATIONS, LARGE STRUCTURE

FIGURE 3
MISSION RANKING SUMMARY

MISSIONS*	PRIORITY†
1- SURVEILLANCE	1
2- COMMUNICATIONS	1
3- DEFENSE	1/2
4- COMMAND AND CONTROL	2
5- ORBITAL SUPPORT	2
6- TERRESTRIAL SUPPORT	2/3
7- SPACE SCIENCE	2/3

* RESEARCH AND DEVELOPMENT WAS CONSIDERED NECESSARY TO ALL OF THE MISSIONS RATHER THAN A SPECIFIC MISSION

† RANKING SYSTEM; 1, 2, 3 WHERE 1 IS THE HIGHEST

FIGURE 4
ASSESSMENT OF PROPULSION RISKS AND NEEDS

Evaluation Factors

Mission	System	Technical Risk 1 = High	Time Period 1 = Near-Term	Priority 1 = High	Weight (klbs)	Orbit 1 = High Orbit	On-Orbit Maneuvering 1 = High	Stationkeeping 1 = High	Pointing 1 = High	Acceleration Limit 1 = High	Figure Control 1 = High	Contamination 1 = High	Fueling Servicing 1 = High
	Large Antenna	2	2/3	2/3	15-40	1/2	3	2	1/2	1/3	3	3	3
Space Science	Optical Systems	1/2	2	2	15-40	1/2	3	1	1	2	2	1	2
	Planetary Station	1	3	3	25-50	1	3	1	1	3	3	3	1
Communications	Large/Multiple Antenna	2	2	$\boxed{1}$	20-30	1/2	3	2	$\boxed{1}$	1/3	1/2	3	3
Command and Control	Command Post	2	2/3	2/3	40-60	1/3	3	2	2	2	3	3	1
Surveillance	Radar	2	1/2	$\boxed{1}$	10-30	2/3	3	2	$\boxed{1}$	1/3	2	3	3
	IR	2	2	2	10-20	1	3	3	3	3	3	1	3
Terrestrial Support	Large/Multiple Antenna	2	2	$\boxed{1}$	20-30	1/2	3	2	$\boxed{1}$	1/3	1/2	3	3
Orbital Support	Power Station	1	2/3	3	100+	1	3	2/3	1/2	1/3	3	3	2
	Space Station*	2	1/2	2/3	100-150	3	3	2	2	3	3	2	1
	Platform†	1/2	1/2	$\boxed{1/2}$	20-40	1/3	3	2	$\boxed{3}$	$\boxed{1/3}$	3	$\boxed{1/3}$	$\boxed{1}$
Defense	Laser 1, (Low Power)	2	2	2	50-100	1/2/3	$\boxed{1}$	2	$\boxed{1}$	3	3	$\boxed{1}$	$\boxed{1}$
	Laser 2, (High Power)	1	3	2	125-150	1/2	3	2	$\boxed{1}$	3	3	$\boxed{1}$	$\boxed{1}$

*Space station manned. Includes fuel depot and maintenance facility.

†Platform unmanned. Large structure for sensor/communication.

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CRYOGENIC ORBIT TRANSFER VEHICLE

W. J. KETCHUM

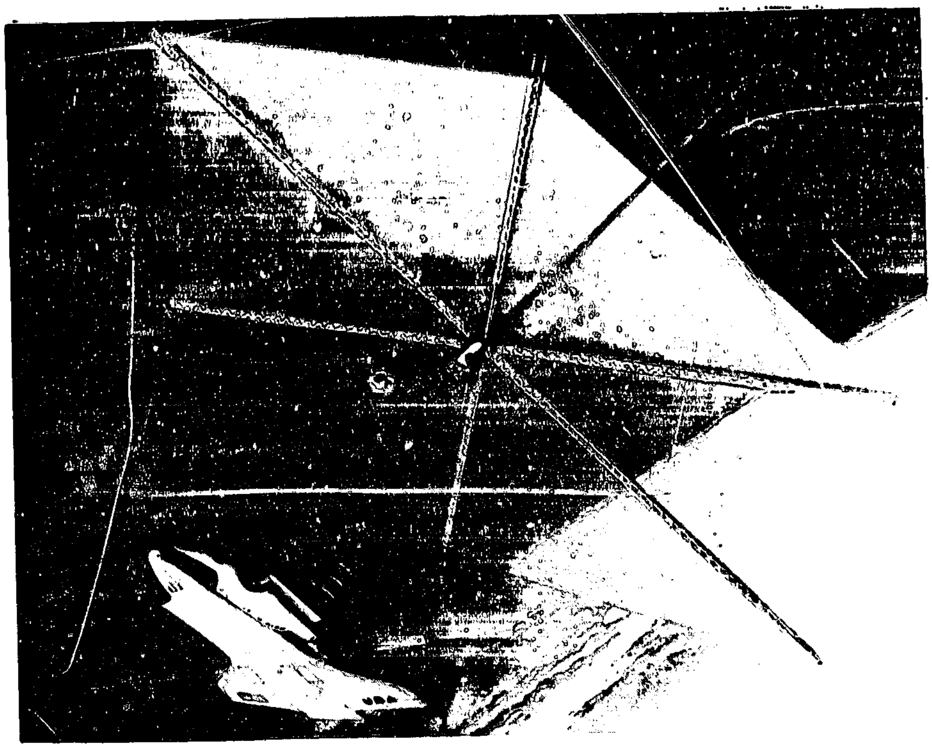
GENERAL DYNAMICS
Convair Division
P.O. Box 80847, San Diego, California 92138

GDC LSS/PROPULSION STUDIES
GDC LOW THRUST VEHICLE STUDIES

1977-78	AF-SAMSO	–	ON-ORBIT-ASSEMBLY DESIGN STUDY
1979-80	NASA/MSFC	–	LOW THRUST VEHICLE CONCEPT STUDY
1979-81	NASA/MSFC	–	GEOSTATIONARY PLATFORM STUDY
1980-81	NASA/LeRC	–	LOW THRUST CHEMICAL PROPULSION SYSTEM PROPELLANT EXPULSION & THERMAL CONDITIONING STUDY
1981-82	AFRPL/RI	–	LARGE SPACE SYSTEM CRYOGENIC DEPLOYMENT SYSTEM

LARGE SPACE SYSTEM
(OOA SBR)

A typical Large Space System shows GDC's concept for a space based radar from our On-Orbit-Assembly design study for the Air Force. GDC has continued to define the structures, and transfer vehicles for such systems.



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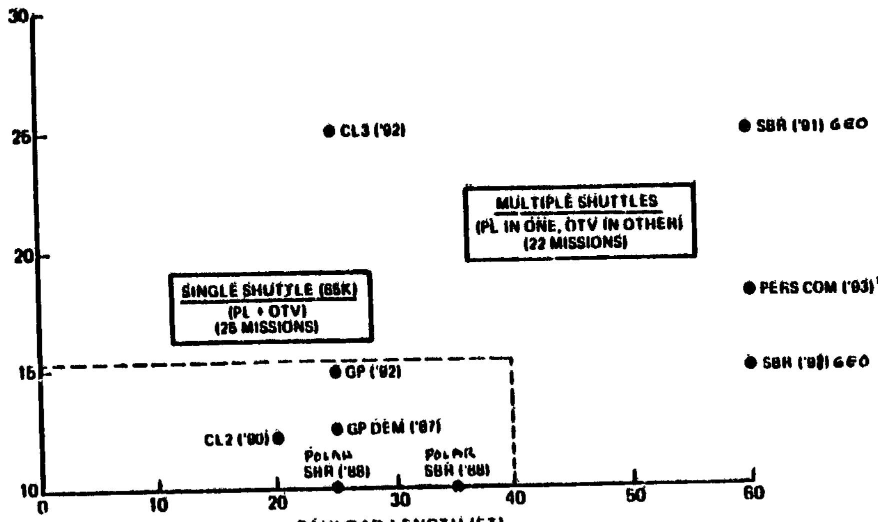
25 OF 47 LSS MISSIONS PLANNED CAN BE ACCOMPLISHED WITH SINGLE STS LAUNCHES



* 64,000 LB (3 PARTS)

A SHUTTLE OPTIMIZED DESIGN GREATLY REDUCES TRANSPORTATION COSTS FOR LSS MISSIONS

GDC studies for NASA MSFC (Geostationary Platform) have determined that a shuttle optimized design (LSS & Transfer Vehicle in one shuttle flight) greatly reduces transportation costs and minimizes orbital operations. Careful attention to design has resulted in efficient payload packaging. A minimum volume, high-energy (LO_2/LH_2) transfer vehicle allows maximum volume for the payload in the STS orbiter cargo bay. Orbiter supported deployment and checkout of the spacecraft insures mission success. Once functioning, the spacecraft is transferred to its operational orbit by the low thrust vehicle.

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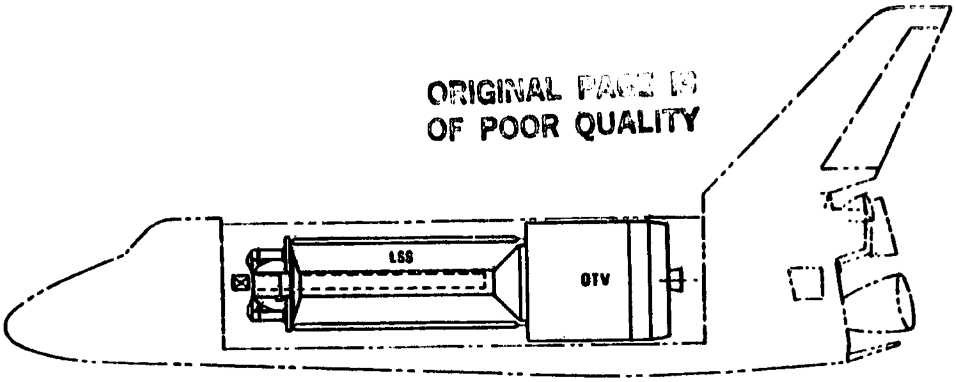
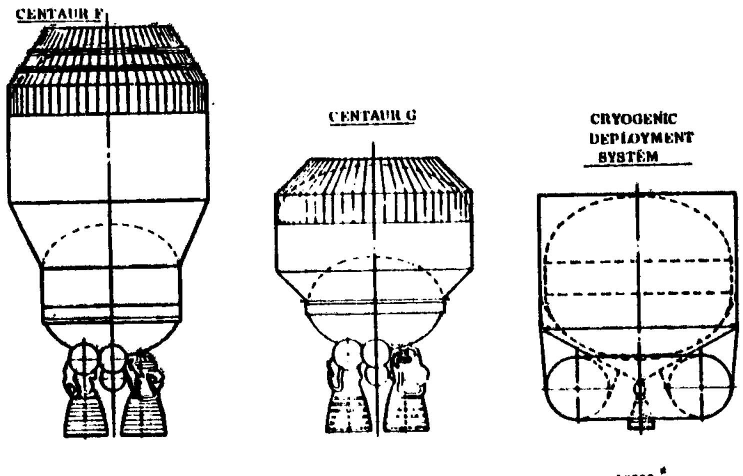


- SINGLE SHUTTLE FLIGHT TO MAXIMIZE ECONOMY AND MINIMIZE ORBITAL OPERATIONS
- LH_2/LO_2 OTV TO MAXIMIZE PAYLOAD
- MINIMUM VOLUME OTV TO MAXIMIZE PACKAGED SPACECRAFT
- ORBITER SUPPORTED DEPLOYMENT AND CHECKOUT OF SPACECRAFT
- LOW ACCELERATION TRANSFER TO MINIMIZE LOADS ON DEPLOYED SPACECRAFT

LOW THRUST LO_2/LH_2 TRANSFER VEHICLES PROVIDE MAXIMUM PERFORMANCE

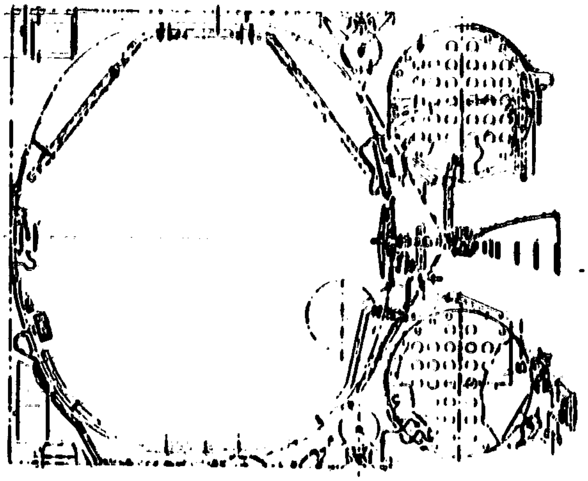
Candidate high-energy low thrust LO_2/LH_2 transfer vehicles include the Centaur F & G, and a more optimized design using a new low thrust engine and toroidal tanks. Geo payload capability up to 16000 lb and 40' long (packaged) payloads are possible with a single 65K STS.



GEO PAYLOAD, LB	12800*	6800	16000*
PAYLOAD LENGTH, FT	30	40	40
THRUST, LB	3000	3000	800
Isp, SEC	430	430	465
IOC	1985	-	-

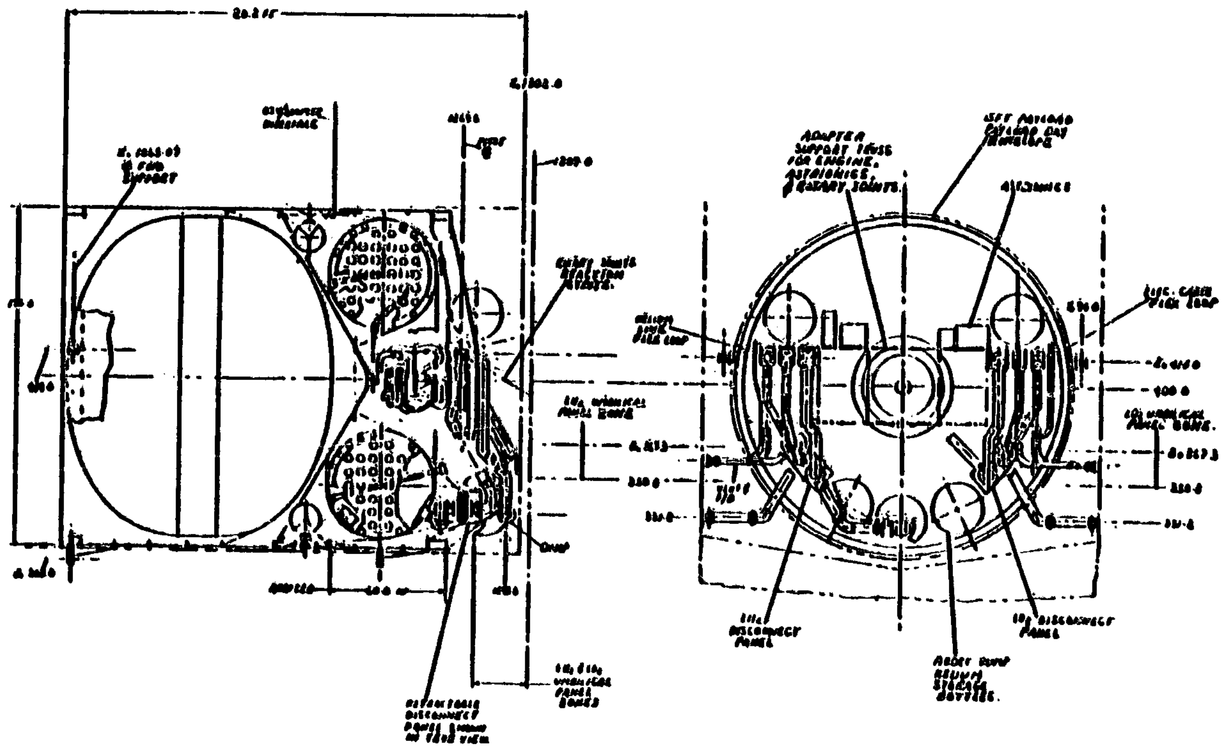
*65K STS

AN OPTIMUM PROPULSION SYSTEM IS BEING DEFINED FOR LSS



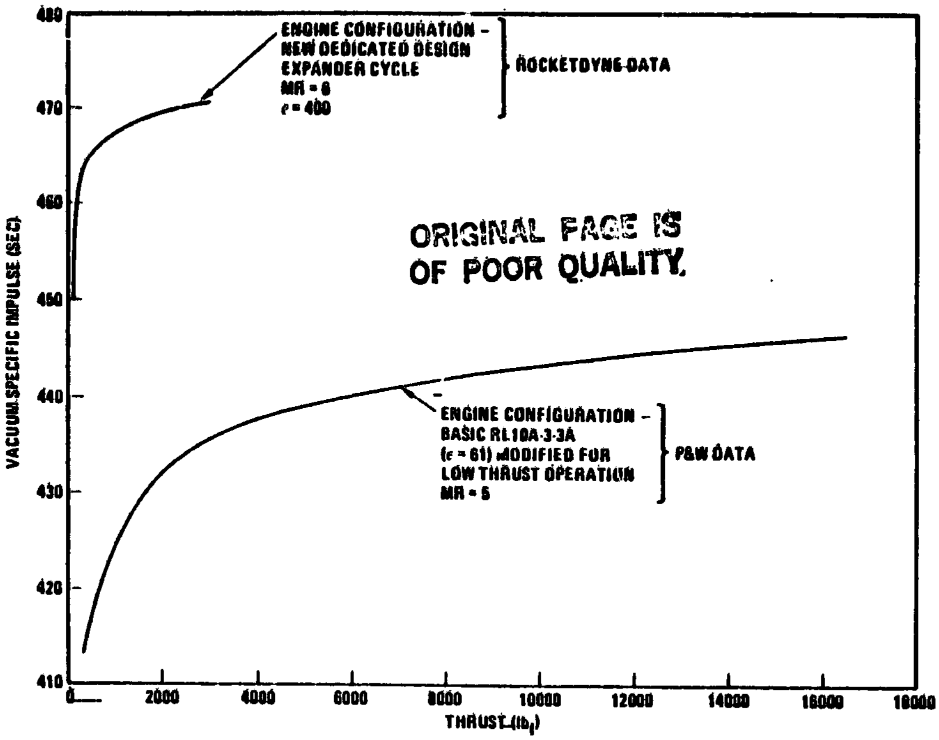
- PUMP FED (.5K) ENGINE
  - ENGINE MOUNTED/DRIVEN PUMPS (NO VEHICLE – MOUNTED BOOST PUMPS)
  - 17 PSIA MIN INLET PRESSURE
  - NPSH
    - LO_2 – 1 PSI
    - LH_2 – 0.8 PSI
  - AUTOGENOUS H_2 BLEED
- COMPOSITE STRUCTURE
- ALUMINUM TANKS
- PROPELLANT ACQUISITION
  - PARTIAL SETTLING
  - SCREENS
- MLI TANK INSULATION (15 LAYERS)
- PRESSURIZATION
  - HELIUM PRE PRESS; O_2 RUN
  - AUTOGENOUS H_2 RUN
- ZERO G VENT/MIXER
- FILL AND DRAIN
- 250 SEC ABORT DUMP
- N_2H_4 ATTITUDE CONTROL
- FUEL CELL POWER (1 KW)

SHUTTLE INTERFACES ARE BEING DEFINED FOR LO_2/LH_2 VEHICLES



A NEW LOW THRUST LO_2/LH_2 ENGINE OFFERS A 40-50 SECOND I_{sp} INCREASE (2200-2800 LB PAYLOAD)

Although the RL-10 engines can run at low thrust, a new low thrust engine offers a 40-50 sec. I_{sp} increase over the RL 10, resulting in an additional payload capability of 2200-2800 lb. Added savings in engine weight and vehicle subsystems make this even more attractive.



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AN OPTIMUM PROPULSION SYSTEM FOR LSS REQUIRES TECHNOLOGY DEVELOPMENT

An optimum propulsion system for LSS requires technology development of the engine, feed systems, and tankage. A 500 lb_f LO_2/LH_2 engine will have to run up to 10 hours, and re-start 10-20 times. Innovations in propellant feed systems at low flow rates and low accelerations are needed for minimizing weight. The torus LO_2 tank allows a very short stage, but requires careful design to minimize weight and residuals.

- ▲ LOW THRUST/HIGH PERFORMANCE ENGINE (> 465 I_{sp})
  - SMALL PUMPS
  - COOLING
  - LONG BURN TIMES
- ▲ LOW ACCELERATION CRYOGENIC PROPELLANT FEED
  - ACQUISITION
  - PRESSURIZATION
  - PU
- ▲ TORUS LO_2 TANK
  - DESIGN (SHAPE, SUPPORT, MFG, COMPONENTS INSTL, ASSY)
  - SLOSHING (C.G., BAFFLES)
  - RESIDUALS (OFFSET C.G., ACQUISITION SYST)
  - INSULATION

SUBSYSTEM ALTERNATIVE TECHNOLOGIES ARE BEING INVESTIGATED

All of these subsystems are being investigated in detail in the AFRPL "Large Space System Cryogenic Deployment System Study." Alternatives are being traded to maximize performance. Results of this study will provide directions for technology development of the engine and vehicle propulsion subsystems.

Subsystem	Alternatives to be Traded
Engine thrust/throttling	Baseline thrust level (500 lbf) Increased thrust levels Constant acceleration vs constant thrust
Pressurization system	Thermal subcoolers Autogeneous pressurization with helium for start-up Autogenous pressurization with sub-coolers for start-up Helium pressurization
Propellant acquisition	Settle propellants Partial capillary acquisition device Total capillary acquisition device
Tank venting	Settled propellants with conventional venting Zero-g thermodynamic vent system
Propellant utilization	Use PU system No PU system (save added complexity & cost)
Propellant gaging	Capacitance Point sensors Nucleonic RF gaging Accurate flowmeter
Abort dump	Optimize sum of pressurant plus line & valve sizes
Propellant mixture ratio	Optimize tank weight & length (coordinate with engine performance)
Tank configuration	Trade LH_2 tank bulkhead shapes Trade LO_2 torus cross-section shape
Tank insulation	Multilayer insulation, vary number of layers Spray-on foam insulation, vary thickness
Tank pressure	Maximum tank pressure vs skin thickness

Trade Study Constraints

- Baseline vehicle definition
- Baseline mission definition
- Shuttle payload constraints
- Shuttle operational constraints

Evaluation Criteria

- Reliability
- Simplicity
- Life cycle costs
- Performance
- Weight & length

Identify beneficial alternatives

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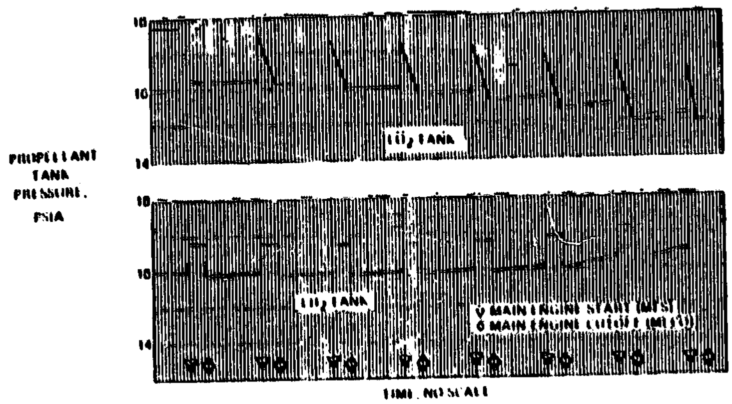
ANALYSIS INDICATES THAT NO VENTING WILL OCCUR

Analysis indicates that no venting of either propellant tank may be required for optimized low thrust LO_2/LH_2 transfer vehicles.



- LO_2 TANK PRESSURIZED WITH HELIUM FOR ENGINE START AND ENGINE BURN
- LH_2 TANK PRESSURIZED WITH HELIUM FOR ENGINE START, AUTOGENOUS PRESSURIZATION FOR ENGINE BURN
- ENGINE NPSP REQUIREMENT
  - 1.0 PSI LO_2
  - 0.6 PSI LH_2

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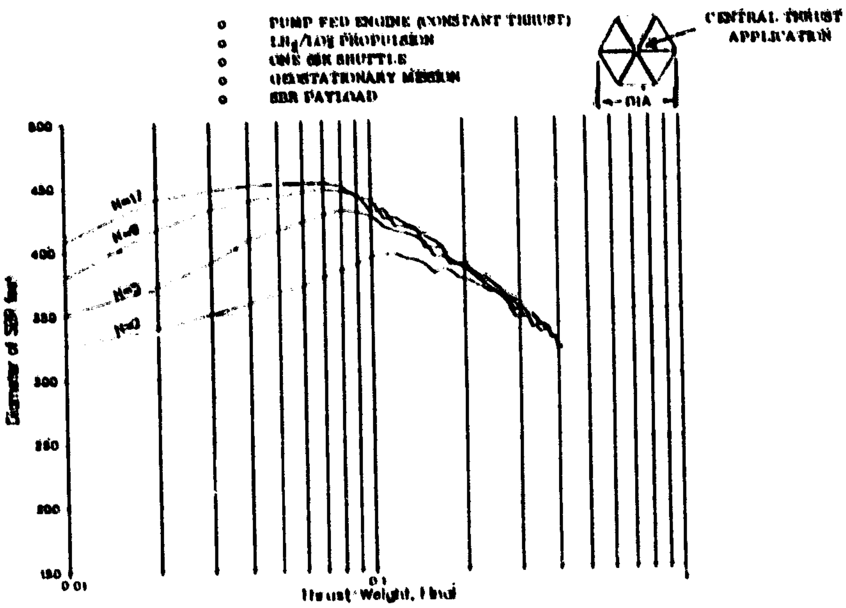
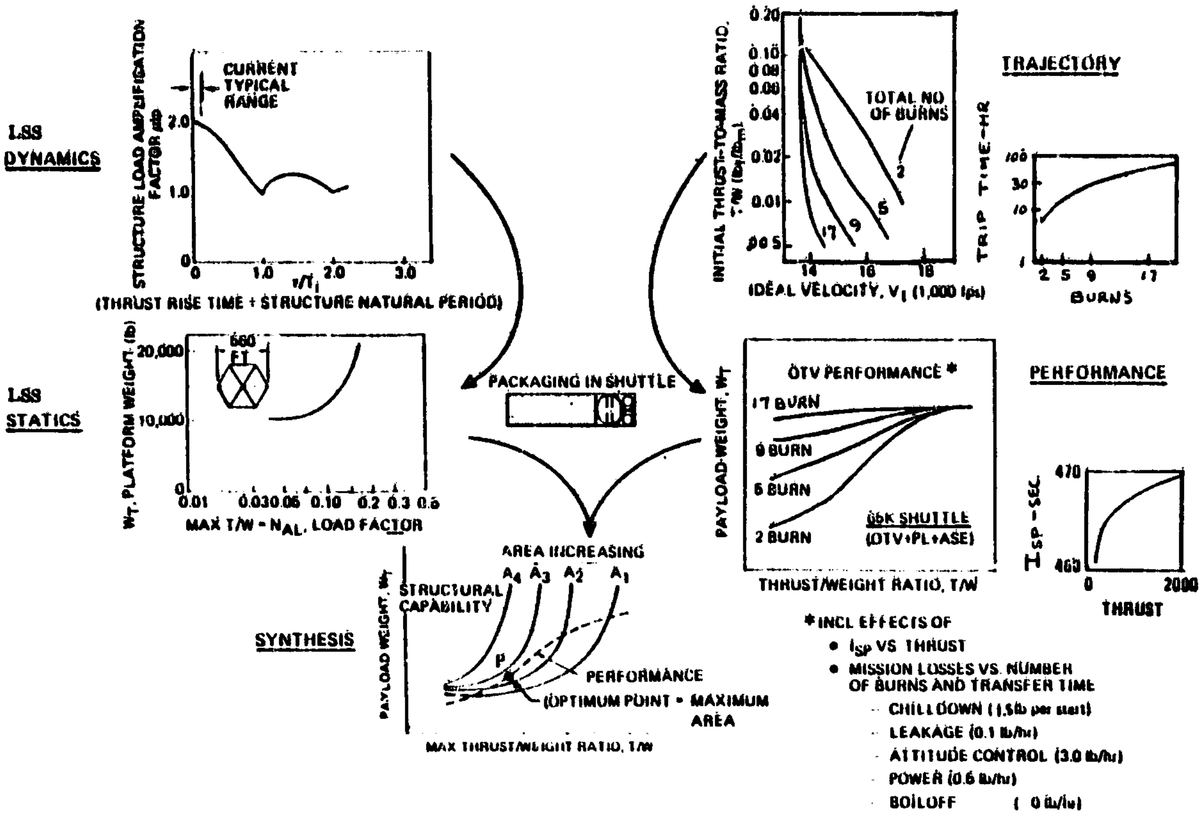
GDC'S OPT OTV PROGRAM EVALUATES LSS/PROPULSION INTERACTIONS

GDC has continued to develop the "OPT OTV" computer program to evaluate LSS/propulsion interactions. The effect of thrust-to-weight and the number of engine burns to obtain the optimum payload is shown. Recent Δ V data from Stanford has been incorporated for more burns (17). Results of recent studies on engine and vehicle performance (I_{sp} and propellant losses) have also been incorporated.



OPT OTV COMPUTER PROGRAM

The GDC "OPT OTV" Computer program considers all factors involved in LSS/propulsion optimization.



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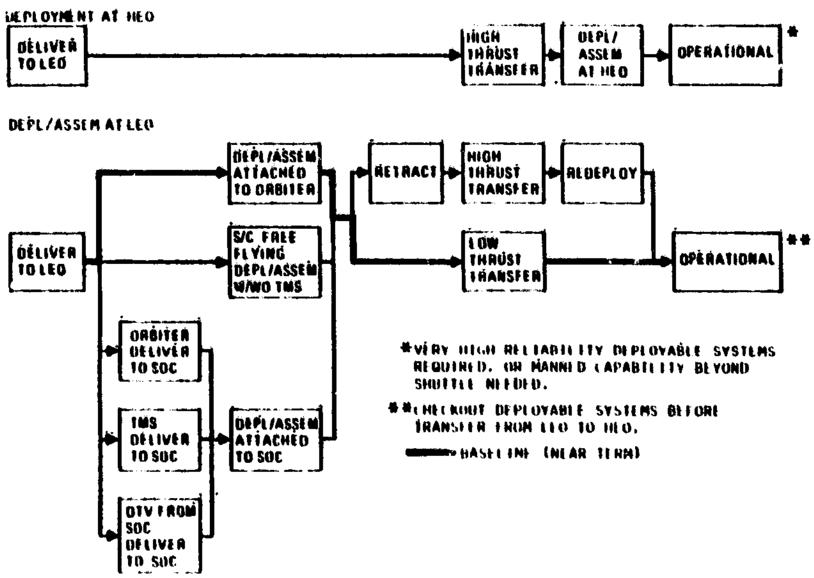
FUTURE STUDIES SHULD CONSIDER ALTERNATE CONSTRUCTION/TRANSPORTATION OPTIONS



FUTURE REQUIREMENTS FOR ORBITAL TRANSFER

- VERY HIGH PERFORMANCE

  15-30,000 POUND PAYLOADS, SERVICING, AND ROUND TRIP

- LOW ACCELERATION

  MAXIMIZE SIZE OF LARGE SPACE STRUCTURES

- REUSABILITY

  REDUCE OPERATING COSTS AND/OR RETURN PAYLOADS TO SHUTTLE

- MANRATING

  ORBIT TRANSFER AND SORTIE FOR CREW MODULES

- AEROMANEUVERING

  PLANE CHANGES, BRAKING TO LOWER ORBIT AND/OR EMERGENCY REENTRY

- SPACEBASING

  FREE FROM SHUTTLE CONSTRAINTS

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Storable Orbit Transfer Vehicle

W. E. Pipes

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Future Air Force and NASA spacecraft will be launched by the Space Transportation System (STS). These missions include communications, meteorological, intelligence gathering, earth resources, and servicing. The satellites will be carried to low earth orbit on Shuttle and will need a vehicle to transport them to their final orbit. The present transfer vehicles (IUS, SSUS, and MMS) have limited delivery capability and cannot meet the needs of the future.

Under Air Force contract we recently completed the development of a mission model for NASA and DOD out to the year 2000. The model contains over 600 spacecraft deliveries including Large Space Systems (LSS). To evaluate the Orbit Transfer Vehicles (OTV) against mission model requirements the model was separated into five categories based on energy required, g-level, spacecraft deploy/return, operational constraints, mission duration, and packaging. Propulsion systems compatible with each mission category were then selected for evaluation. A total of 41 candidates were selected which included storable propellants, cryogenic propellants, and electric propulsion. For the LSS mission category a short LO_2/LH_2 system with torus LO_2 was selected over the storable N_2O_4/MMH OTV based on economics. The lower Life Cycle Cost (LCC) for cryogenics resulted from the very high energy requirements of the proposed future missions. This resulted in increased Shuttle flights for the lower energy storable OTV.

To present the benefits of a storable OTV and its relation to LSS we must first look beyond the LSS missions and assess the high energy missions being flown today. These missions include all present flights to geosynchronous (GEO) and those projected that are 10,000 lbs or less in weight. For these missions (Category II) the storable OTV was the lowest cost and had substantial economic advantage over a cryogenic OTV or the Inertial Upper Stage (IUS). The primary reason is the importance of length. For example, one of the included charts shows a 20 ft long OTV with 12,800 lb delivery to GEO has the same mission capture capability as a 15 ft long OTV with only 7,700 lb delivery capability. Storable OTV's that exhibit this capability include the STS Transtage with over 8,000 lb delivery to GEO and an advanced storable with over 12,000 lb delivery to GEO. Figures showing these vehicles are included. The Transtage was developed during the 1960s and has a demonstrated reliability of 96 percent in its 26 operational missions. For the STS Transtage only mandatory changes have been incorporated to meet program requirements as well as NHB 1700.7A, Safety Policy and Requirements for Payloads Using the STS and ICD-2-19001, Shuttle Orbiter/Cargo Standard Interfaces.

Comparing the delivery capability of the storable OTV to the mission requirements show the lower energy LSS missions can be accommodated very effectively. The storable OTV requirements for LSS mission requirements are presented for both a stowed LSS during transfer and a deployed LSS during transfer. Current studies for some of the lower energy LSS missions are looking very seriously at stowed LSS transfer with a storable OTV. The most significant difference in propulsion requirements for stowed versus deployed LSS transfer is thrust level, number of burns, and gimbal angle. The thrust level is driven by the low g requirement. The number of burns is driven by performance, where multiple perigee burns can reduce the delta velocity to GEO for low thrust from approximately 16,900 to 14,600 fps. The gimbal angle is higher for deployed LSS transfer because the c.g. is not well known prior to deployment since the LSS with less than one g capability can not be deployed in a one g ground environment.

In conclusion the storable OTV is the most cost effective vehicle for current missions. It can also meet low energy LSS requirements. The cryogenic OTV can best meet future high energy missions such as manned missions at GEO. For large traffic of high energy LSS missions it is more cost effective than the storable OTV. Electric propulsion has the potential for economic advantages over the storable or cryogenic OTV when delivery transfer time is not critical and the energy requirements are very large.

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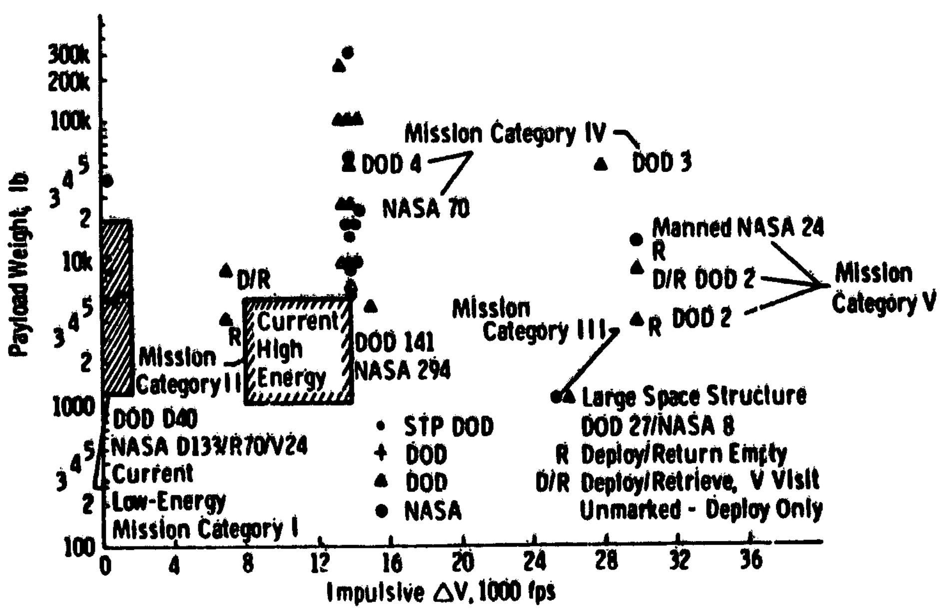
Mission Spectrum Including Future Generic



Propulsion Concept Selection Matrix

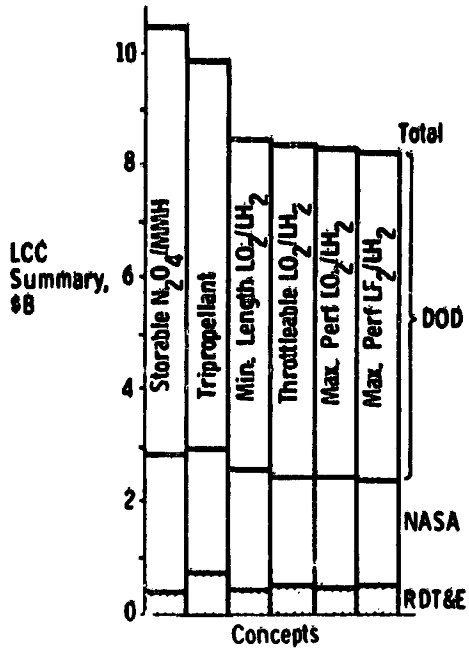
Mission Category	I	II	III	IV	V
Mission Assessment - Energy - g-level - Deploy/Retrieve - Operations/Duration - Packaging	Low Energy - Deploy/Retrieve - $\Delta V < 8K$ fps	High Energy - Deliver to GEO - Wt < 10K lb	LSS* - Wt 6K-300K lb - g-level 0.05-1.0	Heavy Singles - Wt >20K lb - $\Delta V > 14K$ fps	Deploy/Retrieve to GEO - $\Delta V > 28K$ fps
	Single Shuttle Launches		Multishuttle Launches		
- Propulsion - Drivers	- Packaging - Flexibility	- Performance - Packaging	- Performance - g-level - Some Packaging	- Performance - Duration	- Performance - Flexibility - Duration
Electric (5)	- Flexibility - Mission Time	- Transfer Time 30 Days	- 1 Baseline Hg Ion (Solar Power) - 3 Large Inert Gas (Solar Power) - 1 MPD (Nuclear Power) - All Concepts Considered S/C Power		- Transfer Time - Flexibility
Solid (1980 Technology) (2)	- Flexibility - g-level	1 Baseline-IUS - SSUS-PAM A PAM D	- g-level	- Performance	- Performance

Propulsion Concept Selection Matrix (concl)

Mission Category	I	II	III	IV	V
Mission Assess. - Energy - g-level - Deploy/Retrieve - Operations/Duration - Packaging	- Deploy/Retrieve - $\Delta V < 8K$ fps	High Energy - Deliver to GEO - Wt < 10K lb	LSS* - Wt 6K-300K lb - g-level 0.05-1.0	Heavy Single - Wt > 20K lb - $\Delta V \geq 14K$ fps	Deploy/Retrieve to GEO - $\Delta V > 28K$ fps
	Single Shuttle Launches		Multishuttle Launches		
Propulsion Drivers	- Packaging - Flexibility	- Performance - Packaging	- Performance - g-level Some Packaging	- Performance - Duration	- Performance - Flexibility - Duration
Storable - N_2O_4/MMH - ClF_3/B_5H_9 - Triprop (15)	- 1 Baseline MMS - 2 N_2O_4/MMH - 1 ClF_3/B_5H_9	- 1 Baseline - 2 Packaging - 1 Propellant - 1 Reusable - 2 Two Stage	- 1 Baseline - 1 Tripropellant	- 1 Propellant (Alternative - Approaches on Mission 12)	- Not Competitive
Cryogenic - LO_2/LH_2 - LF_2/N_2H_4 - LF_2/LH_2 (16)	- Orbit life - Cost - Packaging	- 1 Baseline - 2 Packaging (IMR) - 2 Propellants - 2 Reusable	- 1 Baseline - 1 Packaging - 1 Propellant - 1 Throttleable Engine	- 1 Baseline - 1 Propellant - 1 Cryo/Storable on Mission 12	- 1 Baseline - 1 Propellant (multitank)
Combination - Liquid/Solid	- 1 N_2O_4/MMH/Solid	- 1 Biprop/Solid - 1 S/C Assembly	- g-level	- Performance	- Performance

*LEO vs GEO Assembly

LSS Conclusions for Liquid Chemical Vehicles



- Low Thrust and High Performance Required
- LCC of Cryogenic Stages Lower Than Storable Combinations
- LCC of All Cryogenic Stages Nearly the Same
- No LCC Advantage for Tripropellant or Advanced Propellants
- No LCC Advantage for Throttleable Engine
- No LCC Advantage or Penalty for Short Stage (Note LSS Mission Definitions)
- DOD Results Unchanged by NASA

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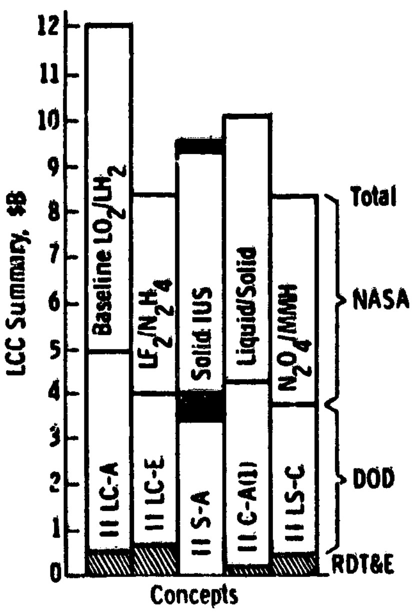
High-Energy Missions - Conclusions (Category II)



Expendable Vehicles

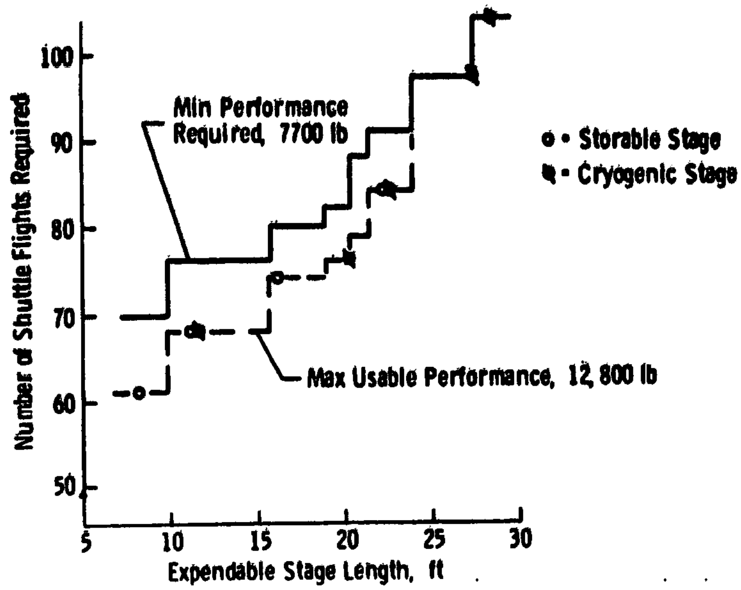
No Advantage for Advanced Propellants

Lowest LCC Is N_2O_4/MMH

DOD Results Unchanged by NASA

Estimated IUS Transportation Cost for 12 Missions Not Captured

Category II - Effect of Stage Length and Performance on DOD Mission Capture



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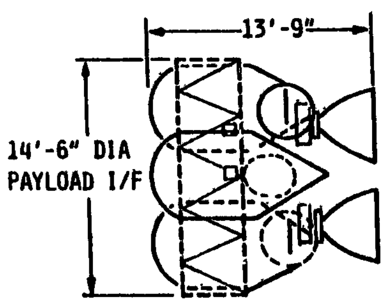
STS TRANSTAGE CONFIGURATION IMPROVED



VEHICLE MODS

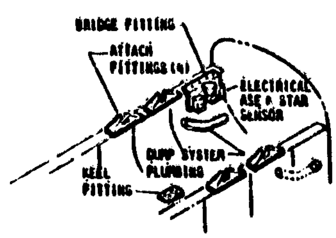
- 4 OXIDIZER TANKS (SHORTENED)
- MIXTURE RATIO 1.65
- INCREASE STRUCTURE SHELL DIA
- REPLACE HELIUM SPHERES (OMS)
- ADD SGLS TT & C (IUS)
- ADD PROPELLANT UTILIZATION
- ADD ADDITIONAL ABLATIVE LINER FOR ENGINES
- REVISE EXISTING IGS-S/W

ORBITER INTERFACE

- ADD STAR TRACKER AND ASE
- DIRECT RAIL ATTACHMENTS (NO CRADLE)
- UTILIZE OMS KIT DUMP PROVISIONS
- RMS FOR DEPLOYMENT

PERFORMANCE OPTION

- 100:1 ENGINE NOZZLE



SATISFIES PLANETARY MISSION REQUIREMENTS WITH ΔV-EGA MISSION DESIGN

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ASDS Minimum Tank Length-Multiengine N_2O_4/MMH Stage for Missions in Category II

ASDS Vehicle Definition

Propulsion System: Storable, Code: II LS-C

Propellant: N_2O_4/MMH, MR: 2.4

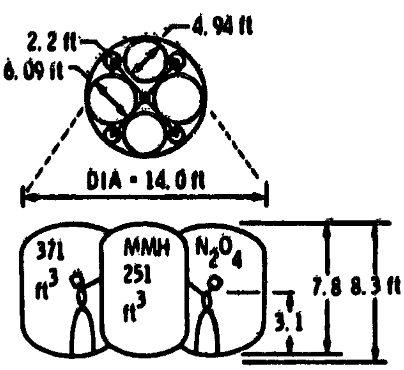
Main Engine: Thrust 15000 lb, I_{sp}: 341.0 sec

Payload Delivery to GEO 12759 lbs, ASE 3000 lb

Item	Weight, lbs		Remarks
Avionics		736	•
Structures		911	•
Thermal		149	•
Propulsion		1260	•
FU Tank	121		AL (2219-T87)
OX Tank	205		SF 1.5, NOF 30%
Propellant Feed	367		
Pressurization	161		
Engine Assy (4 Engines)	406		
8-lb Nozzle Actuator Each			(ALRC) with Actuators
ACPS		359	•
Subtotal		3415	
Contingency 10%		342	
Dry Weight		3757	
Propellant, Nonusable		423	• 0.5% Outage
Burnout Weight		4180	
Vented Propellant		104	2 Burns/4 Engines
Propellant Loaded		45224	
Stage Initial Wt		48981	Max Shuttle Capacity
Propellant, Usable		44697	346 ACS
Mass Fraction		.905	Usable/Initial

• Typical to Tabulated Baseline Storable Stage

Configuration Sketch



Notes:

Engine: 4 Singles @ 3750 lb_f Constant Thrust ϵ = 400:1, Pc = 1430 psi, 60% Retracted Nozzle, 92.5% Eff.

Mission: LEO-GEO, Burns 2, ΔV = 14000 fps, Launch to Burn 1 day, 6 hr transfer time. P/L Reserve Performance 2%, ACPS Propellant Reserve 10%, Propellant Outage 0.5% Regulated Helium, Ambient Storage

Storable OTV LSS Mission Requirements

Requirement	Stowed Transfer	Deployed Transfer	Remarks
Transfer Time (nom)	6 hours	31 hours	8 Perigee Burns (Low Thrust)
Number of Burns (max)	2	9	8 Perigee Burns (Low Thrust)
Launch to First Burn (max)	1 Day	7 Days	STS Capability (7 Days)
Coast Time Outside Orbiter	1 Day	7 Days	STS Capability (7 Days)
Operational Altitude	LEO to HEO	LEO to HEO	High Earth Orbit
Burn Duration (nom)	20 min	9 hours	45,000 lb of Propellant
g-Level (max) Final	3.0 g	0.05/1.0 g	Deployed LSS g-Range
Thrust (max) Final	15,000 lbf	500 lbf	500 lbf (10,000 S/C + Stage)
Gimbal Angle	± 6 deg	± 10 deg	Deployed LSS cg Uncertainty
Vehicle Length (max)	14 ft	14 ft	Based on 42-ft S/C
Safety	NHB 1700.7	NHB 1700.7	Propellant Dump for Large Stages

Conclusions

Storable OTV

- Most Cost Effective Approach for Current Missions
- Can Meet Low-Energy LSS Requirements

Cryogenic OTV

- Best Approach for Future High-Energy Missions, i.e., Manned Mission at GEO
- Cost Effective for Large Quantities of High-Energy LSS Missions

Electric Propulsion

- Can Be Economical for Very High Energy Mission Requirements

ELECTRIC PROPULSION: SYNERGY OF ORBIT TRANSFER AND MAINTENANCE

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TRW
DEFENSE AND SPACE SYSTEMS GROUP

Electric propulsion has the dual capability of transferring large payload mass from LEO to GEO, and providing accurate on-orbit stationkeeping with reduced mass compared to chemical auxiliary propulsion. Integrated propulsion studies have investigated orbit boosting, inclination change, attitude control, stationkeeping, relocation, disposal, and power sharing on-orbit with electric propulsion systems. A solar array pointing strategy has been developed to minimize the effects of atmospheric drag at low altitude, enabling electric propulsion to initiate orbit transfer at Shuttle's maximum cargo carrying altitude. Attitude control requirements are generally met by gimballing the primary orbit transfer thrusters, and using electric auxiliary propulsion thrusters on-orbit. Chemical auxiliary propulsion thrusters only provide a backup function during the mission. Gravity gradient torque is used during ascent to sustain the spacecraft roll maneuvers required for maximum solar array illumination. Stationkeeping is the most demanding on-orbit propulsion requirement. At area densities above 0.5 m^2/kg, East-West stationkeeping requirements from solar pressure exceed North-South requirements from gravitational forces.

Tradeoffs between integrated electric propulsion system mass ratio and transfer time from LEO to GEO were conducted parametrically for various thruster efficiency, specific impulse, and other propulsion parameters. A computer model was developed for performing orbit transfer calculations which included the effects of aerodynamic drag, radiation degradation, and occultation. The electric propulsion system mass included power source, power conditioning and controls, thrusters, gimbals, propellant, propellant storage and distribution system, thermal control, and structure. Significant residual on-orbit power (generally greater than 50 percent) was found to be available for payload utilization. The tradeoff results showed that thruster technology areas for integrated propulsion should be directed towards improving primary thruster efficiency in the range from 1500 to 2500 seconds, and be continued towards reducing specific mass. Comparison of auxiliary propulsion systems showed large total propellant mass savings with integrated electric auxiliary propulsion that results in a leveraged increase in net spacecraft mass.

SUMMARY

- ELECTRIC PROPULSION ADVANTAGES:
  - PUTS LARGE PAYLOAD MASS ON-ORBIT
  - PROVIDES ACCURATE STATIONKEEPING ON LONG LIFE SATELLITES WITH REDUCED MASS
  - PROVIDES ON-ORBIT POWER
- WITH FEATHERED ARRAYS, ELECTRIC PROPULSION INITIATES ORBIT TRANSFER AT SHUTTLE'S MAXIMUM PAYLOAD CARRYING ALTITUDE.
- ELECTRIC APS AND PRIMARY EPS THRUST VECTORING MEET MOST ATTITUDE CONTROL REQ'TS. CHEMICAL AUXILIARY THRUSTERS ONLY PROVIDE BACKUP FUNCTIONS.
- E-W STATIONKEEPING ΔV FROM SOLAR PRESSURE CAN BE GREATER THAN N-S ΔV REQ'D.
- GRAVITY GRADIENT TORQUE IS USED TO SUSTAIN S/C ROLL MANEUVERS FOR MAXIMUM SOLAR POWER DURING ASCENT.

GEO MISSION DEFINITION

NO.	$\left(\frac{M_{EPS}}{M_{PL}}\right)_T$	MISSION TIME (YEARS)
1	1	10
2	2	10
3	3	10
4	4	10
5	2	5

INITIAL ALTITUDE, H_o, 250 Km
FINAL OPERATION ALTITUDE, H_f, GEO
INITIAL INCLINATION, I_o, 28.5°
FINAL OPERATION INCLINATION, 0°
DISPOSAL ALTITUDE, H_d, 40786 Km

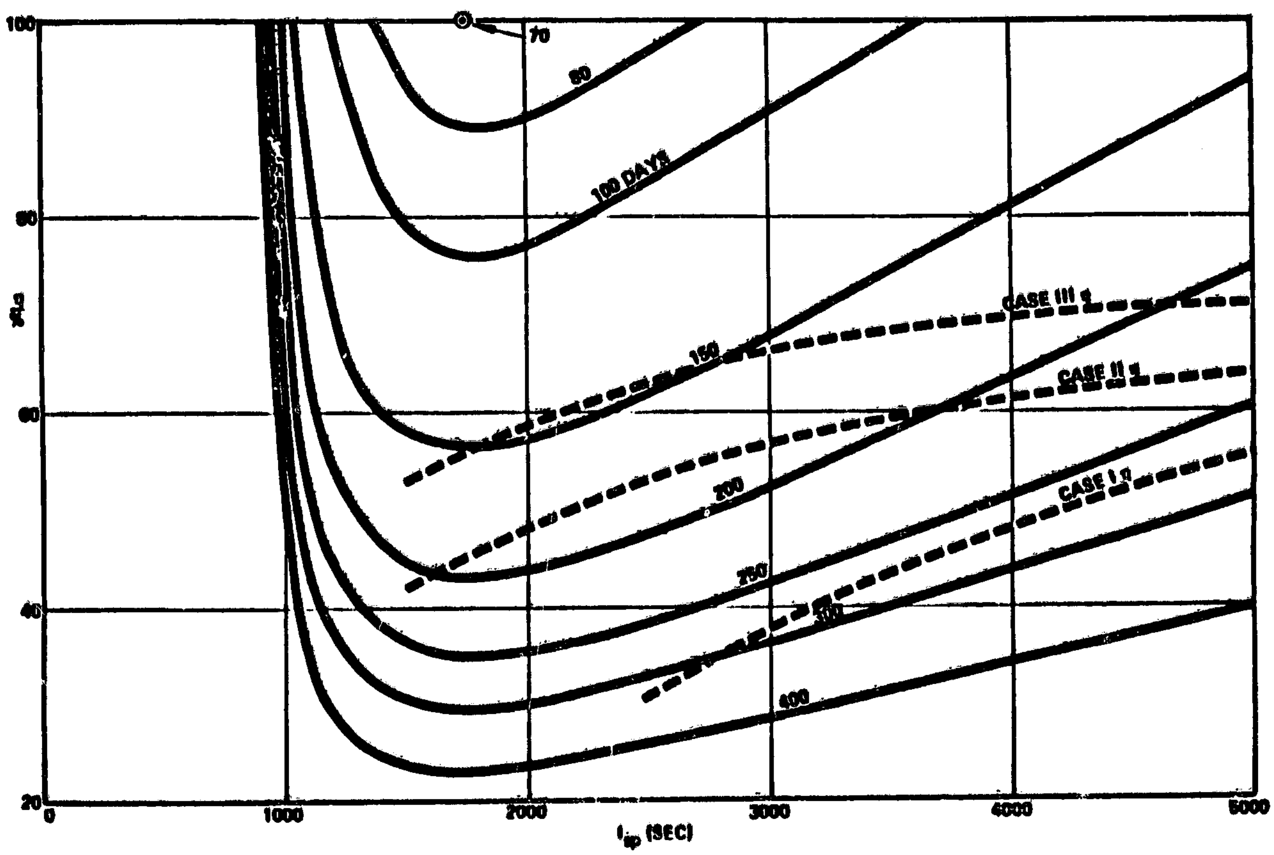
PROPULSION FUNCTIONS INVESTIGATED

- ORBIT BOOSTING
- INCLINATION CHANGE
- ATTITUDE CONTROL
- STATIONKEEPING
- RELOCATION
- DISPOSAL
- POWER SHARING ON-ORBIT

- ORBIT BOOSTING
  - SOLVED GRAVITATIONAL EQUATION
  - ADDED OCCULTATION
  - ADDED AERODYNAMIC DRAG
  - ADDED RADIATION DEGRADATION
  - DEVELOPED OPTIMUM ARRAY ILLUMINATION STRATEGY
- ALL OTHER FUNCTIONS
  - IMPLEMENTED ROCKET EQUATION
- I_{sp} FOR AUXILIARY PROPULSION SET AT 3000 SEC
- PLANE CHANGE INITIATED ABOVE RADIATION BELT

TRANSFER TIME TRADEOFF

$(M_{EPS}/M_{PL})_T = 2.0$



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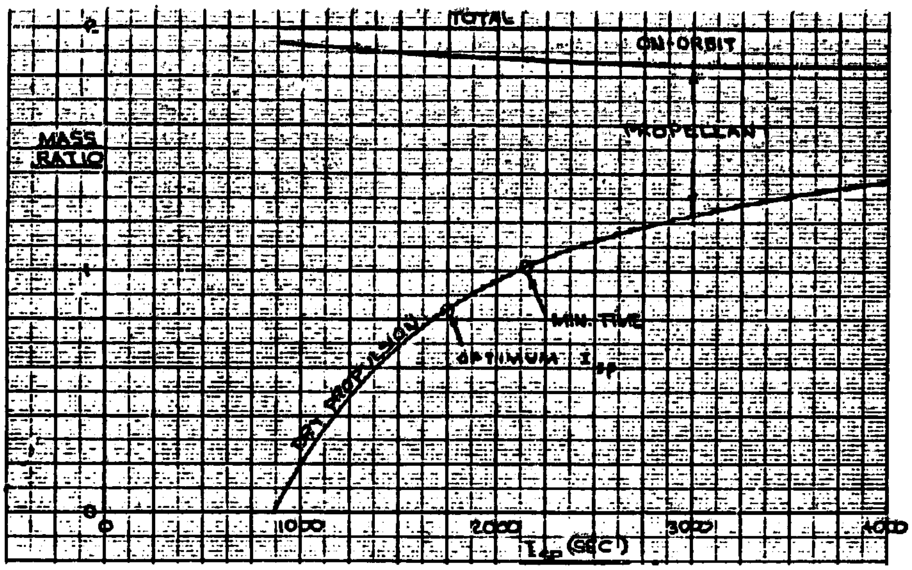
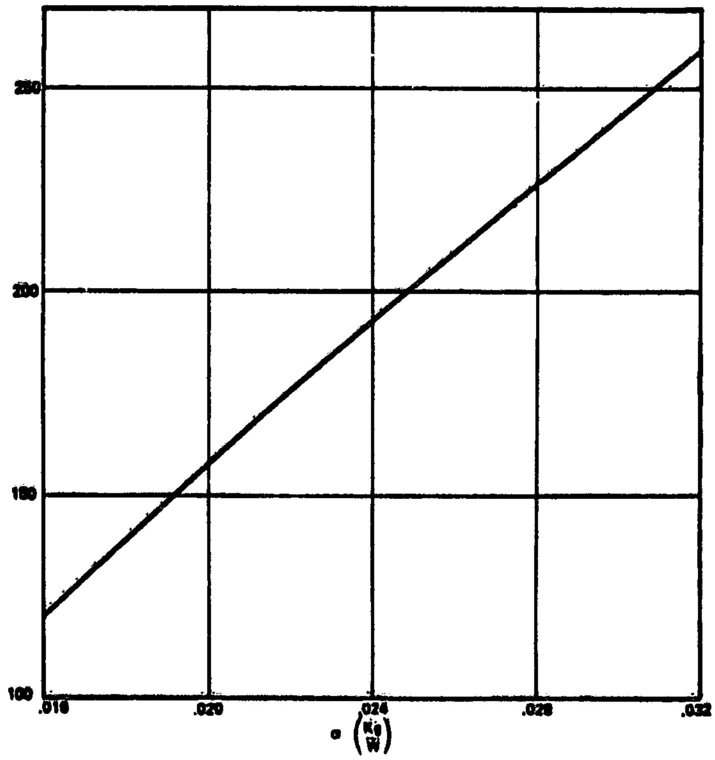
TYPICAL MASS RATIO BREAKDOWN



EFFECT OF SPECIFIC MASS ON TRANSFER TIME

(CASE II η = 45% AT 1750 SEC)



COMPARISON OF ORBIT TRANSFER AND ON-ORBIT CUMULATIVE FLUENCE

$\frac{\text{meV ELECTRONS}}{\text{CM}^2}$

$\left(\frac{M_{EPS}}{M_{PL}}\right)_T$	ORBIT TRANSFER	10-YEAR INCREMENT	SOLAR FLARE	TOTAL	RESIDUAL POWER (%)
1.0	3×10^{16}	1.8×10^{14}	3×10^{14}	3.05×10^{16}	60
2.0	2.2×10^{16}	1.8×10^{14}	3×10^{14}	2.25×10^{16}	58
3.0	1.55×10^{16}	1.8×10^{14}	3×10^{14}	1.60×10^{16}	58
4.0	1.50×10^{16}	1.8×10^{14}	3×10^{14}	1.55×10^{16}	59

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MAJOR AUXILIARY PROPULSION Δ V REQUIREMENTS

IN TEN-YEAR MISSION (M/SEC)

FUNCTION	ELECTRIC APS	CHEMICAL APS	Δ
N–S STATION KEEPING	(P = 0.5) (1)	(P = 0)	
LOW VALUE	(564)	(508)	(56)
HIGH VALUE	639 (2)	575 (2)	64 (2)
E–W STATION KEEPING	(P = 0.3)	(P = 0)	
S/C SIZE d = 20 M M_N = 2700 Kg	(316)	(244)	(72)
d = 60 M M_N = 8100 Kg	623 (2)	431 (2)	192 (2)
REPOSITIONING (5 @ 180°)	257	128	129
DISPOSAL AT END OF MISSION	167	167	–
GEO TO 40, 755 KM ORBIT ALTITUDE			
TOTAL	1686	1301	385

(1) P = ASSUMED DUTY CYCLE

(2) THESE VALUES USED IN TOTAL

EAST-WEST VS NORTH-SOUTH STATION KEEPING REQUIREMENTS

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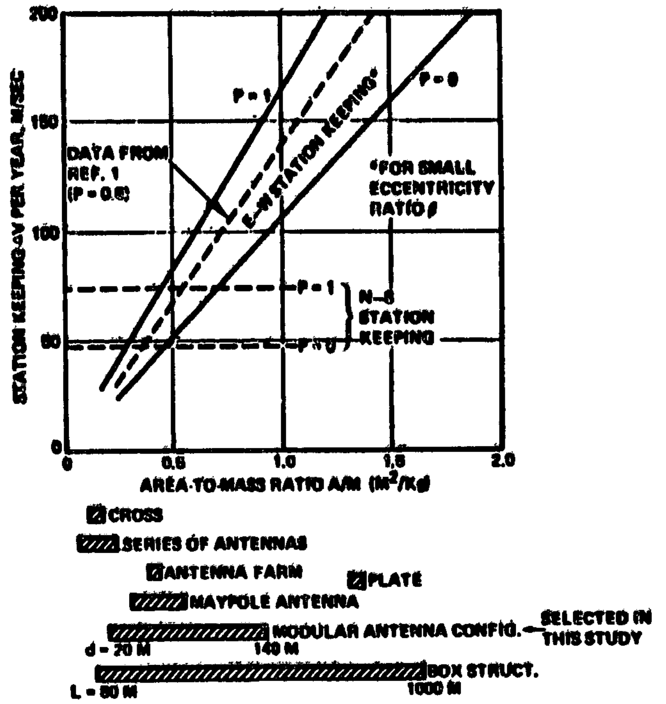
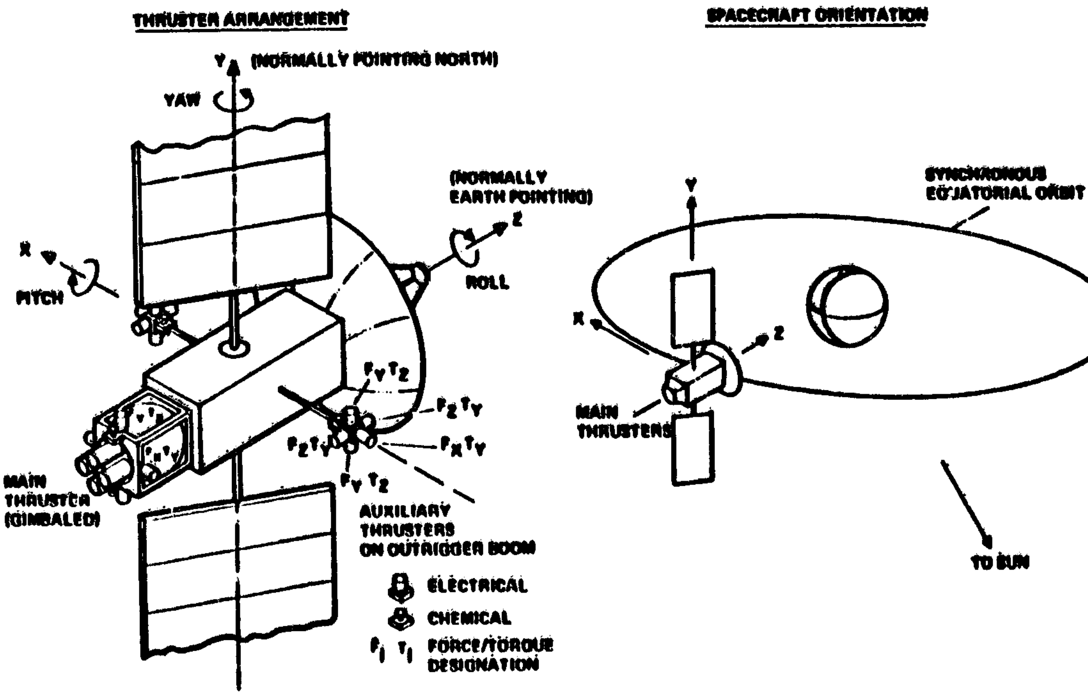
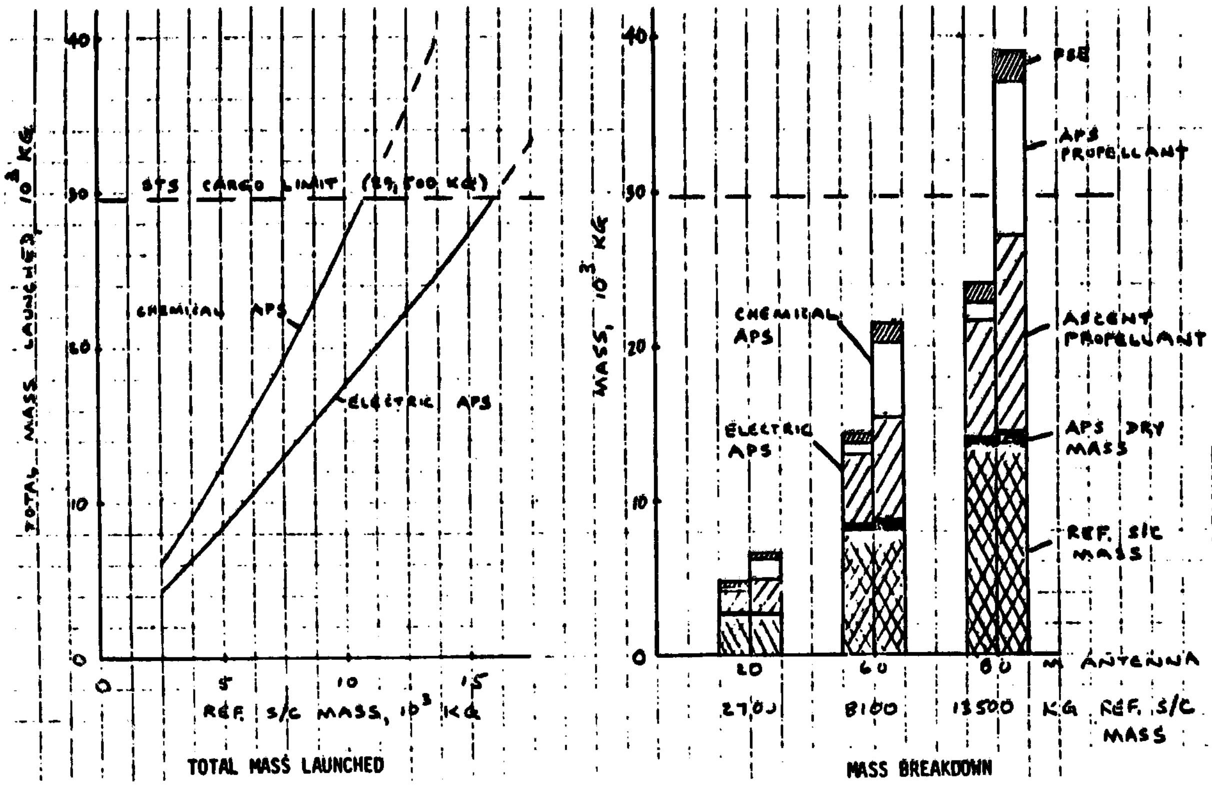


REPRESENTATIVE A/M RANGES OF S/C INVESTIGATED BY BOEING (REFERENCE 1)

MODULAR ANTENNA S/C CONFIGURATION



PERFORMANCE COMPARISON OF CHEMICAL AND ELECTRIC APS



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THRUSTER TECHNOLOGY AREAS

FOR INTEGRATED PROPULSION

- IMPROVE η IN I_{sp} RANGE FROM 1500 TO 2500 SEC
- CONTINUE WORK TOWARDS REDUCING SPECIFIC MASS (α)

CONCLUSION

- LARGE PAYLOAD TRANSFER FROM LEO TO GEO IS POSSIBLE WITH DEMONSTRATED THRUSTER EFFICIENCY
- THRUSTER EFFICIENCY STRONGLY AFFECTS TRANSFER TIME
- ELECTRIC PROPULSION SPECIFIC MASS (INCLUDING SOLAR ARRAY) STRONGLY AFFECTS TRANSFER TIME
- SUSTANTIAL RESIDUAL POWER IS AVAILABLE FOR ON-ORBIT FUNCTIONS
- LARGE TOTAL PROPELLANT MASS SAVINGS VIA INTEGRATED ELECTRIC APS RESULTS IN LEVERAGED INCREASE IN NET S/C MASS
- SOLAR PRESSURE EFFECTS REQUIRE MOST STATIONKEEPING ΔV FOR A/M > 0.5 M^2/KG

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STRUCTURES - PROPULSION INTERACTIONS & REQUIREMENTS

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WIDELY DIFFERENT TYPES OF STRUCTURE

When studying the interaction of a structure with its propulsion system one must consider the wide variety of configuration types that may be involved. While mission analyses may provide a good idea of overall functional requirements of the spacecraft to be involved they give little idea of the actual shapes. In reviewing projected missions one possible breakdown into two general categories is suggested -.

(1) Platforms
(2) "Christmas Trees"

The term "platform" has been widely used in the last several years, particularly in connection with space based radar and power stations. Such a structure gives the impression of a flat disk — it may be curved somewhat, like a saucer, or have a pentagonal outline and be built up with struts and wires and have in-plane arms sticking out, but nevertheless it is disk like — and it will be thrusted from the center with a force perpendicular to its plane.

Vu-graphs (1) and (2) are examples of such structures. Their loadings will be essentially symmetrical and dynamic characteristics can be described roughly in terms of beam or plate modes.

However, a whole class of potential structures exist that cannot be categorized as "platforms." One example is shown in Vu-graph (3), which is a proposed configuration for the Geostationary Communications Platform. The term "Platform" is no longer descriptive here, but is a hang-over from earlier system concepts studies in which little consideration had been given to actual space layouts. Vu-graph (3) is characterized by beams, arms and masts projecting in all three dimensions and carrying antennas or other scientific devices. It is certainly flimsy looking and displays little evidence of symmetry. For want of a better term, such spacecraft have been categorized as "Christmas Trees."

OPTIMIZATION

When optimizing a propulsion system for transportation of a family of large space structures, and only general requirements for the structures are known, one is forced to simplify and idealize. In addition, one must choose a parameter - or parameters - to be optimized. Potential examples are, area, useful weight, deflections, transfer time, etc. Given constraints of the Shuttle Orbitor lift capability and cargo bay size it is easy to choose maximum area as the parameter to be optimized for a "Platform" type.

In the case of the "Christmas Tree" the choice is more difficult though. Certainly area is hardly applicable. In the actual case of the "Geostationary Communications Platform" studies, the required number of antennas and space experiments far exceeded the possibility of a single Shuttle flight, and reduction of the number of flights to a minimum was a major criterion for optimization. However, folding and packaging of the various mec anisms to fit within the cargo bay proved to be the driving problems, while sensitivity to transfer loads seemed to be less provided the thrust to weight ratio was not more than about .02 or .05. If funding were available, it would be desirable to analyze one or two specific "Christmas Tree" configurations in some detail to determine their sensitivity to T/W variations. Because of technical complexity, this would be considerably more costly than for similar work on "Platforms" and may not feasible.

For these reasons it is probable that optimizing for maximum area of a "Platform" type of Large Space Structure is the best possible procedure for establishing thrust characteristics of a new transfer stage.

DYNAMIC INTERACTIONS

Static Loads

For a constant thrust to weight ratio (T/W) the Large Space Structure experiences constant acceleration loads which result from forces which are equal to this acceleration times the mass of each item in the system. If T/W varies then we define as "Static Loads" those loads which come from the instantaneous T/W times the mass of each item.

Thrust Transient Loads

When T/W varies very slowly the static loads approximate the actual loads; but when it varies quickly dynamic transient loads are generated which make the total actual loads considerably greater than the static loads. The ratio of the maximum actual load to the maximum static load is called the "Dynamic Amplification Factor."

Vu-graph No. 5 is an example computer simulation of actual loads experienced during the main engine start transient of a Centaur. The maximum static acceleration is approximately 700 in/sec^2 while the actual load on the engine is seen to be over 800 in/sec^2. Other parts of the system experience other values.

Vu-graph No. 6 shows calculated response of a simple single degree of freedom system to a trapezoidal engine thrust function. The thrust rise time (D) is approximately 1/4 of the natural period of the system. The dynamic amplification factor is 1.9. In the limit, where the rise time approaches zero, the factor for an undamped single degree of freedom system approaches 2. Vu-graph No. 7 shows the effect of increasing the thrust rise time to .8 times the natural period; here the amplification factor has dropped to 1.2. Further increase in rise time would produce further reduction in factor until, in the limit, the factor would become 1.

Vu-graph No. 8 is an example of summary output from an actual dynamic loads analysis for the COA platform. It will be noted that the maximum arm tip acceleration of .111g is 31/4 times the static T/W, namely .034g. However, the bending moment at the root of arm D is only 2.2 times the static moment. For this example the lowest structural natural frequency was .169 Hz and the thrust build up time was .1 sec. Thus the ratio of build up time to natural period is .02 indicating an almost instantaneous build up from the point of view of the structure. If build up times in the order of a few seconds can be achieved then real responses would be more like the examples of Vu-graphs 6 and 7.

Vu-graph No. 9 illustrates the effect of dynamic amplification factor (K_ϕ) upon area optimization from a computor run. Since the factor has been changed 100% - from 1 to 2 - it may seem surprising that the corresponding change in size is only 8%. This is explained qualitatively in Vu-graph No. 10 which does not reproduce actual calculations. The "original" curve is the starting point with, for example, $K_\phi = 1$. If we consider an example point A, where the original T/W = .2, and then arbitrarily halve T/W to .1 while we double K_ϕ to 2, we will not change the loads at all. Thus, if there were no change in engine performance, the structure would not change at all and point A would move horizontally to point A', and in fact the entire curve would simply slide to the left to the position of the "dashed" curve. In this case, the optimum value of T/W would halve but the corresponding size would not change. In practice, however, engine performance will decrease as T/W decreases, and point A' will really drop to a position such as A", and the final curve will look like the "dotted" curve on Vu-graph No. 10. It is clear that the total decrease in optimum size is due to loss in engine performance, while the decrease in size at a fixed T/W (say at T/W = .2) is due to increase in load.

One concludes that the optimum size is not very sensitive to dynamic amplification factor, but that the optimum value of T/W at which it occurs decreases as amplification factor increases. Further, in the region of optimum T/W size is relatively insensitive to K_ϕ, but for higher values of T/W the sensitivity increases. For these reasons it seems justifiable to assume a fixed value for K_ϕ (say 2) for optimization analysis which greatly simplifies the work.

It is important to remember that actual loads on a given spacecraft are directly proportional to K_ϕ and in analyzing design loads actual values of K_ϕ must be determined.

Propellant Accelerations

Vu-graphs 6 and 7 show negative values of acceleration but only after thrust cut-off. For a single degree of freedom system this is the only time when negative acceleration could occur. For real platforms with many degrees of freedom, negative accelerations are possible during burn, as is true for -- all space launch systems. This condition is unlikely in regions near large masses such as the propellant tanks. The condition will depend mainly on the detailed dynamic characteristics of the individual spacecraft relative to the thrust build up function of the engine and will tend to be relatively insensitive to the actual thrust. For a given large space structure and a family of engines with the same thrust build up times the probability of negative propellant accelerations is likely to decrease slightly as thrust decreases. If, as seems probable, the thrust build up time of low thrust engines will be greater than times for conventional high thrust engines then the likelihood of negative propellant accelerations appears to be very low.

As in all configurations past, present, or future, propellant accelerations should be analyzed on an individual basis for each engine/spacecraft combination, but this does not appear to be an important consideration in optimizing the thrust level.

Tolerances

The effects of tolerances fall into two main categories -

(1) c.g offset
(2) rattle in joints

Both of these affect dynamic transient loads and care must be taken to check each spacecraft to assure that it is not overstressed. Their effects, however, are not sensitive to engine thrust or thrust build-up function and therefore they should not take a significant part in decisions on optimization.

Attitude Control

Early analyses on the OOA indicated that engine vectoring should be adequate to control the system during orbit transfer. It is conceivable that a very flexible platform type of design may be developed for which this is not possible. In such a case it is assumed that the on-orbit control system of the spacecraft itself could be used to assist during engine burns.

Distributed Thrust

It is understood that J. Clark will be discussing this subject in detail. Suffice it to say here that distributed thrust can theoretically greatly alleviate spacecraft loads, but that random differences in thrust build up functions of two or more engines may produce intolerable anti-symmetric loading conditions.

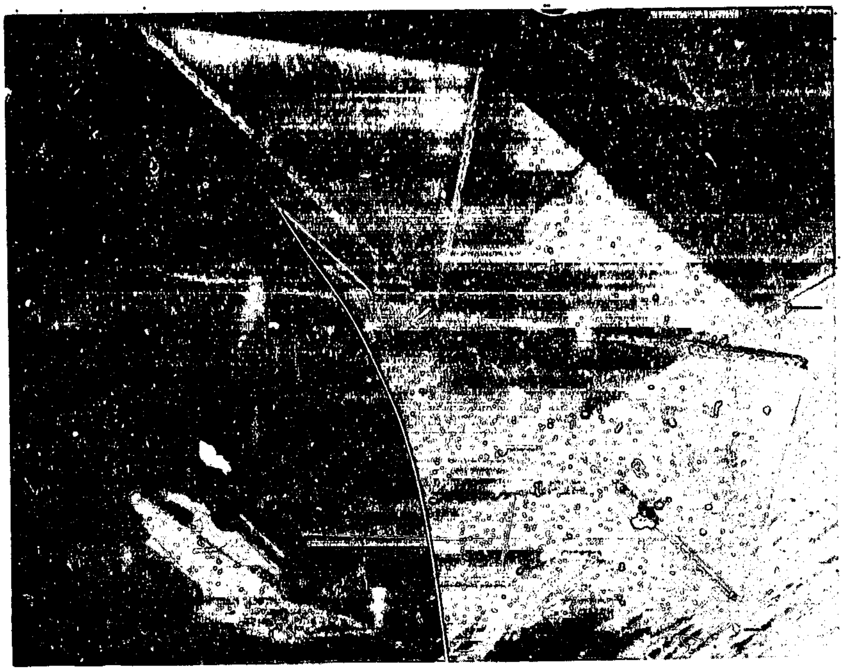
VU - 1 STRUCTURES - PROPULSION INTERACTIONS

- WIDELY DIFFERENT TYPES OF STRUCTURE
  - PLATFORM
  - CHRISTMAS TREE
- OPTIMIZATION
  - PARAMETER TO BE OPTIMIZED

    AREA
USEFUL WEIGHT
DEFLECTIONS
TRANSFER TIME
ETC.
  - CONSTRAINTS

    PACKAGING
TOTAL WEIGHT
MINIMUM GAGE
ETC.
- DYNAMIC INTERACTIONS
  - STATIC LOADS
  - ENGINE THRUST TRANSIENT
  - PROPELLANT ACCELERATION
  - TOLERANCES
  - CONTROL - OTV VS S/C ACS
  - DISTRIBUTED THRUST

VU - 2 LARGE SPACE PLATFORM BEING TRANSFERRED TO HIGH EARTH ORBIT



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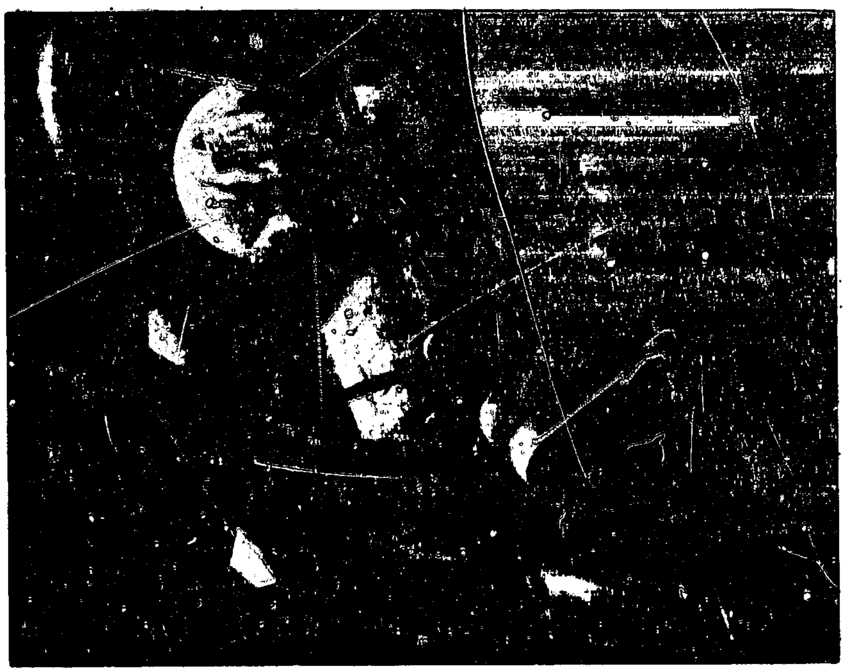
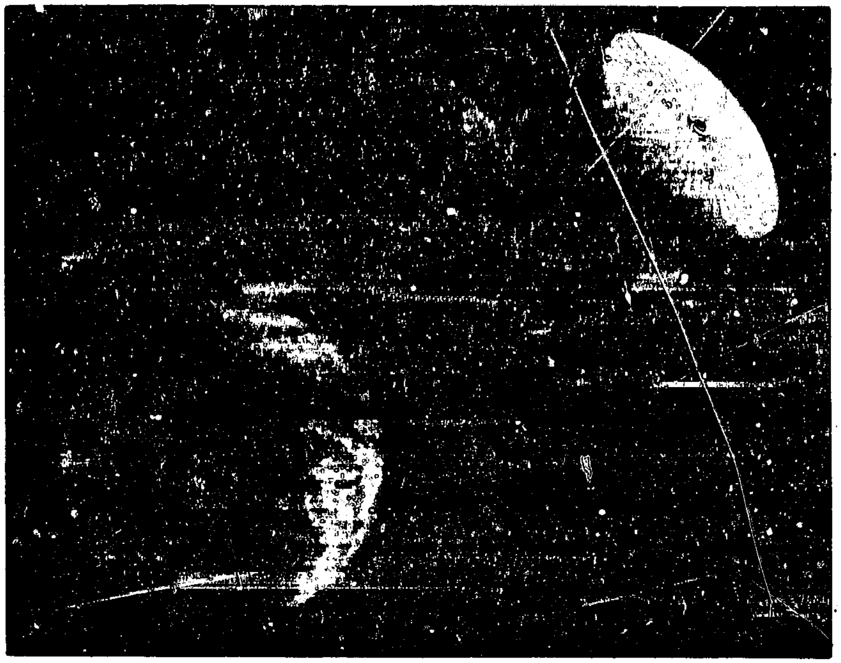
VU - 3 LARGE SPACE PLATFORM BEING RESUPPLIED AT GEOSTATIONARY ORBIT



VU - 4 LARGE SPACE PLATFORM AT GEOSTATIONARY ORBIT



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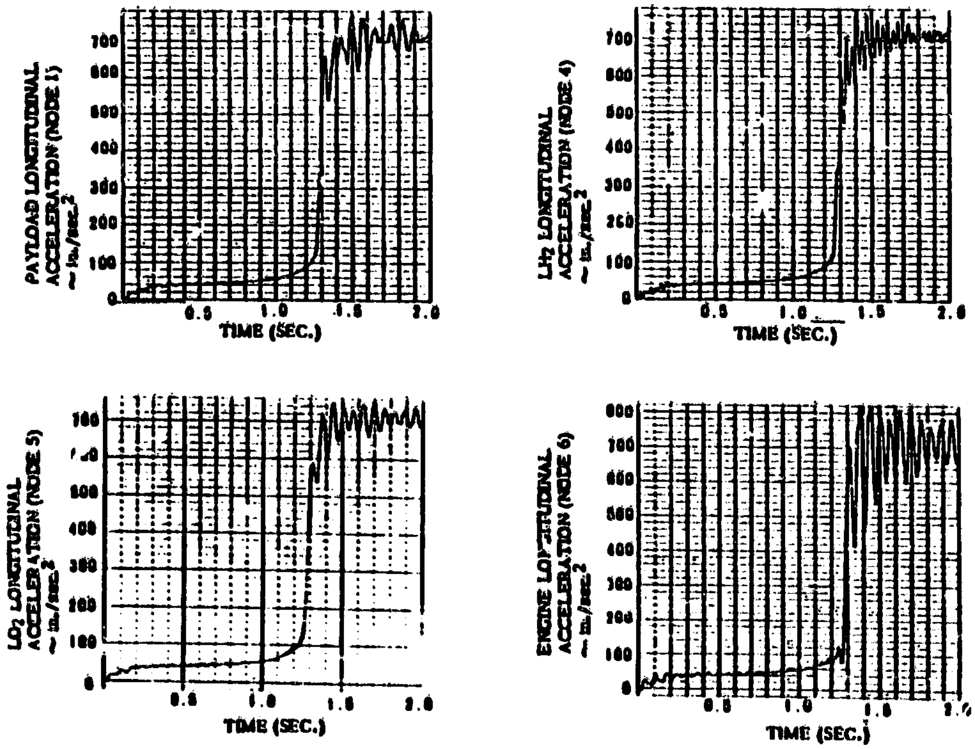
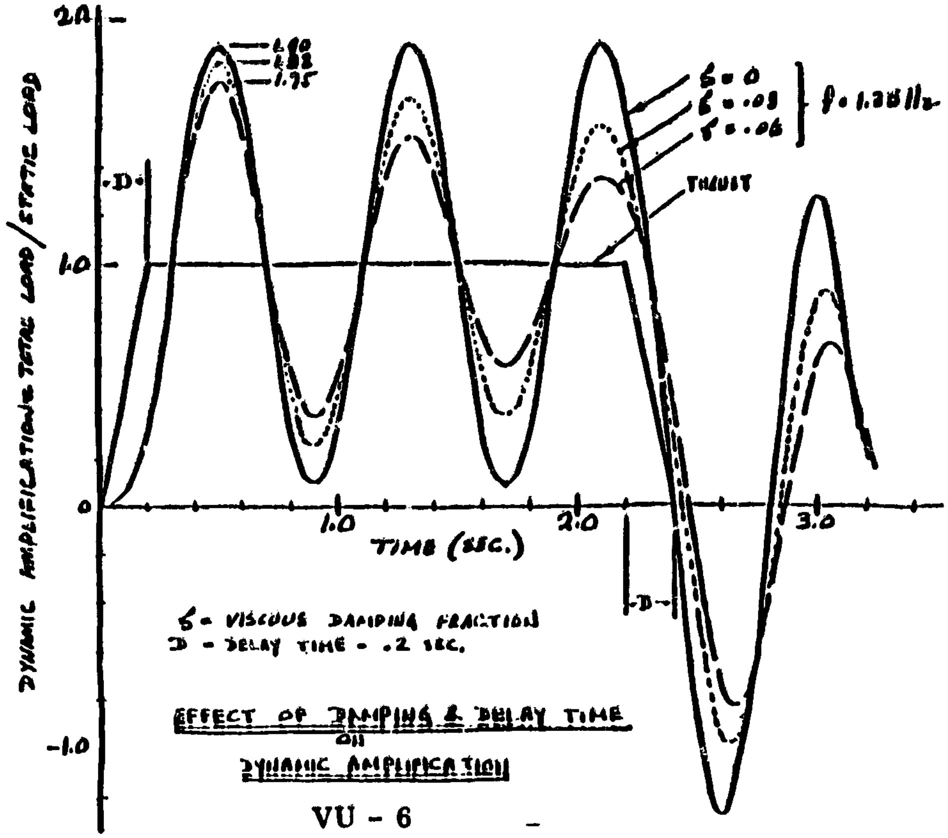
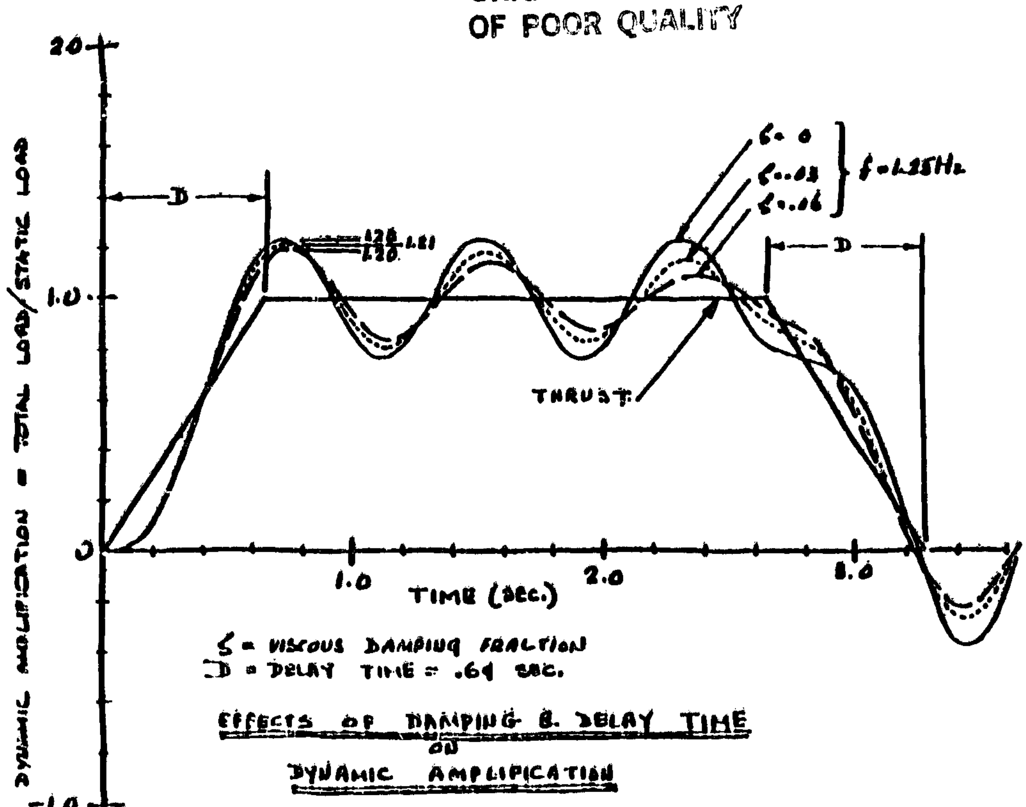


VU - 5 Longitudinal Response Time-Histories of Kick Stage Masses (Mathematical Model No. 37) During Centaur 2nd Main Engine Start



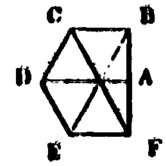
2A
1.80
1.83
1.75
ζ = 0
ζ = .03
ζ = .06
f = 1.38 Hz
THRUST
D
1.0
0
-1.0
1.0
2.0
3.0
TIME (SEC.)
DYNAMIC AMPLIFICATION = TOTAL LOAD/STATIC LOAD
ζ = VISCOUS DAMPING FRACTION
D = DELAY TIME = .2 SEC.
EFFECT OF DAMPING & DELAY TIME ON DYNAMIC AMPLIFICATION

VU - 6



2.0
1.0
0
-1.0
D
1.23
1.21
1.20
ζ = 0
ζ = .03
ζ = .06
f = 1.25 Hz
THRUST
1.0
2.0
3.0
TIME (SEC.)
DYNAMIC AMPLIFICATION = TOTAL LOAD/STATIC LOAD
ζ = VISCOUS DAMPING FRACTION
D = DELAY TIME = .64 SEC.
EFFECTS OF DAMPING & DELAY TIME ON DYNAMIC AMPLIFICATION

VU - 7



	ARM D	ARM E	ARM F	ARM A	ARM B	ARM C
MAX BENDING MOMENT AT ARM ROOT (FT-LB) STATIC	+22,090 10,300	+17,163	+20,607	+35,037	+20,607	+17,163
MAX SHEAR AT ARM ROOT (LB)	101	86	95	190	95	86
MAX ACCELERATION AT ARM TIP	.090	.111	.065	.088	.085	.111

25 PERCENT OF PROPELLANTS REMAINING
SPACECRAFT WEIGHT = 31,359 LB

STEADY-STATE ACCELERATION = 0.034G

VU-8 Dynamic response due to thrust build-up (beginning of apogee burn).

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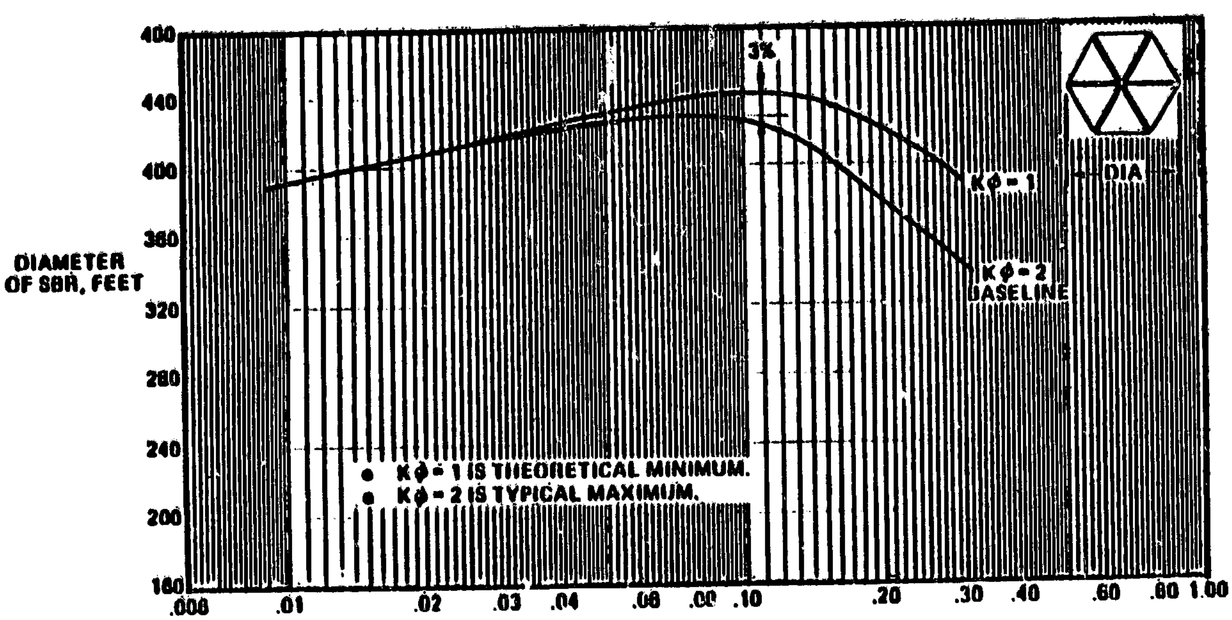
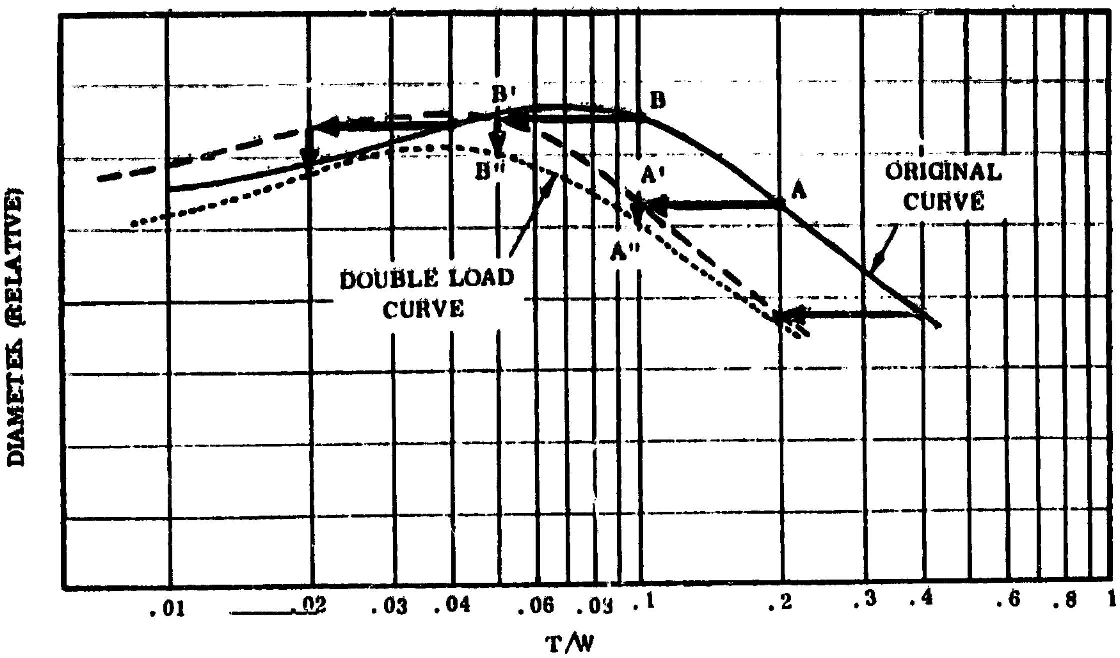
VU - 9 EFFECT OF DYNAMIC FACTOR (KΦ) ON SIZE OF SBR-A



VU - 10 QUALITATIVE RESULT OF DOUBLING LOAD DUE TO DYNAMIC FACTOR



NOTES:

(1) HORIZONTAL ARROWS ARE DUE TO STRUCTURAL REQUIREMENTS.

(2) VERTICAL ARROWS ARE DUE TO ENGINE PERFORMANCE LOSSES.

VU - 11 SUGGESTIONS

- LOW T/W IS MANDATORY - .02 TO .10 RANGE - BUT EXACT VALUE NOT VERY SENSITIVE
  - SLOW THRUST RISE TIME DESIRABLE BUT NOT MANDATORY.
  - ENGINE THRUST SELECTION NOT VERY SENSITIVE
- STRUCTURES WILL LIKELY BE DESIGNED, BUT NOT IMPACTED, BY THIS T/W RANGE.
- PROPELLANT ACCELERATION SHOULD BE INDIVIDUALLY ANALYZED FOR START TRANSIENT BUT VERY UNLIKELY TO GO NEGATIVE.
- "PLATFORMS" MAY SOMETIMES BE REQUIRED TO USE THEIR OWN ATTITUDE CONTROL SYSTEMS DURING THRUST.
- TOLERANCES IN S/C STRUCTURE SHOULD NOT AFFECT CHOICE OF PROPULSION SYSTEM CONFIGURATION, BUT EFFECT UPON LOADS SHOULD BE EVALUATED ON AN INDIVIDUAL BASIS.

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STRUCTURES-PROPULSION INTERACTIONS AND REQUIREMENTS

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The availability of the Space Transportation System (STS) in the 1980's will make it feasible to deploy on-orbit Large Space Systems (LSS). In general terms, large space systems are classified as either deployable or erectable, depending upon the process used to place them into operational configuration. With deployable structures, the entire manufacturing and assembly takes place on the ground and the assembly is flown into space in a high density folded form, where it is then deployed. The concept of erectable structures refers to assembly in space either by a building crew or by remote manipulation. Propulsion system thrust levels required to transfer these general types of structures from LEO to GEO depends upon the load bearing capability of the LSS.

The interaction study determined the effects of low-thrust primary propulsion system characteristics on the mass, area, and orbit transfer characteristics of Large Space Systems (LSS). Three general structural classes of LSS were considered, each with a broad range of diameters and nonstructural surface densitites. While transferring the deployed structure from LEO to GEO, an acceleration range of 0.02 to 0.1 g's was found to maximize deliverable payload based on structural mass impact. After propulsion system parametric analyses considering four propellant combinations produced values for available payload mass, length and volume, a thrust level range which ma[illegible]mizes deliverable LSS diameter was determined corresponding to a structure and propulsion vehicle.

The engine start and/or shutdown thrust transients on the last orbit transfer (apogee) burn can impose transient loads which would be greater than the steady-state loads at the burnout acceleration. The effect of the engine thrust transients on the LSS was determined from the dynamic models upon which various engine ramps were imposed. The ramp times, T_R, were chosen to be a step, 1/2, 2/3, 3/3, and 4/3 of the LSS fundamental period.

Based on a single Shuttle flight with a LEO to GEO orbit transfer, the optimum OTV thrust level range to maximize delivered LSS diameter (if payload mass limited) is between 3100 to 4200N and the maximum performance tankage configuration delivers the maximum deployed LSS diameter. This range is relatively independent of the following:

1) Propellant Combination;
2) Tankage Configuration;
3) Mixture Ratio;
4) Type of LSS; and
5) Type of LSS Nonstructural Surface.

The curve is also relatively flat, implying insensitivity to thrust of a range of thrust.

The results also showed that the transient-induced load is not a significant driver (10% structural mass impact) in defining requirements for the primary propulsion system. If no structural mass impact is desired, the ramp time must be greater than two-thirds of the LSS fundamental period for all LSS considered.

The following stage characteristics are reemphasized which deliver the maximum LSS diameter based on the results of this study. The characteristics are:

1) LO_2/LH_2 propellant combination
2) Pump-fed single engine, and
3) Constant acceleration, 8 perigee burn orbit transfer strategy

However, these results differ from other studies that have determined the optimum thrust level. Also, there are six key factors that may drive the optimum thrust level and must be evaluated prior to stage definition. They are:

- Minimum Strength of Structure
- Non-geosynchronous Operational Altitude (Large Payloads)
- Impact of Subsystem (Mass and Dynamics)
- Larger Capacity Orbiter
- Ultra-Gossamer Structures of the Future
- Ultra-Large Assembled Structures

Another key factor is whether LSS deployment in low earth orbit is required and whether LSS servicing and repair can be done in LEO. If LSS designers do not include servicing provisions in their design, then why impose low thrust requirements on the stage designer?

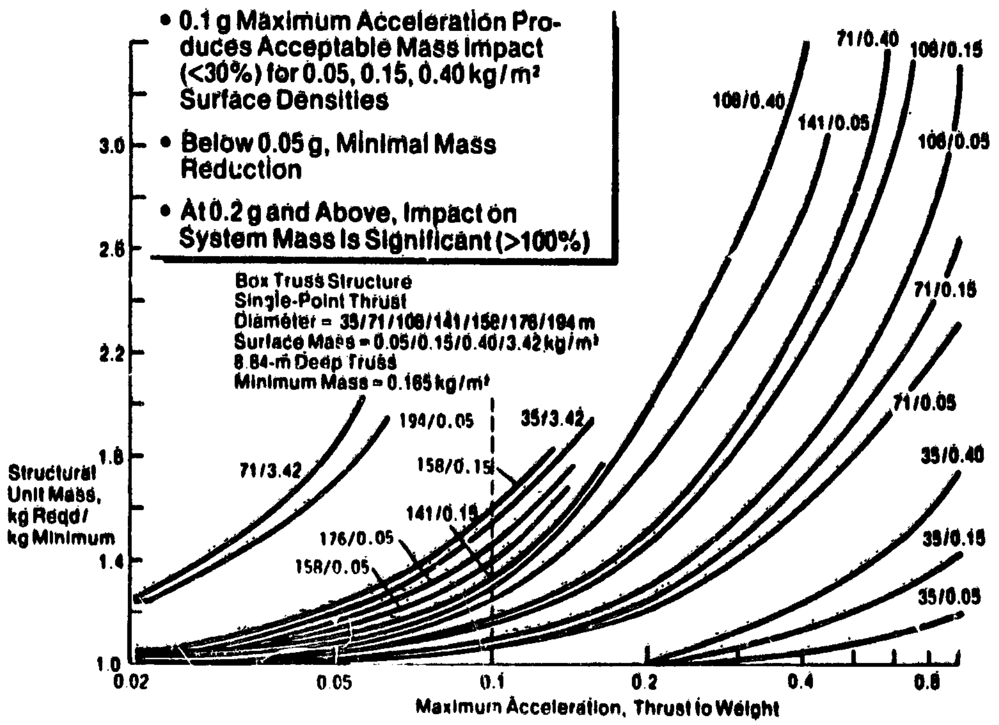
PP/LSSI Program Summary

The primary objective of the Primary Propulsion/Large Space System Interaction Study program is to determine the effects of low-thrust primary propulsion system thrust-to-mass ratio, thrust transients, and performance on the mass, area, and orbit transfer characteristics of large space systems.

LSS Configurations

Structural Concept	Diameter Range, m	Surface Densities, Kg/m²	Potential	Point of Thrust Application	Thrust to Mass Ratio, g
Expandable Box Truss	30-200	0.05, 0.15, 0.40, 3.42	Antennae Across Frequency Range, Power Generation, Solar Cells	Center of Structure Normal to Plane	0.02-1.0
Wrap Radial Rib	30-200	0.05, 0.15	Low Frequency Antennae	Center of the Hub	0.02-1.0
Hoop and Column	30-200	0.15, 0.40	Low Frequency Antennae, Power Generation Solar Cells	Aft End of Telescoping Mass	0.01-1.0

Expandable Box Truss—Unit Mass vs Thrust to Weight



Average Structural Mass Impact for Start Transient Effects

Ramp Time T_R,	Box Truss Average kg Reqd/kg Steady State	Radial Rib, Average kg Reqd/kg Steady State	Hoop/Column, Average kg Reqd/kg Steady State
Step	1.10	1.10	1.07
$1/3 f_1$	1.03	1.05	1.01
$2/3 f_1$	1.00	1.00	1.00
$3/3 f_1$	1.00	1.00	1.00
$4/3 f_1$	1.00	1.00	1.00
T_R Range for No Structural Impact	0.2 to 2 s	0.5 to 10 s	0.3 to 11 s

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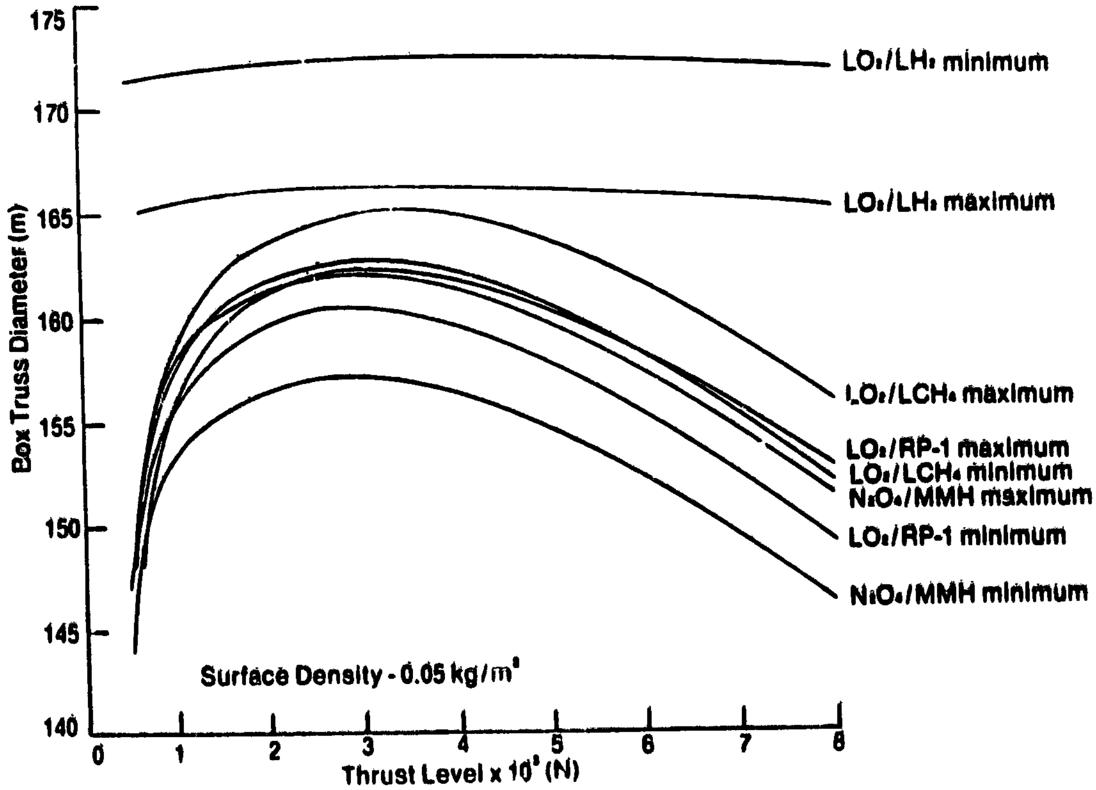
Effect of Thrust Level on Box Truss Diameter for Various Propellant Combinations and Tankage Configurations



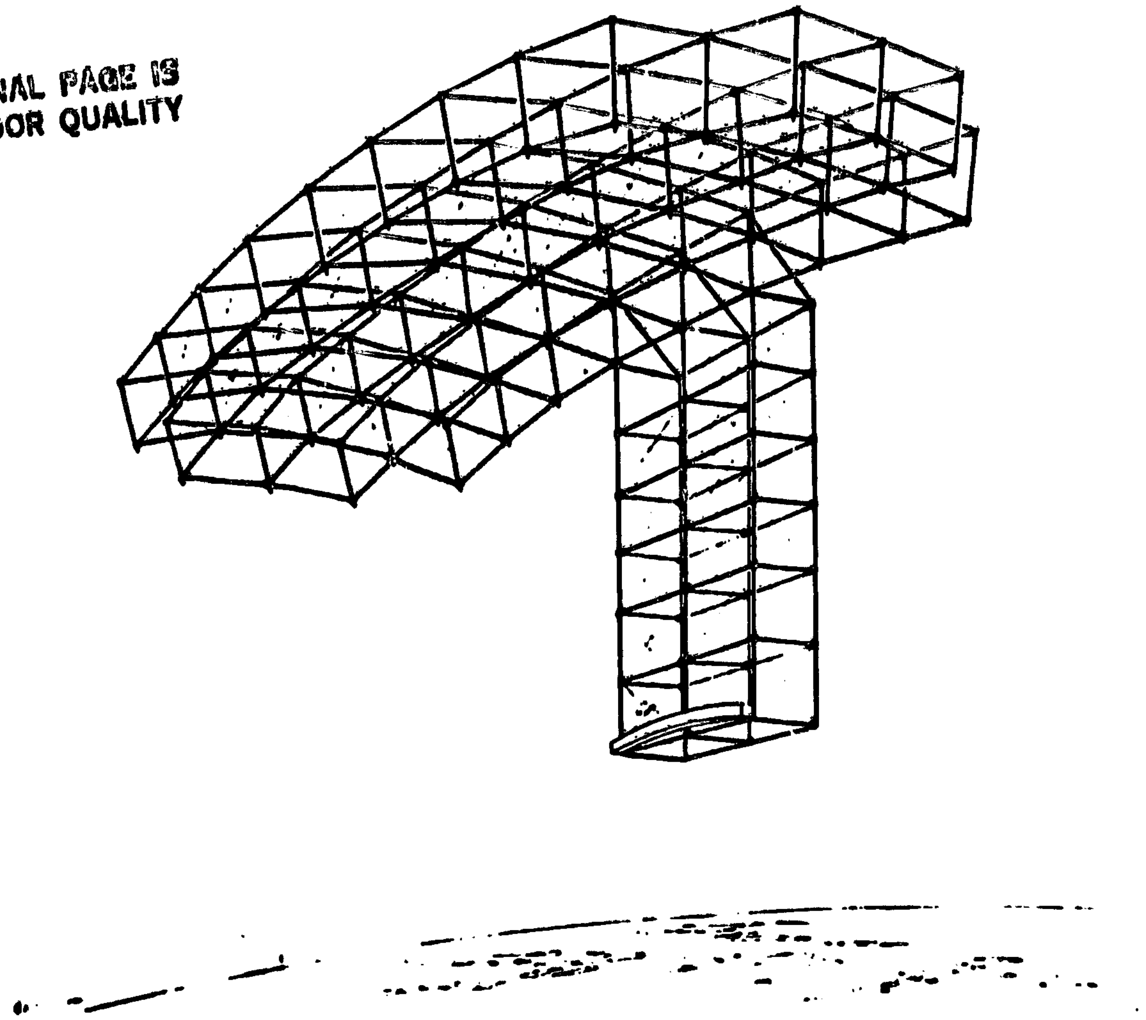
Technical Issues for LSS-OTV Design

- Do Large Space Systems (LSS) drive the design of OTV?
- Stage Length
  - LSS Package Density is 24 to 48 kg/m^3
  - 6820 kg Payload Occupies 17.22 m at 24 kg/m^3 and 8.63 m at 48 kg/m^3
  - 9090 kg Payload Occupies 23.01 m at 24 kg/m^3 and 11.49 m at 48 kg/m^3
  - Orbiter Cargo Bay Length = 18.3 m
  - Short Dense Stages Are Required If Payload Is To Be Mass Limited and Not Volume Limited
- OTV Thrust Level
  - Will LSS be deployed in low Earth orbit (LEO) before transfer?
  - What contingency operations can be accomplished in LEO?
  - What is the optimum thrust level if the LSS is deployable?

Thrust Level Considerations

- Previous Studies Indicate Two Different Optimum Thrust Ranges
  - Less Than 4400 N (Martin Marietta)
  - Greater than 4400 N (General Dynamics)
- What factors could affect this optimal thrust level?
  - Minimum Strength of Structure
  - Non-geosynchronous Operational Altitude (Large Payloads)
  - Impact of Subsystem (Mass and Dynamics)
  - Larger Capacity Orbiter
  - Ultra-Gossamer Structures of the Future
  - Ultra-Large Assembled Structures

120 Meter x 60 Meter Radiometer



CENTRALIZED VERSUS DISTRIBUTED PROPULSION

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SUMMARY

Several related topics are addressed. First, the functions and requirements of auxiliary propulsion systems are reviewed. None of the three major tasks - attitude control, stationkeeping and shape control - can be performed by a collection of thrusters at a single central location. The need, in general, for torques about all three axes requires that propulsion units be physically separated. If a centralized system is defined as a collection of separated clusters, made up of the minimum number of propulsion units, then such a system can provide attitude control and stationkeeping for most vehicles. A distributed propulsion system can provide shape control for flat plate type structures and it can also perform attitude control and stationkeeping.

A review of various proposed large space systems leads to the conclusion that centralized auxiliary propulsion is best suited to vehicles with a relatively rigid core. These vehicles may carry a number of flexible or movable appendages. This group includes many of the proposed deployable concepts. A second group, consisting of one or more large flexible flat plates, may need distributed propulsion for shape control. There is a third group, consisting of vehicles built up from multiple shuttle launches, which may be forced into a distributed system because of the need to add additional propulsion units as the vehicles grow.

The results of a study to examine the effects of distributed propulsion on a beam-like structure are reported. The deflection of the structure under both translational and rotational thrusts is shown as a function of the number of equally spaced thrusters. When two thrusters only are used it is shown that location is an important parameter.

The possibility of using distributed propulsion to achieve minimum overall system weight is also examined. It is shown that under certain conditions the additional weight of distributed propulsion can be more than offset by reduced structural weight made possible by reducing stiffness.

Finally, an examination of active damping by distributed propulsion is described.

Although flight experience to date has been exclusively with centralized propulsion systems, distributed propulsion will be needed in the not too distant future. A number of issues should be addressed. These include: an examination of the optimum number and distribution of thrusters for active shape control of uniform and non-uniform one and two dimensional plate like structures; the problem of control of evolving structures that may experience drastic changes in configuration and mass properties; and the sensor placement, control law development, computation and other considerations needed to implement distributed propulsion systems.

C-2

TOPICS

- o AUXILIARY PROPULSION SYSTEMS
  - o FUNCTIONS
  - o REQUIREMENTS
  - o DISTRIBUTED AND CONCENTRATED SYSTEMS
- o APPLICATIONS
- o PERFORMANCE OF DISTRIBUTED SYSTEMS
  - o NUMBER OF THRUSTERS
  - o LOCATION
- o MINIMUM WEIGHT SYSTEMS
- o ACTIVE DAMPING

AUXILIARY PROPULSION SYSTEM FUNCTIONS

ATTITUDE CONTROL	
o POINTING	ROTATION
o MANEUVER	ROTATION
o DISTURBANCE CANCELLATION	ROTATION; TRANSLATION
STATIONKEEPING	TRANSLATION
o NORTH-SOUTH	TRANSLATION
o EAST-WEST	TRANSLATION
SHAPE CONTROL	

AUXILIARY PROPULSION REQUIREMENTS

ATTITUDE CONTROL - PROVIDE TORQUES
- o THRUSTERS AWAY FROM CENTER OF MASS
- o PAIRS TO PROVIDE PURE COUPLE AND AVOID TRANSLATION

STATIONKEEPING - PROVIDE ΔV THRUSTS
- o CORRECT ORIENTATION
  - o ATTITUDE CONTROL
- o NET THRUST THROUGH CENTER OF MASS
  - o GIMBALLED THRUSTERS
  - o FIXED THRUSTER(S) PLUS TORQUE CAPABILITY

SHAPE CONTROL - ZERO NET ROTATION AND TRANSLATION
- o DISTRIBUTED THRUSTERS
- o MODULATION
- o PRECISE TIMING

AUXILIARY PROPULSION SELECTION PROBLEMS

WEIGHT MINIMIZATION

PLACE UNITS CLOSE TOGETHER
- o SHARE ANCILLIARY EQUIPMENT

PLACE UNITS FAR APART
- o INCREASE TORQUES
- o DECREASE PROPELLANT REQUIRED

SOLUTION - MOUNT UNITS IN SEPARATED CLUSTERS

REDUCE NUMBER OF THRUSTERS
- o MINIMIZE TOTAL THRUSTER WEIGHT

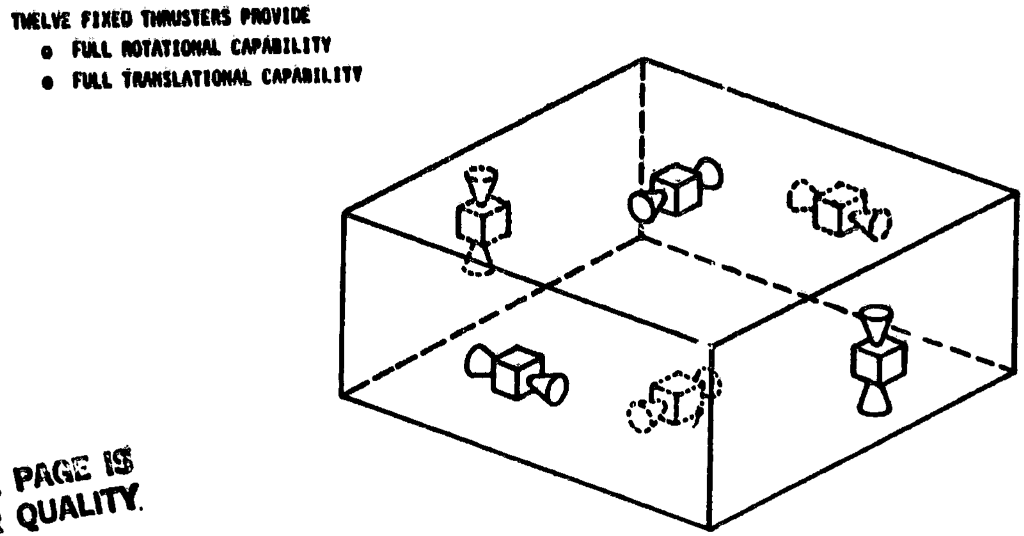
INCREASE NUMBER OF THRUSTERS
- o DISTRIBUTED THRUSTERS MAY PERMIT LOWER STRUCTURAL STIFFNESS AND REDUCE STRUCTURE WEIGHT

SOLUTION - MINIMIZE TOTAL STRUCTURE PLUS AUXILIARY PROPULSION WEIGHT

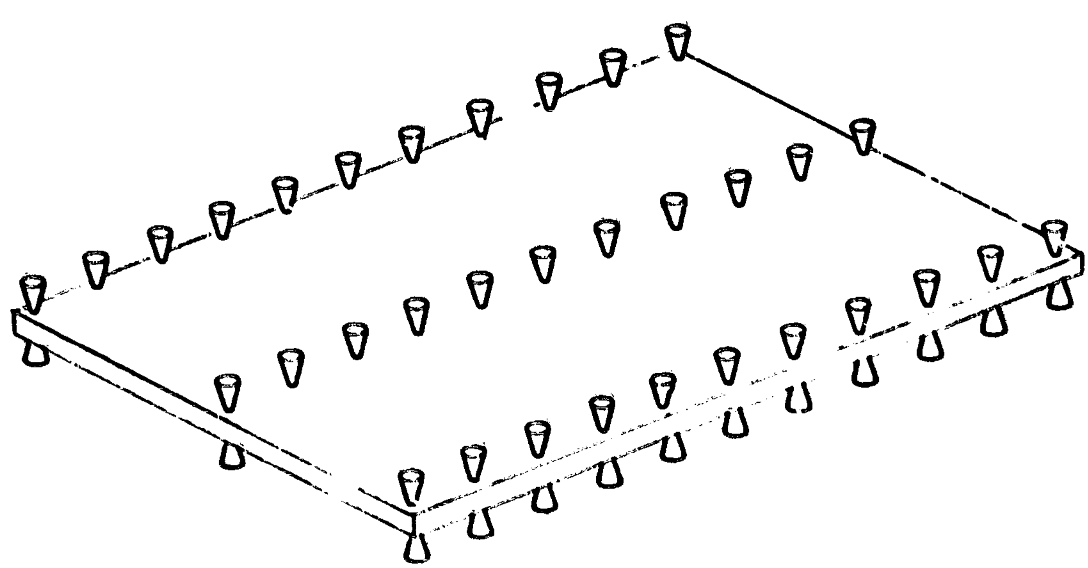
CONSTRAINTS

- o STRUCTURE TOO WEAK TO SUPPORT THRUST LOADS
- o PACKAGING AND DEPLOYMENT
- o CONFIGURATION
- o CONTAMINATION

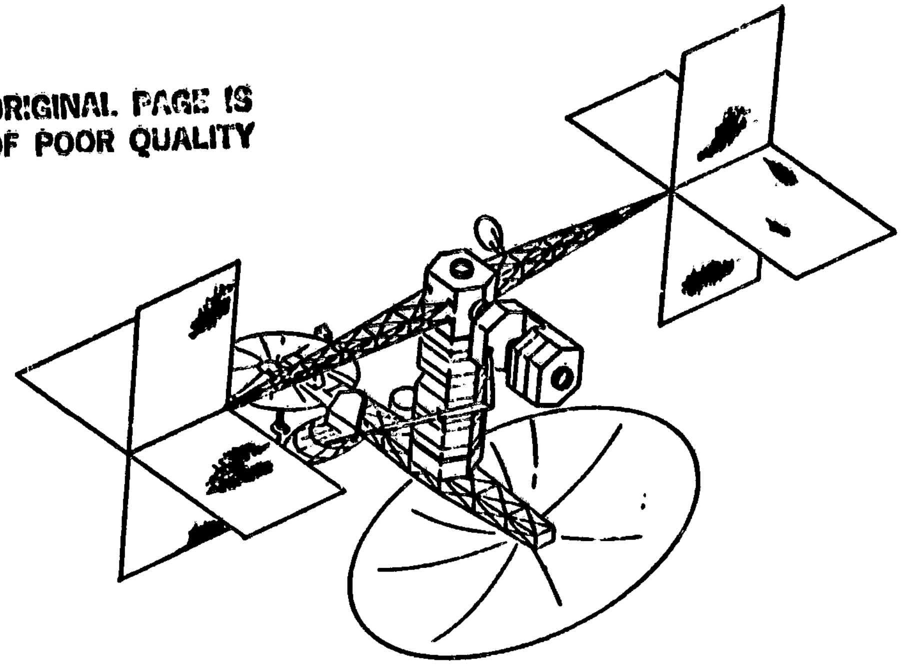
CONCENTRATED AUXILIARY PROPULSION



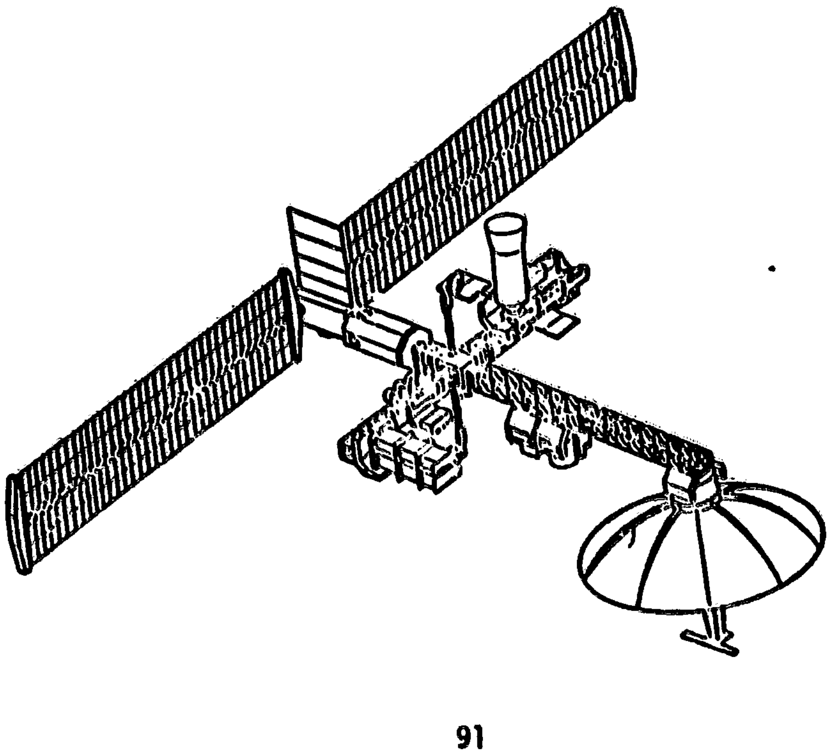
DISTRIBUTED THRUSTERS



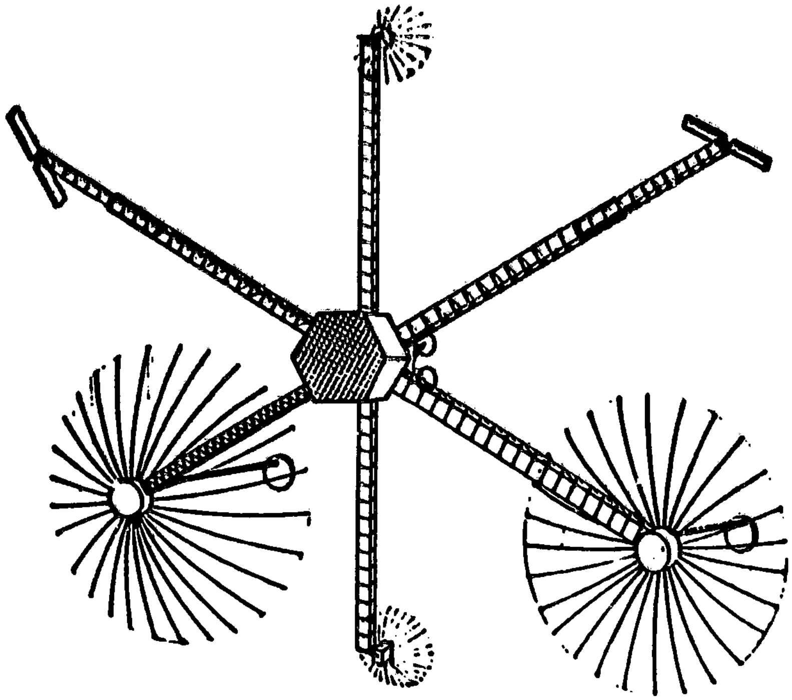
JET PROPULSION LAB. OAF AMERICA



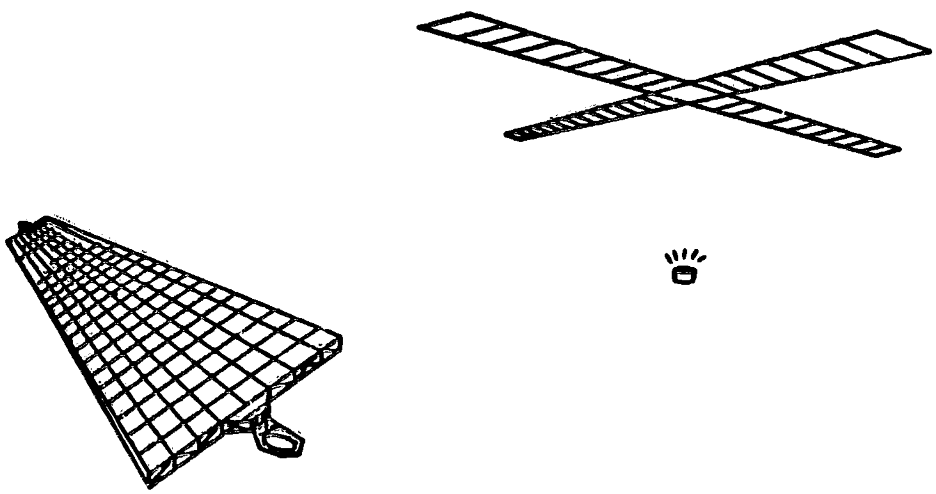
McDONNELL-DOUGLAS SCIENCE AND APPLICATIONS SPACE PLATFORM



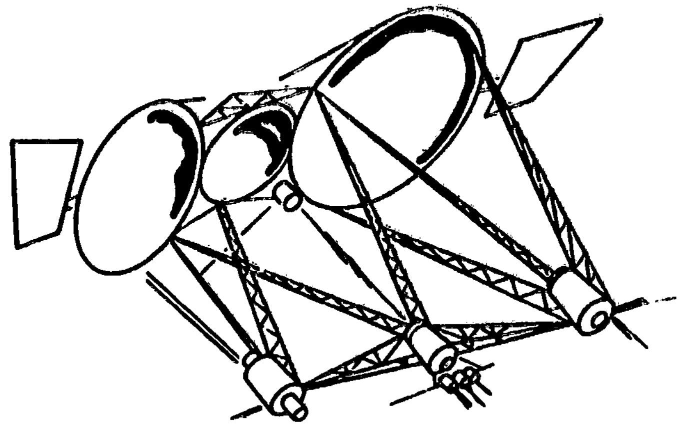
GENERAL DYNAMICS MULTIPLE BEAM GEOPLATFORM



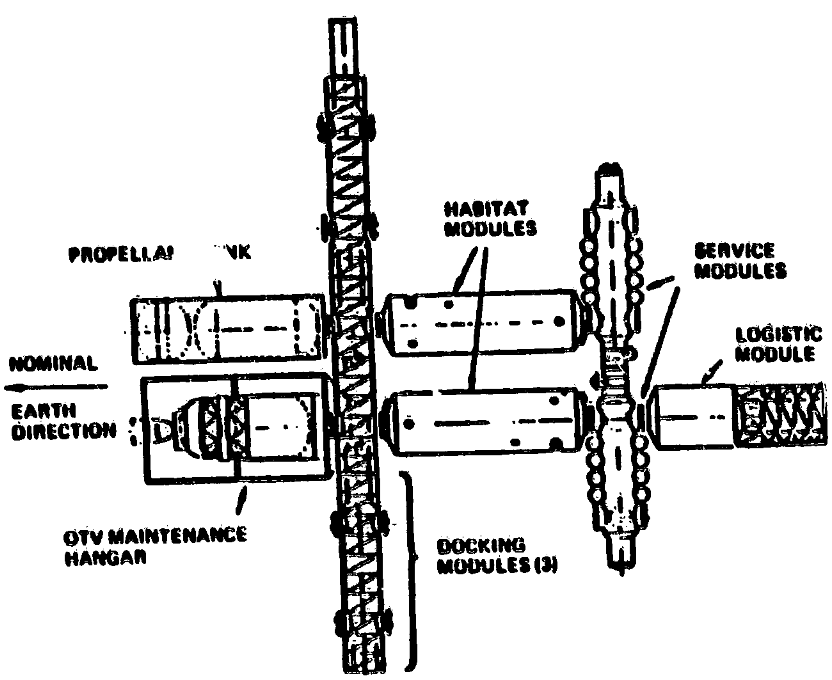
FLAT PLATE STRUCTURES



GRUMMAN PUBLIC SERVICE PLATFORM



SPACE OPERATIONS CENTER



APPLICATIONS OF CENTRALIZED AND DISTRIBUTED PROPULSION

CENTRALIZED PROPULSION

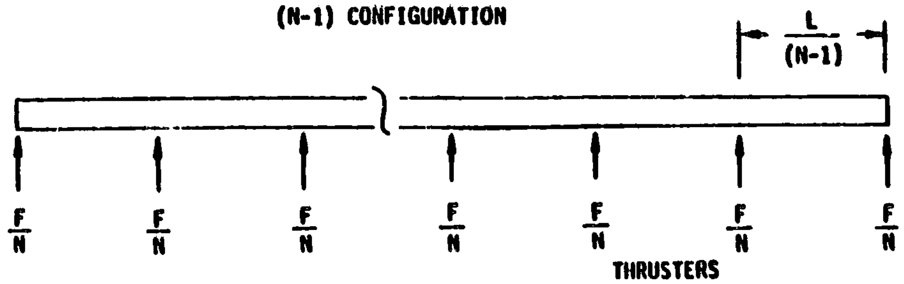
- MINIMUM OR NEAR MINIMUM NUMBER OF THRUSTERS
- RIGID VEHICLES
- VEHICLES WITH RIGID CORE AND FLEXIBLE APPENDAGES
- SHAPE CONTROL NOT REQUIRED
- SHAPE CONTROL NOT FEASIBLE WITH THRUSTERS

DISTRIBUTED PROPULSION

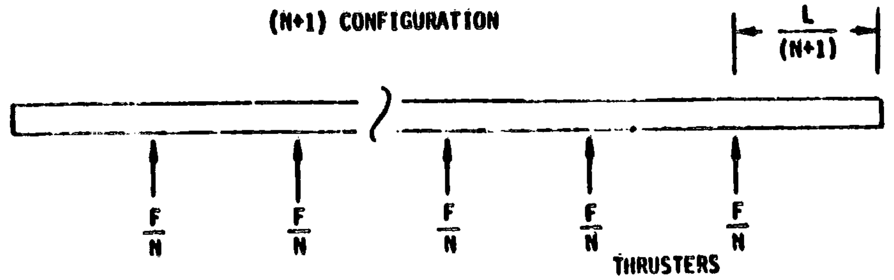
- MORE THAN MINIMUM NUMBER OF THRUSTERS
- FLEXIBLE HOMOGENEOUS VEHICLES (FLAT PLATES)
- SHAPE CONTROL DESIRABLE AND POSSIBLE WITH THRUSTERS
- VEHICLES ASSEMBLED FROM MULTIPLE SHUTTLE LAUNCHES

MULTIPLE THRUSTER DISTRIBUTIONS

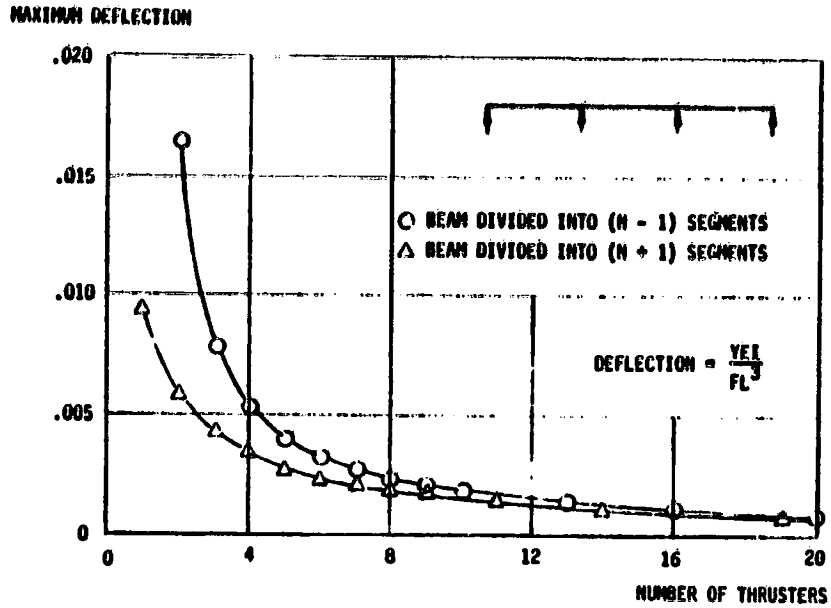
(N-1) CONFIGURATION



(N+1) CONFIGURATION



BEAM DEFLECTION DUE TO TRANSLATION



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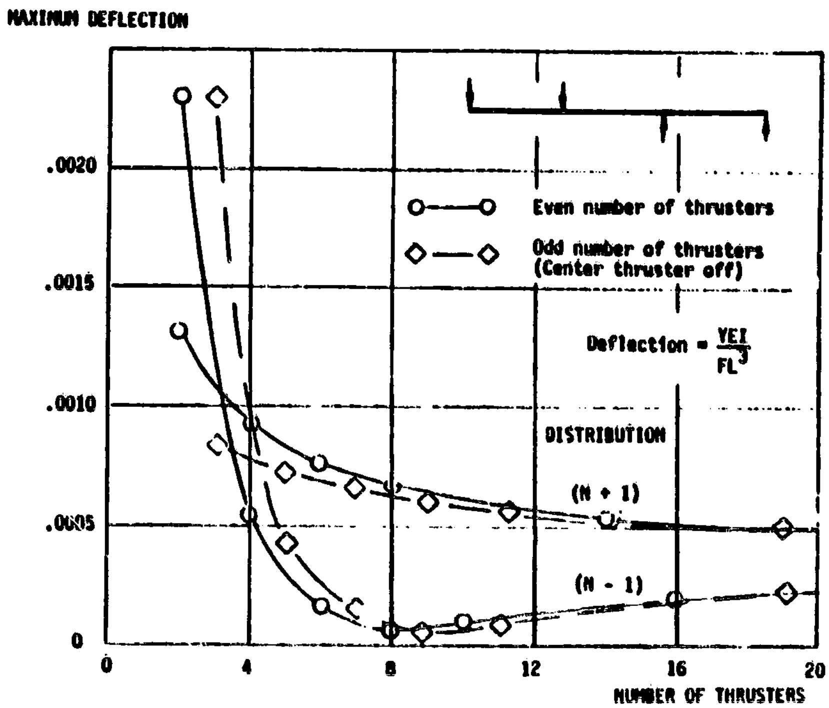
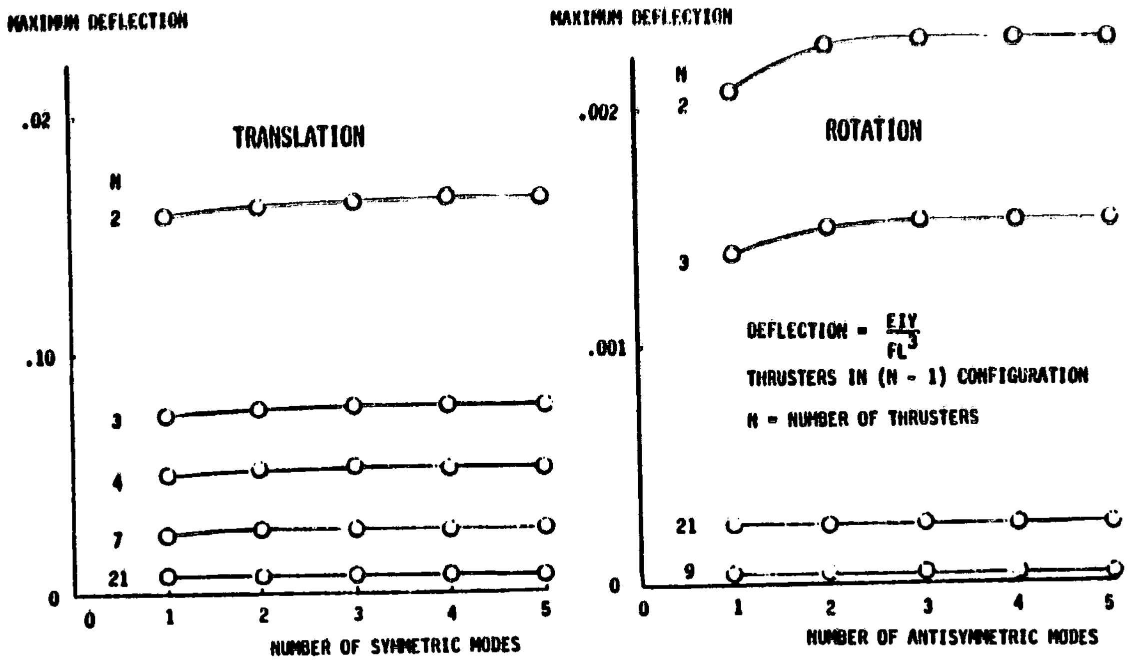
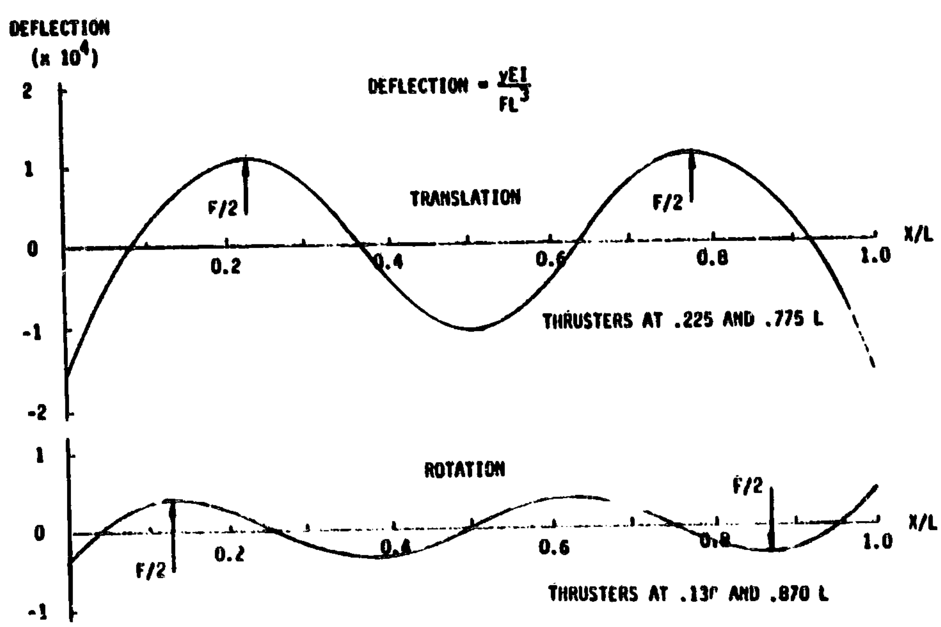
BEAM DEFLECTION DUE TO ROTATION



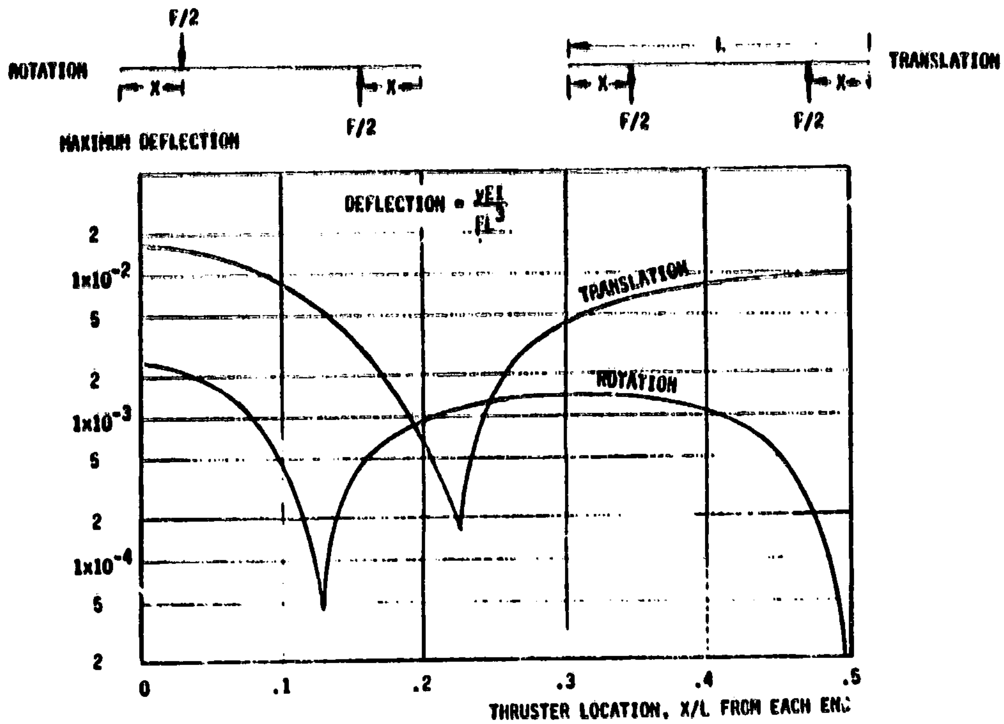
LOW ORDER MODES ONLY EXCITED BY DISTRIBUTED THRUSTERS



SHAPE OF BEAMS WITH MINIMUM DEFLECTION UNDER TWO THRUSTERS



BEAM DEFLECTIONS WITH TWO THRUSTERS ONLY



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SYSTEM WEIGHT MINIMIZATION USING DISTRIBUTED PROPULSION

ASSUME: $W_{system} = \underbrace{W_{propulsion} + W_{structure}}_{\text{MINIMIZE}} + W_{fixed}$

$W_p = KN\left(\frac{F}{N}\right)^x$ | $\bar{y} = \frac{yEI}{FL^3}$ | $W_{PN} = K'N\left(\frac{1}{N\bar{y}_N}\right)^x$

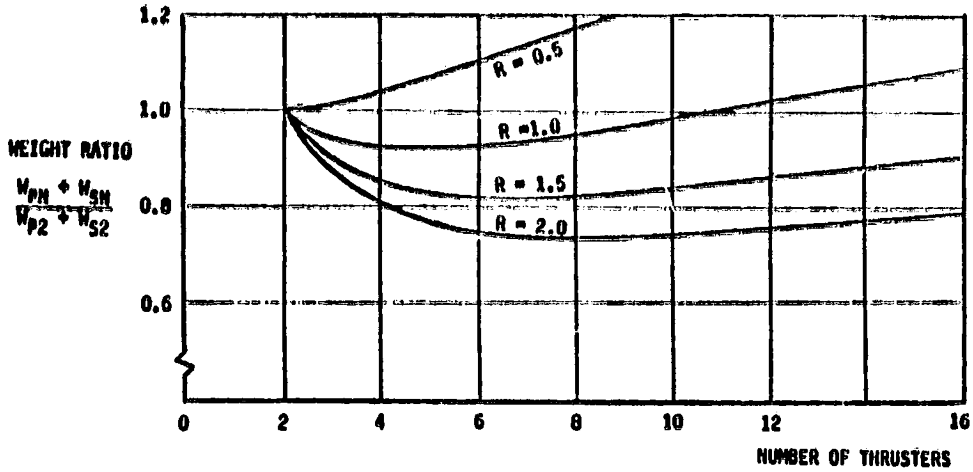
$W_S = CI$ | | $W_{SN} = C'y_N$

$$\frac{W_{PN}}{W_{P2}} = \left(\frac{N}{2}\right)^{1-x}\left(\frac{y_2}{y_N}\right)^x \qquad \frac{W_{SN}}{W_{S2}} = \left(\frac{y_N}{y_2}\right)$$

$$\frac{W_{PN} + W_{SN}}{W_{P2} + W_{S2}} = \left[\left(\frac{N}{2}\right)^{1-x}\left(\frac{y_2}{y_N}\right)^x + \left(\frac{y_N}{y_2}\right)R\right]\frac{1}{(1+R)} \qquad \text{where } R = \frac{W_{S2}}{W_{P2}}$$

IF $y_2 = y_N$ $\qquad \frac{W_{PN} + W_{SN}}{W_{P2} + W_{S2}} = \left[\left(\frac{N}{2}\right)^{1-x} + R\right]\frac{1}{(1+R)}$

MINIMUM WEIGHT SYSTEM IN TRANSLATION, ELECTRIC THRUSTERS



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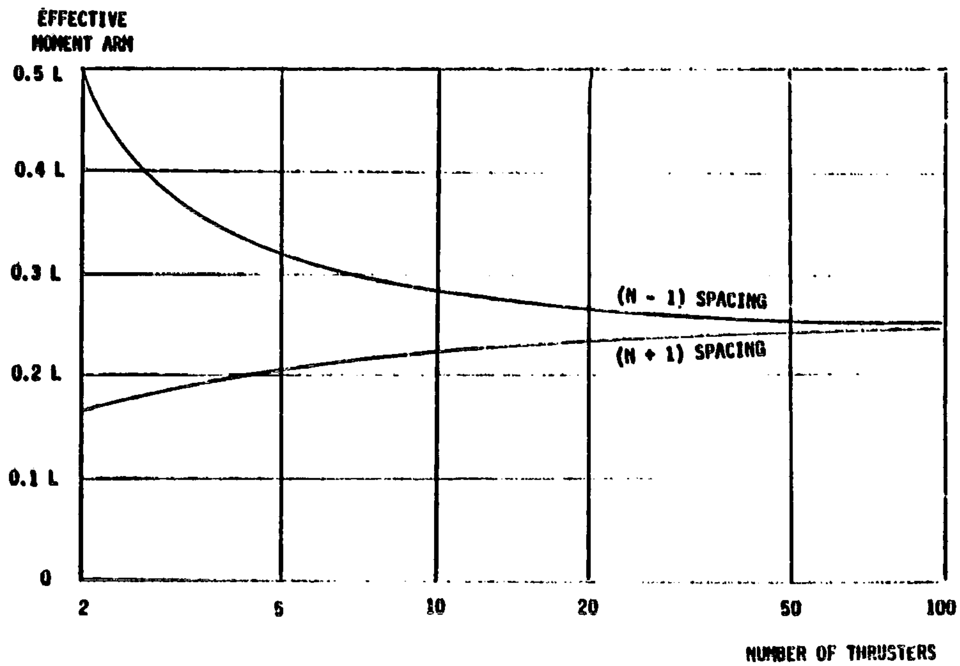
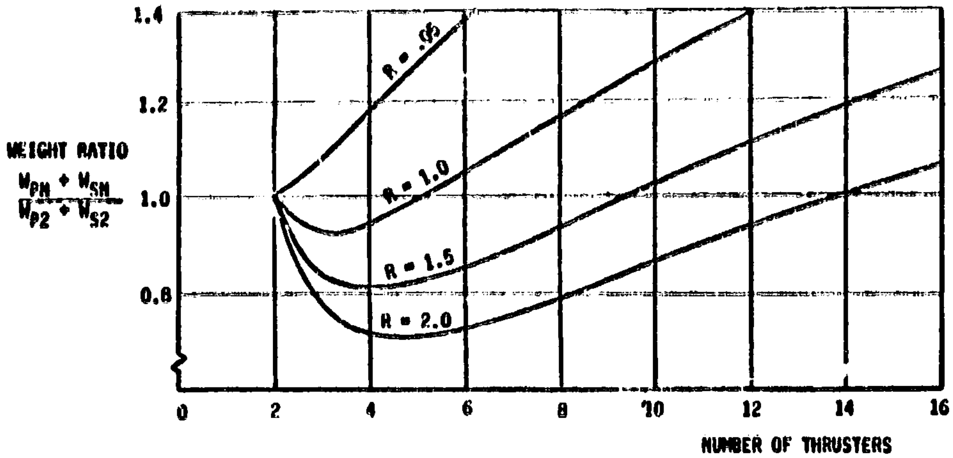
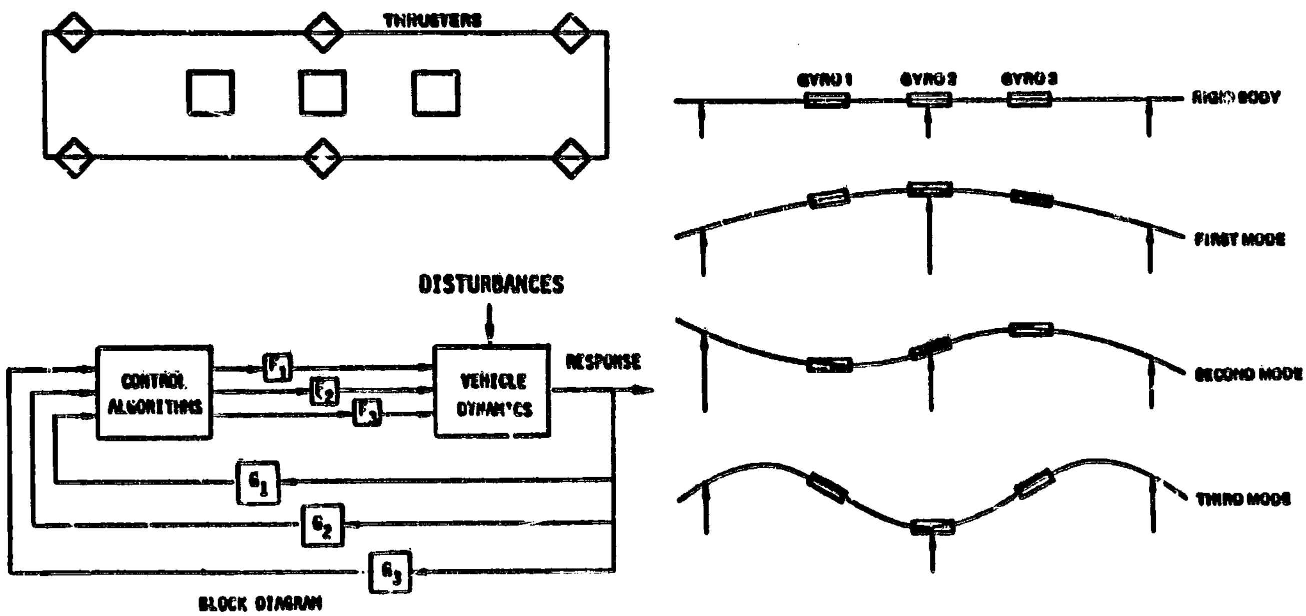
EFFECTIVE MOMENT ARM OF DISTRIBUTED THRUSTERS



MINIMUM WEIGHT SYSTEM IN ROTATION, ELECTRIC THRUSTERS



ACTIVE DAMPING USING DISTRIBUTED THRUSTERS



RESULTS OF COMBINED ATTITUDE AND SHAPE CONTROL STUDY

ATTITUDE CONTROL
- o TWO RIGID MODES
- o FULL CANCELLATION OF GRAVITY GRADIENT AND SOLAR RADIATION TORQUES

FIGURE CONTROL
- o THREE TWO-DIMENSIONAL FLEXIBLE MODES
- o NO EXCITATION OF SECOND MODE
- o SMALL EXCITATION OF FIRST AND THIRD MODES
- o GOOD FIRST AND THIRD MODE DAMPING

SENSITIVITY
- o SYSTEM STABLE AND INSENSITIVE TO MISPLACEMENT OF SENSORS

ISSUES

- o OPTIMUM DISTRIBUTION OF THRUSTERS
  - o NUMBER
  - o LOCATION
- o THREE DIMENSIONAL SYSTEMS
- o MINIMUM WEIGHT SYSTEMS
- o CONTROL SYSTEM MECHANIZATION
  - o SENSOR TYPES
  - o SENSOR LOCATION
  - o CONTROL LAWS
  - o REQUIREMENTS AND CONSTRAINTS
  - o SENSITIVITY
- o CONTROL OF VARIABLE CONFIGURATION VEHICLES

SYSTEM REQUIREMENTS

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George C. Marshall Space Flight Center
Marshall Space Flight Center, AL 35812

Requirements of future space systems, including large space systems, that operate beyond the space shuttle are discussed in this paper. Typical functions required of propulsion systems in this operational regime include: payload placement, retrieval, observation, servicing, space debris control and support to large space systems. These functional requirements are discussed in conjunction with two classes of propulsion systems: (1) primary or orbit transfer vehicles (OTV) and (2) secondary or systems that generally operate within or relatively near an operational base orbit. Three propulsion system types are described in relation to these requirements: Cryogenic OTV, Teleoperator Maneuvering System (TMS) and a solar electric OTV.

Operations purely in low earth orbit are described in relation to the TMS since LEO does not lend itself to an effective application of a solar electric type of system. Both TMS and solar electric systems are candidates in high or geosynchronous orbits. Payload placements and retrieval, as well as sub-satellite operations are considered early operational requirements for a TMS. Later operations may include payload servicing for both conventional and large space systems, and space debris control. Servicing operations range from module changeout on small spacecraft and changeout of experiment pallets on a space platform in LEO to replenishing expandables and updating operational capabilities of large space platforms in GEO.

Space debris control can be divided into two major categories: (1) low and intermediate orbits, and (2) geosynchronous orbit. Effective space debris control in low and intermediate orbits poses a very high energy requirement due to the varying orbital parameters of every object in this regime. Early debris control in this region may be accomplished with a TMS for objects that are near co-orbital with the TMS. GEO debris control is considerably more manageable since all the objects are near co-orbital. The integrated solution of this problem tends to favor a high energy propulsion system like a solar electric OTV although it is within the general capability of a conventional system like a TMS.

Large space systems such as advanced communication platforms, science and application platforms, and space-based radar will pose two primary classes of requirements on primary propulsion system (OTV): (1) high performance and (2) possible low thrust acceleration. These systems, as currently envisioned, are to be deployed in LEO, checked out and then transferred to GEO or other high orbits. It is this aspect of these systems and their operational modes that currently infers low thrust acceleration for the OTV. There is need for considerable debate in this area since the resultant OTV program would of necessity be required to include the capability for a wide throttle range on a high thrust engine or two engines - a low thrust and a high thrust engine. Since these large space systems also impose high performance requirmeents on the OTV, the likely choice will be to have two separate engines in the program. The low thrust engine would be single purposed and due to the very nature of the large space system approach, would be used infrequently. The cost leverage of this approach requires careful scrutiny by the large space systems and propulsion system communities. It should be pointed out that the structural loads associated with: transport to LEO, deployment and checkout in LEO, and/or assembly in LEO may impose a structural capability in the system to withstand transfers to high Earth orbits with high thrust engines with reasonable throttle ranges that maintain high performance.

In summary, the key interactions between the future large space systems and propulsion systems are: growth capability in performance, maximizing length available for payloads in shuttle cargo bay, capability to provide servicing and debris control, and very importantly transportation economy.

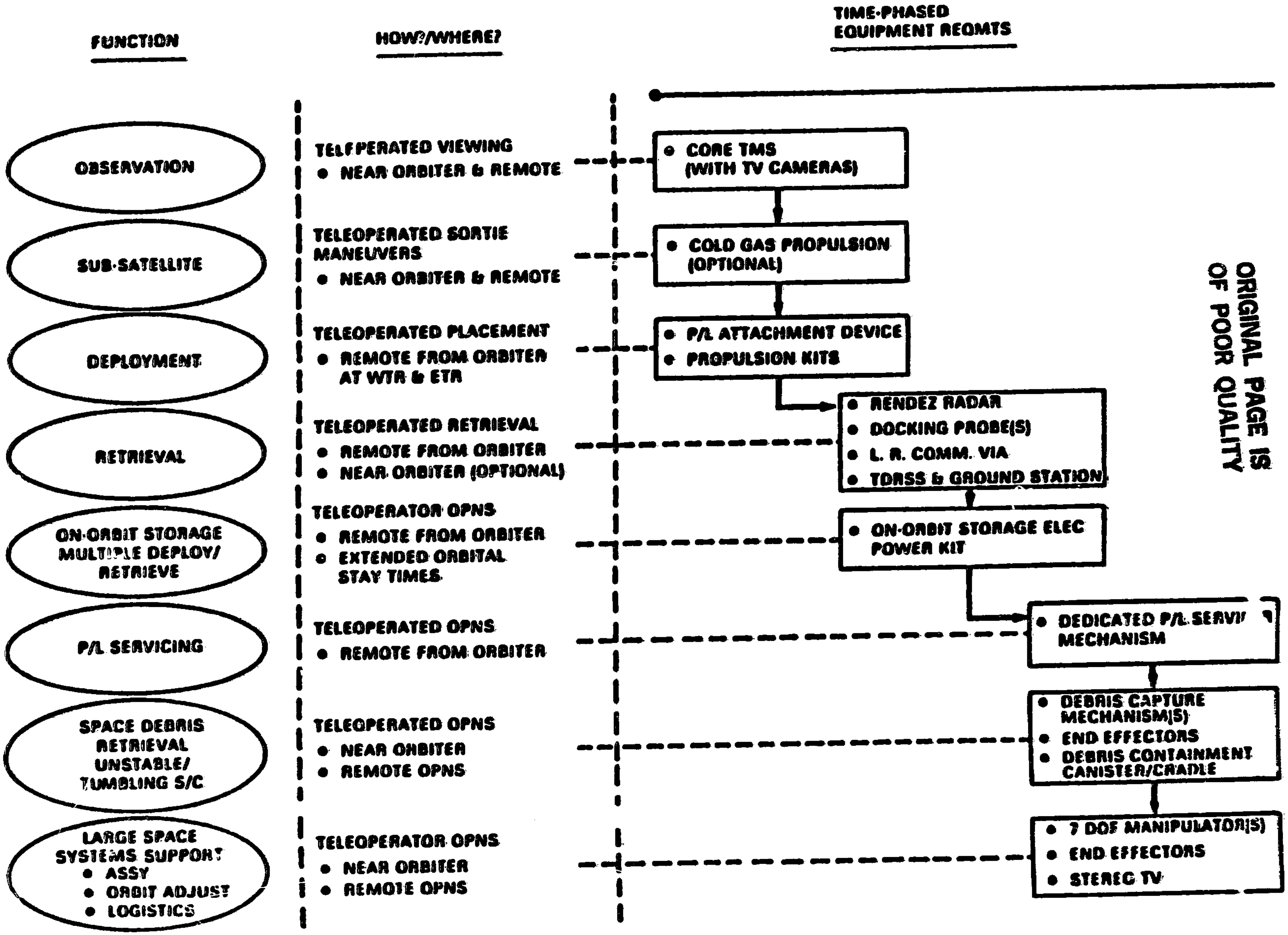
DISCUSSION TOPICS

- ORBITAL SERVICES REMOTE TO THE ORBITER
  - FUNCTIONS
  - RATIONALE
- ORBITAL SERVICES VIA TELEOPERATORS MANEUVERING SYSTEM (TMS)
  - TMS DESCRIPTION
  - EVOLUTION OF CAPABILITIES
- SPACE DEBRIS AT GEOSYNCHRONOUS
- POTENTIAL LARGE SPACE SYSTEM APPLICATIONS
- GEOSTATIONARY PLATFORM
  - DEPLOYMENT IN LEO
  - TRANSFER OPTIONS TO GEO (ORBIT TRANSFER VEHICLE)
- SHUTTLE UPPER STAGE REQUIREMENTS (NASA MISSION MODEL)
- SUMMARY

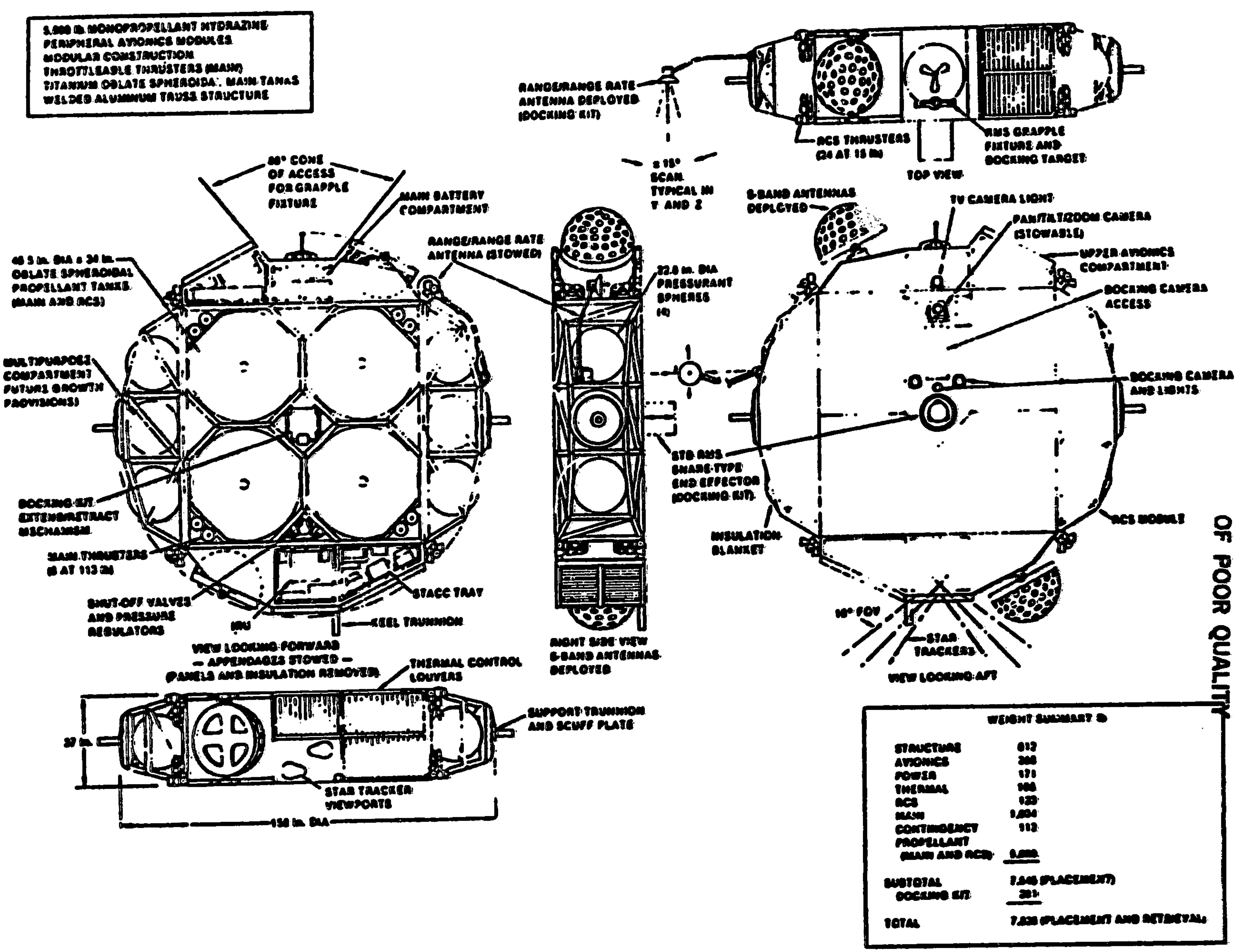
ORBITAL SERVICES

TYPES	FUNCTIONS	WHY NEEDED
SATELLITE SERVICES– REMOTE	• DEPLOYMENT	– REUSEABLE PLACEMENT OF PAYLOADS UP TO 1800 KM FROM ORBITER, OR ANYWHERE IN GEOSTATIONARY ORBIT
	• RETRIEVAL	– REUSEABLE REMOTE RETRIEVAL REDUCING INTEGRAL S/C PROPULSION BURDEN
	• OBSERVATION	– REMOTE AND NEAR OBSERVATION FOR OPS, SAFETY, AND DAMAGE ASSESSMENT
	• SUB-SATELLITE OPS	– IN-ORBIT SCIENTIFIC INVESTIGATIONS
	• MULTIPLE P/L DEPLOY/RETRIEVE	– PERMITS MANIFESTING FLEXIBILITY AND REDUCES S/C INTEGRAL PROPULSION BURDEN
	• SERVICING	– MAINTAIN REMOTE ORBITAL SERVICES ON-LINE, UPGRADING.
	• SPACE DEBRIS CONTROL	– CLEAR SPACE LANES OF LARGE DEBRIS
	• LARGE SPACE SYSTEMS	– SUPPORT ASSEMBLY LOGISTICS, ORBITAL

SATELLITE SERVICES REMOTE— TELEOPERATOR MANEUVERING SYSTEM (TMS)



TMS General Arrangement



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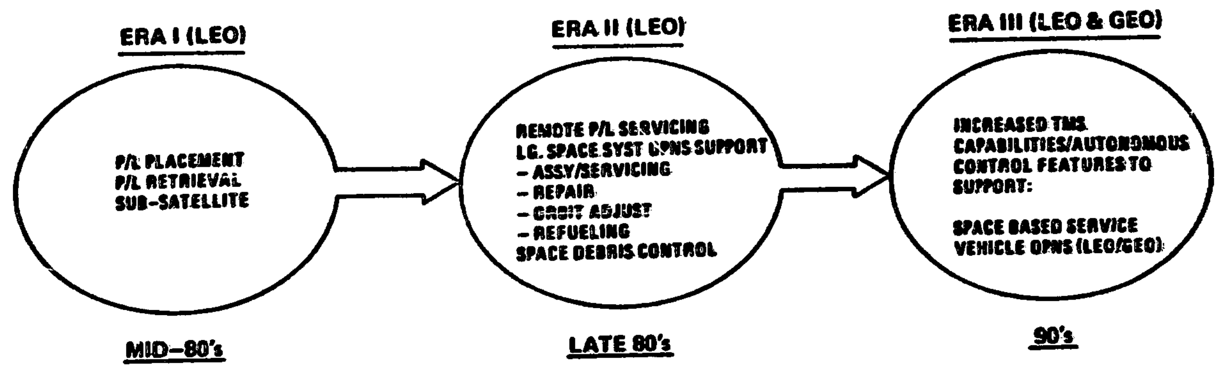
REMOTE SATELLITE SERVICES PROGRAM EVOLUTION
TELEOPERATOR MANEUVERING SYSTEM (TMS) PROGRAM SCOPE



TMS SYSTEMS APPROACH

- TMS MODULAR DESIGN ("KITTABLE") TO SUPPORT EVOLVING NEEDS
  - –GENERAL PURPOSE UTILITY VEHICLE, TMS WILL INTERACT WITH MANY FUTURE SPACE PROGRAM HARDWARE ELEMENTS

TMS BENEFITS

- TMS COMPLEMENTS STS SERVICES TO USERS:
  - –PROPULSIVE PERFORMANCE
  - –CONTINGENCY RETRIEVAL
  - –REMOTE SPACECRAFT SERVICES
  - –OPERATIONS SUPPORT WHERE STS RMS REACH/PLUME IMPINGEMENT CONSTRAINTS PREVAIL

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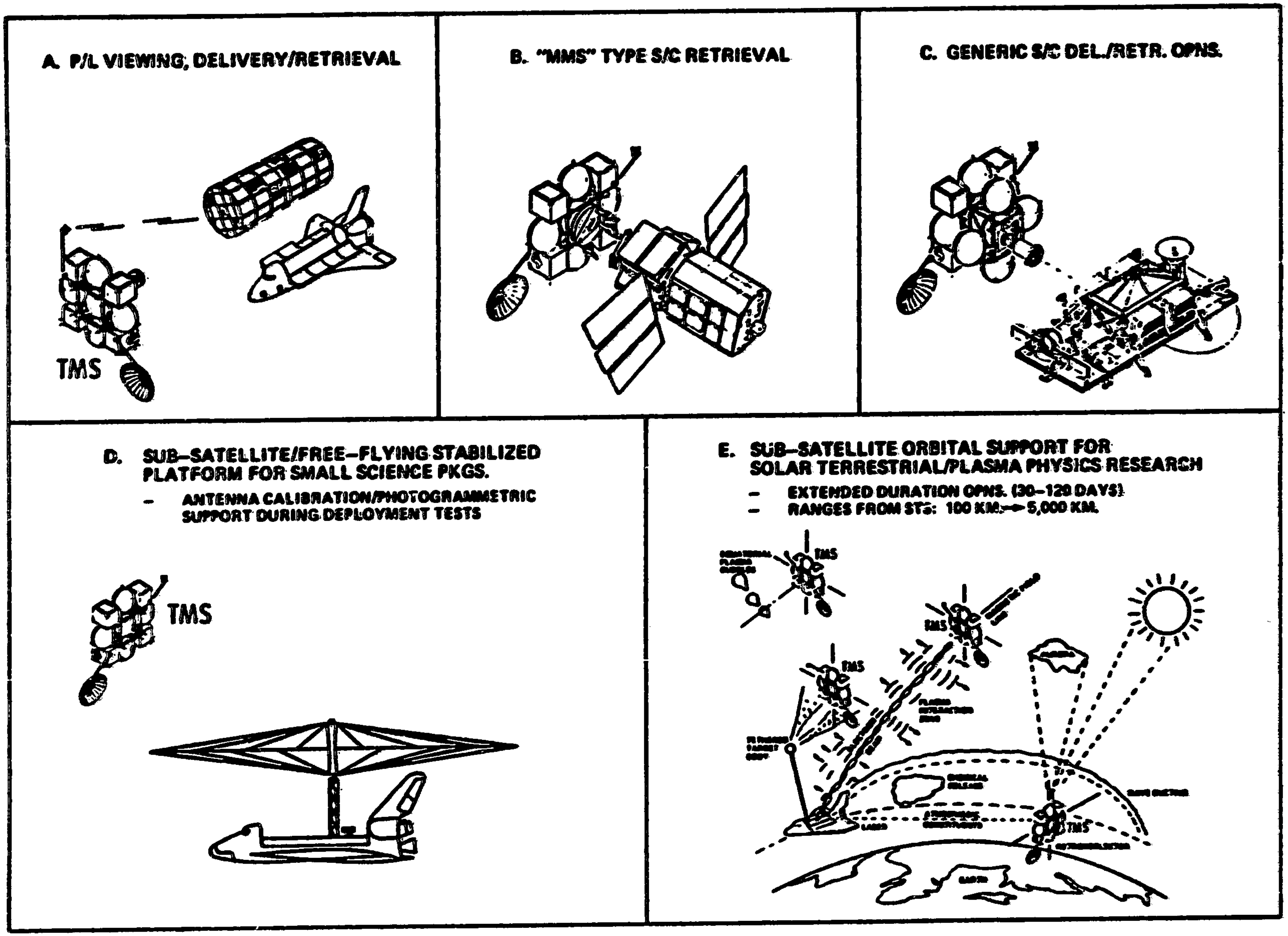
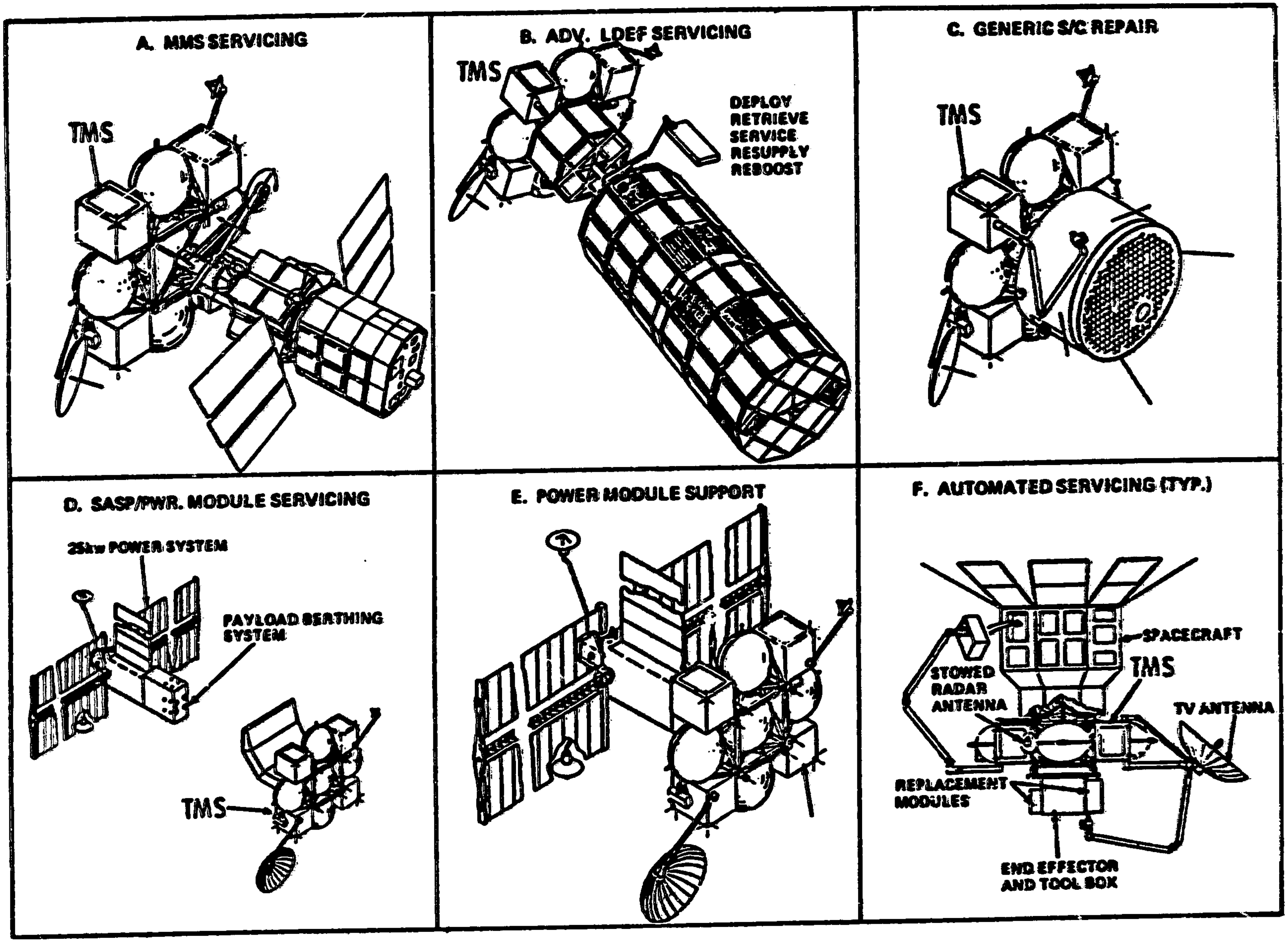
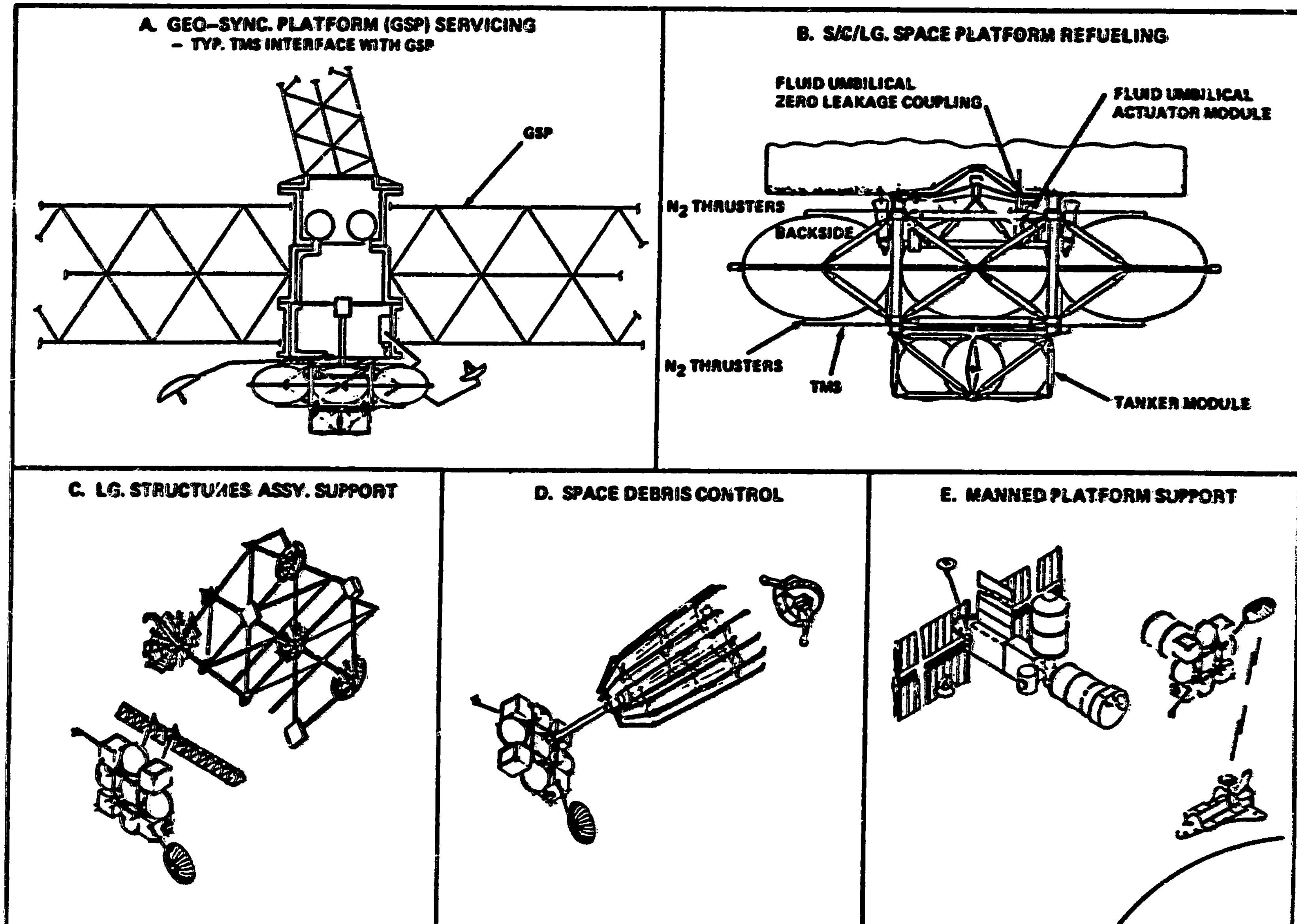
ERA I: REPRESENTATIVE MISSION APPLICATIONS FOR TMS (MID 80'S)



ERA II: REPRESENTATIVE MISSION APPLICATIONS FOR TMS (MID-LATE 80's)



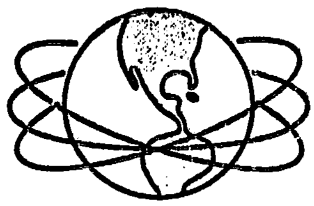
ERA II/III: REPRESENTATIVE MISSION APPLICATIONS FOR TMS (LATE 80'S - 90'S)



SPACE DEBRIS PROBLEM AT GEOSYNCHRONOUS ORBIT

- BY 1996 APPROXIMATELY 620 SPACECRAFT AND PIECES OF DEBRIS WILL BE IN SYNCHRONOUS ORBIT
- THE GREATEST DENSITY IS IN NORTH AMERICAN SECTOR (~ 240 IN THE 80° – 140° W LONGITUDE SEGMENT)
- PAYLOADS HAVE NO COLLISION ANTICIPATION/AVOIDANCE CAPABILITY
- ALL SPACECRAFT AND DEBRIS WITHOUT ACTIVE STATION KEEPING WILL CONSTITUTE A MOVING HAZARD IN SYNCHRONOUS ORBIT

EARTH TRIAXIALITY



SUN-MOON PERTURBATION

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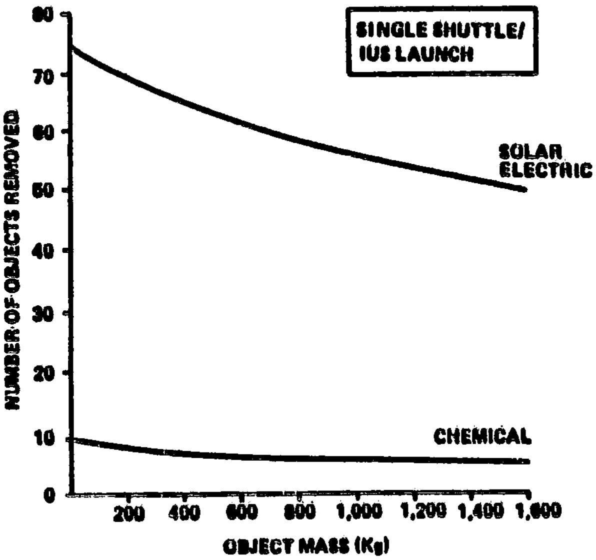
SPACE DEBRIS CONTROL AT GEOSYNCHRONOUS ORBIT



SPACE DEBRIS CONTROL

- CONTROL INCREASING HAZARD OF COLLISION AT GEOSYNCHRONOUS ORBIT
  - NUMBER OF OBJECTS IN GEOSYNCHRONOUS ORBIT IS INCREASING
  - INACTIVE OBJECTS HAVE DRIFT RATES
  - NORTH AMERICAN SECTOR BECOMING CROWDED
- MOVE INACTIVE OBJECTS TO HIGHER ORBIT (NON INTERFERING)

ALTERNATIVE VEHICLES

- LIQUID CHEMICAL (STORABLE) SYSTEM
  - SINGLE SHUTTLE/IUS LAUNCHED SYSTEM LIMITED TO 6 TO 10 OBJECTS
- SOLAR ELECTRIC
  - SINGLE SHUTTLE/IUS LAUNCHED SYSTEM PROVIDES FACTOR OF TEN INCREASE IN CAPABILITY OVER CHEMICAL SYSTEM
  - LONG TERM RESISTANCY: 5 OR MORE YEARS
  - ELECTRICAL POWER AND OTHER SERVICES FOR OBSERVATION EQUIPMENT

POTENTIAL LARGE SPACE SYSTEM APPLICATIONS

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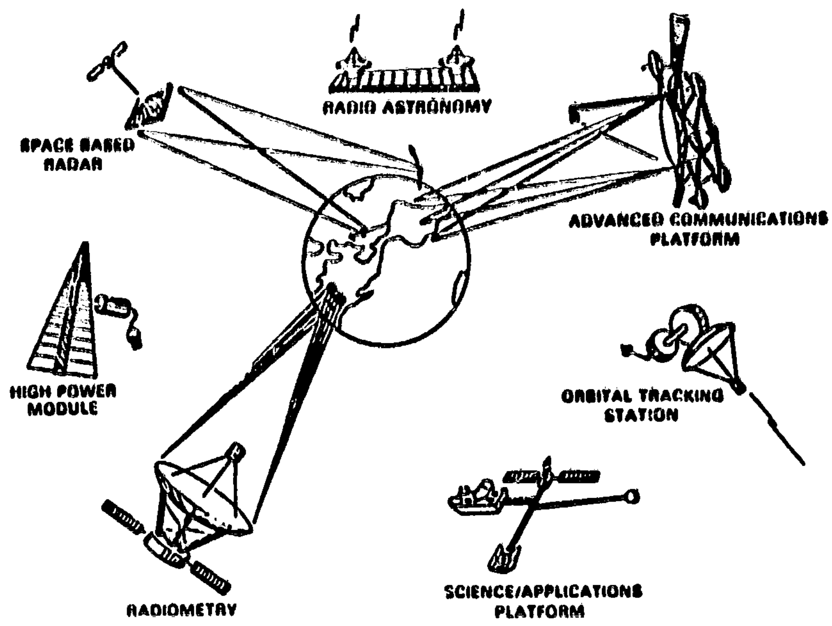
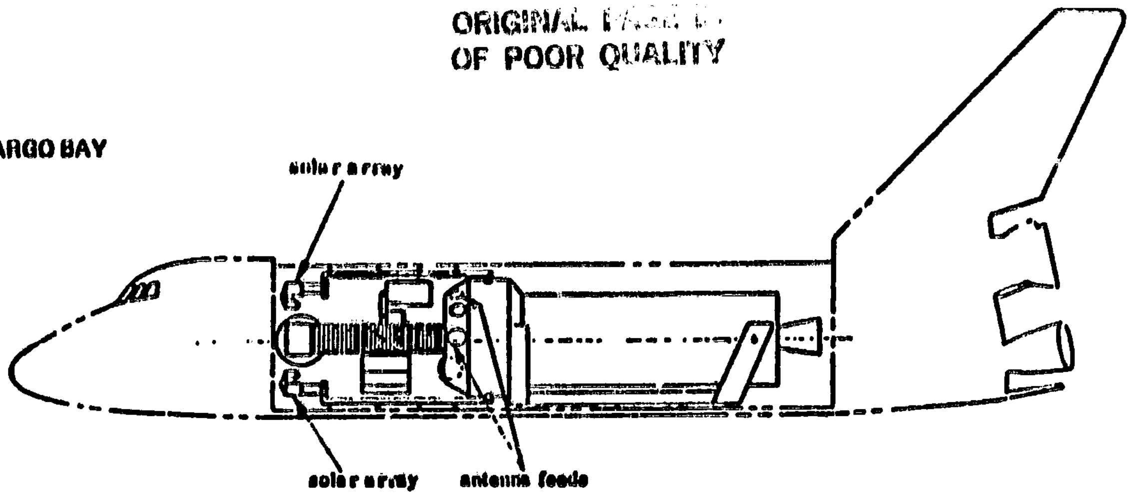


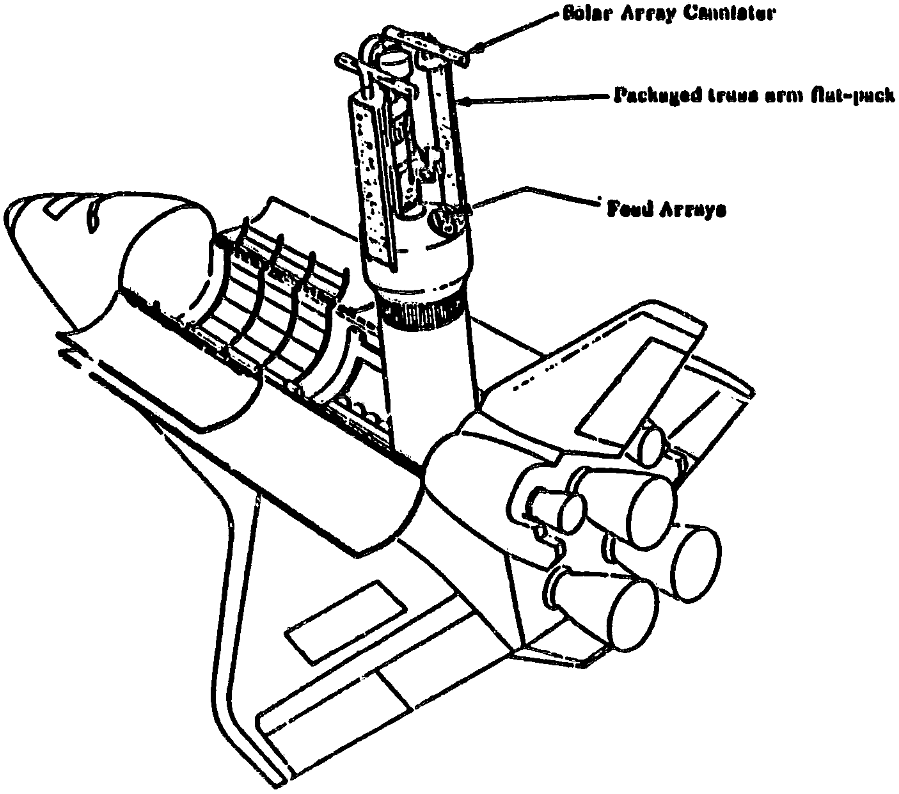
EXPERIMENTAL GEOSTATIONARY PLATFORM

GEOSTATIONARY PLATFORM DEPLOYMENT SEQUENCE

1. PACKAGED IN CARGO BAY



2. ROTATED OUT OF CARGO BAY



GEOSTATIONARY PLATFORM DEPLOYMENT SEQUENCE (CONT'D)

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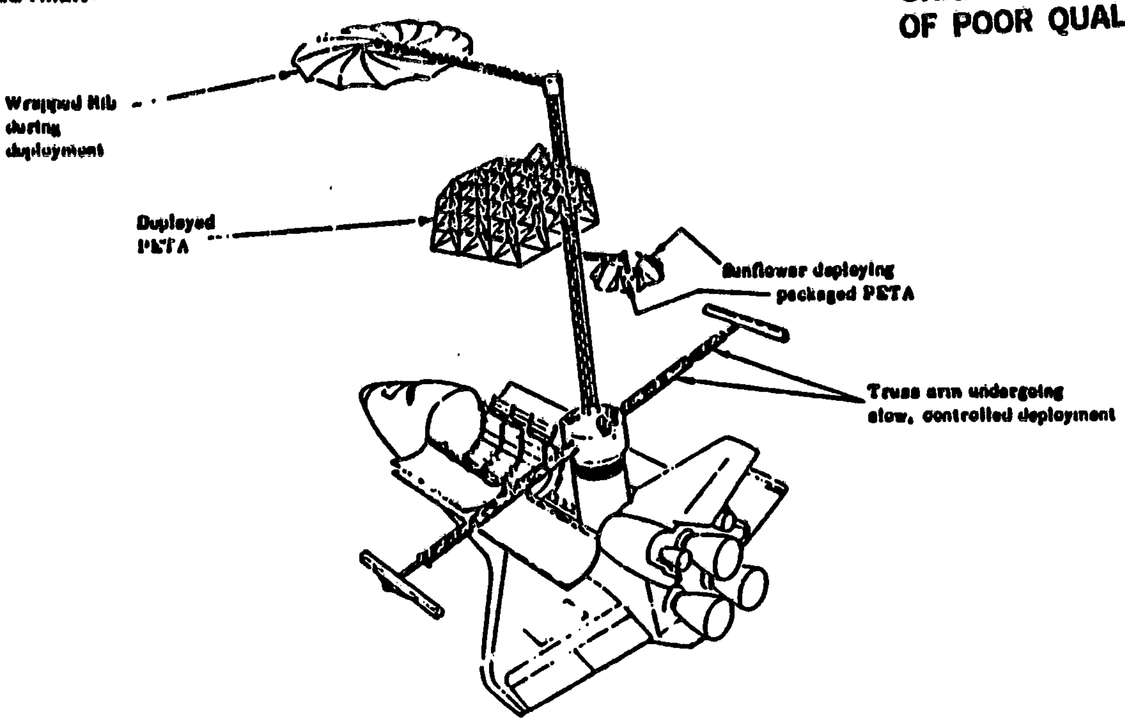
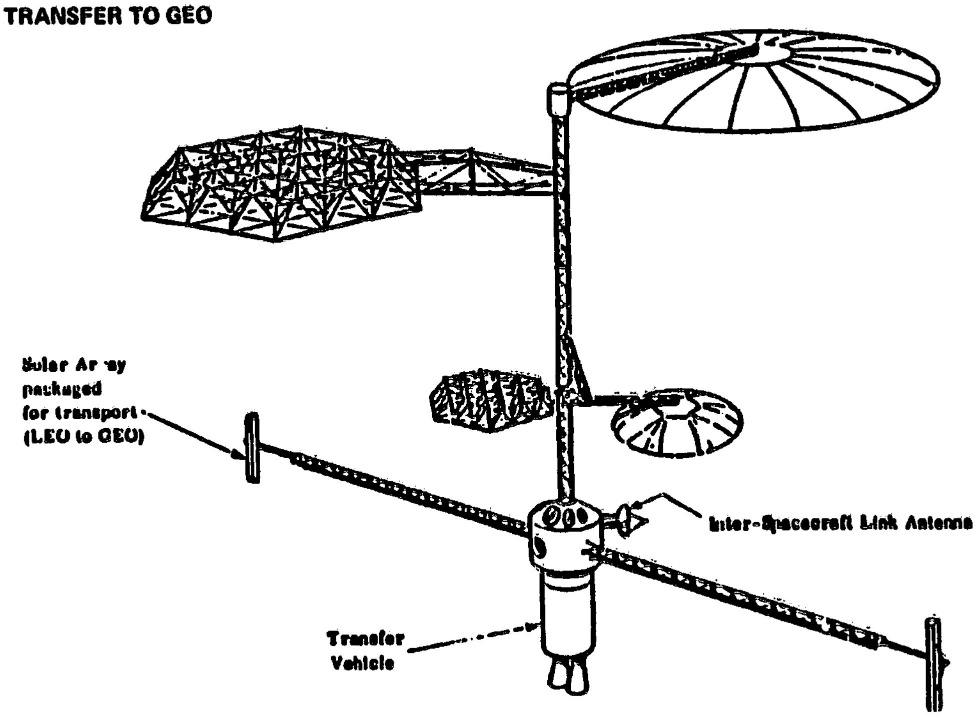
3. PLATFORM DEPLOYMENT



4. READY FOR TRANSFER TO GEO



GEOSTATIONARY PLATFORM
OTV PERFORMANCE REQUIREMENTS

- PLATFORM DELIVERY MISSION
  - ▲ 6895 kg DELIVERY CAPABILITY, LEO TO GEO
    14,000 FT/SEC ΔV
    ±0.1 km OF TARGET POSITION
    0.07 G T/W MAX

- SERVICING MISSION
  - ▲ 2267 kg DELIVERY CAPABILITY, LEO TO GEO
    14,000 FT/SEC ΔV
    ±0.1 km OF TARGET POSITION
    NO T/W CONSTRAINT

  - ▲ 822 kg RETURN CAPABILITY, GEO TO LEO
    14,500 FT/SEC ΔV
    NO T/W CONSTRAINT

GEOSTATIONARY PLATFORM
OTV SUBSYSTEMS SUPPORT REQUIREMENTS

- ENVIRONMENTAL CONTROL
  NO REQUIREMENT

- POWER
  700 WATTS DURING PRELAUNCH (GYRO ACTIVATION)

- COMMUNICATIONS
  I/F BETWEEN PLATFORM AND ORBITER; S-BAND
  32 KBPS UPLINK, 6.4 KBPS COMMAND
  192 KBPS DOWNLINK, 128 KBPS DATA

- PROPULSION
  6895 kg CAPABILITY FROM LEO TO GEO, PLATFORM DELIVERY MISSION
  14,500 FT/SEC ΔV FOR TRANSFER, +415 FT/SEC INTO DEBRIS ORBIT
  MAX T/W: 0.07
  ELEVEN BURNS MAX
  Isp > 415 SEC

  2267 kg CAPABILITY FROM LEO TO GEO, SERVICING MISSION
  28,500 FT/SEC ΔV FOR TRANSFER & RETURN
  10,000 TO 20,000 LB THRUST
  Isp > 460 SEC

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GEOSTATIONARY PLATFORM

OTV SUBSYSTEMS SUPPORT REQUIREMENTS

GUIDANCE & NAVIGATION

- PLATFORM DELIVERY MISSION
  - ▲ DELIVERY CAPABILITY FROM LEO TO 96 km BEHIND, 16 km BELOW GEOSYNCHRONOUS TARGET LOCATION, ±8 km (RENDEZVOUS POINT)
  - ▲ APPROACH TO TARGET LOCATION
    - ± 0.1 km
    - < 3 cm/SEC RESIDUAL TRANSLATIONAL VELOCITY
    - ±0.05°/SEC ROTATIONAL VELOCITY
    - ±1° OF REQUIRED ATTITUDE
- SERVICING MISSION
  - ▲ DELIVERY CAPABILITY FROM LEO TO 96 km BEHIND, 16 km BELOW GEOSYNCHRONOUS TARGET LOCATION, ±8 km (RENDEZVOUS POINT)
  - ▲ APPROACH TO TARGET LOCATION (CENTER OF CONSTELLATION)
    - ±0.1 km
    - ±1 m/SEC RESIDUAL TRANSLATIONAL VELOCITY
    - ±1.0° OF REQUIRED ATTITUDE
  - ▲ STATIONKEEP
  - ▲ RENDEZVOUS AND REDOCK WITH SERVICE MODULE
    - ±1° SUN ORIENTATION
    - ±3 cm/SEC RESIDUAL TRANSLATIONAL VELOCITY
    - ±0.05°/SEC ROTATION
  - ▲ RENDEZVOUS WITH ORBITER
    - RETURN CAPABILITY FROM GEO TO 96 km FWD, 16 km ABOVE THE ORBITER, ±8 km (RENDEZVOUS POINT)
  - ▲ APPROACH TO ORBITER (SAME TOLERANCES AS GEO TARGET)
  - ▲ STATIONKEEP
    - ±1° SUN ORIENTATION
    - ±3 cm/SEC RESIDUAL TRANSLATIONAL VELOCITY
    - ±0.05°/SEC ROTATION
  - ▲ PASSIVE DURING FINAL DOCKING

GEOSTATIONARY PLATFORM

TMS SUBSYSTEMS SUPPORT REQUIREMENTS

• AVIONICS	COMMAND & CONTROL (VIA GROUND-PLATFORM-TMS RF LINK) VIDEO (STEREO) – ARTICULATION AND PAN LIGHTING – WITH VIDEO REQUIREMENT, FOR LOGISTICS RESUPPLY AND FOR DOCKING RENDEZVOUS RADAR – TMS, ACTIVE; OTV AND PLATFORM, PASSIVE
• PROPULSION/RCS	42 FT/SEC ΔV FOR OTV/PLATFORM TRANSLATION ATTITUDE CONTROL FOR TRANSLATION AND ROTATION IN 3 AXES MINIMUM EXHAUST PRODUCT CONTAMINATION – COLD GAS PREFERRED
• ENVIRONMENTAL CONTROL	NO REQUIREMENT
• STABILIZATION & CONTROL	±3 cm/SEC TRANSLATIONAL VELOCITY ±2° CENTERLINE ORIENTATION ±5° ROLL ORIENTATION

DESIRED FEATURES
SHUTTLE UPPER STAGE

- SHUTTLE COMPATIBILITY
- CAPABILITY TO MEET MISSION REQUIREMENTS
- FLEXIBILITY FOR MISSION OPERATIONS
- HIGH RELIABILITY
- CAPABLE OF EVOLUTIONARY GROWTH
- LOW COST

OTV DESIGN CRITERIA
MISSION REQUIREMENTS

- PERFORMANCE AND FUNCTIONAL CAPABILITY WITH MARGINS TO PERFORM PLANETARY MISSIONS
- HIGH PERFORMANCE TO GEO AND OTHER HIGH ENERGY ORBITS
- CAPABLE OF SUBSTANTIAL PAYLOAD IN REUSABLE DELIVERY MODE (POTENTIALLY 36% OF MISSION MODEL)
- CAPABILITY TO DELIVER LARGE FLEXIBLE SPACECRAFT WITH LOW ACCELERATION (POTENTIALLY 25% OF MISSION MODEL) - LOW THRUST, MULTIPLE BURN, LONG DURATION
- MINIMUM PRACTICAL STAGE LENGTH WITH FULL PROPELLANT LOADING AND/OR ADAPTABLE TO SHORT-LENGTH VERSION - MAXIMUM PAYLOAD LENGTH
- T/W AND NUMBER OF ENGINES CONSISTENT WITH PERFORMANCE AND MISSION SUCCESS REQUIREMENTS
- THERMAL CONTROL CONSISTENT WITH MISSION DURATIONS
- ENABLE ROUNDTRIP SORTIE MISSIONS TO HIGH EARTH ORBITS
- MUST BE OPERATIONALLY AVAILABLE WHEN NEEDED
- PROVIDE SPACECRAFT 3-AXIS STABILIZATION

OTV DESIGN CRITERIA
HIGH RELIABILITY

- PROVIDE HIGH RELIABILITY, HIGH ACCURACY AUTONOMOUS OPERATIONS
  - DUAL STRING AVIONICS WITH COMPUTER CAPABLE OF HANDLING REDUNDANCY MANAGEMENT, ADVANCED GUIDANCE SCHEMES, AND SPACECRAFT INTERFACE SUPPORT
  - BODY SHELL (METEOROID PROTECTION)
- CURRENT STATE OF THE ART
  - AVIONICS
  - RCS
  - MAIN ENGINE

OTV DESIGN CRITERIA
EVOLUTIONARY GROWTH

- ADAPTABLE TO SHORT STAGE FOR LONG PAYLOADS
- CAPABLE OF INCORPORATING RETROFIT AEROASSIST KIT
- CAPABLE OF INCORPORATING ADVANCED ENGINE
- CAPABLE OF ACHIEVING HIGHER PERFORMANCE WITH GROWTH IN SHUTTLE LIFT CAPABILITY
- CAPABLE OF PERFORMING HEAVY DELIVERY AND MANNED MISSIONS TO HEO EITHER THRU MULTIPLE STAGING ON A 65K STS OR WITH A SINGLE STAGE ON A 100-110K STS
- READILY MAN-RATED
- CAPABLE OF EXTENDED DURATION IN SPACE

MISSION DERIVED REQUIREMENTS
NOMINAL MISSION MODEL, REVISION 4

PAYLOAD CHARACTERISTICS / MISSIONS	MASS (LBS)	LENGTH	"G" LIMIT	ON-ORBIT Δ V	NUMBER	TIMING
LONG GEO SATELLITES	5,000 – 10,000 VARIES INVERSELY W/LENGTH	42' – 30'	N/A	---	12	1987 → 2000
SHORT GEO SATELLITES	3,000 – 7,000	≦ 25'	N/A	---	36	1987 → 2000
EXPERIMENTAL GEO PLATFORM	12,500	≦ 25'	0.15 – 1.0G	---	1	1989
OPERATIONAL PLATFORMS	10,000 – 15,000	≦ 30'	0.05 – 0.2G	≤ 65 FPS	24	1988 → 2000
VERY LARGE PLATFORMS	15,000 – 30,000	≦ 2 X 60'	0.05 – 1.0	---	9	1990 → 2000
UNMANNED SERVICING SORTIE	6,000 → 8,000 UP 0 → 2,000 DOWN	< 25'	N/A	600 FPS	29	1991 → 2000
MANNED SORTIE	UP TO 12,000	< 20'	N/A	115 FPS	11	1994 → 2000

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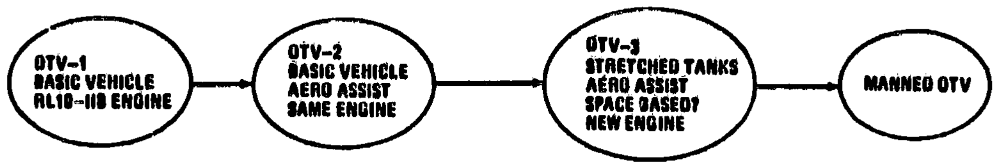
OTV DESIGN CRITERIA
MISSION DURATION

MISSION TYPE/MODE	DURATION REQUIRED	NO. OF MISSIONS
PLANETARY INJECTION	22 HR	9
MULTIPLE PAYLOAD DELIVERY	39 HR	10
EXPENDABLE GEO DELIVERY	41 HR*	(12)
REUSABLE GEO DELIVERY	62 HR	38
LOW THRUST GEO DELIVERY	90 HR	33
UNMANNED SERVICING	194 HR	29
MANNED SORTIE (7 DAYS)	150 HR	11
MANNED SERVICING (21 DAYS)	500 HR	0
MULTIPLE LAUNCH (3 LAUNCH)	>200 HR	1
		131

*54 HRS IF STAGE IS DISPOSED OF AT GEO + 2000 NM

REUSABLE ORBIT TRANSFER VEHICLE (OTV) EVOLUTION

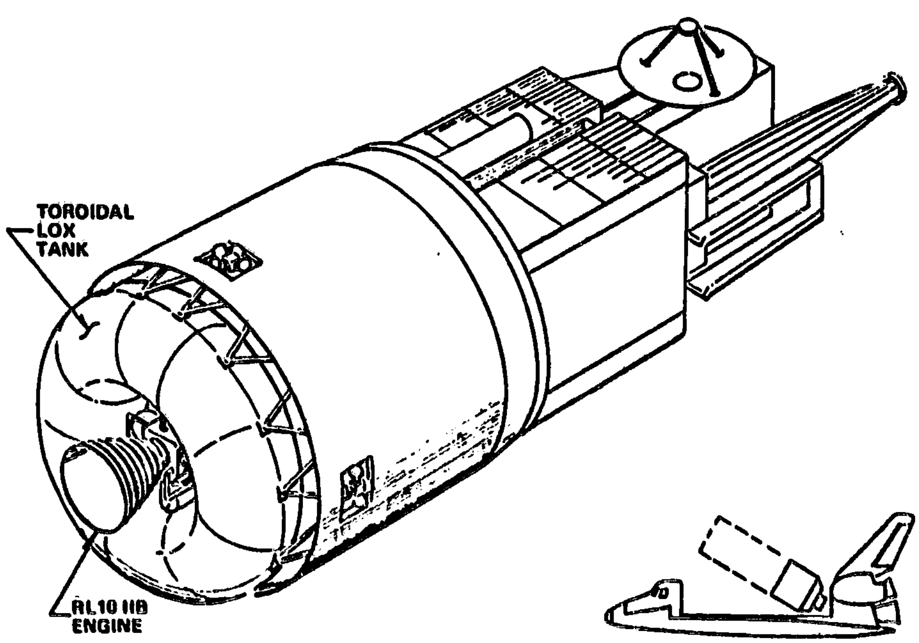


•OTV & PAYLOAD: SAME LAUNCH	•OTV & PAYLOAD: SAME LAUNCH	•OTV & PAYLOAD: SPLIT LAUNCH	•2 MAN SORTIE TO GEO
•SIZED FOR 65K STS	•SIZED FOR 65K STS	•SIZED FOR AUGMENTED STS	•AUGMENTED STS
•DELIVERY ONLY	•AEROBRAKE: DELIVERY ONLY	•COULD BE SPACE BASED	•SPACE BASED

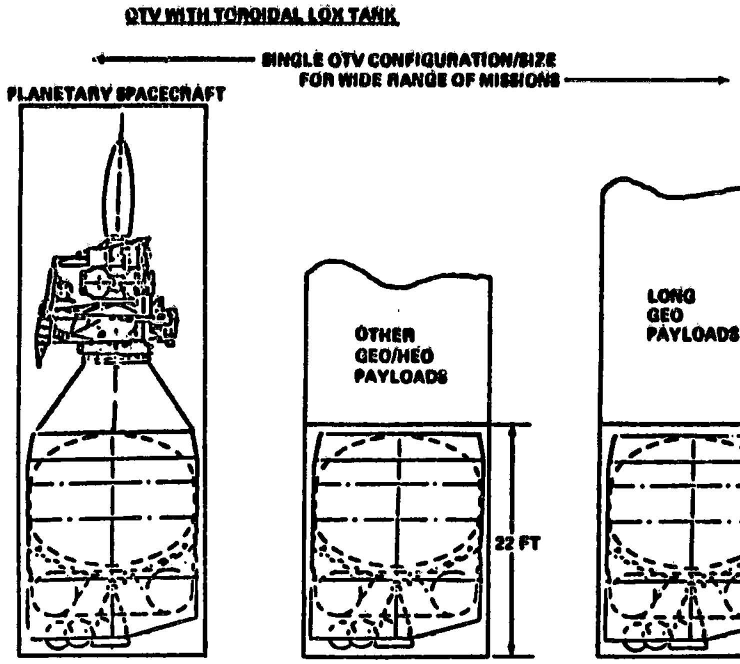
PAYLOAD POTENTIAL (POUNDS)

	OTV-1	OTV-2	OTV-3	MOTV
DELIVERY	6K	12K	14K	–
RETRIEVE	–	–	16K	–
ROUNDTRIP	–	–	8K	12K
EXPEND	16K	17K	27K	–

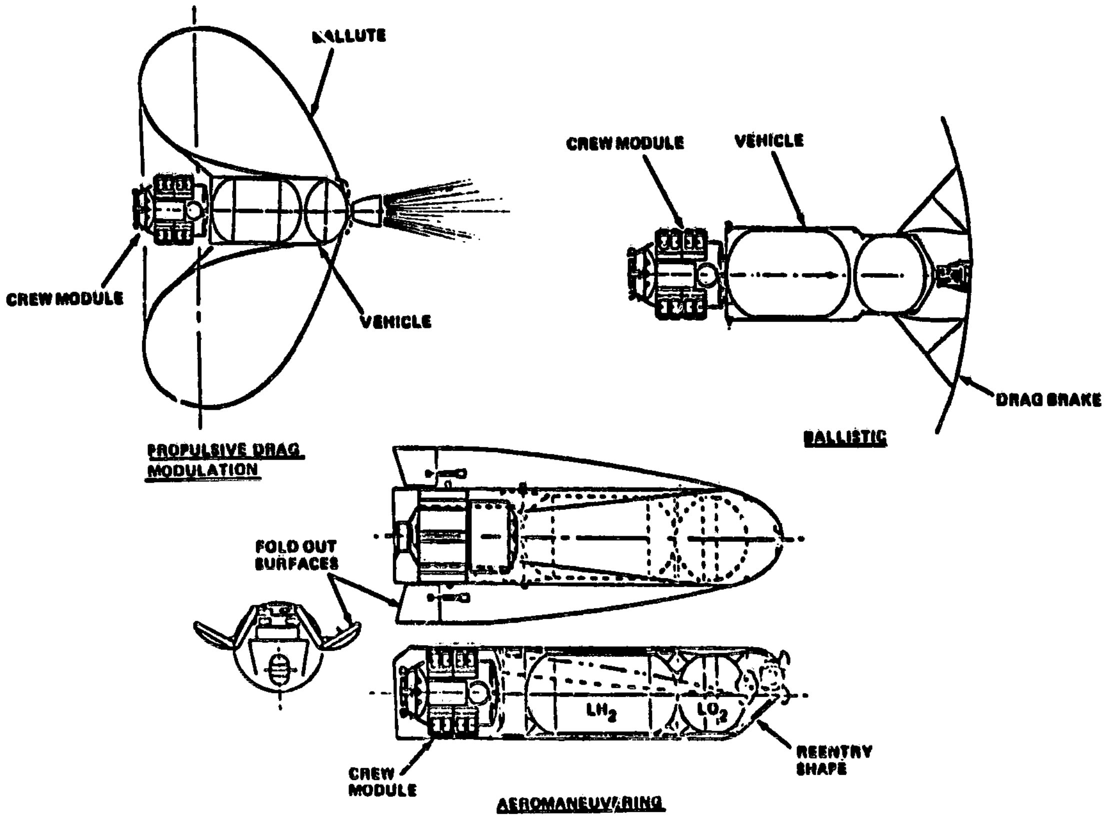
REUSABLE CRYOGENIC ORBIT TRANSFER VEHICLE (OTV-1) 22 FEET



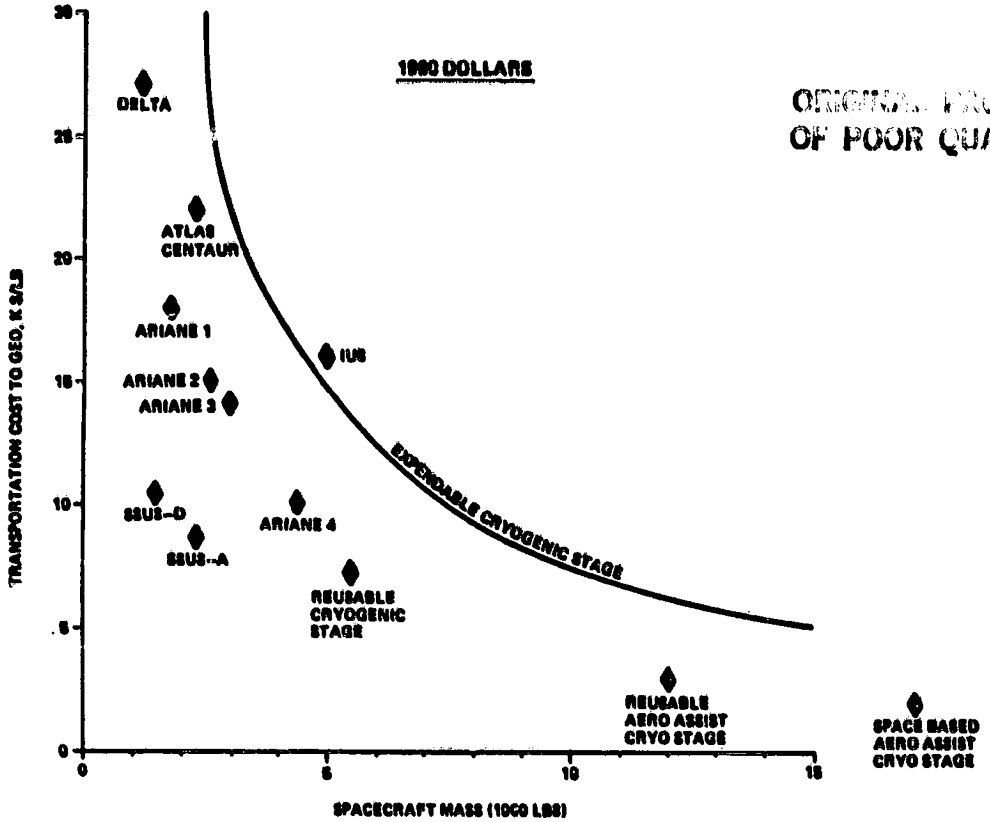
ORBITER TRANSFER VEHICLE (OTV)



AEROASSISTED ORBITAL TRANSFER VEHICLES



TRANSPORTATION COST COMPARISON TO GEOSYNCHRONOUS EQUATORIAL ORBIT (GEO)



LARGE SPACE SYSTEMS REQUIREMENTS ON PROPULSION SYSTEMS

<u>SUMMARY</u>

- PERFORMANCE
  - 12–15 klb OF PAYLOAD TO GEO
  - THRUST ACCELERATION 0.1 – 0.15g CAN BE TOLERATED
  - AERO ASSISTED RETURN TO LEO MAJOR LEVERAGE ITEM
- PAYLOAD PACKAGING
  - PLATFORM PACKAGING DENSITY IN STS LOW
  - UPPER STAGE LENGTH SHOULD BE MINIMIZED
- SERVICING/DEBRIS CONTROL
  - TELEOPERATOR MANEUVERING SYSTEM: LEO AND GEO SERVICING/DEBRIS CONTROL
  - SOLAR ELECTRIC OTV: DEBRIS CONTROL
- ECONOMY
  - TRANSPORTATION ECONOMY CAN BE A MAJOR ENABLING FACTOR FOR LARGE SPACE SYSTEMS – PARTICULARLY AT GEO
  - REUSABILITY FOR STS UPPER STAGES: VITAL
  - CRYOGENIC OTV SYSTEM WITH AERO ASSIST FOR RETURN: ENABLES 65K STS TO PERFORM EARLY PLATFORM MISSIONS

SYSTEMS INTEGRATION

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Lewis Research Center
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Cleveland, OH 44135

WHAT IS SYSTEMS INTEGRATION?

HOW IS IT TRADITIONALLY ACCOMPLISHED?

HOW DOES ITS CHARACTER CHANGE WITH THE ADVENT OF -

- THE STS?
- PROPULSION OUT OF THE ORBITER?
- LSS?

WHAT DEMANDS, IF ANY, WILL IT IMPOSE ON TECHNOLOGY?

INTEGRATION AS A TASK

DEFINING, UNDERSTANDING, AND ACCOUNTING FOR INTERACTIONS BETWEEN THE MAJOR SYSTEMS OF A SPACE MISSION.

ASSURING THAT THESE INTERACTIONS DO NOT VIOLATE THE MISSION'S SAFETY, RELIABILITY, PERFORMANCE, SCHEDULE AND COST STANDARDS.

HISTORICAL APPROACH TO INTEGRATION

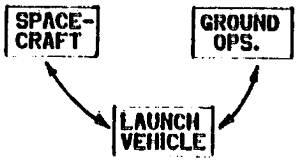
- IDENTIFICATION OF INTERFACES
- ESTABLISHMENT OF STANDARDS AND INTERFACE SPECS.
- DESIGN REVIEWS
- CHARACTERIZATION OF PROTOTYPES
- CONFIGURATION CONTROL
- QUALIFICATION AND ACCEPTANCE TESTS
- ANALYSIS AND DISPOSITION OF NON-CONFORMANCES
- FLIGHT-READINESS AND POST-FLIGHT REVIEWS

OBSERVATIONS

- MAJOR BURDEN OF CONFORMANCE HISTORICALLY PLACED ON SPACECRAFT.
  - VEHICLE - A DEVELOPED ENTITY W/CERTAIN OPTIONS
  - SPACECRAFT - AN EVOLVING ENTITY
- INTEGRATION IS A MAJOR FRACTION OF TOTAL LIFE-CYCLE COST.
- THE LATER INTEGRATION BEGINS IN THE LIFE-CYCLE PROCESS, THE HIGHER THE LIFE-CYCLE COST.
- TECHNOLOGY NOT NORMALLY VIEWED AS A METHOD OF ENHANCING OR ENABLING THE INTEGRATION TASK.
  - INSTEAD -
    - TASK MADE MORE COMPLEX BY IT
    - SOLUTIONS ARE ENGINEERED
- SYSTEMS INTEGRATION CONTINUALLY INCREASES IN COMPLEXITY AND COST.
  - MORE SYSTEMS - MORE INTERFACES THAN BEFORE
  - SYSTEMS MORE COMPLEX THAN BEFORE
  - MORE CONFORMANCE CRITERIA THAN BEFORE
- LSS FURTHER AGGREVATES THIS PROBLEM.
- LSS MISSIONS POSSIBLY PREVENTED BY IT.

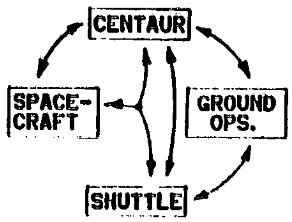
EXAMPLE - MORE INTERFACES

THEN -
ATLAS/CENTAUR



TWO INTERFACES

NOW -
STS/CENTAUR



FIVE INTERFACES

EXAMPLE - MORE CONFORMANCE CRITERIA

THEN - ATLAS/CENTAUR	NOW - STS/CENTAUR
RELIABILITY	RELIABILITY
PERFORMANCE	PERFORMANCE
COST	COST
SCHEDULE	SCHEDULE
	+
	• SAFETY
	• ENVIRONMENTAL

- SAFETY ALONE PLACES SIGNIFICANT DEMANDS ON FUTURE SPACECRAFT.

  FROM NHB 1700.1A, 'SAFETY POLICY AND REQUIREMENTS FOR PAYLOADS USING THE STS' -

  MINIMUM OF TWO INHIBITS ON THE FOLLOWING -

  PREMATURE ROCKET IGNITION

  PREMATURE S/C DEPLOYMENT OR SEPARATION

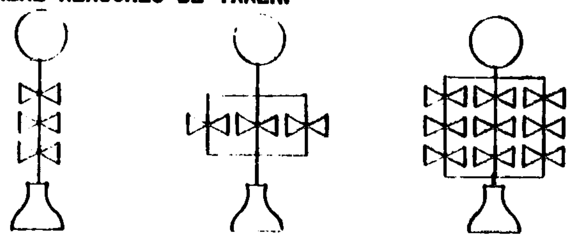
  DEPLOYMENT/EXTENSION PREVENTING PAYLOAD DOOR CLOSURE

IN THE CASE OF LSS -

ALL OF THE ABOVE CIRCUMSTANCES EXIST

ALONG WITH ADDED THREAT OF ORBIT DECAY/REENTRY DUE TO ATM. DRAG ON THE DEPLOYED LSS.

- SAFETY AND RELIABILITY OFTEN HAVE OPPOSITE INFERENCE ON SYSTEM DESIGN - MUTUAL CONFORMANCE REQUIRES THAT EXTREME MEASURES BE TAKEN.



SAFE BUT UNREL. REL. BUT UNSAFE SAFE AND REL.

IN THE CASE OF LSS -

A DISABLING PENALTY MAY EXIST BECAUSE OF THE NUMBER OF REPETITIVE ELEMENTS IN THE SYSTEM.

E.G. - MANY HINGES, MANY ROCKETS, ETC.

- THE UNIQUE NATURE OF PROPULSION REQUIRED BY LSS COULD FURTHER COMPLICATE INTEGRATION.

  PROPULSION POSSIBLY DISTRIBUTED THROUGHOUT THE LSS

  PROPULSION MAY LOOK MORE LIKE AN LSS SUBSYSTEM THAN A SHUTTLE UPPER STAGE

  IF SO, A GROSSLY DIFFERENT SET OF INTEGRATION RULES PERTAIN TO PROPULSION -

PROPULSION AS AN UPPER STAGE	PROPULSION AS AN LSS SUBSYSTEM
• ENTIRE STS RE-CERT.	• PROPULSION IS STS CARGO
• MINIMAL SHUTTLE/UPPER STAGE INTEGRATION AFTER THAT	• PROPULSION INTEGRATION REQUIRED FOR EACH MISSION

WHAT LSS/PROPULSION TECHNOLOGIES WOULD EASE THE BURDEN OF STS INTEGRATION?

ANSWER - UNKNOWN AT THIS TIME

  CRITICAL TO THE SUCCESS OF FUTURE LSS MISSIONS

  COULD GROSSLY INFLUENCE FUTURE COSTS

- WORTHWHILE STEPS NOW -

  DEFINE AN LSS MISSION FROM THE STANDPOINT OF INTEGRATION

  INVOLVE STS OPERATIONS MANAGEMENT

  APPLY CURRENT STS CONSTRAINTS AND EXPECTED LSS TECHNOLOGY

  IDENTIFY MISSION LIMITATIONS BECAUSE OF THE ABOVE

  DETERMINE NEEDED CHANGES TO - STS OPERATIONS
  - LSS TECHNOLOGY

GROUND- VERSUS SPACED-BASED ORBITAL TRANSFER VEHICLE

Joe Rehder

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A space-based OTV (SBOTV) is one that is delivered to orbit empty and remains there for its operational lifetime. Propellant and payload are brought up on subsequent launches. Ground-based OTV's (GBOTV) are launched with propellant and payload and are returned to the ground after each mission. Key issues in the comparison of these systems include debris protection, space-based OTV maintenance provisions, flight performance, on-orbit refueling, and launch and return operations.

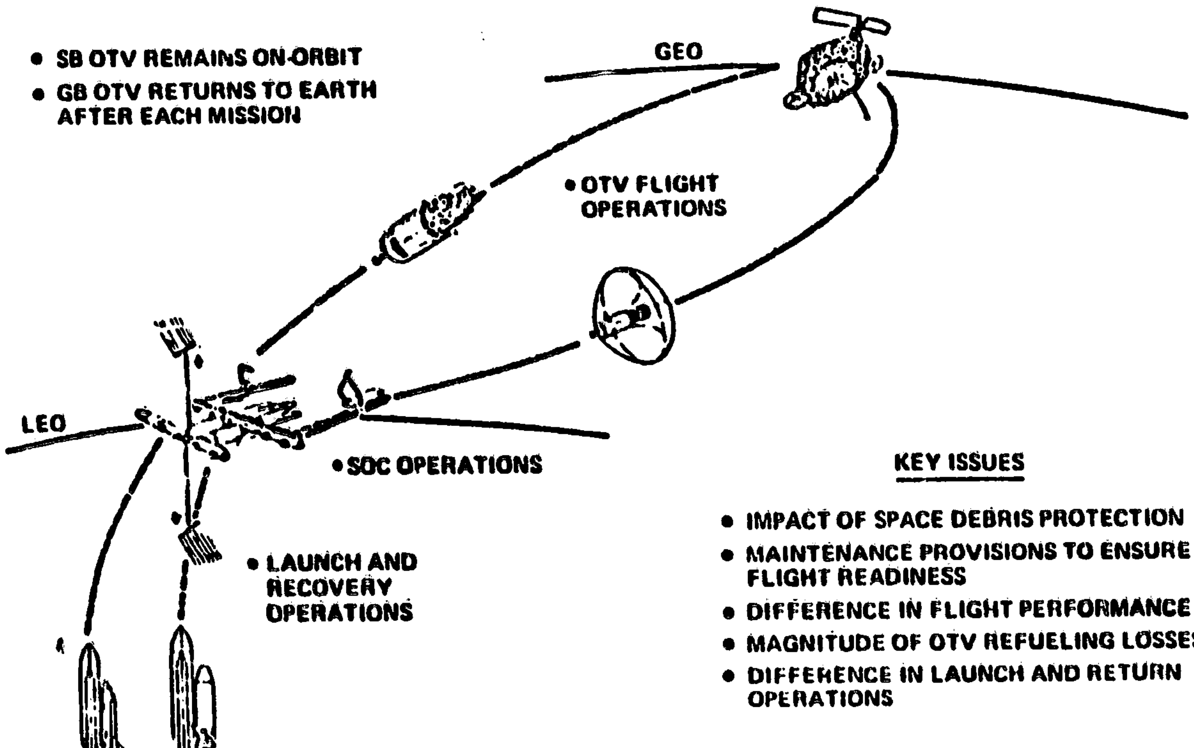
Debris protection has a severe impact on the SBOTV. The increasingly hostile man-made debris environment and the more stringent requirement of no tank impact require more protection than previous open truss designs could provide. The penalty for the GBOTV is much less severe, but still significant. A key technology issue is the protection capability of composite materials.

On-orbit maintenance is critical for SBOTV. If no maintenance is performed, the vehicle would have less than a 90-percent chance of success by the tenth mission. With full restoration of key components the frequency of return to Earth for major maintenance is decreased to once every 29 flights. The mass penalty and burden on the space station seem acceptable.

The main problem with refueling a SBOTV is reduction of losses during the various transfers. One concept, a compromise between no propellant recovery and reliquification results in a total loss rate of 12.8 percent. Zero-g propellant storage and transfer is an important technology area for SBOTV.

The configurations and masses of the SBOTV and GBOTV become very similar when the penalties are added in, the major difference being the maintenance provisions. In their most efficient modes of operation, both depend heavily on the space station. The GVOTV is used in two sizes and payloads are mated to the vehicles on-orbit. A reusable shroud must be developed to return GBOTV's if a Shuttle derivative vehicle is used. The advantage of space-basing lies in more efficient use of the launch vehicle. Since most of the mass going to LEO is OTV propellant, and the launches to deliver the SBOTV propellant are generally mass limited, substantially fewer launches are required to support the SBOTV.

OTV BASING CONCEPTS
INTEGRATED TRANSPORTATION OPERATIONS



KEY ISSUES

- IMPACT OF SPACE DEBRIS PROTECTION
- MAINTENANCE PROVISIONS TO ENSURE FLIGHT READINESS
- DIFFERENCE IN FLIGHT PERFORMANCE
- MAGNITUDE OF OTV REFUELING LOSSES
- DIFFERENCE IN LAUNCH AND RETURN OPERATIONS

SPACE DEBRIS PROTECTION FACTORS

VEHICLE AREA

LONGER EXPOSURE TIME FOR SPACE-BASED OTV

MAN-MADE DEBRIS PROBLEM GETTING WORSE

CRITERIA OF NO TANK IMPACT

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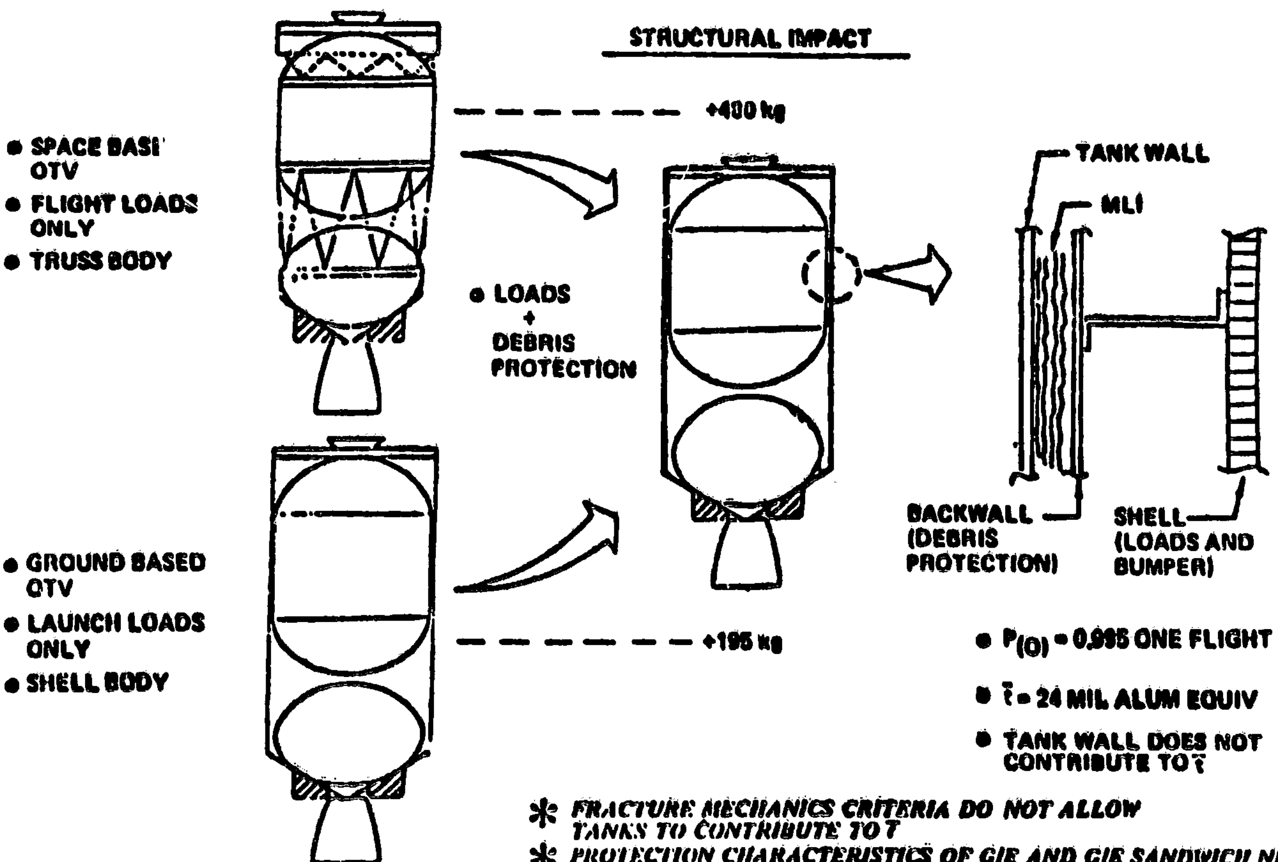
DEBRIS PROTECTION (METEOROID) IMPACT



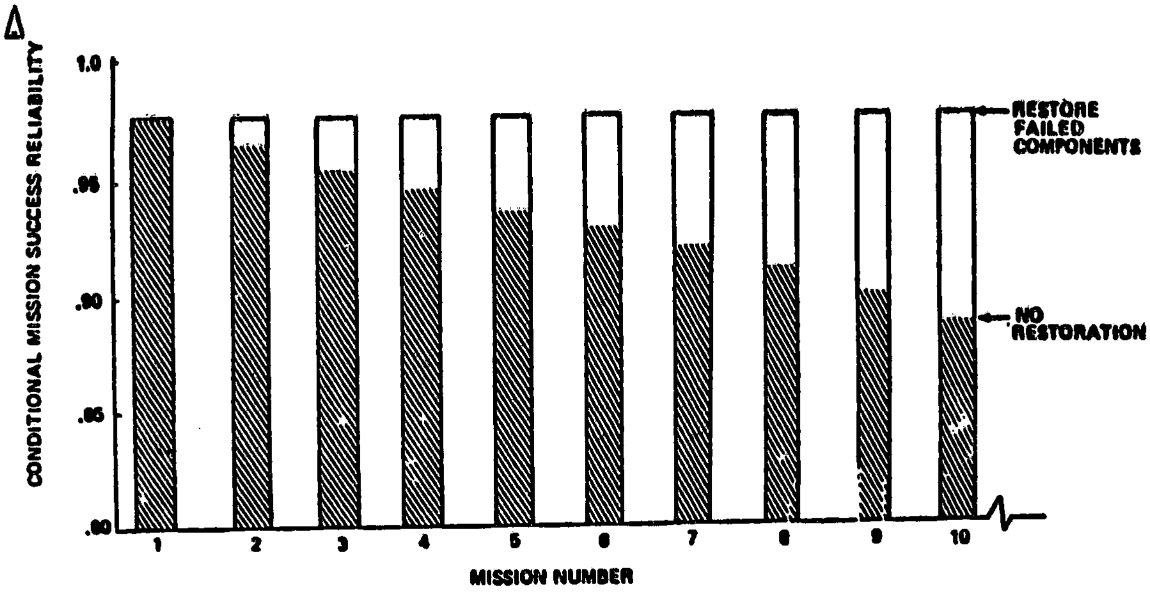
- $P_{(0)}$ = 0.995 ONE FLIGHT
- $\bar{t}$ = 24 MIL ALUM EQUIV
- TANK WALL DOES NOT CONTRIBUTE TO $\bar{t}$

* *FRACTURE MECHANICS CRITERIA DO NOT ALLOW TANKS TO CONTRIBUTE TO $\bar{t}$*

* *PROTECTION CHARACTERISTICS OF G/E AND G/E SANDWICH NEEDS TO BE ESTABLISHED*

* *HANGAR NECESSARY FOR SB OTV PROTECTION DURING ON-ORBIT STORAGE ($\bar{t}$ = 18 MIL)*

PREDICTED RELIABILITY FOR INDIVIDUAL MISSION
SPACE BASED OTV



▷ ASSUMES OTV HAS FUNCTIONED SUCCESSFULLY TO START OF Nth MISSION

UNSCHEDULED MAINTENANCE
SPACE BASED OTV

ISSUES . . . IS MAINTENANCE NECESSARY?
. . . WHAT'S REQUIRED TO ENSURE THE SAME DEGREE OF READINESS AS POSSIBLE WITH A GB OTV?

LEVEL OF SPACE MAINTENANCE	FREQUENCY OF EARTH RETURN (NO. OF MISSIONS)	TOTAL MASS IMPACT (kg) ▷1	MAINT TIME (HRS) (PER UNIT) ▷2
• NONE	1.06	-	-
• ACS THRUSTER MODULES	4.78	12	2.0
• PLUS FUEL CELLS	7.38	12	4.0
• PLUS MAIN ENGINES	11.7	104	7.0
• PLUS AVIONICS MODULES	29.07	42	1.5

▷1 PLUS BUILT IN TEST EQUIP (84 kg)
TOTAL MASS IMPACT = 270 kg

▷2 CREW OF 3

✱ *ON ORBIT MAINT. IS MOST EFFECTIVE METHOD TO ASSURE READINESS*
✱ *HANGAR WOULD BE BENEFICIAL*
✱ *NO KEY TECHNOLOGY ISSUES . . . DEMONSTRATION IS NECESSARY*

OTV REFUELING FACTORS

PROPELLANT REQUIREMENTS

DELIVERY	TRANSFER	STORAGE	TRANSFER	MISSION PROPELLANT
• BOILOFF	• SOC CHILLDOWN	• BOILOFF	• OTV CHILLDOWN	• ∇V
• RESIDUALS	• PRESSURIZATION	• RESIDUALS	• PRESSURIZATION	• INFLIGHT LOSSES
				• RESIDUALS

SOC RELATED

NON RECURRING

- ELECTRICAL POWER
- RADIATOR
- STORAGE TANKS
- SUPPORT EQUIPMENT REFRIG, RELIQUIFIER ETC

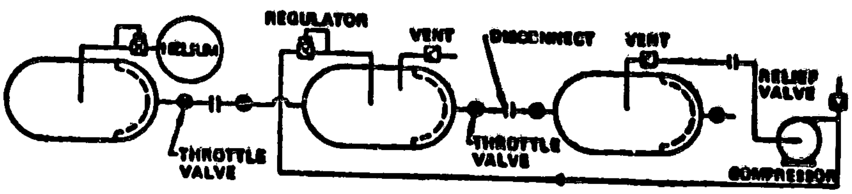
RECURRING

- ORBIT MAINTENANCE PROPELLANT
- PRESSURIZATION GAS

SELECTED OTV REFUELING CONCEPT

RECOVERED VAPOR PRESSURIZATION

- ISSUE: THE AMOUNT OF LOSSES ASSOCIATED WITH DELIVERY, STORAGE AND TRANSFER



TANKER — SOC STORAGE TANK — OTV PROPELLANT TANK (ON SOC)

SOC STORAGE TANK FEATURES

- QUANTITY (2)
- CAPACITY 59000Kg* OF LO_2/LH_2
- 50 LAYERS OF MLI
- FULL SCREEN ACQUISITION

TANK SET UTILIZATION

FLT	DAY	TS 1	TS 2
–	0	FULL	F
1	20	1/2	F
2	40	0	F
3	60	F	1/2
4	80	F	0

PROP. FLOW (Kg)

TANKER 60090 → STORAGE 56390 → OTV 52350

	LOSSES (TANKER → STORAGE)	LOSSES (STORAGE → OTV)
RESID	1340	1800
BOIL-OFF	120	1600
CHILL-DOWN RCVR	640	1240

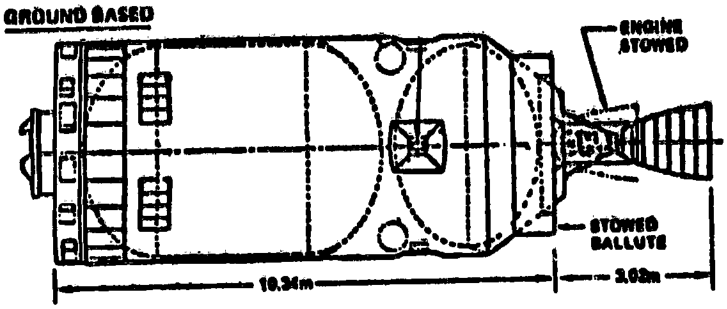
- TOTAL LOSSES = 6740 KG

* *TOTAL PROPELLANT REQUIREMENT IS 12.8% GREATER THAN OTV FLIGHT REQUIREMENT*

* *DEMONSTRATION OF ZERO G PROPELLANT STORAGE AND TRANSFER IS KEY PRIOR TO SB OTV COMMITMENT*

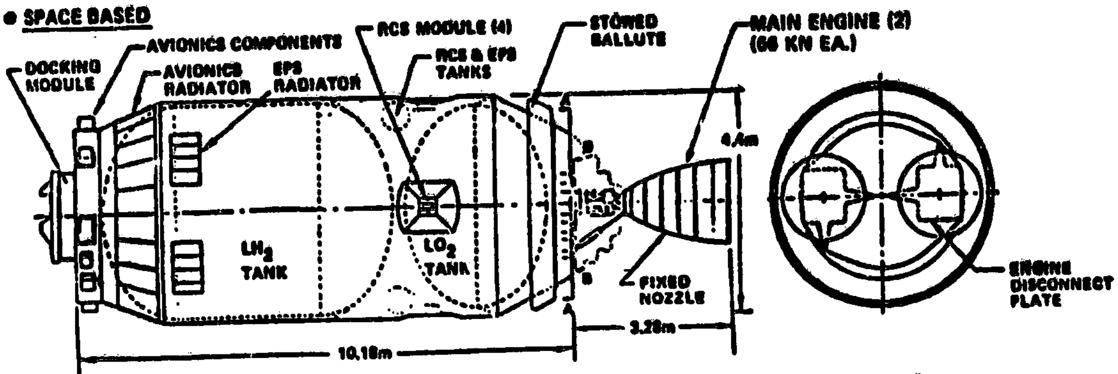
LO_2/LH_2 OTV CONFIGURATION
NORMAL GROWTH

● GROUND BASED



MASS (MT)	GB	SB
● DRY	3.9	3.6
● PROP	33.4	32.5
● GROSS	38.0	37.7
● MASS FRACTION	0.856	0.8638
● PAYLOAD		
• ROUND TRIP	7.6/8.0 MT	
• DELIVERY (0.2g)	12.7	13.0

● SPACE BASED



* NO MAJOR DIFFERENCE BETWEEN CONFIGURATIONS
* SB OTV REQUIRES 3% LESS PROP. FOR FIXED PAYLOAD: PROVIDES 6% MORE PAYLOAD FOR FIXED PROP. LOAD

OTV BASING MODE IMPACT ON SOC

IMPACT	GROUND BASED OTV (2 SIZES)	SPACE BASED OTV
● HANGAR • DEBRIS PROTECTION • MAINTENANCE CAPAB.	● NONE, UNLESS OTV STAYS AT BASE MORE THAN 3 DAYS (DEBRIS PROTECTION)	● 4 (ONE FOR EACH OTV) ● ONLY ONE WITH MAINTENANCE CAPABILITY
● MAINTENANCE CAPAB. ● CHECKOUT CAPAB.	● NONE ● OTV/PAYLOAD	● SCHEDULED & UNSCHED. ● OTV ● OTV/PAYLOAD
● REFUELING	● NONE	● (2) 52 MT TANK SETS AND ALL ASSOCIATED PLUMBING & CONTROL SYSTEMS
● DOCKING PORTS	● OTV (3) ● PAYLOADS (3)	● OTV (4) ● TANKER (1) ● PAYLOADS (3)
● HANDLING (MATING) PROVISIONS FOR:	● OTV/OTV (11) ● OTV/PAYLOAD (135) ● OTV/RECOVERY VEHICLE (193)	● OTV/OTV (11) ● OTV/PAYLOAD (182) ● OTV/RECOV. VEH (6)
● PERSONNEL	● 1-2, 10% DUTY CYCLE	● 3; 40% DUTY CYCLE

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OTV RECOVERY OPERATIONS

PROBLEM

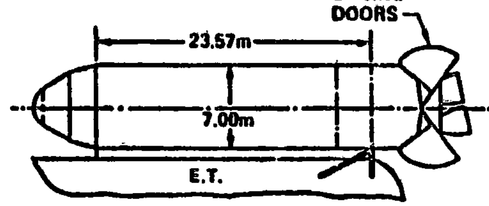
- INSUFFICIENT STS (ORBITER) FLIGHTS (OR SPACE WITHIN ORBITER) TO RETURN OTV ELEMENTS
  - 72 ORBITER FLIGHTS
    - 182 GB OTV's
    - 100 TANKERS FOR SB OTV
- MOST COST EFFECTIVE CARGO LAUNCH VEHICLE DOES NOT HAVE RETURN CARGO CAPABILITY

OPTIONS

- GB OTV
  1) DEDICATED LAUNCH VEHICLES (STS GROWTH)
  2) SDV WITH REUSABLE P/L SYS
- SB OTV
  1) EXPENDABLE TANKER
  2) SDV WITH REUSABLE PAYLOAD SYS (RPS)

SELECTED OPTION

- REUSABLE PAYLOAD SYSTEM (RPS)



- PAYLOAD REDUCED TO 60 MT
- △DDTE $100M

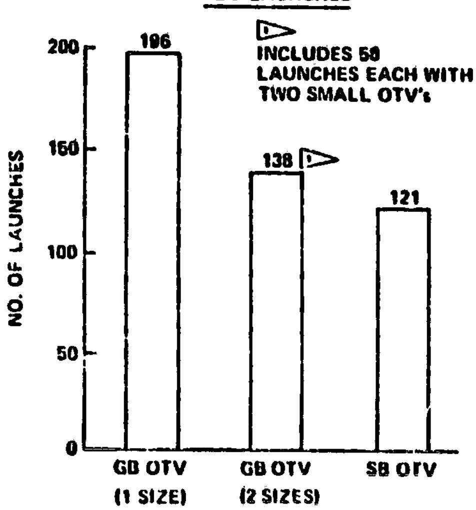
✱ *LARGE SCALE GEO OPERATIONS SUGGEST EXTENSIVE RETURN OPERATIONS*

✱ *REUSABLE PAYLOAD SYSTEM IS NEW TECHNOLOGY*

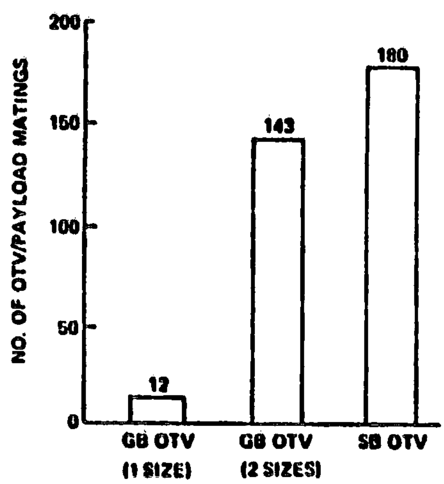
OTV LAUNCH AND PAYLOAD OPERATIONS

- 182 OTV FLIGHTS

SDV LAUNCHES



OTV/PAYLOAD MATINGS



✱ *GB OTV CONCEPT IMPROVED WHEN USING TWO SIZES*

✱ *SOC IS USEFUL FOR EITHER BASING MODE*

FINDINGS
SPACE vs GROUND BASED OTV'S

THE FOLLOWING STATEMENTS ARE MADE ASSUMING A GROUND BASED REUSABLE LO_2/LH_2 OTV WITH AERO ASSIST CAPABILITY AS THE POINT OF DEPARTURE FOR FUTURE OTV'S

- NO CLEAR CUT WINNER – – – VERY DEPENDANT ON EXISTING LAUNCH VEHICLE AND SPACE BASE
- CONFIGURATION, DESIGN FEATURES AND PERFORMANCE VERY SIMILAR
- SB OTV DOES PROVIDE COST BENEFITS, ADDITIONAL FLEXIBILITY AND MORE RAPID GEO ACCESS FOR AN ADVANCED SPACE SCENARIO INVOLVING SHUTTLE DERIVATURE VEHICLES AND SOC
- OTV ACCELERATED TECHNOLOGY TENDED TO IMPROVE SB OTV'S MORE THAN GB OTV BUT IN EITHER CASE IMPROVEMENT WAS SMALL
- ACCELERATED TECHNOLOGY WOULD BE BENEFICIAL FOR SB OTV PROPELLANT STORAGE/TRANSFER (REDUCE LOSSES FROM 12 TO 4%)
- LAUNCH SYSTEM EMPLOYED IS SINGLE MOST DOMINATING FACTOR – – – ADDITION OF SHUTTLE DERIVATIVE CARGO LAUNCH VEHICLE (SDCLV) SIGNIFICANTLY REDUCES TRANSPORTATION LCC
- OTV/TANKER RETURN NEEDS ARE KEY CONSIDERATION IN LAUNCH SYSTEM SELECTION
- MISSION MODEL COULD HAVE 50% REDUCTION OF 100% GROWTH AND NOT ALTER BASING MODE RESULTS BECAUSE LAUNCH VEHICLE SELECTION WOULD REMAIN THE SAME
- SPACE BASED OTV IMPACT ON SOC APPEARS ACCEPTABLE – – –CREW OF 2-3, 30% DUTY CYCLE; HANGAR
- A SPACE BASE WILL HAVE A VALUABLE ROLE WITH HYBRID GB OR SB OTV'S
- MOST SIGNIFICANT NEW TECHNOLOGY ISSUES FOR FUTURE OTV'S INCLUDE:
  - SPACE DEBRIS PROTECTION
  - REFUELING

* *SB OTV'S CAN PROVIDE REDUCED COST AND IMPROVE OVERALL SPACE TRANSPORTATION OPERATIONS*

* *A SDCLV IS THE MOST IMPORTANT FACTOR IN REDUCING NEAR TERM SPACE TRANSPORTATION COST*

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"SYSTEM REQUIREMENTS AND OPERATIONS" PANEL WORKSHOP SUMMARY

Fred R. Schwartzberg

MARTIN MARIETTA AEROSPACE

DENVER DIVISION POST OFFICE BOX 179 DENVER, COLORADO 80201

Emphasis in this panel's deliberations was primarily on (1) issues germane to the large-area systems, such as deployment, altitude, orbit transfer, and on-orbit operation; (2) issues strongly propulsion oriented, such as the orbit transfer vehicle and the auxiliary propulsion systems; and (3) programmatic issues.

The principal discussion associated with-large area systems for the late 1980's and early 1990's concerned the question of LEO versus GEO deployment. This issue is a significant driver on propulsion requirements for an orbit transfer vehicle and is closely associated with concerns for reliability of yet untested systems and concepts.

It was concluded that early systems would be deployed at LEO. This deployment would be near the Shuttle, but sufficiently remote to avoid interaction with the Shuttle. Hence, the Shuttle Remote Manipulator System (RMS) would not be involved in early demonstration activities. The panel noted the pressing need for a Teleoperator Maneuvering System (TMS) and recognized the potential of the Manned Maneuvering Unit (MMU). However, it was generally agreed that man's role in the actual deployment would be minimal. His principal role would be as an observer, particularly for sequentially deployed systems, and that activity would normally be observation from the aft flight deck to control and/or provide information to assist in future designs. Man's role in deployment assistance or repairs would have to be extremely simple and would have to satisfy complex personnel safety requirements.

In order to proceed into the LSS age, a number of LEO versus GEO trade studies would have to be performed. These include a study of load capability during launch, deployment, orbit transfer, and operation. Recognizably, these capability levels vary significantly from the very high launch vibration loads to operational forces developed principally from thermal influences. Although LEO deployment was clearly the choice for early technology, it was apparent that partial deployment was indeed likely. Certain spacecraft already utilize deployment at operational altitude of subsystems such as solar panels with satisfactory reliability. Trade studies to determine which appendages could be reliably deployed at altitude should be performed. Another trade study suggested by the panel dealt with anticipated requirements for deployment assistance, repair, and checkout.

Following deployment at low Earth orbit, issues associated with orbit transfer must be addressed. One of the suggested studies deals with the question of central versus distributed propulsion, which was well paraphrased as "Where do you push on it?" It was generally agreed that the many advantages of distributed propulsion had attendant hurdles but that the technology was manageable. In order to subsequently address the question of the nature of the orbit transfer vehicle, it is essential to establish acceptable thrust-acceleration levels for various types of payload systems. The behavior of the system during transfer through the Van Allen belts and the time-dependent hazards to various subsystems were identified as studies requiring attention. Additionally, the needs to provide control during transfer were recognized.

During on-orbit operation servicing, resupply and capability to update were identified as major systems concerns requiring trade studies. Other considerations requiring study include integration of auxiliary requirements into the structural system, particularly with respect to volume, weight, center of gravity, and support; system operational requirements; control-system interactions with the auxiliary propulsion system; and servicing-vehicle needs.

Discussion of Space Operations Center (SOC) type systems for the 1990's identified that the needs will be governed by the transition from early deployables to the ability to erect and ultimately to construct structures in space. It was agreed that operational needs have not yet been addressed.

Propulsion oriented issues addressed by the panel were (1) orbit transfer vehicles, and (2) auxiliary propulsion systems. Discussion of the orbit transfer vehicle for the late 1980's addressed whether the propulsion system should be cryogenic, storable, or possibly electric, as well as the required thrust level and the question of reusable versus expendable vehicles. The panel agreed that the packaged length of the stage should be as small as possible and that to satisfy the projected need date, development should start immediately. It was concluded that there would be only a single new OTV to serve LSS and other spacecraft requirements.

Auxiliary propulsion system discussion revolved around the question of chemical versus electric systems. Considerations of significance were attitude, shape, station keeping, and desaturation. An important feature of APS was deemed to be evolutionary capability or the growth of the system as a function of time.

A discussion of programmatic issues concluded that the need exists for a better understanding of realistic capability and schedule. The overall conclusions of the panel were that an integrated large space systems/propulsion approach is essential and that improved dialogue between these two communities is mandatory. They also concluded that the LSS community needs to establish propulsion development requirements as soon as possible.

CONFIGURATION DEVELOPMENT OF THE LAND MOBILE SATELLITE SYSTEM (LMSS) SPACECRAFT

C. T. GOLDEN, J. A. LACKEY & E. E. SPEAR

BOEING AEROSPACE COMPANY
P.O. Box 3999
Seattle, Washington 98124

LSST/LMSS SYSTEM STUDY OVERVIEW

BOEING WAS SELECTED BY THE JPL LSST ANTENNA ORGANIZATION TO PROVIDE CONFIGURATIONS AND SYSTEM/SUBSYSTEM REQUIREMENTS FOR LARGE DEPLOYABLE ANTENNAS. INITIALLY THE STUDY WAS TO COMBINE THE NASA FOCUSED MISSIONS AND THE LSST GENERIC LARGE ANTENNAS TO DEVELOP A SYSTEM DESIGN AND SUBSYSTEM REQUIREMENTS. EARLY IN THE STUDY IT BECAME EVIDENT THAT THE FOCUS MISSIONS AND THE GENERIC ANTENNAS WOULD NOT GENERATE THE DESIRED DATA AT A MEANINGFUL LEVEL OF DETAIL. THE LAND MOBILE SATELLITE SYSTEM WAS SELECTED AS THE MISSION REQUIREMENTS BASELINE AND TWO ANTENNA SYSTEM DESIGNS (WRAP RIB AND HOOP COLUMN) INITIATED TO SATISFY THE MISSION REQUIREMENTS.

- CONTRACT AWARD: JULY 22, 1980
  "A STUDY OF SUBSYSTEM INTERFACES OF DEPLOYABLE ANTENNAS FOR THE LARGE SPACE SYSTEMS TECHNOLOGY (LSST) PROGRAM"
- INITIAL MEETING INTRODUCING LMSS SEPTEMBER 15, 1980
- INITIAL MEETING WITH LOCKHEED ON WRAP RIB LMSS FEBRUARY 24, 1981
- CONFIGURATION 1 OF WRAP RIB LMSS MARCH 18, 1981
- INITIAL MEETING WITH THE HARRIS CORP. ON HOOP COLUMN LMSS MAY 18, 1981
- CONTRACT EXTENSION: JUNE 22, 1981
- WRAP RIB CONFIGURATION 4A & HOOP COLUMN CONFIGURATION 2 COMPLETE AUG. 17, 1981
- PRESENT AT LSST THIRD ANNUAL REVIEW LaRC NOVEMBER 16, 1981
  SYSTEM/SUBSYSTEM REQUIREMENTS WRAP RIB LMSS
  CONFIGURATION 3 OR 4 HOOP COLUMN LMSS

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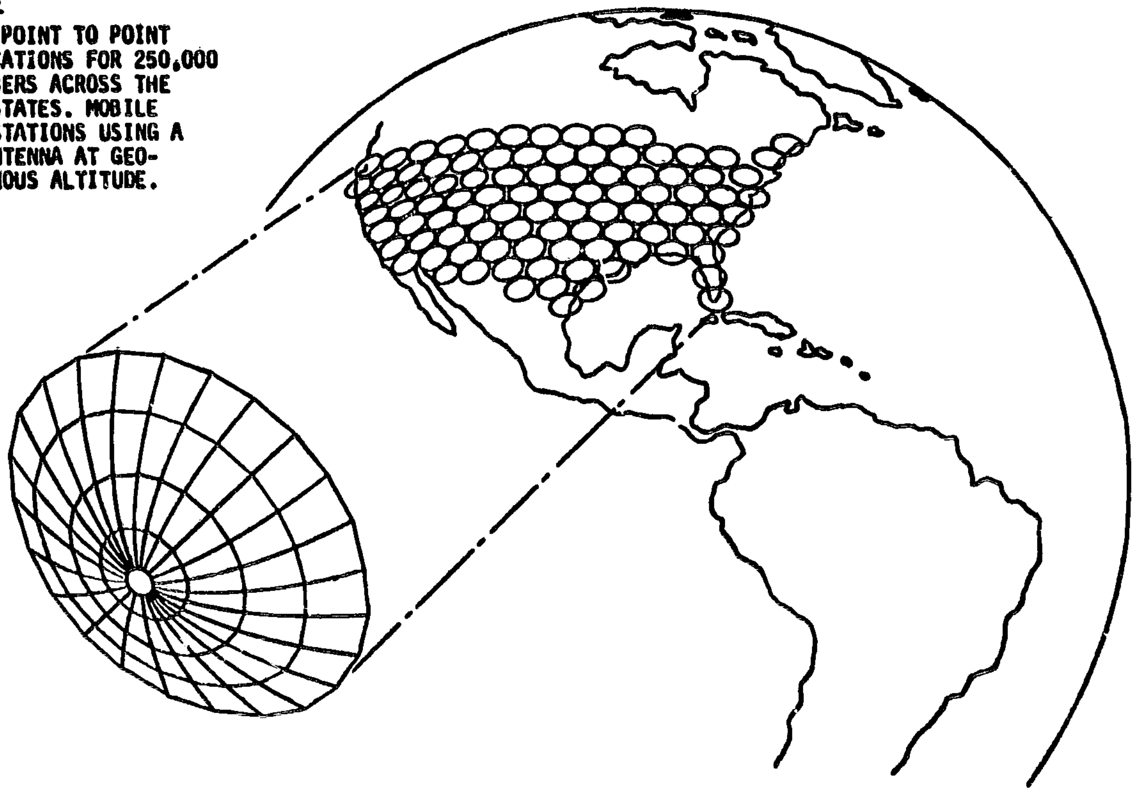
LAND MOBILE SATELLITE SYSTEM (LMSS)

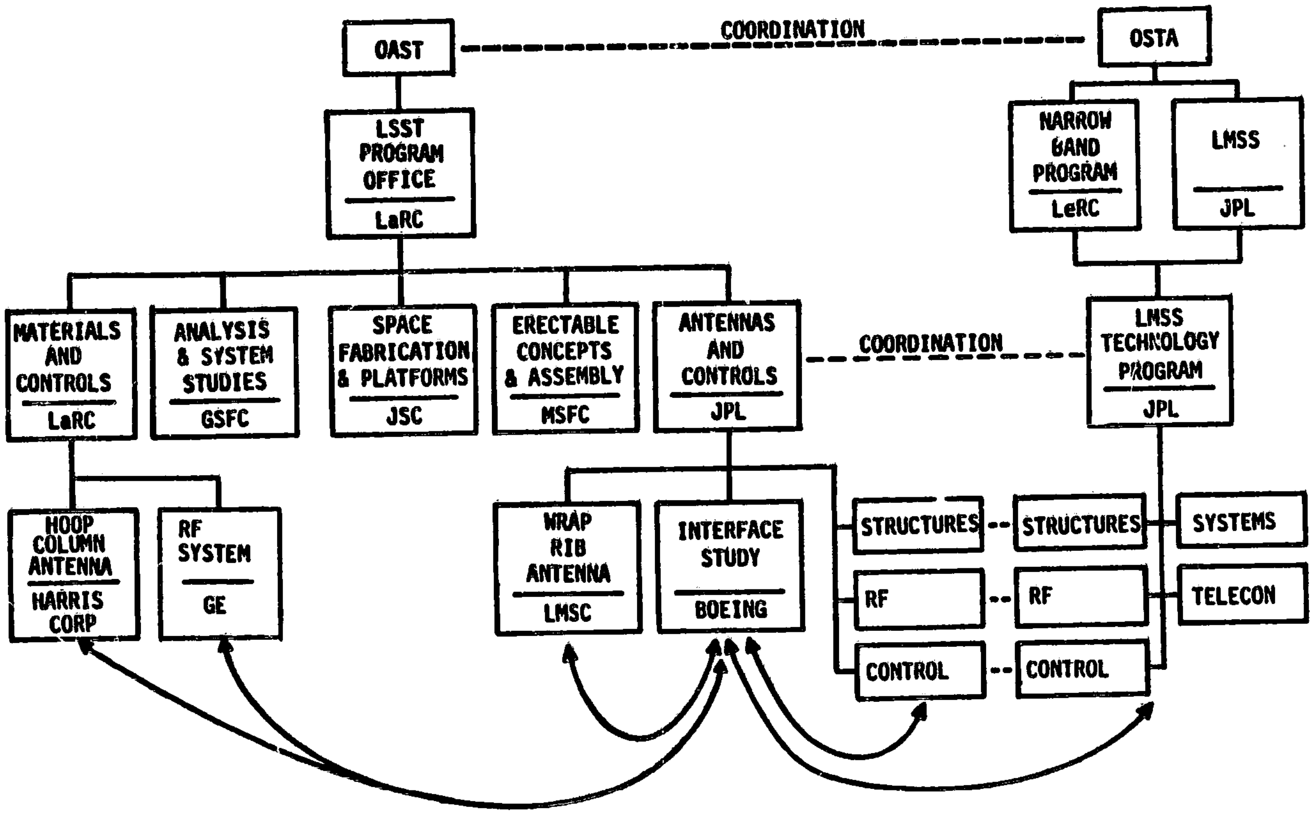
<u>LMSS</u>

PROVIDE POINT TO POINT COMMUNICATIONS FOR 250,000 SUBSCRIBERS ACROSS THE UNITED STATES. MOBILE GROUND STATIONS USING A RELAY ANTENNA AT GEO-SYNCHRONOUS ALTITUDE.



LSST/LMSS STUDY PARTICIPANTS

THE LSST ORGANIZATION, LMSS ORGANIZATION AND SOME OF THEIR SUBCONTRACTORS ARE SHOWN. BOEING'S ROLE IS TO COLLECT THE DATA GENERATED BY THE RESPONSIBLE DESIGN OR ANALYSIS GROUP TO SYNTHESIZE THE SYSTEM CONFIGURATION. THE MAJOR WRAP RIB CONTRIBUTORS ARE: LOCKHEED FOR THE BOOM, DEPLOYMENT, AND REFLECTOR; AND JPL FOR THE RF SYSTEM DESIGN, CONTROLS AND STRUCTURE. THESE LATTER THREE GROUPS CONSIST OF INDIVIDUALS THAT ARE EITHER SUPPORTING THE LSST WORK, LMSS WORK OR IN SOME CASES BOTH. THE MAJOR HOOP COLUMN CONTRIBUTORS ARE: THE HARRIS CORPORATION FOR THE STRUCTURE AND REFLECTOR; GENERAL ELECTRIC FOR THE RF SYSTEM DESIGN; AND JPL FOR THE STRUCTURAL ANALYSIS AND CONTROLS SUBSYSTEM DEFINITION. THROUGH MEMOS, TELECONS, ACTION ITEMS, AND COORDINATION MEETINGS, BOEING HAS ORCHESTRATED A SERIES OF ITERATIONS THAT WILL RESULT IN A POINT DESIGN FOR EACH ANTENNA REFLECTOR CONCEPT TO PERFORM THE LMSS MISSION.



LMSS CONFIGURATION STUDY
MISSION REQUIREMENTS

- FREQUENCY -- 821 → 831 & 2650 → 2690 MHz UPLINK
  866 → 876 & 2550 → 2590 MHz DOWNLINK
- BEAM TO BEAM ISOLATION -- 25 dB GOAL
- BEAMWIDTH -- 0.4 → 0.5^{o} NOMINALLY
- NUMBER OF USERS -- 250,000 NOMINALLY
- USER DISTRIBUTION -- POPULATION DENSITY RELATED (WITH VOX)
- COVERAGE -- CONUS
- POLARIZATION -- CIRCULAR
- POINTING -- (ABSOLUTE $\pm 0.10^{o}$, STABILITY $\pm 0.03^{o}$)
- ORBIT -- GEOSYNC AT 110^{o} LONG
- LAUNCH YEAR, 1995; VEHICLE, SHUTTLE; LIFETIME, 10 YEARS
- COMPATIBILITY WITH ATT CELLULAR SYSTEM -- YES
- MOBILE G/T -- -20 → -25 dB
- BASE STATION G/T -- 11 dB

THE GOAL OF THE STUDY IS NOT TO COMPARE THE TWO CONCEPTS, BUT THE PARTICIPANTS AGREED THAT THERE SHOULD BE SOME COMMONALITY IN SUBSYSTEM REQUIREMENTS. THE MISSION REQUIREMENTS COVER MOST OF THESE AND THE ONES CITED BELOW COMPLETE THEM. ALTHOUGH THE REQUIREMENTS ARE THE SAME, THE DESIGN THAT IMPLEMENTS THEM NEED NOT BE. DIFFERENCES SUCH AS THE SINGLE APERTURE VS. THE QUAD APERTURE ARE EXPECTED THROUGHOUT THE SUBSYSTEM DESIGNS.

LMSS CONFIGURATION STUDY
SUBSYSTEM REQUIREMENTS

POWER	10 KW BEGINNING OF LIFE
ATTITUDE CONTROL	STATION KEEPING PROPULSION 300 ISP
MESH CONTOUR	12 MM
RF FEED	POWER AMPLIFIER EFFICIENCY 50%

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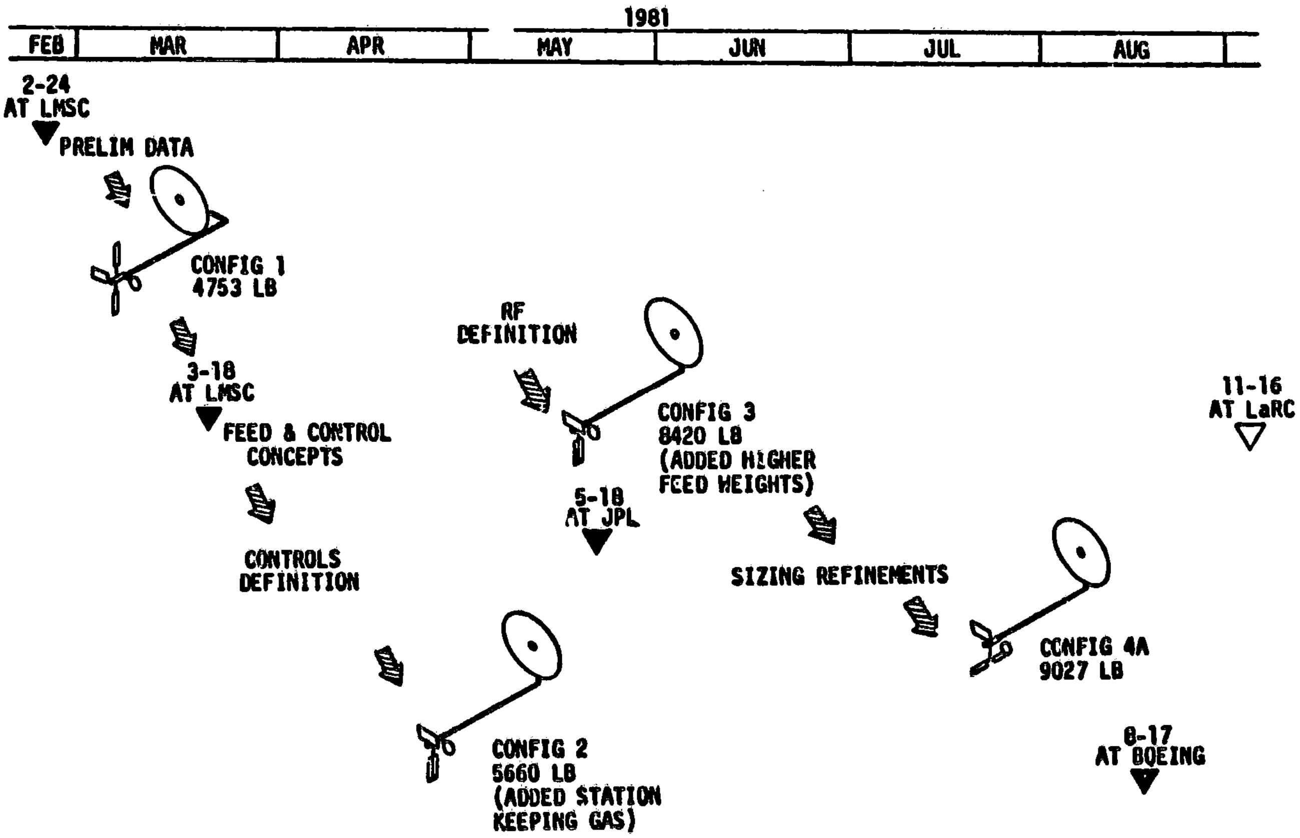
LMSS - WRAP RIB CONFIGURATION

SHOWN BELOW IS THE EVOLUTION OF THE STUDY AND THE FOUR MAJOR CONFIGURATIONS. AS SHOWN, THE OVERALL GEOMETRY HAS REMAINED FAIRLY CONSTANT, BUT THE WEIGHT HAS ESSENTIALLY DOUBLED AND THE OTHER MASS PROPERTIES INCREASED ACCORDINGLY.

LMSS WRAP RIB
BOEING STUDY HISTORY



LAND MOBILE SATELLITE SYSTEM SPACECRAFT
55 METER OFFSET WRAP RIB CONCEPT

This is a preliminary configuration of an offset wrap rib reflector spacecraft capable of relaying radio messages to land mobile units throughout the continental United States. It is intended to service units such as ambulances, police cars, taxis -- in essence all radio dispatched vehicles. Its position in geosynchronous orbit avoids present radio interferences caused by tall buildings, hills, and other factors. It is one of numerous concepts for extending current space technology made feasible by the Space Shuttle. The spacecraft is sized for a single Shuttle launch and a 10-year life beginning in the mid-1990's.

The long boom points at the earth's center, which is up and to the left. The shorter, vertical boom at the right points up to the north supporting the antenna reflector. The large panel at the left is the ultra-high-frequency feed. It and the 55 meter diameter wire mesh reflector are angled to point at the center of the United States near Kansas City. Multiple beams emanating from the feed panel are arranged to cover all contiguous 48 states with a potential in the design for communication with Alaska, Hawaii and parts of Canada.

Major contributors to development of this configuration are:

System Design	The Boeing Aerospace Co.
Wrap Rib Design (UHF Reflector/Booms)	Lockheed Missile & Space Co.
RF Subsystem	Jet Propulsion Laboratory
Controls Subsystem	Jet Propulsion Laboratory
Structural Dynamics	Jet Propulsion Laboratory

LMSS WRAP RIB
WEIGHT STATEMENT

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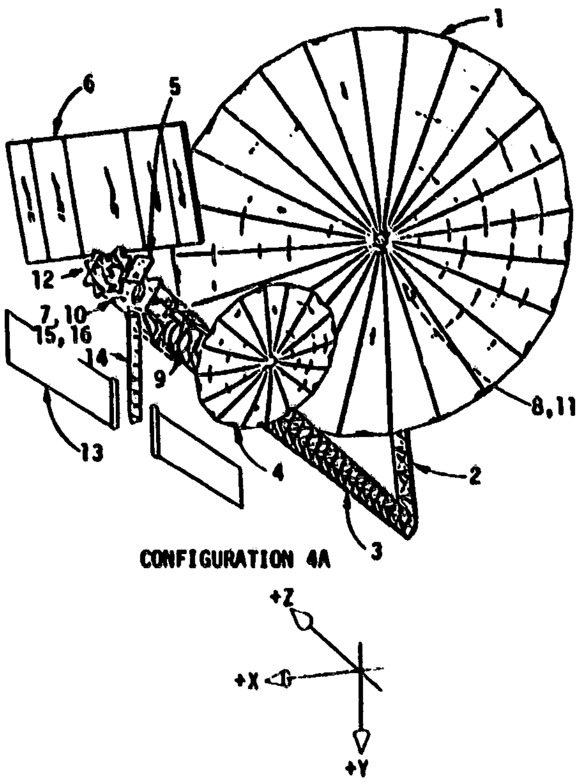
THE WEIGHT STATEMENT FOR THE LATEST WRAP RIB LMSS CONFIGURATION IS SHOWN BELOW. IT SHOULD BE NOTED THAT THE WEIGHTS ARE NOT GROUPED BY SUBSYSTEM, BUT BY LOCATION. THIS WAS DONE PURPOSELY TO IMPLEMENT MASS PROPERTIES DETERMINATION. A SUBSYSTEM BREAKDOWN AND REDISTRIBUTION OF CONTINGENCY WEIGHTS WILL BE MADE TO REFLECT SUBSYSTEM REQUIREMENTS.

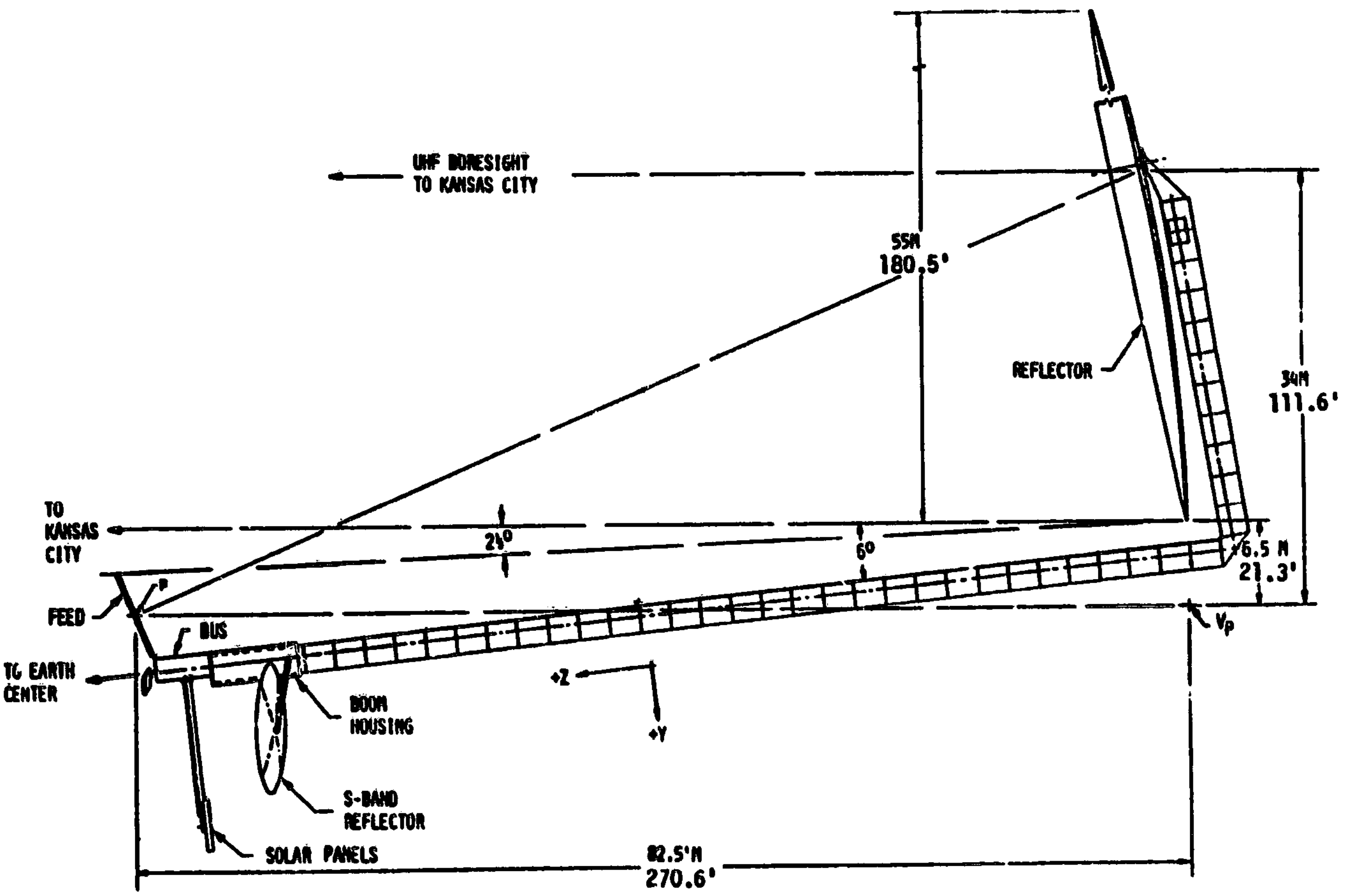


CONFIGURATION 4A

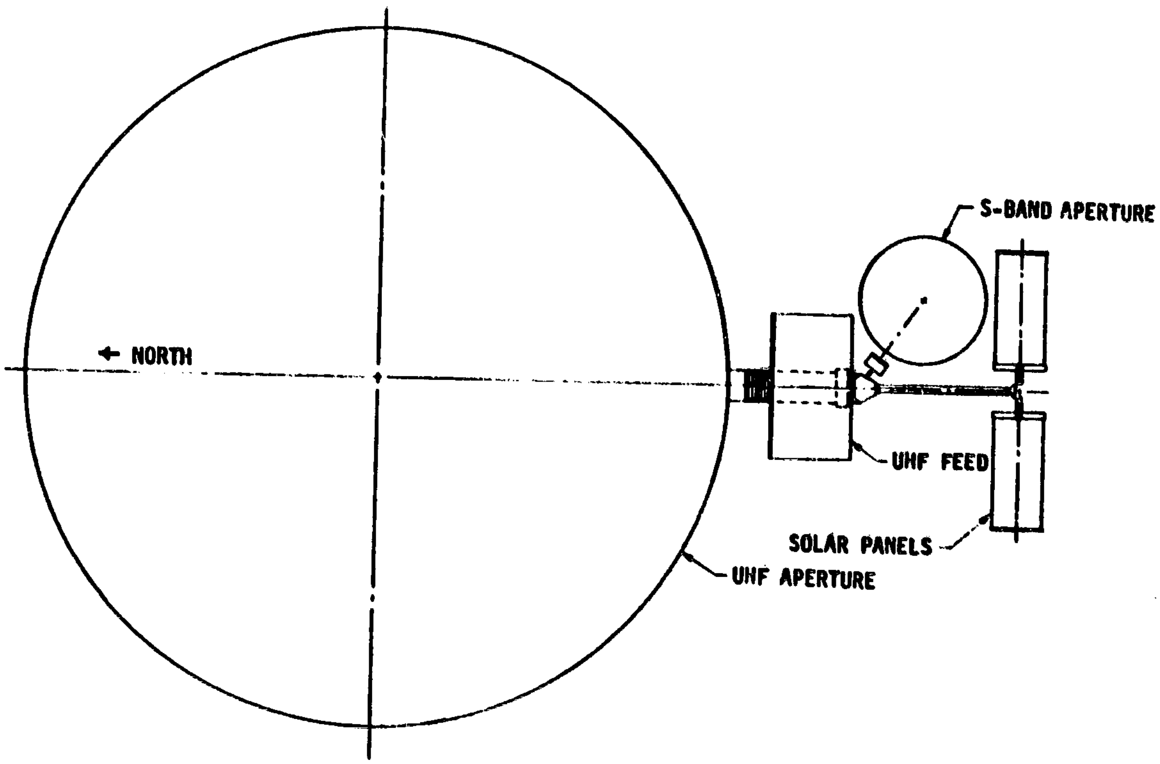
ITEM	WT~LB
1. UHF REFLECTOR & BOOM ATTACH	740
2. UHF BOOM-OUTBD & CABLE	90
3. UHF BOOM-INBD & CABLE	210
4. S-BAND BOOM & REFLECTOR	151
5. S-BAND FEED	255
6. UHF FEED	2667
7. RF ELECTRONICS IN BUS	490
8. REACTION WHEELS & SENSORS - OUTBD	88
9. REACTION WHEELS - INBD 10. CONTROL EQUIP & SENSORS - BUS	581
11. TANKAGE, FUEL, & THRUSTERS - OUTBD	449
12. TANKAGE, FUEL, & THRUSTERS - INBD	1691
13. SOLAR PANELS	367
14. SOLAR PANEL MAST & MECH	110
15. BATTERIES & POWER COND	288
16. BUS STRUCT, CABLING, T/C, & CAGE	850
TOTAL	9027

LMSS WRAP RIB SPACECRAFT

THE LMSS SPACECRAFT WILL BE POSITIONED ON ORBIT AT 110° WEST LONGITUDE WITH THE MAIN BOOM POINTED AT THE EARTH'S CENTER (EQUATOR). THE ANTENNA BORE SIGHT IS RAISED 6° ABOVE THE BOOM TO POINT TO THE LATITUDE OF KANSAS CITY. THE 1° POINTING CORRECTION TO THE EAST (TO POINT TO KANSAS CITY) IS NOT BEING CONSIDERED AT THIS TIME, BUT WILL BE INCLUDED IN ANY FINAL DESIGN.



THE WRAP RIB LMSS AS VIEWED FROM EARTH IS SHOWN ROTATED 90°. THE N-S AXIS OF THE SPACECRAFT RUNS IN THE LONG DIRECTION OF VIEW WITH THE UHF REFLECTOR TO THE TOP (NORTH). THE CLEARANCES BETWEEN APERTURES (UHF & S-BAND) WITH RESPECT TO THE UHF FEED AND SOLAR PANELS ARE SHOWN. THE SOLAR PANELS ROTATE ABOUT THE N-S AXIS.



LMSS WRAP RIB
LAUNCH CONFIGURATION

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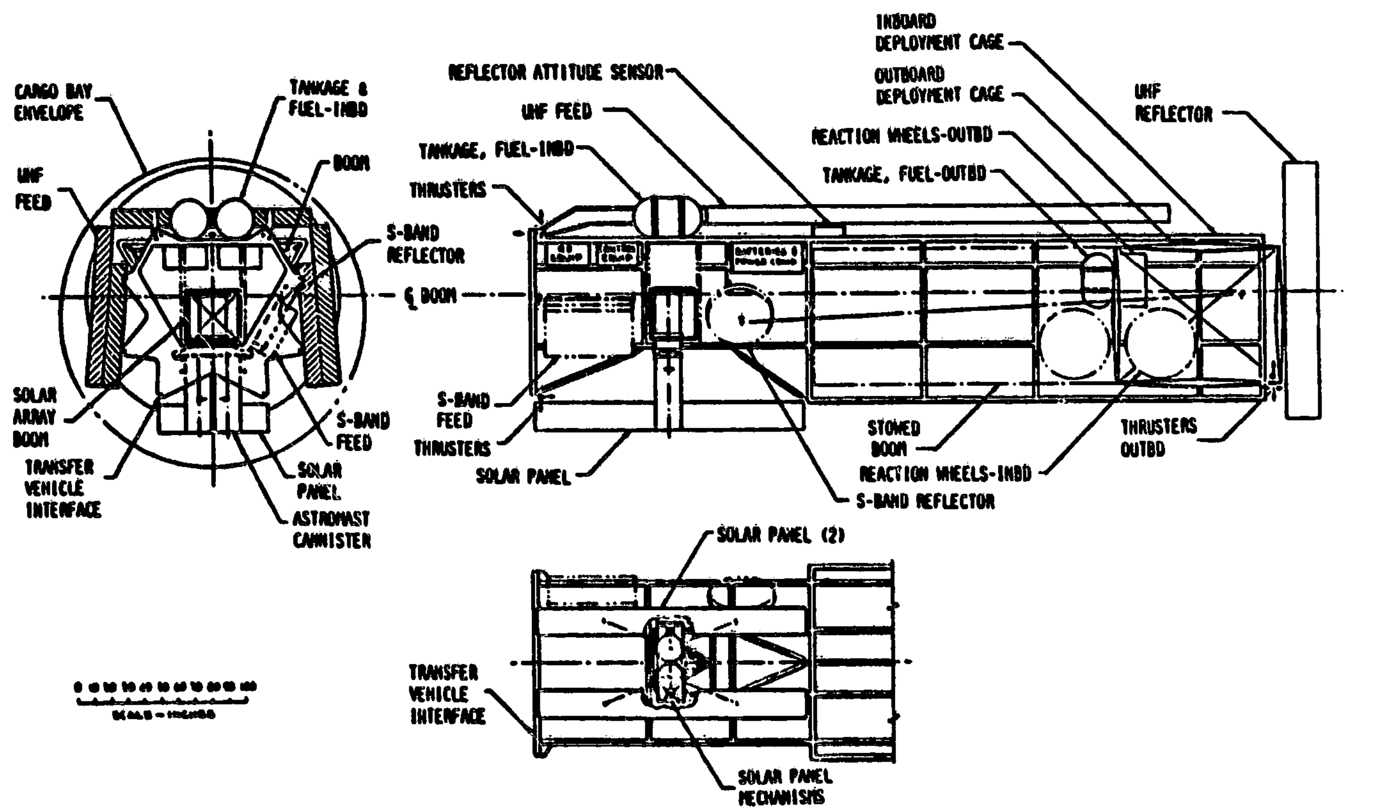
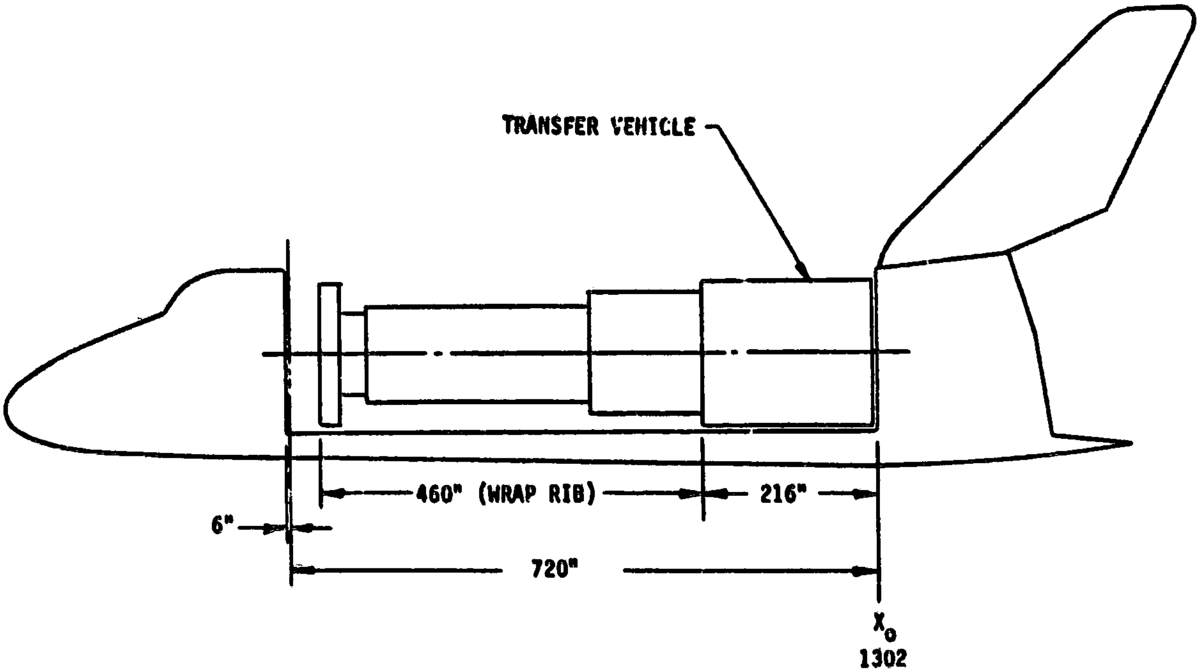
THE LAUNCH CONFIGURATION IS CONSTRAINED BY THE 180" DIAMETER DYNAMIC ENVELOPE WITHIN THE SHUTTLE CARGO BAY AND THE LENGTH OF BAY AVAILABLE AFTER INSTALLING AN APPROPRIATE SIZED ORBIT TRANSFER STAGE. FROM THE WEIGHT STATEMENT A 9,000 LB. CAPABILITY TRANSFER STAGE IS REQUIRED. THE PRIME CONFIGURATION DRIVER WITH RESPECT TO STOWING THE SPACECRAFT IN THE SHUTTLE IS THE UHF FEED. THE FEED MUST BE FOLDED IN 5 SECTIONS (THUS COMPLICATING THE FEED STRUCTURE AND RF CIRCUITRY) AND EXTENDS OVER 80% OF THE LENGTH. THIS REQUIRED THAT THE BOOM CROSSECTION BE REDUCED FROM THE MAXIMUM "PACKAGABLE" SIZE OF 120" TO 95". AS SHOWN IN THE SIDE VIEW, THE SPACECRAFT BUS (FEED END) IS ATTACHED TO THE TRANSFER VEHICLE AND THE LMSS CANTILEVERED FROM THE INTERFACE. THE SECONDARY ATTITUDE CONTROL EQUIPMENT IS SHOWN MOUNTED IN THE DEPLOYMENT CAGE ADJACENT TO THE UHF REFLECTOR.



LMSS WRAP RIB IN CARGO BAY

THE OVERALL STOWAGE OF THE WRAP RIB LMSS IS SHOWN. THE STOWED LENGTH OF 460" FITS EASILY WITHIN THE CARGO BAY. ASSUMING THE 9,000 LB. CAPACITY TRANSFER VEHICLE REQUIRES 216" THERE IS A 44" CLEARANCE. IN ADDITION, THERE IS A POTENTIAL OF SHORTENING THE WRAP RIB LMSS BY AN ADDITIONAL 40" TO 44" BY REDUCING THE LARGE DIAMETER (CENTER) OF THE LONGITUDINAL STRUTS FROM 4" TO 3". THUS A TRANSFER VEHICLE LENGTH OF UP TO 400" CAN BE ACCOMMODATED.



LMSS WRAP RIB

REFLECTOR HUB AREA

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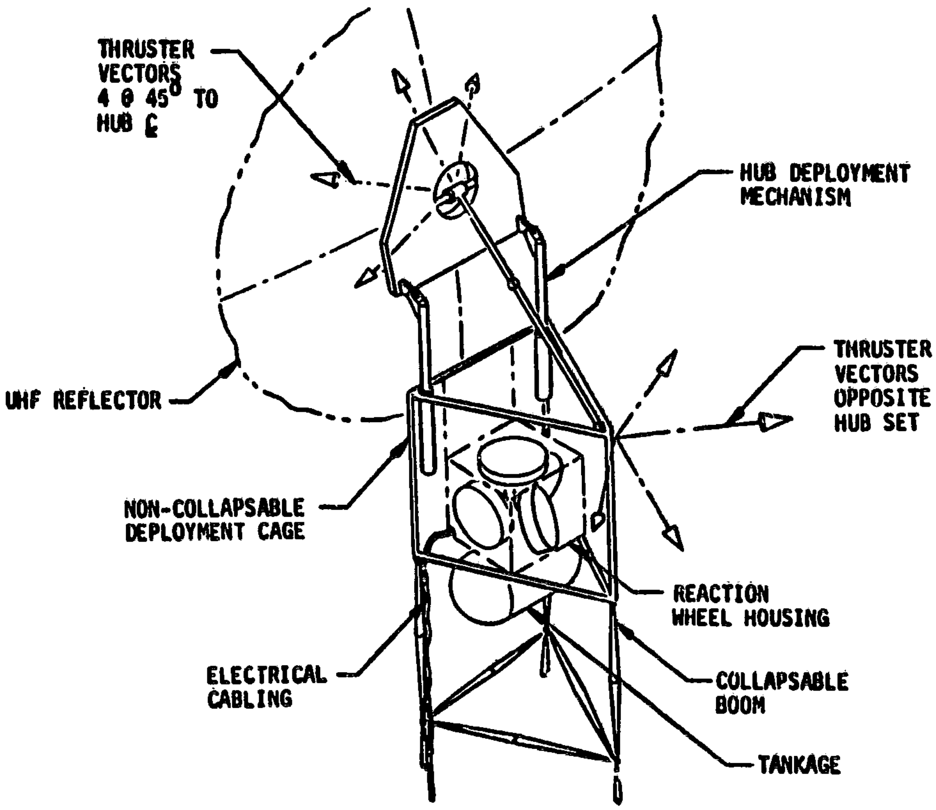
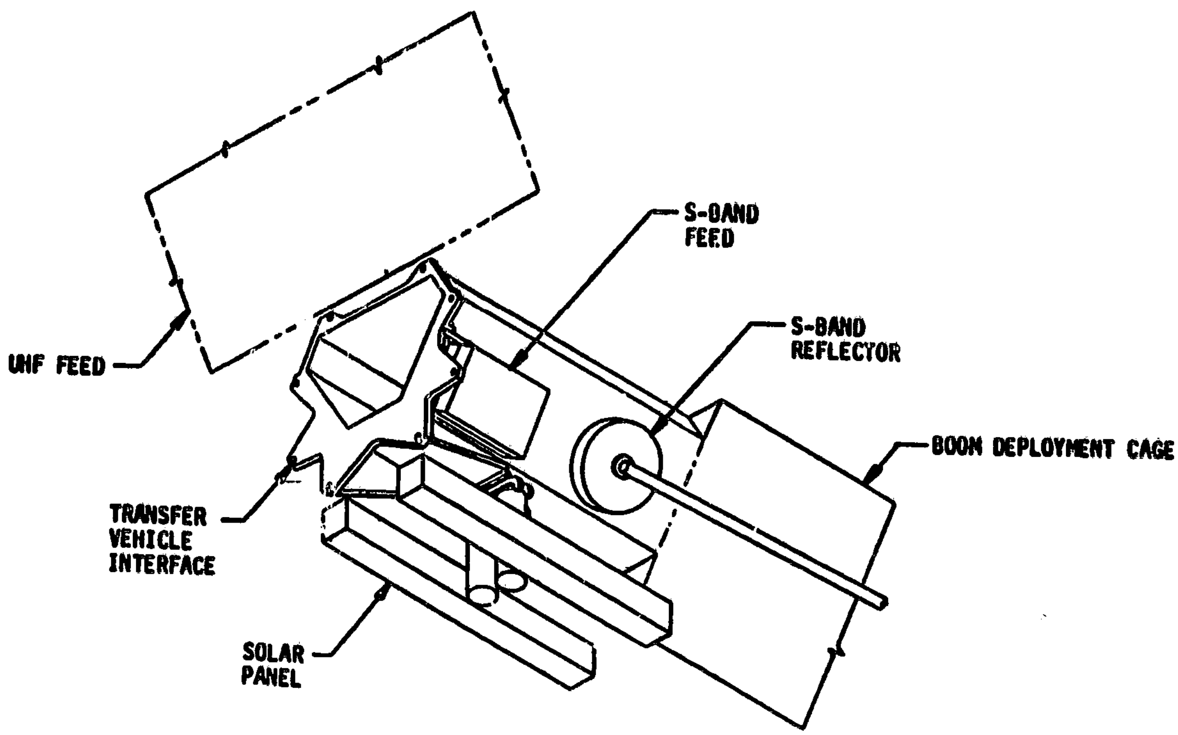
THE CONFIGURATION OF THE LMSS ADJACENT TO THE REFLECTOR HUB IS SHOWN ON THIS CHART. POINTING CONTROL OF THE DEPLOYED UHF REFLECTOR IS NOT SHOWN BUT MAY BE REQUIRED. THE SECONDARY ATTITUDE CONTROL EQUIPMENT IS LOCATED AS SHOWN.

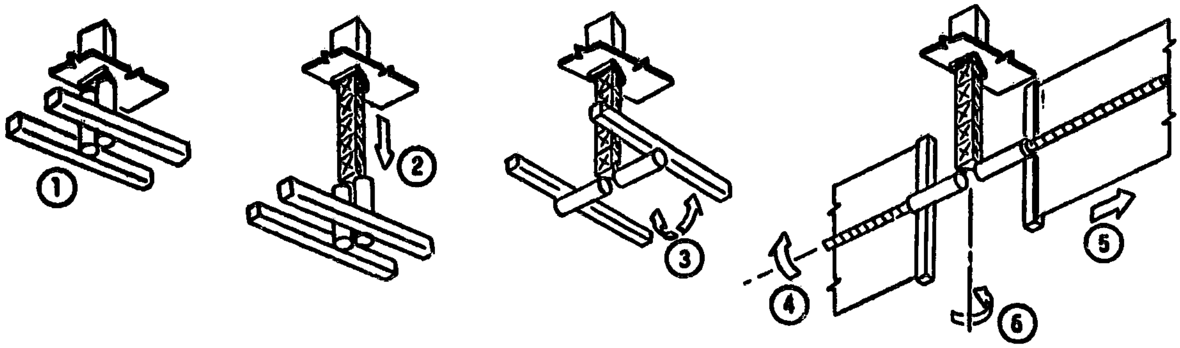


LMSS WRAP RIB
APPENDAGE STOWAGE



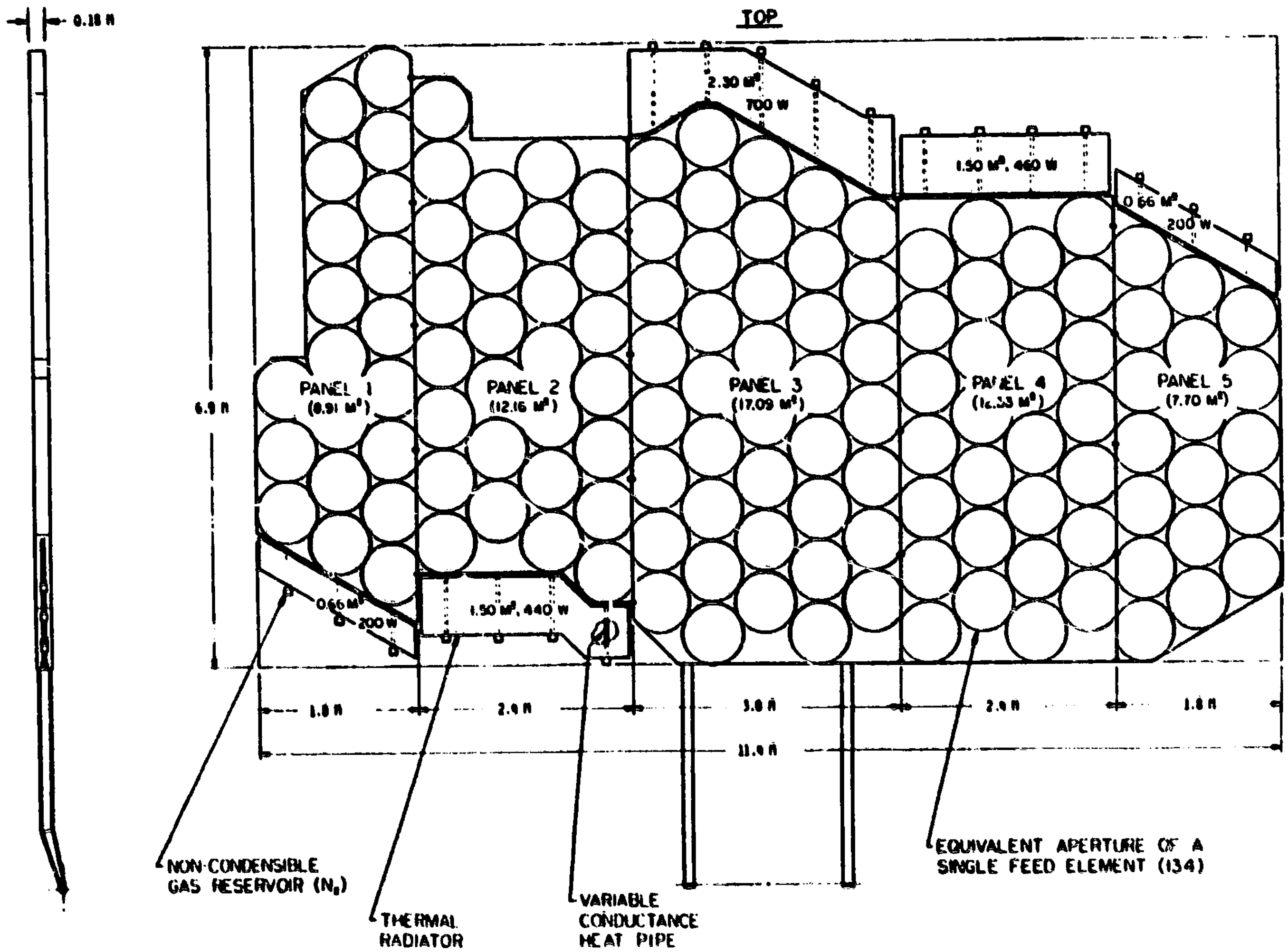
LMSS WRAP RIB
SOLAR PANEL DEPLOYMENT

SHOWN IS THE PROPOSED SOLAR PANEL DEPLOYMENT. A TIME LINE FOR THIS IN THE OVERALL DEPLOYMENT SEQUENCE HAS NOT BEEN ESTABLISHED. IT APPEARS TO BE AN INDEPENDENT EVENT THAT CAN BE SEQUENCED AT THE MOST DESIRABLE TIME. THE DEPLOYED SOLAR PANELS WILL REQUIRE "UNWINDING" ONCE A DAY TO PRECLUDE CABLE TWISTING. THIS MAY TAKE UP TO 30 MIN. TO ACCOMPLISH (A TIME EQUIVALENT TO THE MAXIMUM SHADOWING TO BE EXPERIENCED).



1 FULLY STOWED
2 MAST EXTENDED
3 CANNISTERS DEPLOYED
4 CANNISTERS ROTATED
5 ARRAY BLANKETS EXTENDED
6 ARTICULATED THROUGHOUT MISSION

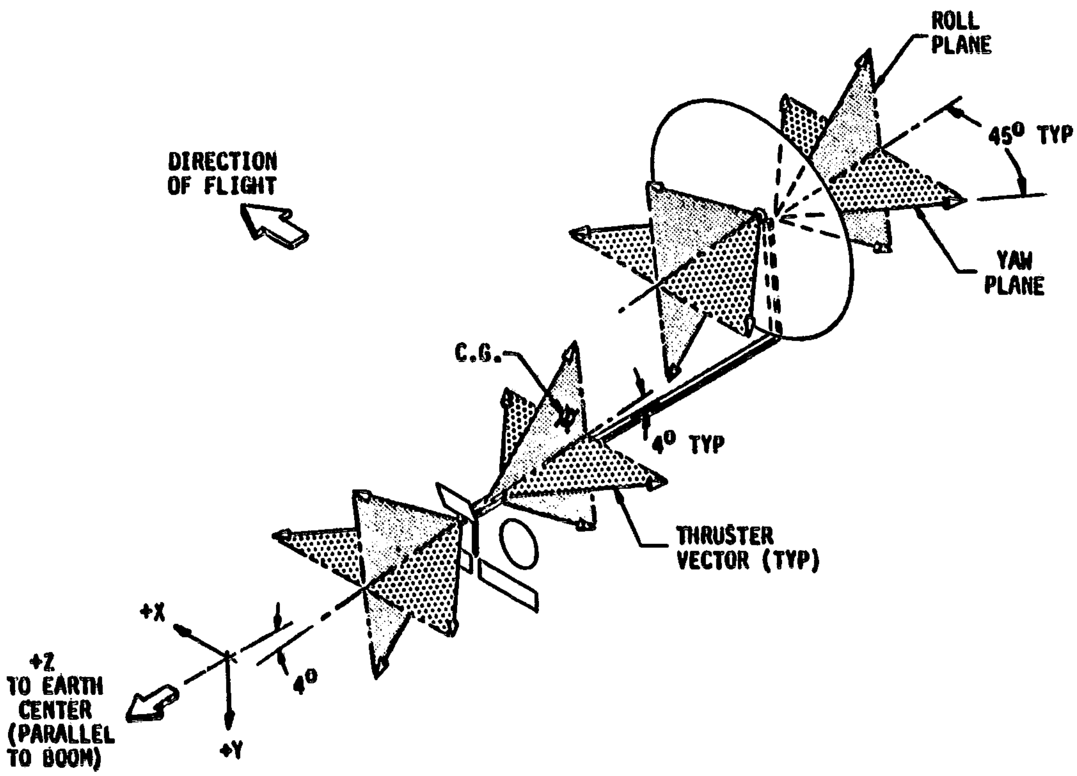
THE FACE OF THE FEED VIEWING THE 55 METER REFLECTOR IS SHOWN. THERE ARE 134 INDIVIDUAL PLANAR ARRAYS (87 TO COVER CONUS AND 47 AROUND THE PERIPHERY TO PROVIDE THE REQUIRED RF BEAM ISOLATION). THE FEED IS 6.9 M X 11.4 M X .018 M AND WEIGHS 1212 KG (2667 LBS). IN ADDITION TO THE PLANAR FEEDS, EACH FEED SEGMENT HAS A PASSIVE THERMAL RADIATOR ATTACHED TO DISSIPATE HEAT PRODUCED BY THE HIGH POWER AMPLIFIERS (ABOUT 2000 W TOTAL). THE ACTUAL AREA REQUIRED IS ONLY THE PLANAR ARRAYS AND RADIATORS. TO FACILITATE THE STRUCTURAL DESIGN AND MECHANISM IMPLEMENTATION 5 RECTANGULAR SEGMENTS MAY BE USED. WITHIN THE THICKNESS OF THE FEED ARE THE GROUND PLANES, BEAMFORMING NETWORK, AND ELECTRONICS (DIPLEXER, HIGH POWER AMPLIFIER, LOW NOISE AMPLIFIER, THERMAL CONTROL HEAT PIPES, ETC.).



LMSS WRAP RIB
PROPOSED THRUSTER LOCATIONS

ORIGINAL PAGE IS OF POOR QUALITY

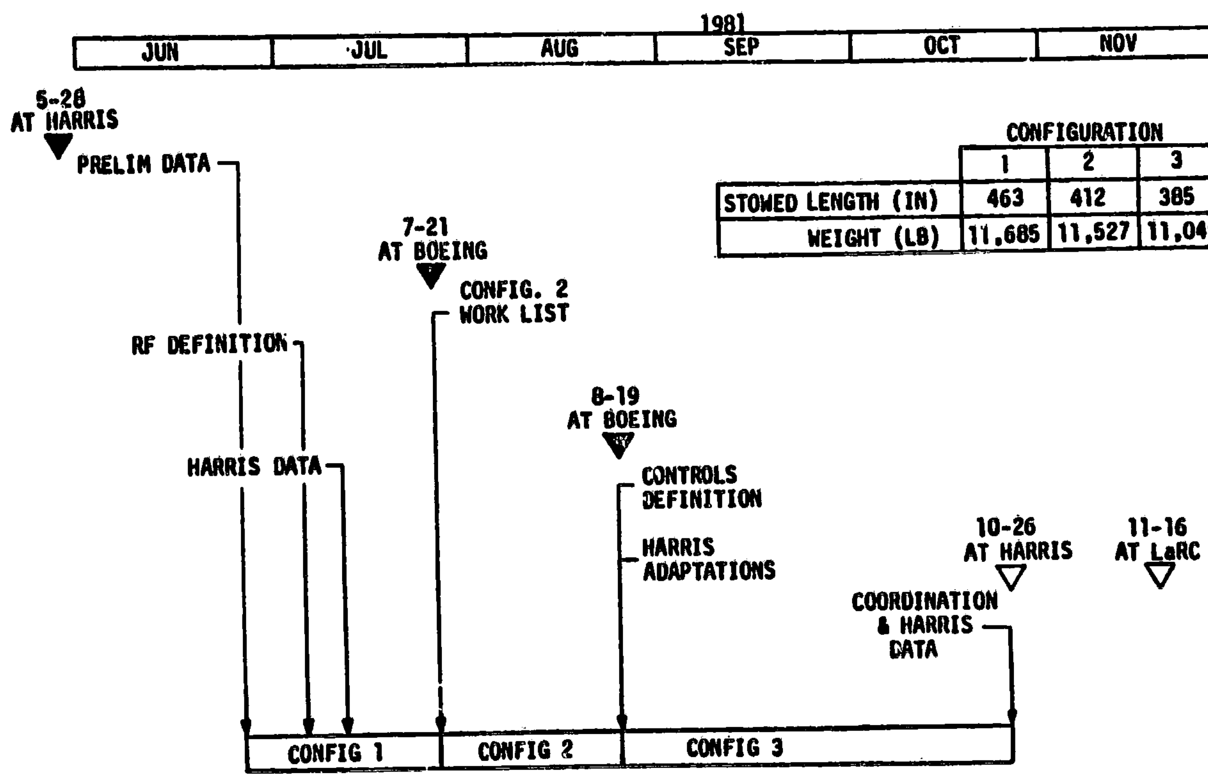
THE PROPOSED THRUSTER LOCATIONS ARE SHOWN ON THIS CHART. TO HAVE THE THRUSTERS IN THE USUAL ORIENTATION (ALONG PRIMARY AXES) IS NOT PRACTICAL FOR THIS CONFIGURATION. THE RESULT IS THRUSTER PLUME AND THRUST IMPINGEMENT ON MOST RADIATING AND STRUCTURAL MEMBERS. IT WAS DECIDED THAT A 30° HALF CONE SHOULD DESCRIBE THE PLUME AND WE WOULD USE A 45° CENTERLINE CLEARANCE. RIGID ADHERANCE TO THIS RESULTS IN CANTING THE THRUSTERS AT THE SPACECRAFT BUS (FEED AREA) 4° IN THE Y PLANE TO CLEAR THE FEED. (THIS MAY BE CHANGED DURING DETAIL DESIGN.) DURING DEPLOYMENT (STOWED PACKAGE CONFIGURATION SEPARATED FROM THE TRANSFER VEHICLE) TWO SETS OF THRUSTERS ARE OPERABLE: (1) THE +Z SET OF 4 THRUSTERS AS SHOWN; AND (2) THE SET OF 4 THRUSTERS FROM THE CENTER OF THE 55 M REFLECTOR WHICH WILL BE POINTED IN THE -Z DIRECTION AT THE -Z END OF THE STOWED PACKAGE. IN THIS POSITIONING ROLL CONTROL IS QUESTIONABLE, AND ADDITIONAL THRUSTERS MAY BE REQUIRED FOR THE INITIAL DEPLOYMENT EVENTS.



LMSS - HOOP COLUMN CONFIGURATION

THE EVOLUTION OF THE HOOP COLUMN IS SHOWN. TO DATE THE TREND HAS BEEN TO IMPROVE THE DESIGN, SHORTEN THE STOWED LENGTH AND REDUCE THE WEIGHT WITH EACH REVISED CONFIGURATION. THE STOWED LENGTH AND WEIGHT ARE NOW WITHIN THE RANGE OF THE LARGER PROPOSED 1995 TRANSFER VEHICLES.

LMSS HOOP COLUMN
BOEING STUDY HISTORY



	CONFIGURATION 1	2	3
STOWED LENGTH (IN)	463	412	385
WEIGHT (LB)	11,685	11,527	11,045

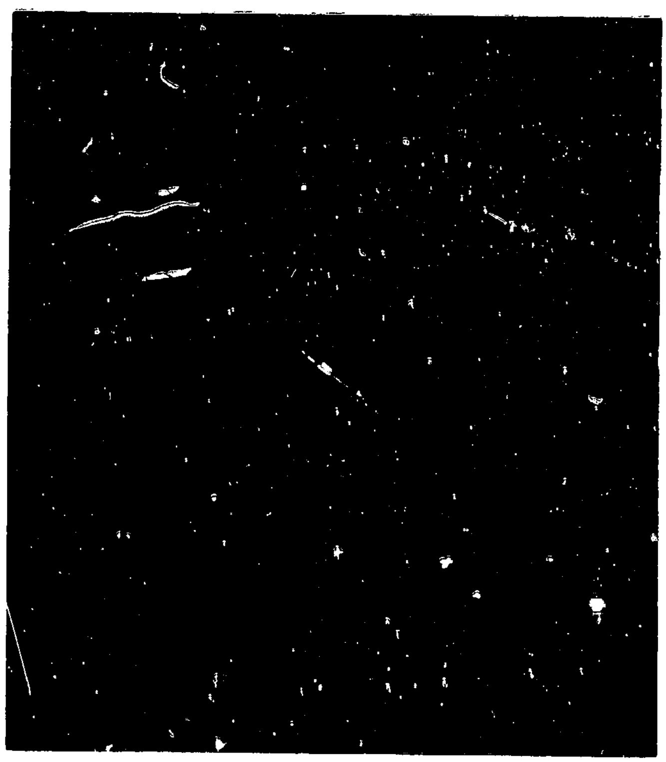
LAND MOBILE SATELLITE SYSTEM SPACECRAFT
120 METER HOOP COLUMN CONCEPT

This is a preliminary concept of a quad aperture reflector spacecraft capable of relaying radio messages to land mobile units throughout the United States. It is intended to service units such as ambulances, police cars, taxis -- in essence any radio dispatched vehicles. Its position in geosynchronous orbit avoids present radio interferences caused by tall buildings, hills, and other factors. It is one of numerous concepts for extending current space technology made feasible by the Space Shuttle. The spacecraft is sized for a single Shuttle launch and a 10-year life beginning in the mid-1990's.

The central column points at the center of the United States near Kansas City. Each of the four feed panels at the upper left projects a multiple beam pattern onto its assigned quadrant on the large, molybdenum-mesh reflector. There are uplink and downlink feeds for both the eastern and western halves of the country. The radio beams are arranged to cover all contiguous 48 states with a potential in the design for communication with Alaska, Hawaii, and parts of Canada.

Major contributors to the development of this configuration are:

System Design	The Boeing Aerospace Co.
Hoop Column Design	The Harris Corp.
RF Subsystem	General Electric
Controls Subsystem	Jet Propulsion Laboratory
Structural Dynamics	The Harris Corp.



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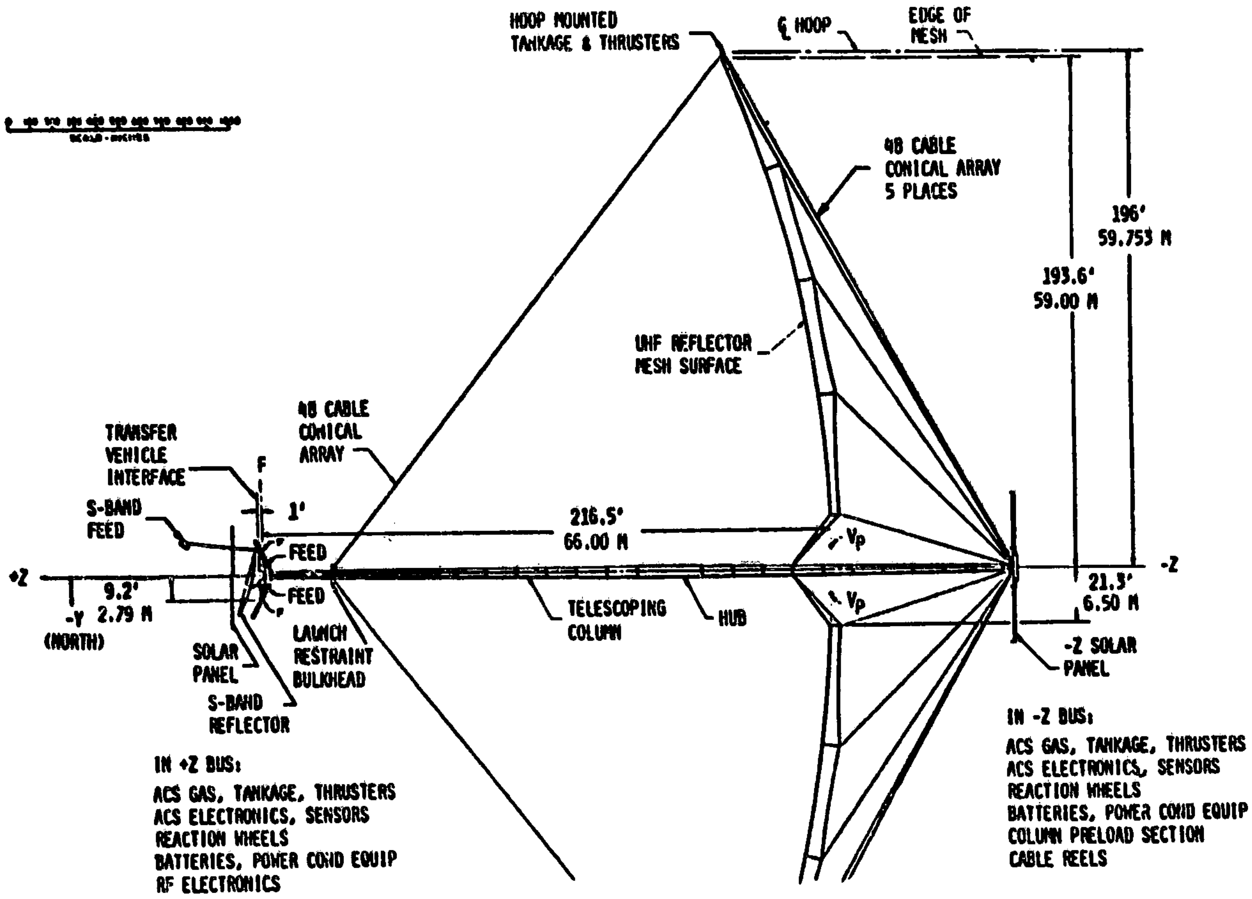
THIS SECTIONAL SIDE VIEW THROUGH THE HOOP COLUMN N-S PLANE SHOWS THE OVERALL SIZE OF THE SPACECRAFT. THE REFLECTOR IS ABOUT 120 METERS IN DIAMETER, AND THE COLUMN INCLUDING S-BAND FEED AND -Z BUS IS 97 M. THE WEIGHT STATEMENT ASSOCIATED WITH THIS CONFIGURATION IS SHOWN ON THE SECOND CHART.



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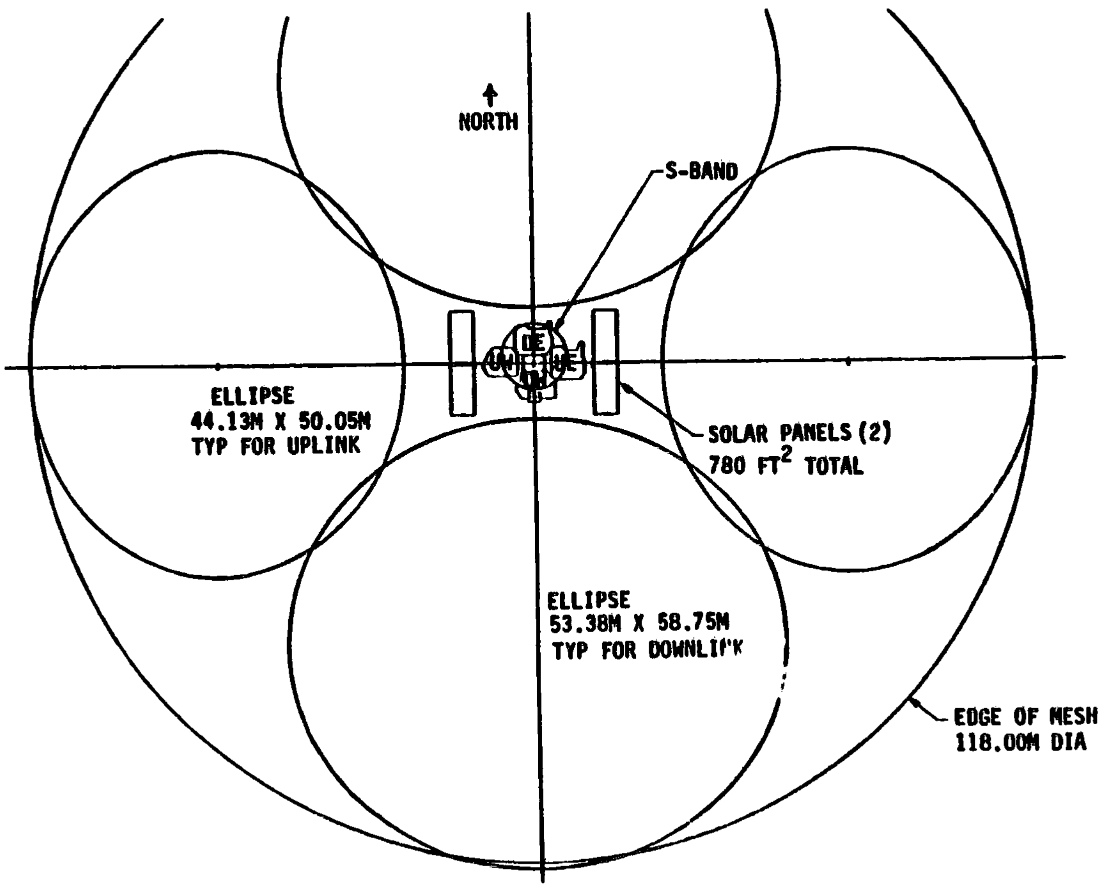
LMSS HOOP COLUMN WEIGHT STATEMENT

'OOP, COLUMN, REFLECTOR STRUCTURE TOTAL		(3201)	
1	COLUMN		2067
2	HOOP		729
3	MESH & CABLES		405
INSULATION		(238)	
4A	MLI, +Z BUS		35
4B	MLI, -Z BUS		23
4C	MLI, HOOP		180
S-BAND ASSY.		(385)	
5	S-BAND REFLECTOR & BOOM		130
6	S-BAND FEED, BOOM, COAXES		255
UHF FEEDS, ELECTRONICS		(2267)	
7	FEEDS (INCLUDES STRUCT, TC, CABLING)		2060
8	ELECTRONICS (TELEMETRY)		207
ELEC. POWER SUBSYSTEM		(891)	
9A	+Z SOLAR PANELS, BOOMS, MECHANISMS		378
9B	-Z SOLAR PANELS, BOOMS, MECHANISMS		185
10A	+Z BATTERIES, POWER COND EQUIP.		235
10B	-Z BATTERIES, POWER COND EQUIP.		93
CONTROL SUBSYSTEM		(3439)	
11A	+Z REACTION WHEELS & SENSORS, ELECTRONICS		497
11B	-Z REACTION WHEELS & SENSORS, ELECTRONICS		240
12A	+Z TANKAGE, GAS, THRUSTERS		1399
12B	-Z TANKAGE, GAS, THRUSTERS		953
12C	HOOP MOUNTED TANKAGE, GAS, THRUSTERS		350
BUS STRUCTURE		(430)	
13A	+Z BUS STRUCTURE		280
13B	-Z BUS STRUCTURE		150
ELEC. CABLING (EXCEPT RF) & MISC.		(194)	
14	CABLING (IN BUSSES & COLUMN)		150
15	MISC. (INSTRUMENTATION, OPTICAL ALIGNMENT)		44
		TOTAL	11,045 LB.
			(5,009 KG)

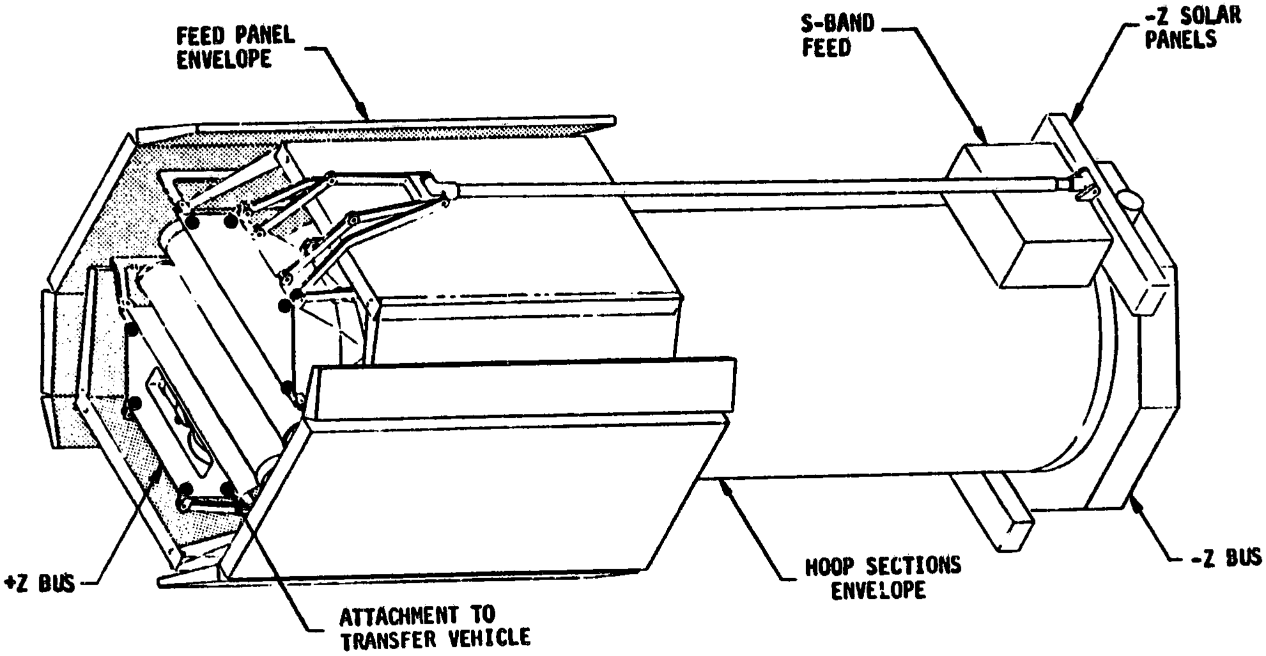
LMSS HOOP COLUMN
QUAD APERTURES

A VIEW OF THE QUAD-APERTURE HOOP COLUMN IS SHOWN LOOKING FROM THE EARTH AT THE UHF REFLECTOR. THE FOUR SLIGHTLY ELLIPTICAL SPOTS ARE THE ANTENNA APERTURES. THE TWO LARGER ONES IN THE N-S PLANE (VERTICAL) ARE THE DOWNLINK FOR THE EASTERN AND WESTERN HALVES OF THE UNITED STATES. THE SLIGHTLY SMALLER APERTURES IN THE E-W PLANE (HORIZONTAL) ARE FOR UPLINK COMMUNICATIONS. THE SOLAR PANELS, FEEDS (4), AND S-BAND ANTENNA ARE CLUSTERED IN THE CENTER. THIS PARTICULAR CONFIGURATION REQUIRES SOLAR PANEL ARTICULATION ABOUT BOTH AXES. IF THE PATTERN WERE ROTATED 90^0, SINGLE AXIS ARTICULATION OF THE SOLAR PANELS WOULD BE ADEQUATE. AN ANALYSIS ON THE IMPACT ON RF PERFORMANCE IS BEING MADE TO SEE IF THIS IS PRACTICAL.



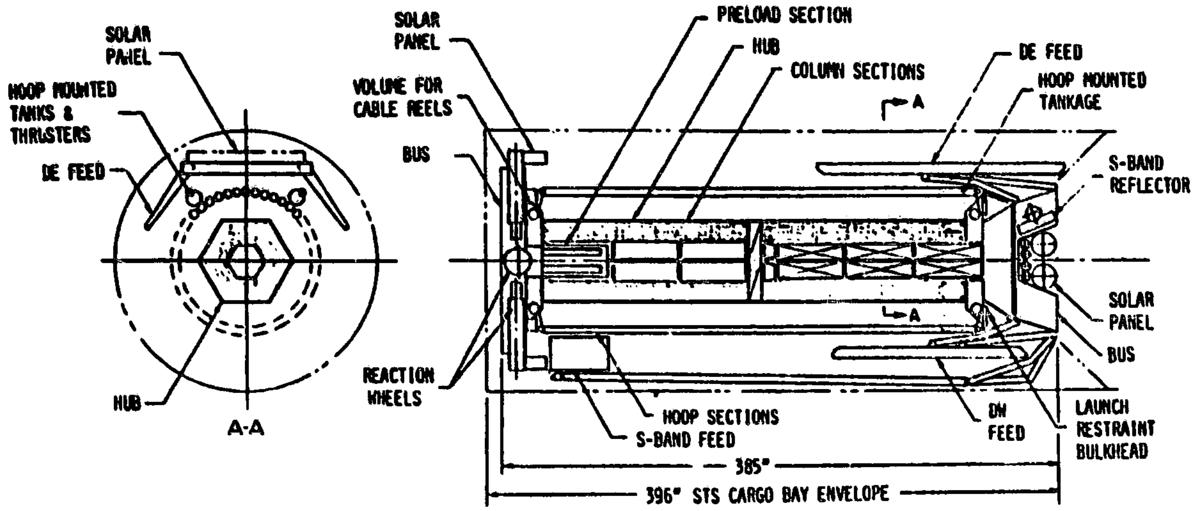
LMSS HOOP COLUMN
LAUNCH CONFIGURATION

THIS CHART SHOWS AN ISOMETRIC OF THE STOWED HOOP COLUMN. THE ATTACHMENT TO THE TRANSFER VEHICLE IS AT THE +Z BUS.



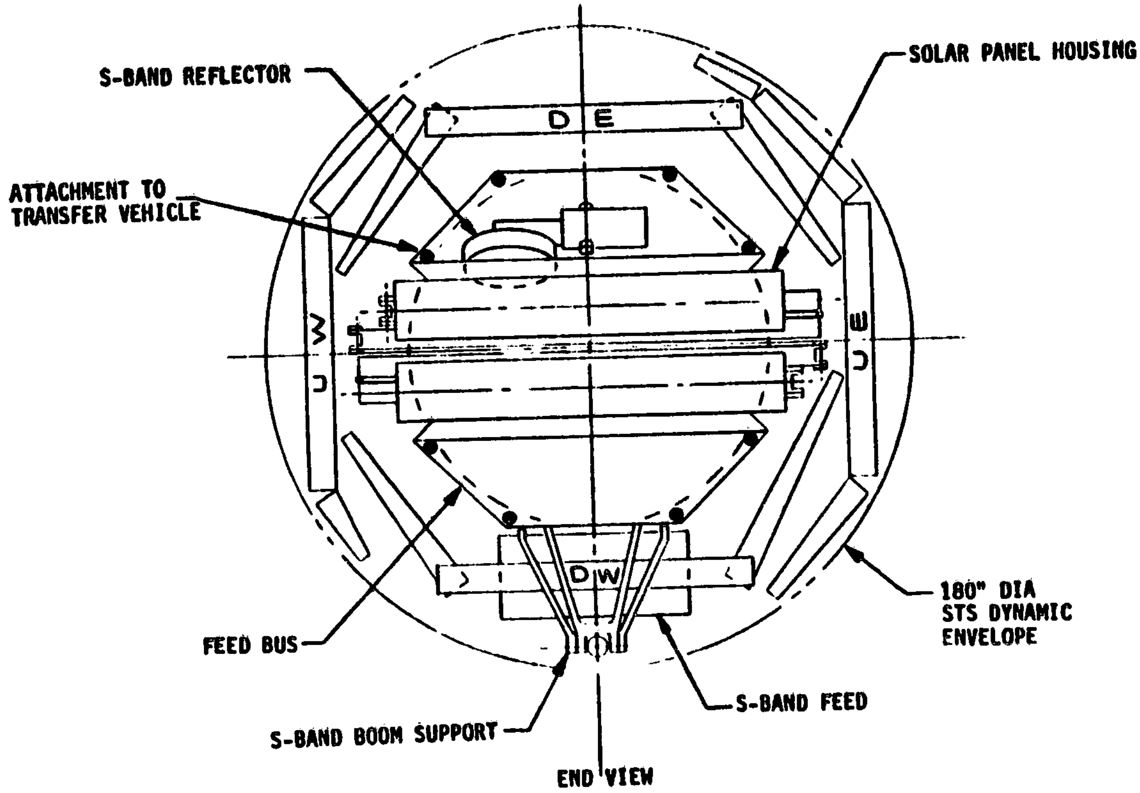
LMSS HOOP COLUMN
LAUNCH CONFIGURATION

THE STOWED CONFIGURATION IS CONSTRAINED BY THE 180" DIAMETER OF THE CARGO BAY AND THE LENGTH AVAILABLE AFTER INSTALLING AN APPROPRIATE SIZED ORBIT TRANSFER STAGE. FROM THE WEIGHT STATEMENT AN 11,000 LB. CAPABILITY TRANSFER STAGE IS REQUIRED. THERE ARE THREE CONFIGURATION DRIVERS WITH RESPECT TO THE HOOP COLUMN LAUNCH CONFIGURATION:
(1) DIAMETER REQUIRED TO STOW THE 4 FEED ARRAYS; (2) THE LENGTH REQUIRED FOR THE HOOP COLUMN MEMBERS PLUS THE +Z AND -Z BUSSES; AND (3) THE LENGTH REQUIRED FOR THE S-BAND FEED BOOM. AT THIS TIME ALL FIT, BUT THE MARGIN OR POTENTIAL FOR GROWTH OR REDUCING THE SIZE OF THE LAUNCH CONFIGURATION IS SLIGHT. ASSUMING THE 12,000 LB. CAPABILITY TRANSFER STAGE HAS A LENGTH OF 324", THERE IS ONLY 11" CLEARANCE IN THE LENGTH OF THE SHUTTLE CARGO BAY.



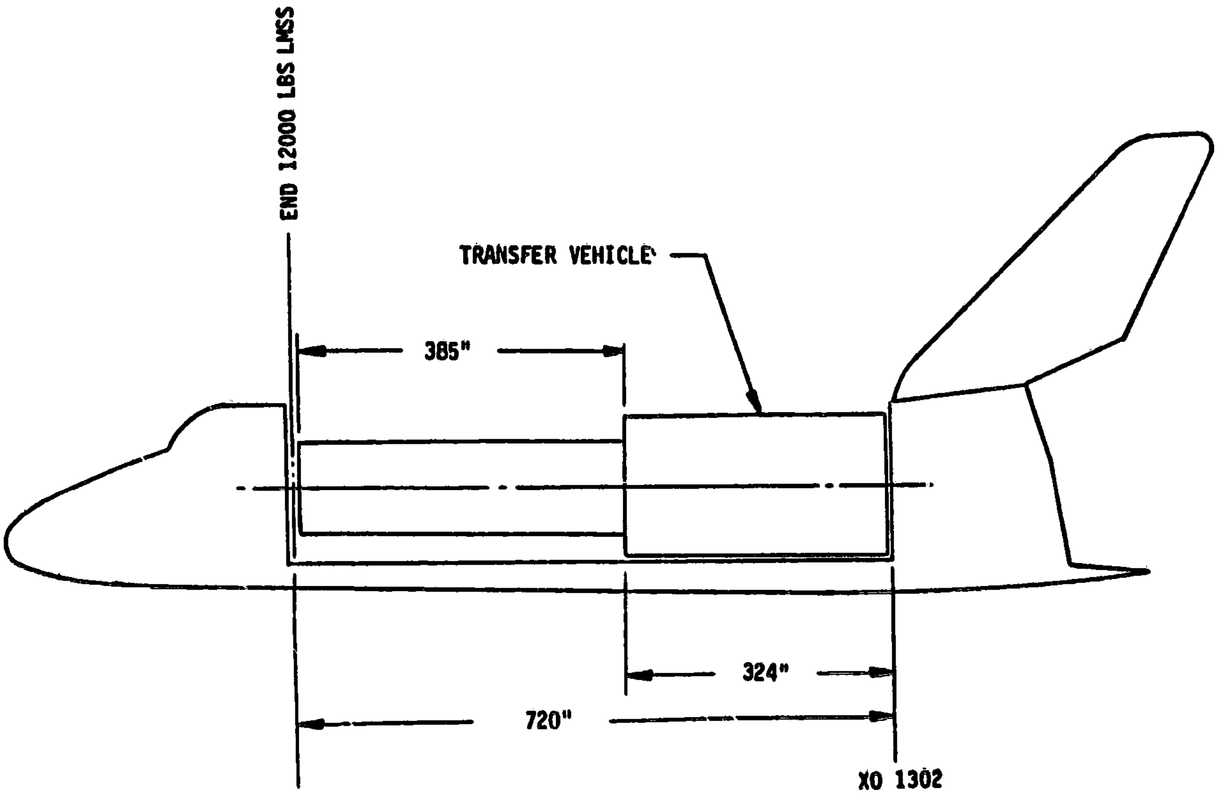
LMSS HOOP COLUMN
APPENDAGE STOWAGE

THIS CHART SHOWS AN END VIEW OF THE HOOP COLUMN LAUNCH CONFIGURATION AT THE INTERFACE WITH THE TRANSFER STAGE.



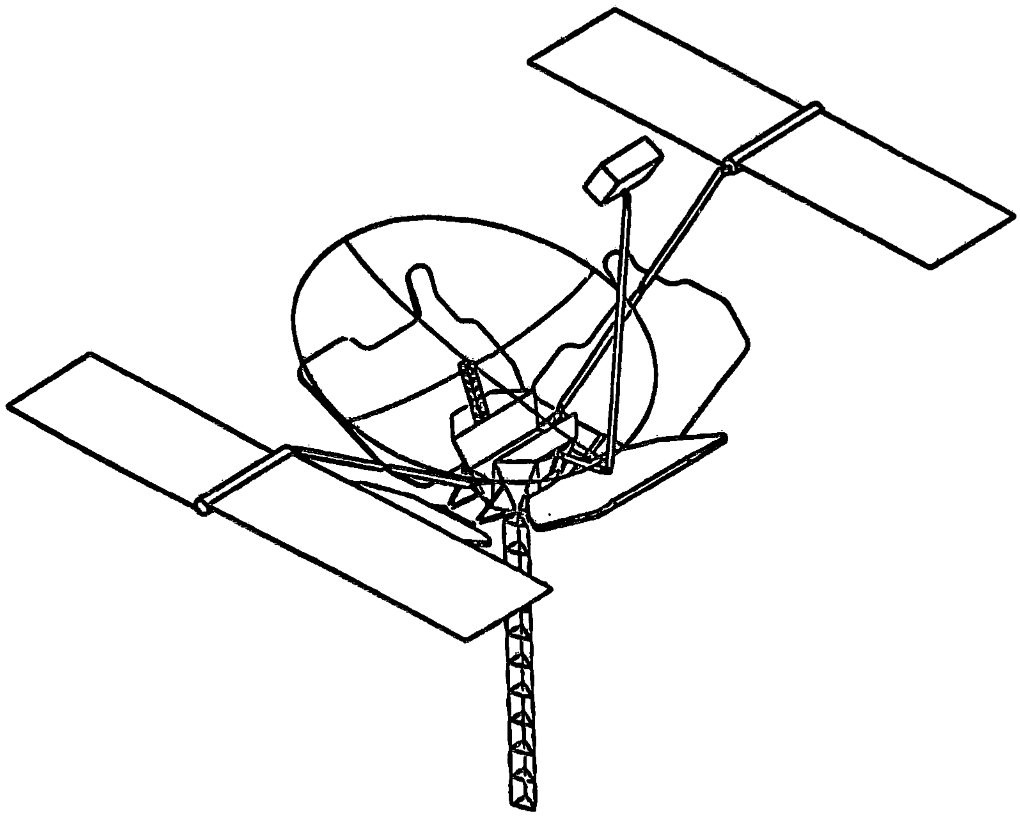
LMSS HOOP COLUMN IN CARGO BAY

THE HOOP COLUMN LMSS/TRANSFER STAGE VEHICLE COMES WITHIN 11" OF FILLING THE STS CARGO BAY LENGTH. THE 324" TRANSFER VEHICLE LENGTH IS THE GREATEST OF ANY PROPOSED, BUT SHORTER TRANSFER VEHICLES DO NOT HAVE THE CAPABILITY TO INSERT AN LMSS OF ABOUT 11,000 LBS. INTO GEOSYNCHRONOUS ORBIT.



LMSS HOOP COLUMN
FEED AREA

THIS CHART SHOWS THE DEPLOYED ELEMENTS AT THE FEED END (+Z BUS) OF THE COLUMN. THE SUBSEQUENT CHARTS IN THIS SERIES SHOW CROSSECTIONS IN THE N-S AND E-W PLANES OF THE FEED AND THEN A VIEW OF THE FEEDS AND OTHER APPENDAGES LOOKING TOWARD EARTH.



LMSS HOOP COLUMN
NORTH-SOUTH PLANE

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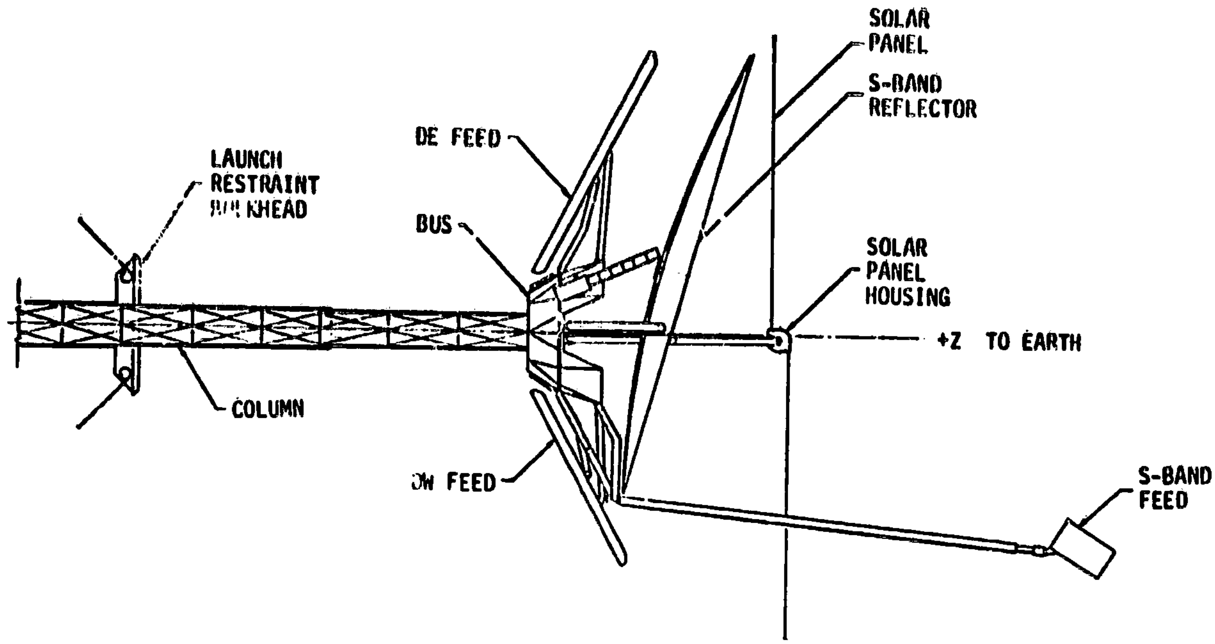
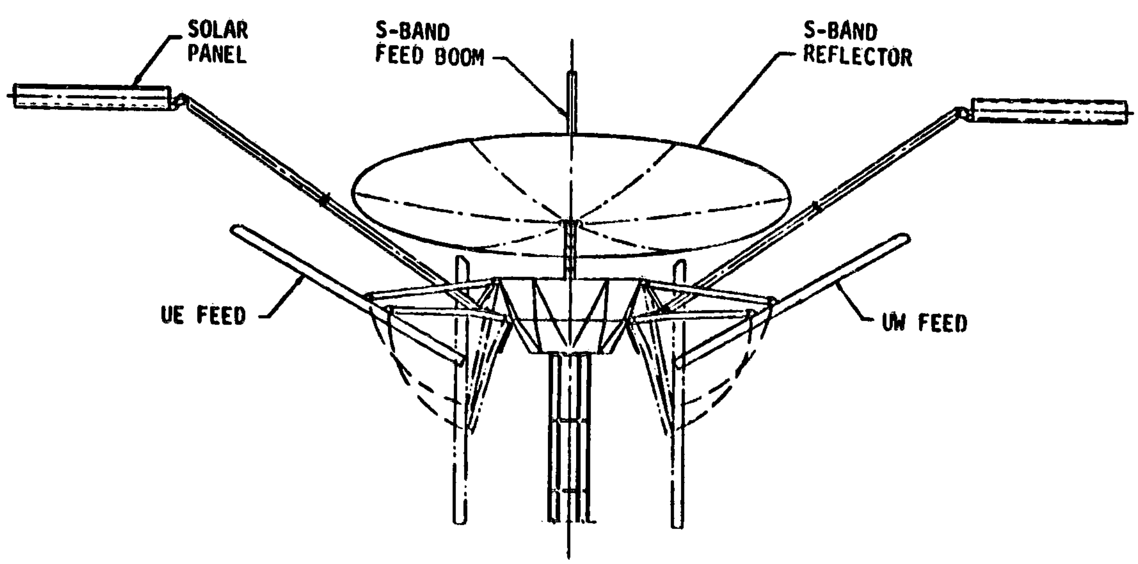


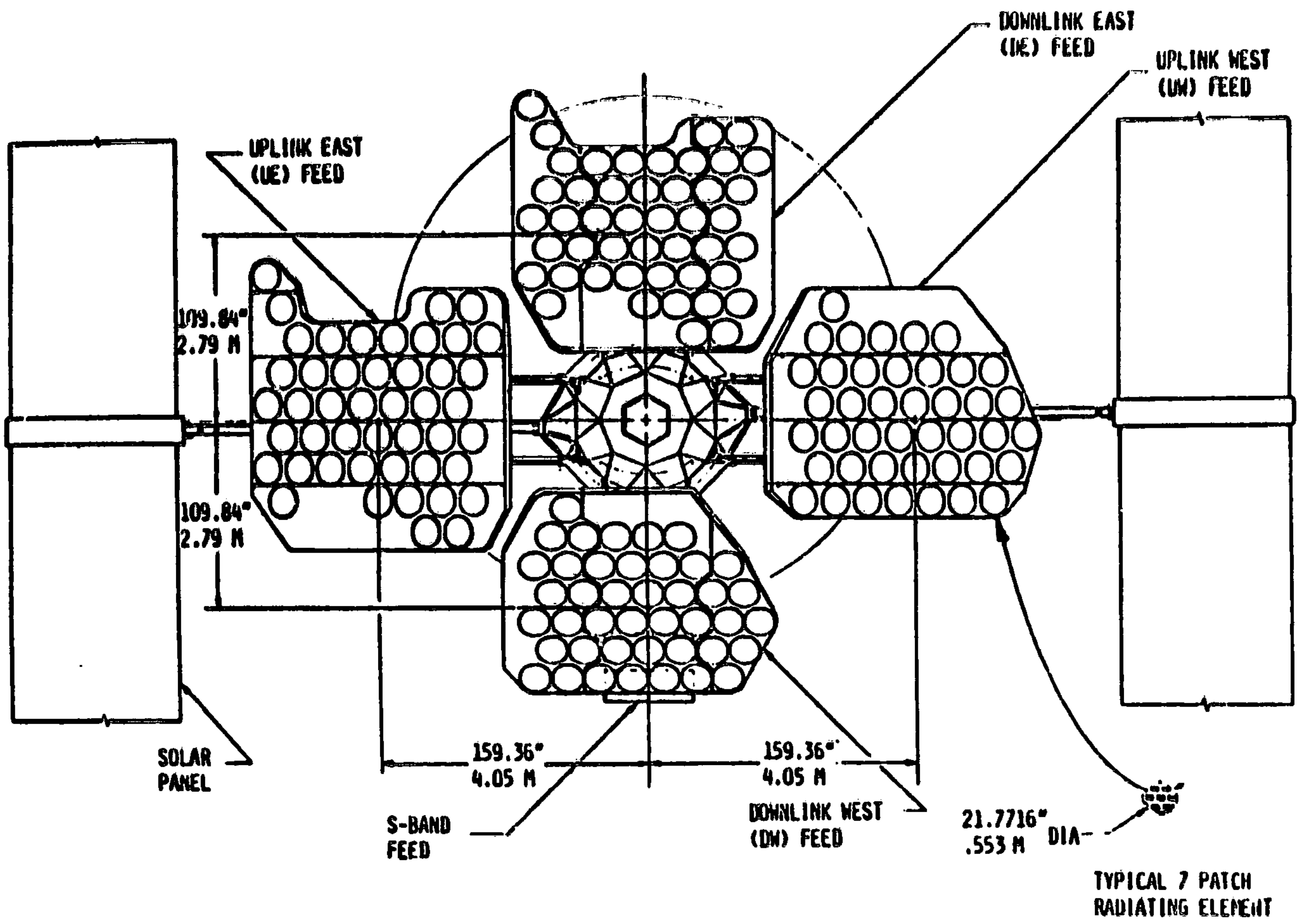
FEED AREA

LMSS HOOP COLUMN
EAST-WEST PLANE



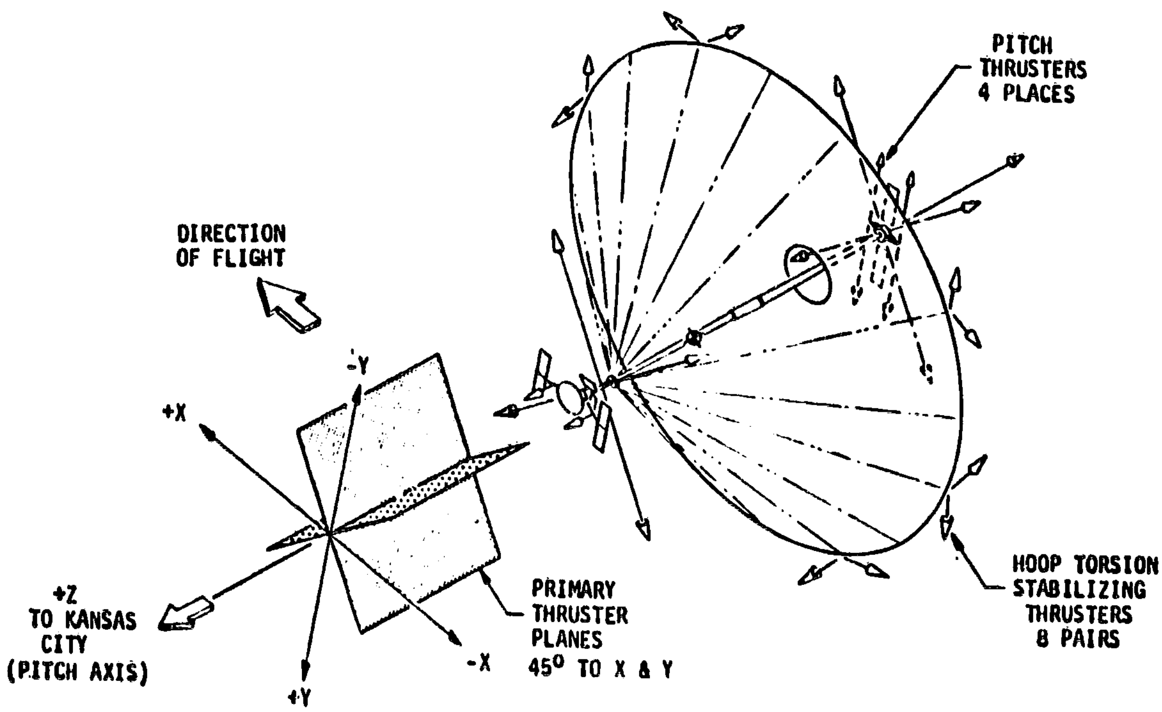
FEED AREA

LMSS HOOP COLUMN
FEED AREA - VIEW TOWARD EARTH

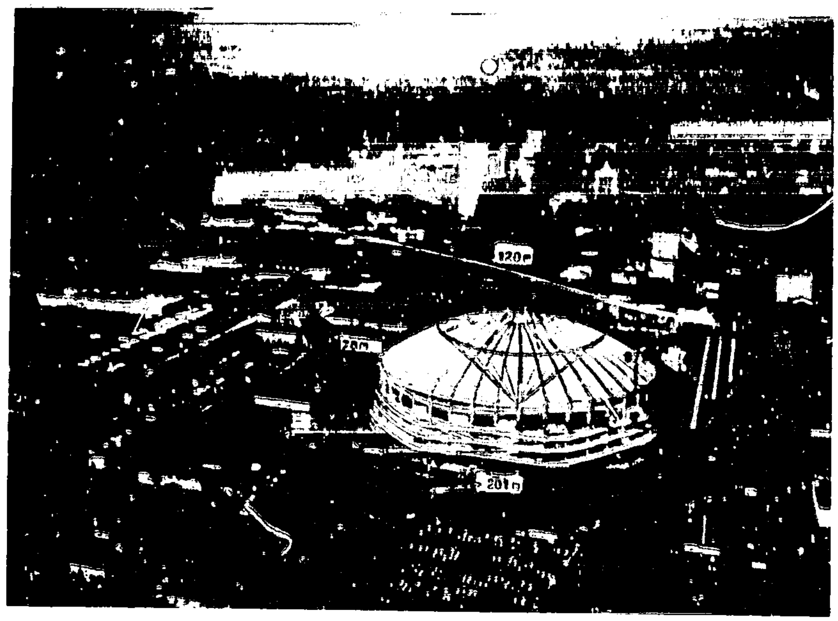


LMSS HOOP COLUMN
PROPOSED THRUSTER LOCATIONS

THE PROPOSED LOCATION FOR ATTITUDE CONTROL THRUSTERS IS SHOWN. THE HOOP COLUMN CONFIGURATION IS ESSENTIALLY SYMMETRICAL ALONG THE COLUMN AXIS WITH THE CG DISPLACED SLIGHTLY IN THE +Z DIRECTION. DUE TO ITS GEOMETRY, THERE ARE FEWER PROBLEMS WITH RESPECT TO PLUME IMPINGEMENT ON ADJACENT MEMBERS. AN AREA OF CONCERN IS PITCH CONTROL OF THE UHF REFLECTOR. THE TENSION TIES TO THE HOOP IN THE FORM OF THE QUARTZ AND GRAPHITE EPOXY CABLES DO NOT PROVIDE ANY APPRECIABLE TORSIONAL STIFFNESS. AT THIS TIME THE CONTROLS GROUP (JPL) AND THE HOOP COLUMN DESIGNERS (THE HARRIS CORP.) ARE EXAMINING PLACING 8 PAIRS OF THRUSTERS ON THE PERIPHERY OF THE HOOP TO PROVIDE CONTROL CAPABILITY OF THE REFLECTOR.



THE DEPLOYED HOOP COLUMN LMSS IS SHOWN RELATIVE TO THE SEATTLE KINGDOME (SPORTS STADIUM). ALTHOUGH THE KINGDOME IS A LARGE STADIUM, WE WOULD HAVE TO RAISE THE ROOF ~40 M OR DIG A DEEP HOLE IN THE CENTER OF THE FOOTBALL FIELD TO BE ABLE TO DEPLOY THE HOOP COLUMN LMSS INSIDE. AS FAR AS WE KNOW, THERE IS NO FACILITY CAPABLE OF 0G TESTING OF THESE TYPE AND SIZE STRUCTURES.



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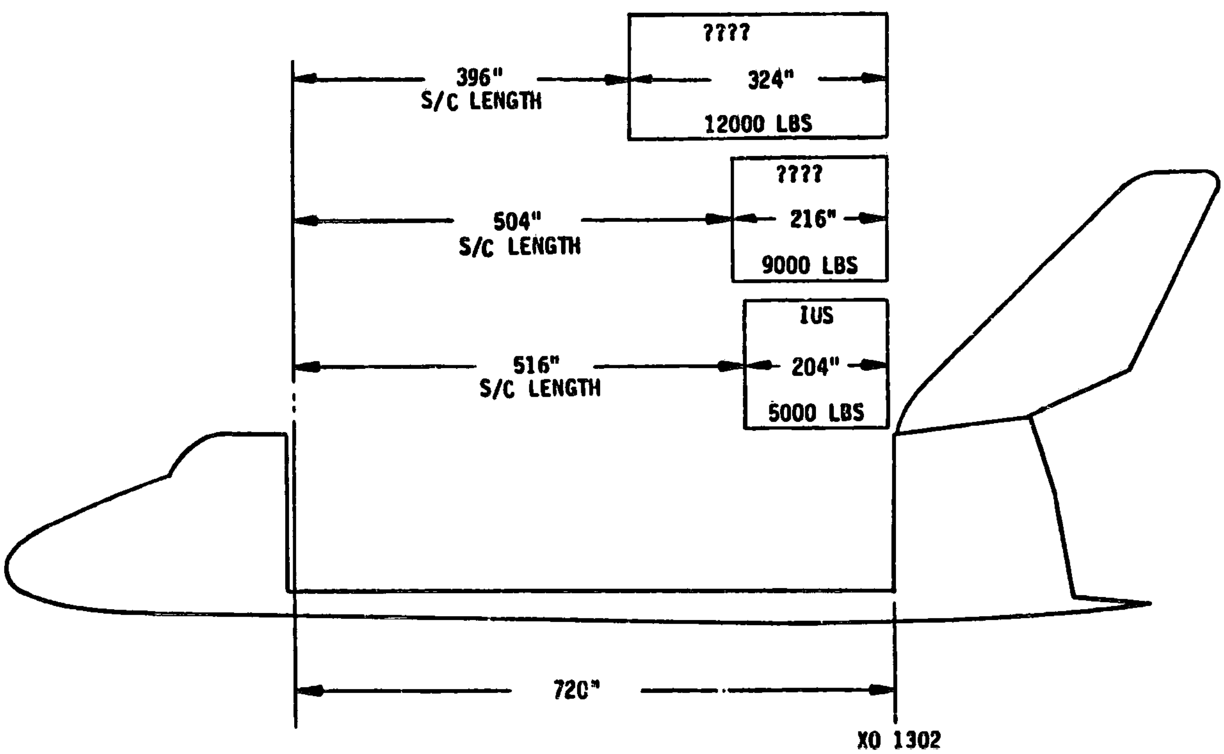
INTEGRATION RESULTS

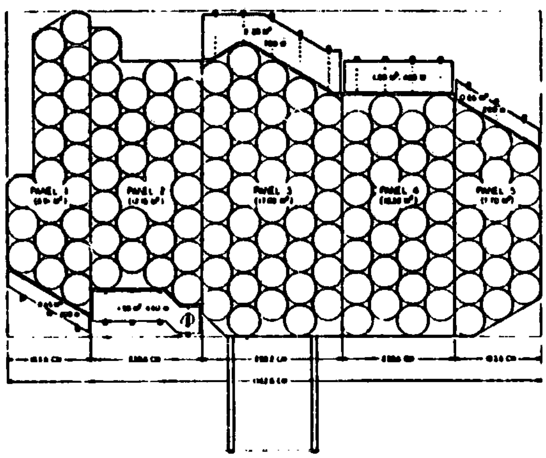
AS A RESULT OF THE WORK CONDUCTED TO DATE THERE ARE SEVERAL AREAS OF CONCERN. THESE VARY FROM HAVING A SLIGHT ADVERSE EFFECT (INCREASED WEIGHT) TO EVENTS THAT CAN SCRUB THESE MISSIONS AS NOW CONCEIVED (NO TRANSFER VEHICLE WITH COMPATIBLE SIZE AND PERFORMANCE). PERHAPS THE MOST SIGNIFICANT OF THESE IS PLANNED TRANSFER CAPABILITY AND SIZE IN THE 1985 TO 2000 TIMEFRAME. SOME DEFINITIVE PLAN IS NEEDED.

STOWAGE VOLUME
SPACECRAFT & TRANSFER VEHICLE

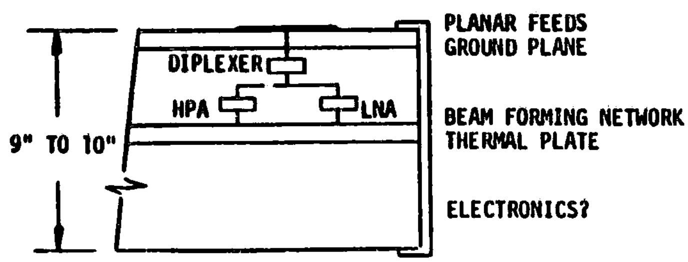


FEED ASSEMBLY
TECHNOLOGY DEVELOPMENT AREAS

THE PROPOSED FEED FOR THE WRAP RIB LMSS IS SHOWN, BUT THE TECHNOLOGY DEVELOPMENT REQUIRED APPLIES TO THE HOOP COLUMN FEED AND PARTIALLY TO THE S-BAND FEEDS FOR BOTH. THERE IS A HIGH DEGREE OF CONFIDENCE THAT THE CONCEPTS PROPOSED AND THE WORK ACCOMPLISHED WILL LEAD TO VIABLE DESIGN SOLUTIONS. BUT THE CRITICALITY OF THE FEED SIZE, WEIGHT, HINGING, PACKAGING AND PERFORMANCE WITH RESPECT TO THE TOTAL SYSTEM CANNOT BE OVERSTATED. IF THE LMSS IS GOING TO WORK, THE FEED CONCEPTS MUST BE DEVELOPED AND VERIFIED EARLY (ASAP).



- RF DESIGN
  - FREQUENCY/POLARIZATION PLAN
  - RADIATING ELEMENTS
  - BEAM FORMING NETWORK (BFN)
- STRUCTURAL/MECHANICAL DESIGN
  - DEPLOYMENT
  - HINGE LINES
    - COAXIAL CABLE RUNS
    - BFN
  - WEIGHT
  - THERMAL CONTROL

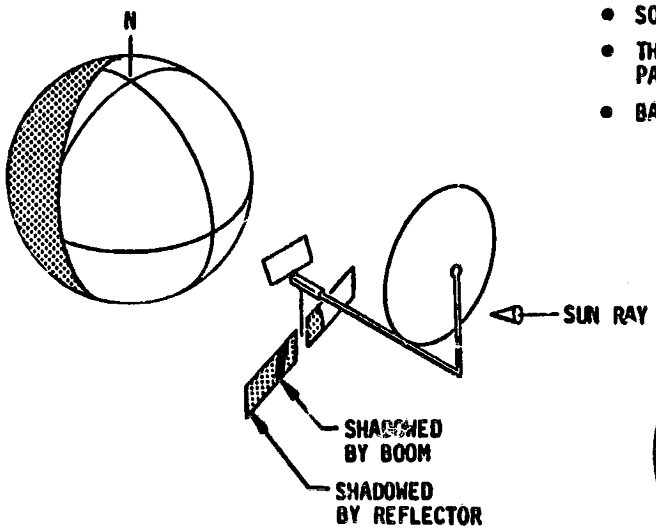


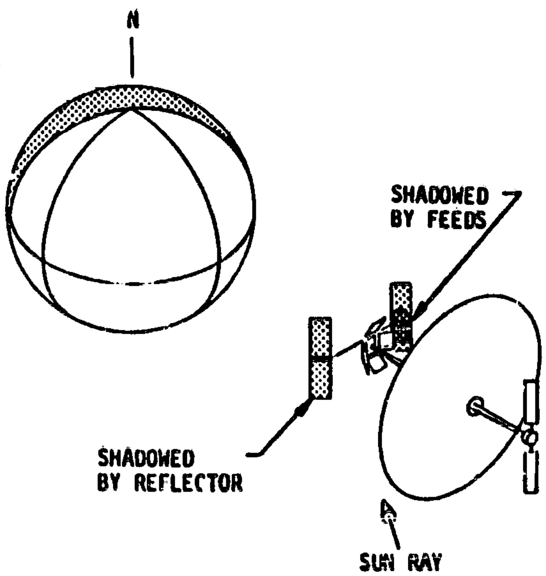
PARTIAL X SECTION

LMSS WRAP RIB & HOOP COLUMN
SOLAR PANEL LOCATIONS

THE SIZE AND PACKAGING CONSTRAINTS OF BOTH LMSS CONCEPTS SEEM TO REQUIRE PARTICULAR SOLAR PANEL PLACEMENT THAT DOES NOT PROVIDE CONTINUOUS SUN ILLUMINATION. AT THE EQUINOX THE PANELS WILL BE IN AN ECLIPSE CONDITION FOR ABOUT 7^{o} (30 MIN.); THUS BATTERIES ARE BEING PROVIDED. THESE WILL ALSO COME INTO PLAY DURING THE DAILY SOLAR PANEL UNWIND. THE PROBLEM IS THAT DURING OTHER TIMES IN THE ORBIT, THE ANTENNAS AND STRUCTURAL MEMBERS WILL CAST SHADOWS ON THE SOLAR ARRAYS. TO DATE THE SEVERITY OF THIS HAS NOT BEEN INVESTIGATED. AN ANALYSIS SHOULD BE MADE SOON, SINCE THIS MAY ADVERSELY AFFECT WEIGHT (SOLAR PANELS AND BATTERIES) AND PACKAGING.

- SOLAR PANELS TO BE LOCATED NEAR PRIME USER
- THESE EARTH POINTING SATELLITES WILL ENCOUNTER SOLAR PANEL SHADOWING
- BATTERIES REQUIRED DURING ECLIPSE PERIODS





STRUCTURAL SUBSYSTEM

ALTHOUGH SEVERAL STRUCTURAL MODIFICATIONS WERE MADE ON BOTH CONFIGURATIONS, NONE OF THESE POSES A SIGNIFICANT PROBLEM. A POTENTIAL STRUCTURAL PROBLEM MAY WELL BE IN THE INHERENT STIFFNESS OF BOTH CONFIGURATIONS. WITHIN THE PACKAGING CONSTRAINTS IT DOES NOT APPEAR THAT ANY APPRECIABLE INCREASE IN STIFFNESS IS POSSIBLE.

- REDUCED WRAP RIB BOOM CROSSECTION FROM 120" TO 108" TO 95" TO ACCOMMODATE FEED STOWAGE
- LOCATED TANKAGE AND SOLAR PANELS TO REDUCE GRAVITY GRADIENT TORQUES
- REVISED COLUMN DESIGN TO SHORTEN STOWED LENGTH

CONTROL SUBSYSTEM

THE ATTITUDE CONTROL SUBSYSTEM FOR BOTH CONFIGURATIONS REQUIRES AN ADVANCEMENT IN TODAY'S TECHNOLOGY. THE POINTING STABILITY REQUIREMENT OF 0.03° COMBINED WITH THE COMPARATIVELY LOW STRUCTURAL FREQUENCY AND LARGE SIZE RESULT IN A NEED TO CONTROL BOTH ENDS OF THE CONFIGURATION. THE REQUIREMENT FOR ABSOLUTE POINTING OF 0.10° REQUIRES TIGHT IF NOT IMPRACTICAL STRUCTURAL ALIGNMENT OF FEEDS, BOOMS AND REFLECTORS. THIS MAY LEAD TO A REQUIREMENT FOR ON ORBIT SPATIAL SENSING AND ADJUSTMENT CAPABILITY. A CONTROL CONCEPT HAS BEEN PROPOSED AND SUPPORTING ANALYSIS MUST BE COMPLETED TO VERIFY THE DESIGN SOLUTION.

- IDENTIFIED THE NEED FOR TWO POINT CONTROL (SPACECRAFT AND WRAP RIB HUB OR SPACECRAFT AND OPPOSITE END OF COLUMN)
- GROUND TEST NOT PRACTICAL, SO FABRICATION & ASSEMBLY CRITICAL IN OBTAINING ABSOLUTE POINTING OF $< 0.10^{\circ}$
- MAY NEED TO SENSE AND ADJUST CRITICAL SPATIAL PARAMETERS ON ORBIT
- MUST CONTROL SPACECRAFT FROM PLACEMENT ON ORBIT THROUGH DEPLOYMENT AND OPERATIONAL LIFE (ORDERS OF MAGNITUDE CHANGE IN INERTIA PROPERTIES)
- LOCATION AND ORIENTATION OF THRUSTERS CRITICAL WITH RESPECT TO GAS IMPINGEMENT ON MESH AND STRUCTURAL ELEMENTS

LSST/LMSS CONFIGURATION STUDY
CONCLUSIONS TO DATE

THE RESULTS OF THE STUDY TO DATE INDICATE THAT FINITE ANSWERS FOR THE FOUR MAJOR POINTS ON THE CHART ARE NEEDED. THE FIRST THREE ARE BEING DEVELOPED, AND THE ANALYSIS IDENTIFIED IN THE FOURTH COULD BE STARTED AT ANY TIME.

- NEED A SCHEDULE/CAPABILITY PLAN FOR OTV'S
- PLANAR ARRAY FEED TECHNOLOGY NEEDS DEVELOPMENT
- NEED TO DEVELOP A SENSING/ADJUSTMENT CAPABILITY TO ACTIVELY CONTROL SPATIAL POSITIONING OF POINTS ON LARGE STRUCTURES IN ORBIT
- SOLAR PANEL SHADOWING IN THIS APPLICATION SHOULD BE ANALYZED

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100-meter Radiometer Spacecraft Study

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A rigid body analysis of a baseline Large Space System (LSS) which is to function as a radiometer is presented. The LSS is placed in circular orbit about the Earth at an altitude of 650km, subjected to environmental and vehicle interaction forces and torques, without an active control system of any type on board. The environment forces and torques are gravity gradient, solar radiation, and aerodynamic. Normal operation is in nadir pointing alont the Z-local vertical axis. Orbital velocity is assigned to the x-axis of the spacecraft. The analysis is then used to demonstrate the ability or lack of the gravity gradient torques to stabilize the LSS over one complete orbit. The results generated by the analysis can then be used to size an active control system consisting of thrusters, momentum exchange devices, or a combination of the various active control devices, when used in conjunction with gravity gradient torques that would be required for altitude control and stationkeeping.

The baseline LSS which is an Electrostatically Charged Membrane Mirror (ECMM) radiometer placed in an orbit at an inclination angle of 60 degrees. When this angle is added to the 23 degree angle the Earth's equatorial plane makes with the ecliptic plane it causes that orbit to move through ± 83 degrees with respect to the sunline. For the purpose of the analysis an inclination angle of 67 degrees was chosen thus making the orbit move through ± 90 degrees and makes checking the computer simulation results much easier. Shown in Figure 1 is the orbit plane with respect to the inertial reference frame and the sunline for various beta angles. Figure 2 illustrates the spacecraft for a beta angle of zero degrees for which there exists an occulated region during the orbit. Using the physical properties of the ECMM spacecraft and aerodynamic model parameters as shown in Figures 3 and 4 respectively, in the rigid body dynamic model as shown in Figure 5, a computer simulation was performed for one complete orbit for beta angles of 0, 45, and 90 degrees. The results shown in Figures 6 through Figure 17 illustrate the various torques and angular momentum as a function of the anomaly angle, which then will be used to size an active control system to maintain the altitude and stationkeeping requirements of the spacecraft.

Presented in Figures 18 and 19 is a summary of possible control torque actuators that may be used for altitude control and stationkeeping requirements.

An analysis of the results concluded that momentum exchange devices are not practical control actuators to be used on the ECMM. The facts used to reach such a conclusion are as follows:

(1) The spacecraft would require a minimum of two momentum exchange devices per axis, with their respective spin vectors aligned opposite one another. The weight of the system becomes prohibitive.

(2) For the case of $\beta = 90$, there exists at the end of the orbit a residual momentum of 1.6426×10^3 n-m-sec (1.2168×10^3 ft-lb-sec).

(3) There does not exist for any time during the orbit an occulated region which would be required in order to perform a desaturation of the momentum control device.

(4) The size of the device was such that when it was attached to the box ring truss of the ECMM spacecraft, the packaging problems were such that the volume requirements of the Space Shuttle cargo bay were exceeded.

(5) The average life of a momentum exchange device is in the order of 3 years, which in the case of the ECMM spacecraft the mission time is 10 years, would require either a redundant system of the devices or an orbital refurbishment every 3 years.

The entire control of the spacecraft now depends solely on the same form of thruster system. Presented in Figure 20 and 21 are the candidates for the microthrusters that were considered for use as the active control system. The magnitudes of the environmental forces and torques which the control system must compensate are summarized in Figures 22 and 23 respectively.

Considering only the flight-qualified and high-Isp thrusters, the Pulse Plasma Thruster (PPT) and Mercury Ion Thruster (HgIon) were selected as candidates to be used for the comparative analysis. The basic differences between the PPT and HgIon systems are summarized in Figure 24.

A comparison of the two thruster systems which are considered for use as the ECMM spacecraft altitude control system is presented in Figure 25. Thrusters are to be positioned on the ECMM structure to overcome the external forces and torques. The maximum restoring torques are created when the thrusters fire in a direction perpendicular to the longest moment arm. The maximum force occurs when as many thrusters as possible fire in the proper direction. Finally, the strength of the structure, packaging considerations, and wiring problems also have to be considered to determine the positions of the thrusters.

Due to these considerations, thrusters are placed on the lower ring formed by the ECMM's box trusses, at locations 1 to 8 as shown in Figures 26 to 28 Thrusters were put on the feed beam, positions 9 and 10, to stabilize the vehicle and to create more restoring torque.

The orbit transfer propulsion consists of four hydrazine, blowdown, monopropellant thrusters. They are throttleable from 4.1 to 0.7 lb. These thrusters provide the impulse required to transfer from the deployment altitude of 240km to the operating altitude of 650km. They are in pairs for redundancy.

The suggested propulsion system consists of four space-qualified MRE-4-A-1 hydrazine thrusters produced by TRW and RCA. These thrusters have an I_{sp} of about 220-230 s. These thrusters will be hard mounted on the upper ring formed by the ECMM's box trusses as shown in Figures 26 to 28.

The expression defined by equation shown on the graph present in Figure 28 is used in this form to evaluate the time to decay from one altitude to another. The accuracy in predicting orbit lifetime is quite good as long as the altitude increments used are from 25 to 50km. The total orbit lifetime is obtained by adding the Δts. Shown in Figure 29 is Δt as plotted as a function of Δh, for the spacecraft shown in Figure 3. For the purpose of comparison, the membrane material was replaced with a mesh material. Presented in Figure 30 are the values that were used to determine Δt for a Δh equal to 50km increments.

There will exist times during the mission, in the case of large orbit inclination angles, for which no occulted region exists. This case is illustrated in Figure 31. The inclination angle, as a function of orbit altitude, will produce an occulted region.

Shown in Figure 32 is a plot of orbit inclination as a function of orbit altitude. Orbits with no occulted region are subjected to continuous solar radiation pressure force and torque over a long period. This presents difficult problems in a momentum management scheme if momentum exchange devices are considered for use in trying to compensate the torque due to solar radiation pressure.

Figure 1 **Orbit Plane with Respect to the Inertial Reference Frame and Sun Lines**

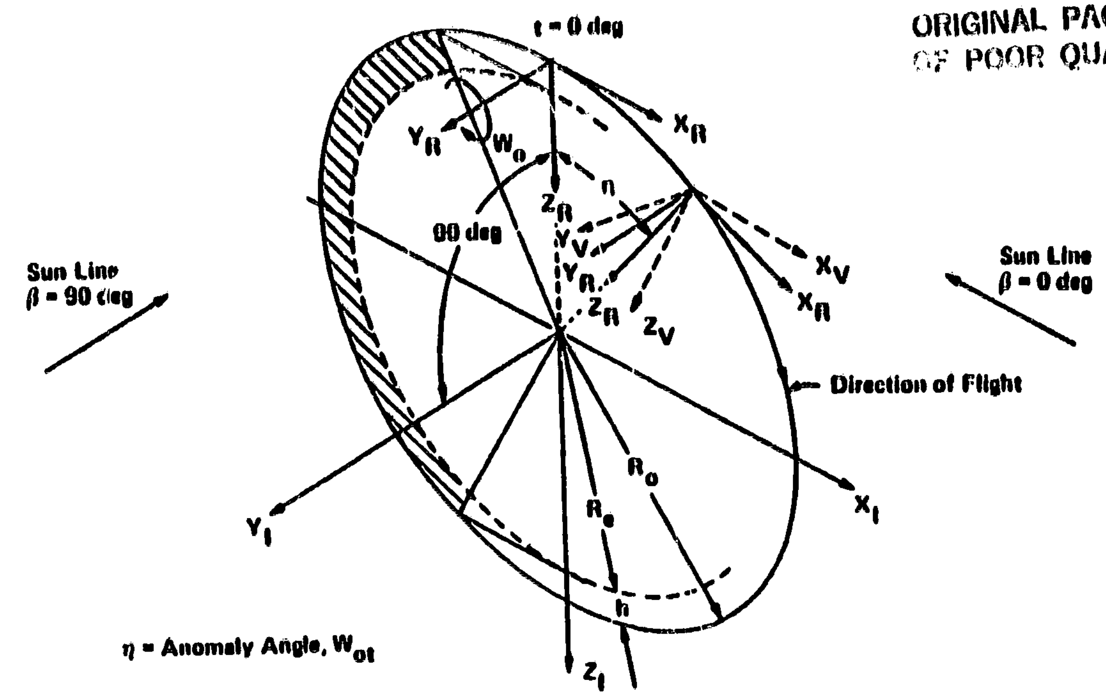

Figure 2 **Spacecraft Orbit for $\beta = 0$ deg**

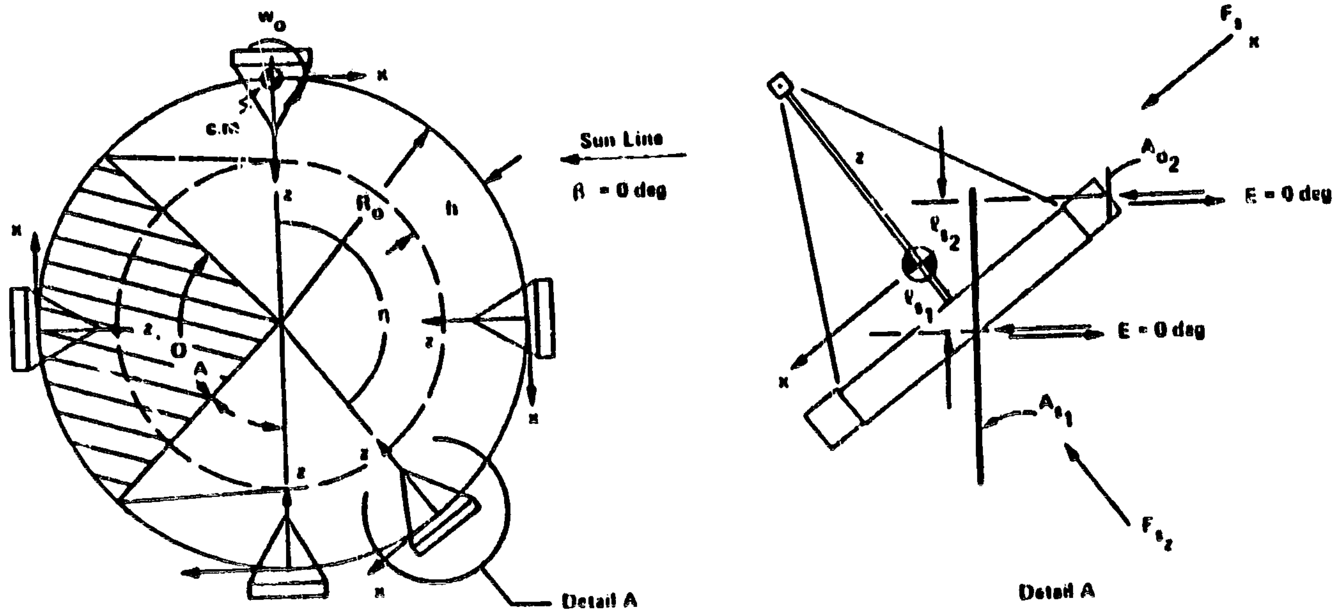

Figure 3 ECMM Spacecraft Physical Dimensions and Properties

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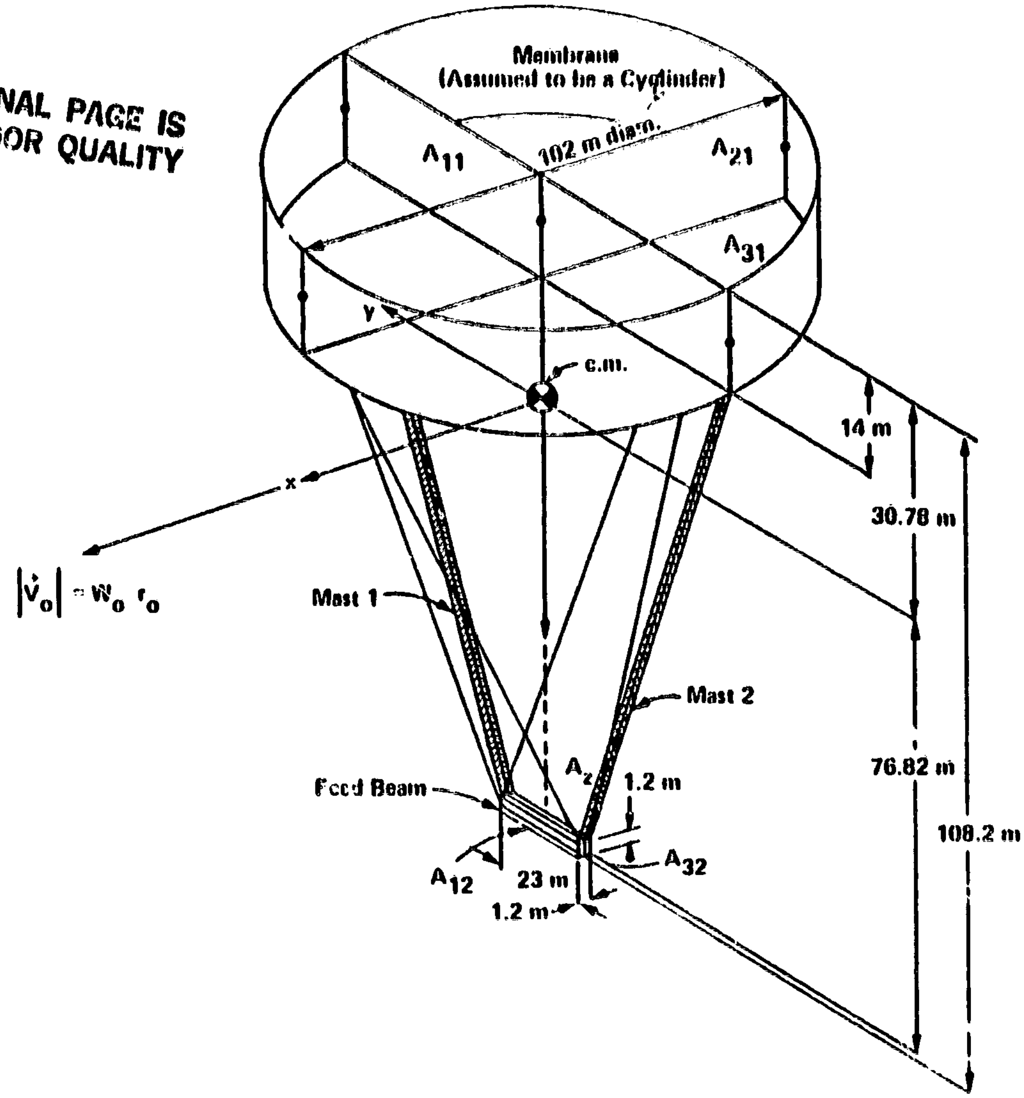

Box truss ring that supports the membrane is omitted.

Mass = 6800 kg

$I_{xx} = 1.2.8 \times 10^6$ kg·m^2

$I_{yy} = 1.218 \times 10^6$ kg·m^2

$I_{zz} = 0.959 \times 10^6$ kg·m^2

Cross products of inertia are zero.

Figure 4 **Aerodynamic Model Data Parameters**

Geometric shape	Projected areas $(A_{jk})\ m^2$		Drag coefficient, CD_{jk}		Distance from c.m. to c.p. of respective projected areas, (l^3_{cpjk}), m		
					x	y	z
Cylinder	A11	1442	CD11	2.5	0	0	-23.78
Cylinder	A21	1442	CD21	2.5	0	0	-23.78
Flat circular disk	A31	7850	CD31	4.0	0	0	-23.78
Flat plate	A12	28	CD12	4.0	0	0	76.82
Flat plate	A22	28	CD22	4.0	0	0	76.82
Flat plate	A32	1.44	CD32	4.0	0	0	76.82

Figure 5 **Block Diagram Rigid Body Dynamic Model**

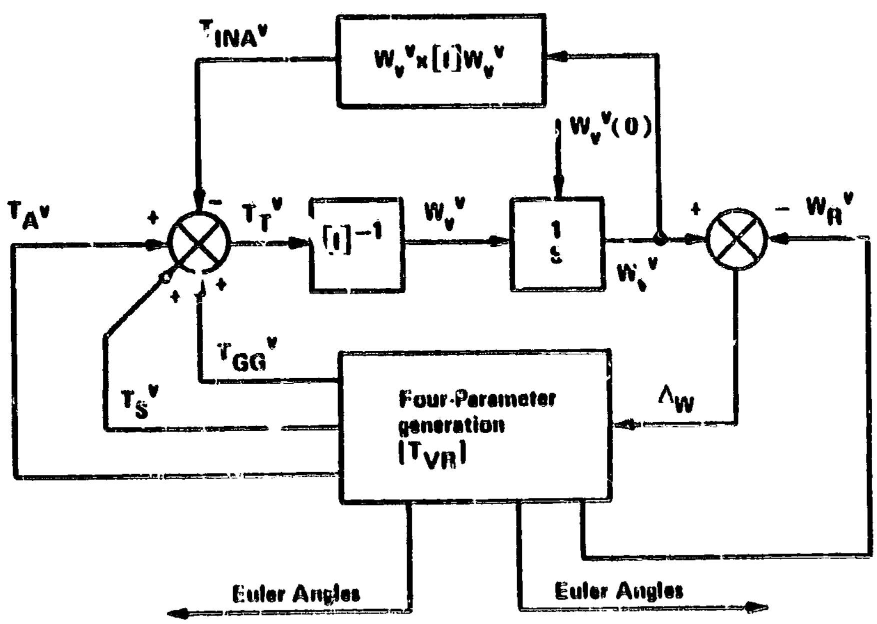

Figure 6 Total Torque Components Versus Anomaly Angle, $\beta = 0$ deg

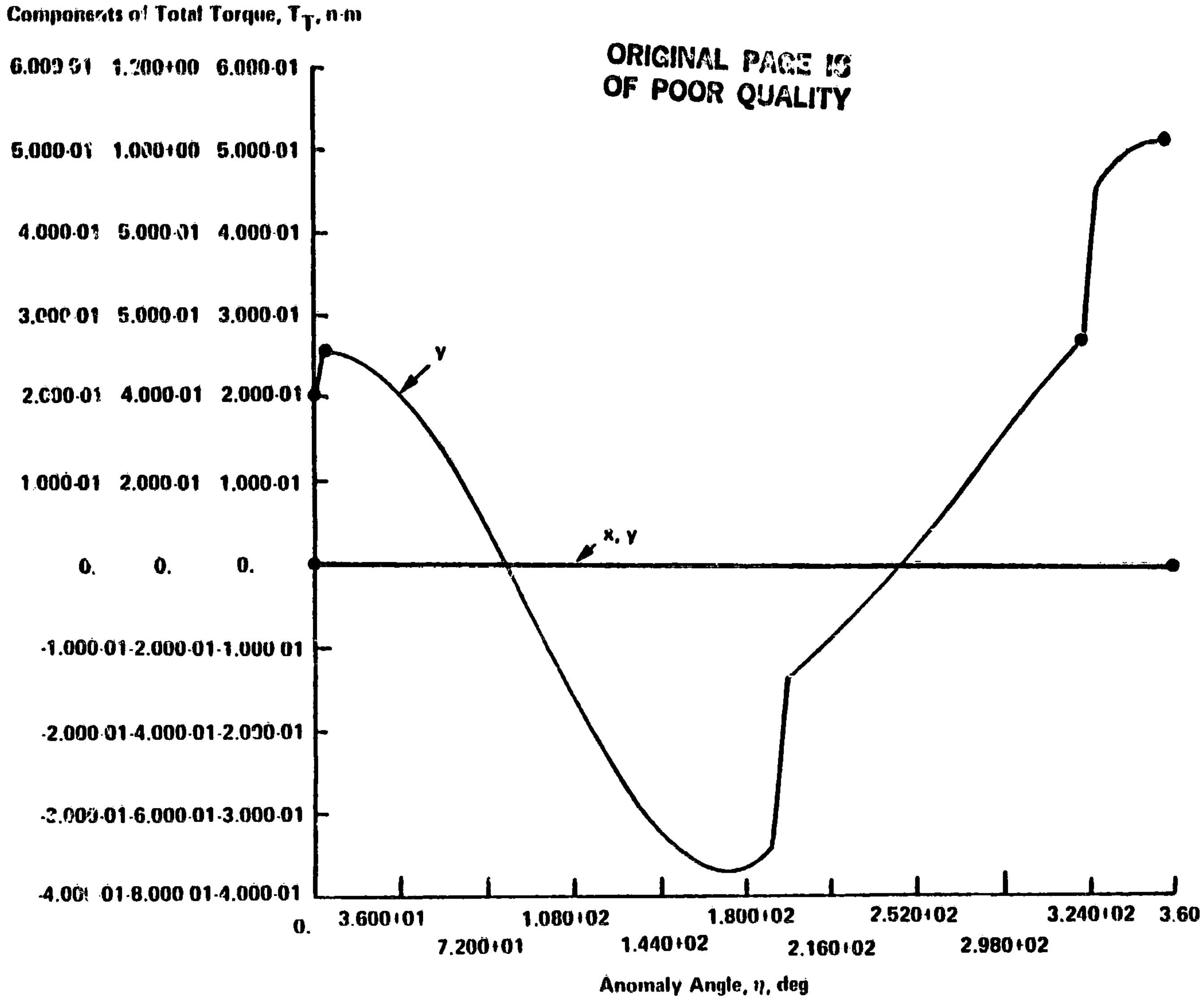

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Figure 7 **Magnitude of Total Torque Versus Anomaly Angle, β = 0 deg**

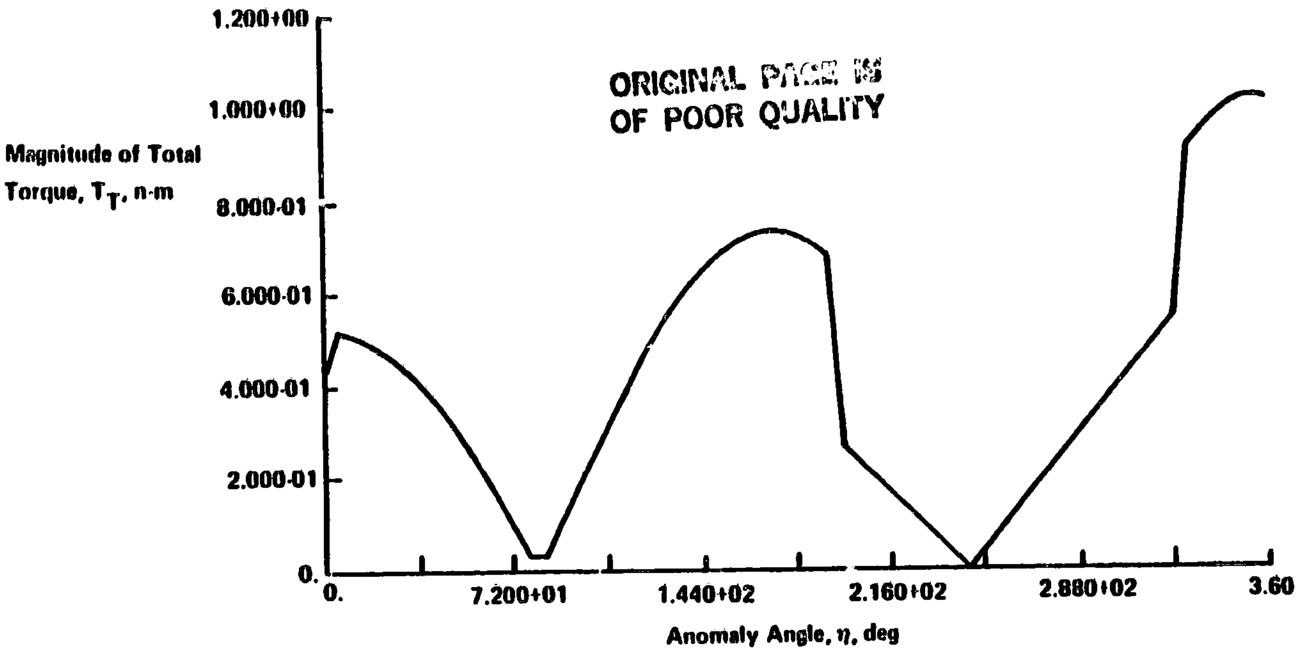

Figure 8 **Momentum Components Versus Anomaly Angle in Inertial Frame, β = 0 deg**

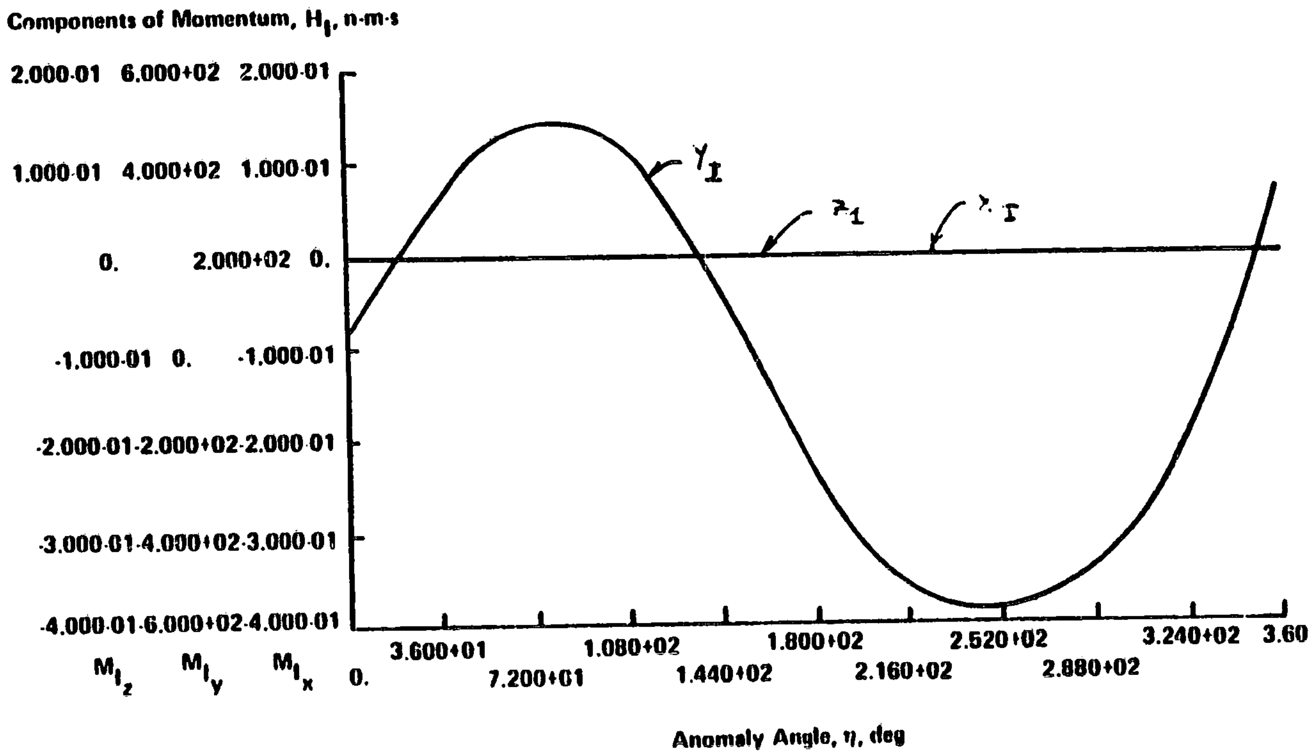

Figure 9 **Magnitude of Momentum Versus Anomaly Angle In Inertial Frame, $\beta = 0$ deg**

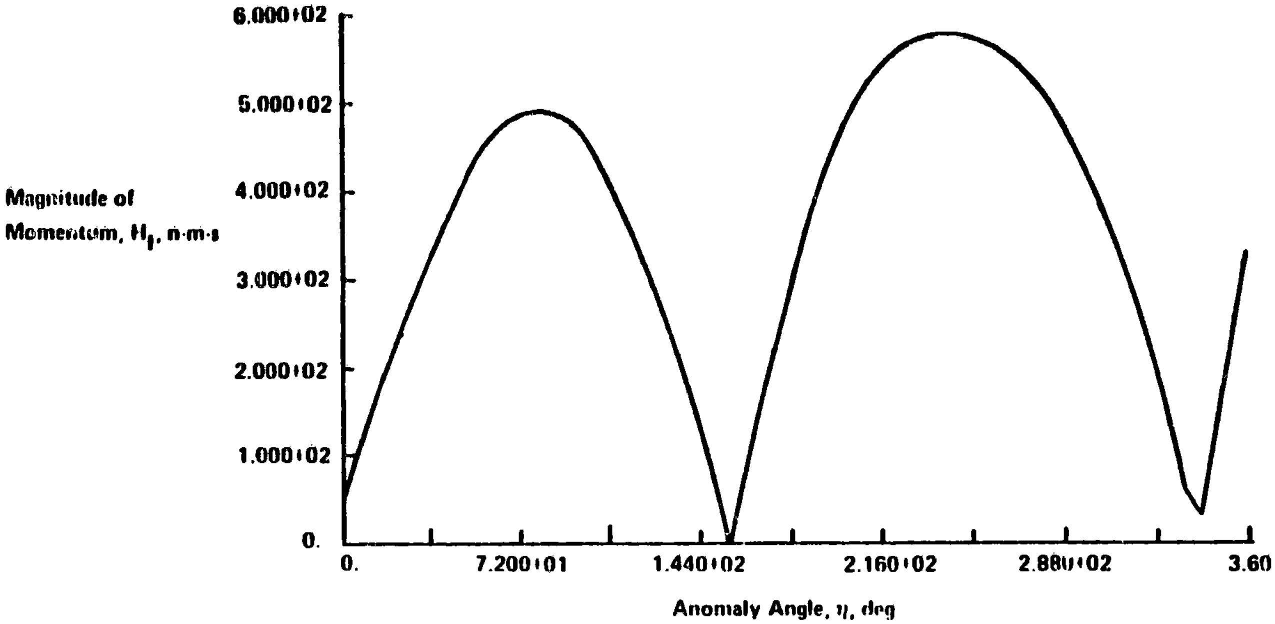

Figure 10 **Total Torque Components Versus Anomaly Angle, $\beta = 45$ deg**

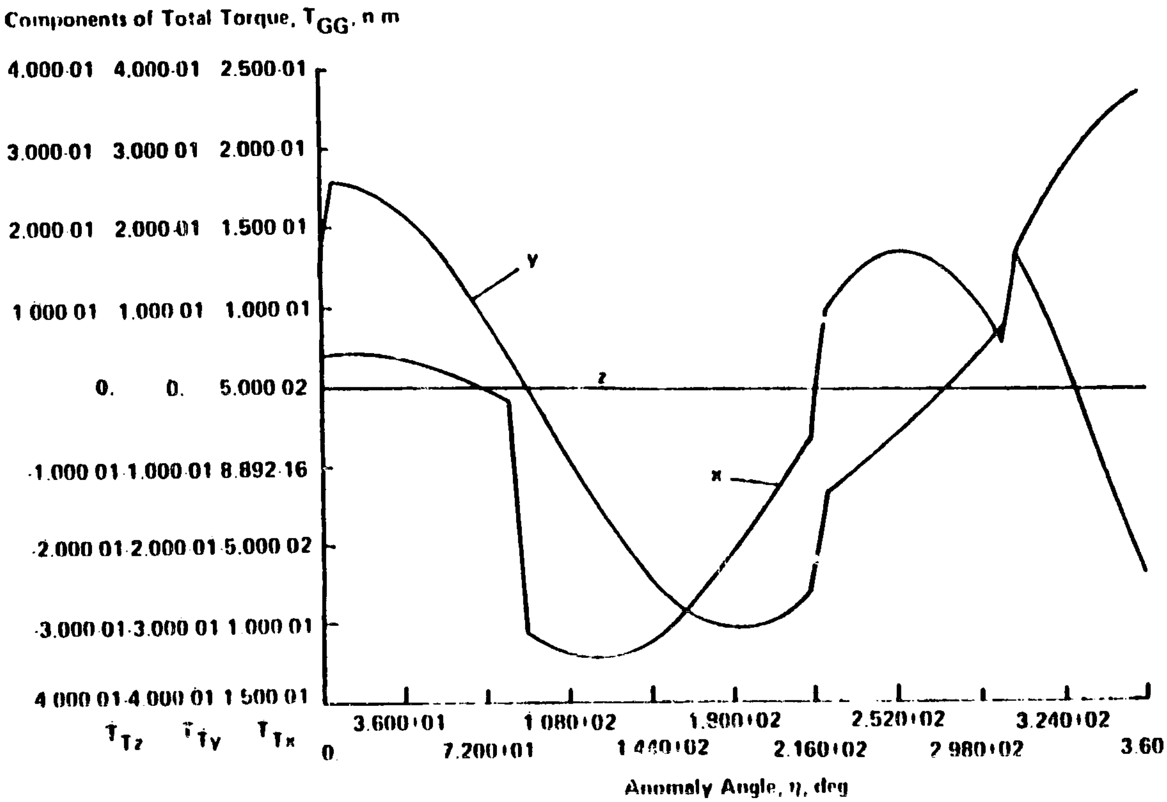

Figure 11 **Magnitude of Total Torque Components Versus Anomaly Angle, $\beta = 45$ deg**

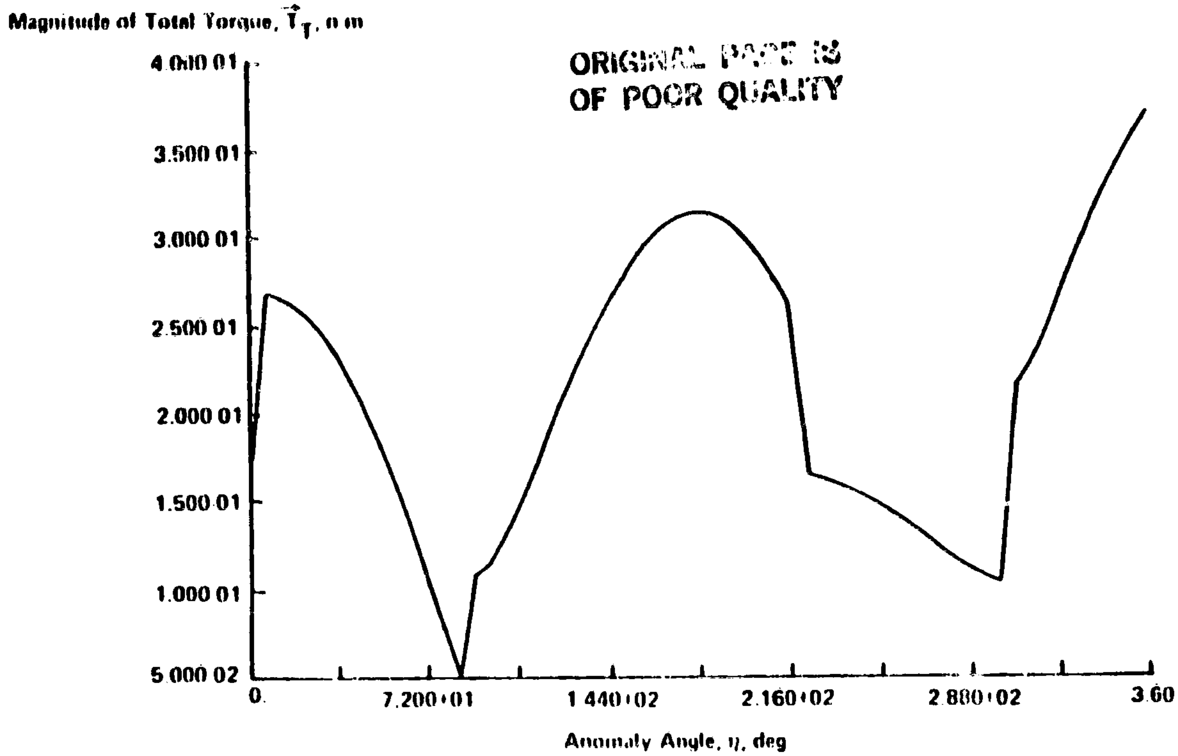

Figure 12 **Momentum Components Versus Anomaly Angle in Inertial Frame, $\beta = 45$ deg**

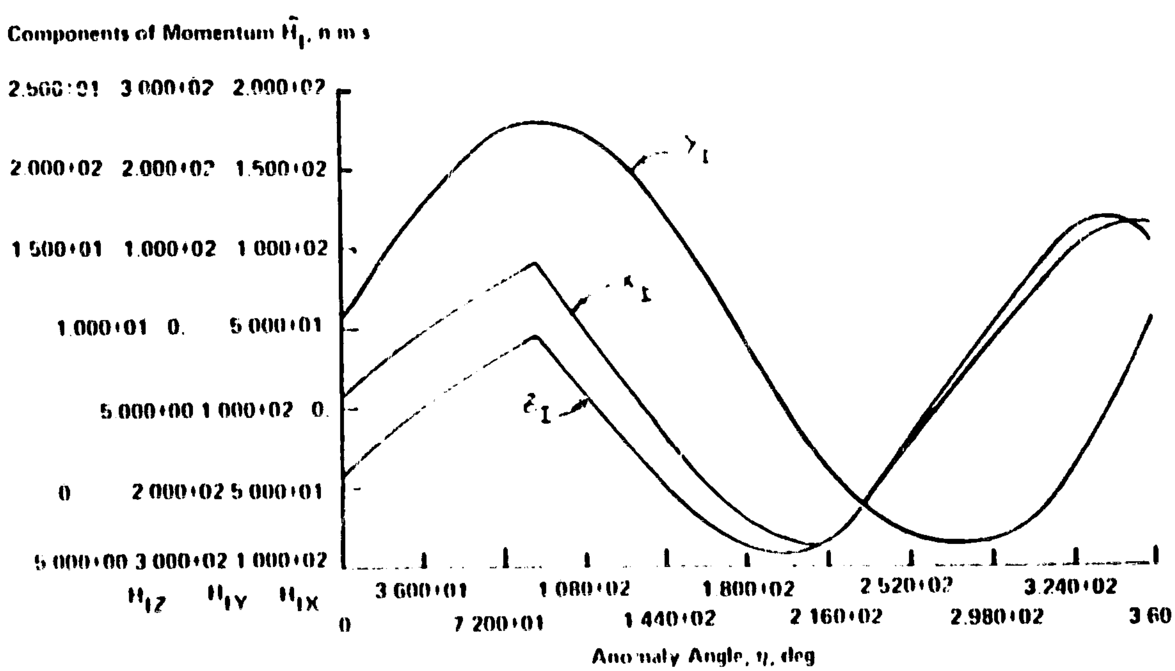

Figure 13 **Magnitude of Momentum Versus Anomaly Angle in Inertial Frame, β = 45 deg**

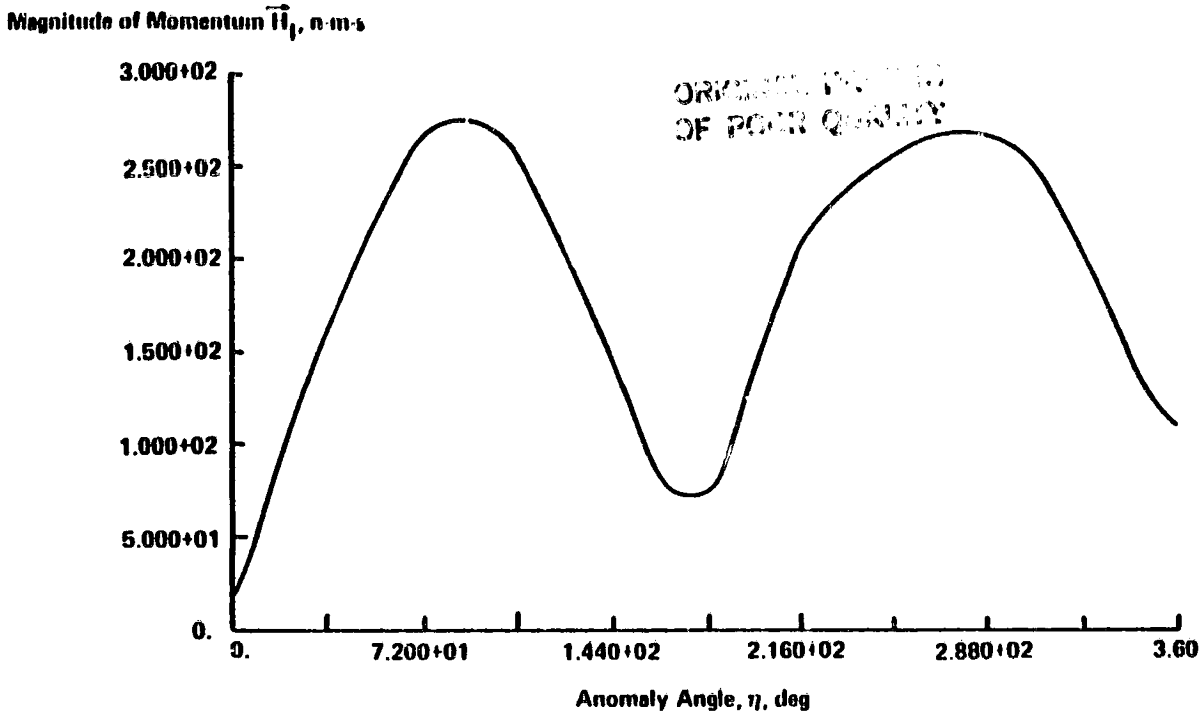

Figure 14 **Total Torque Components Versus Anomaly Angle, β = 90 deg**

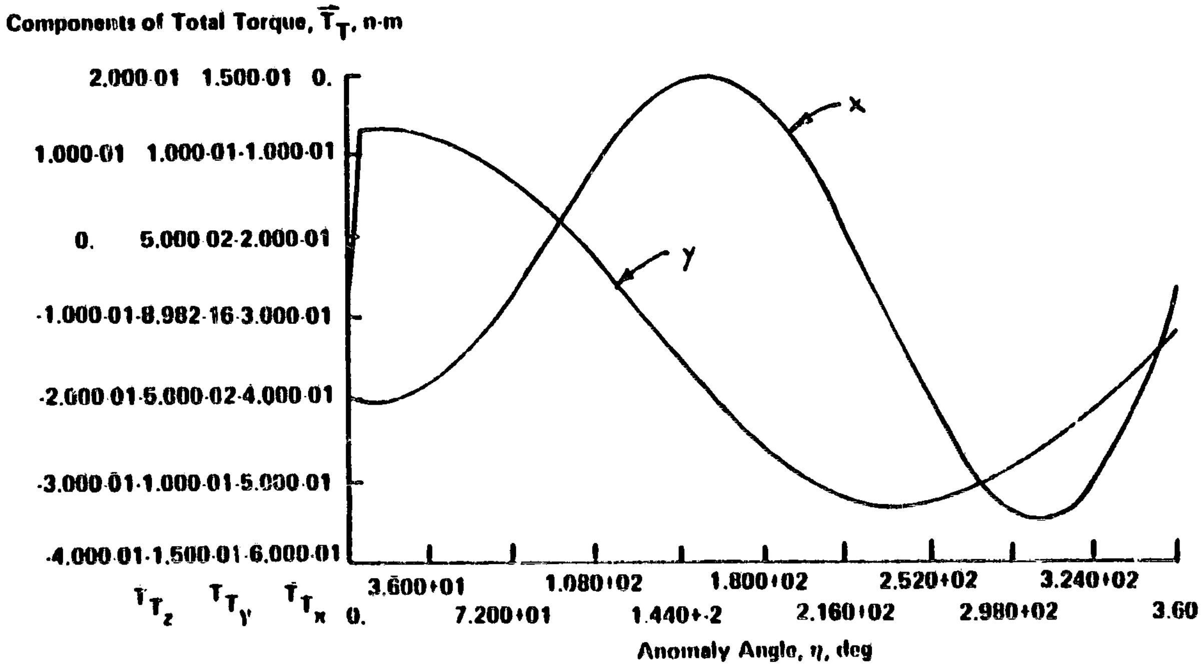

Figure 15 **Magnitude of Total Torque Versus Anomaly Angle, β = 90 deg**

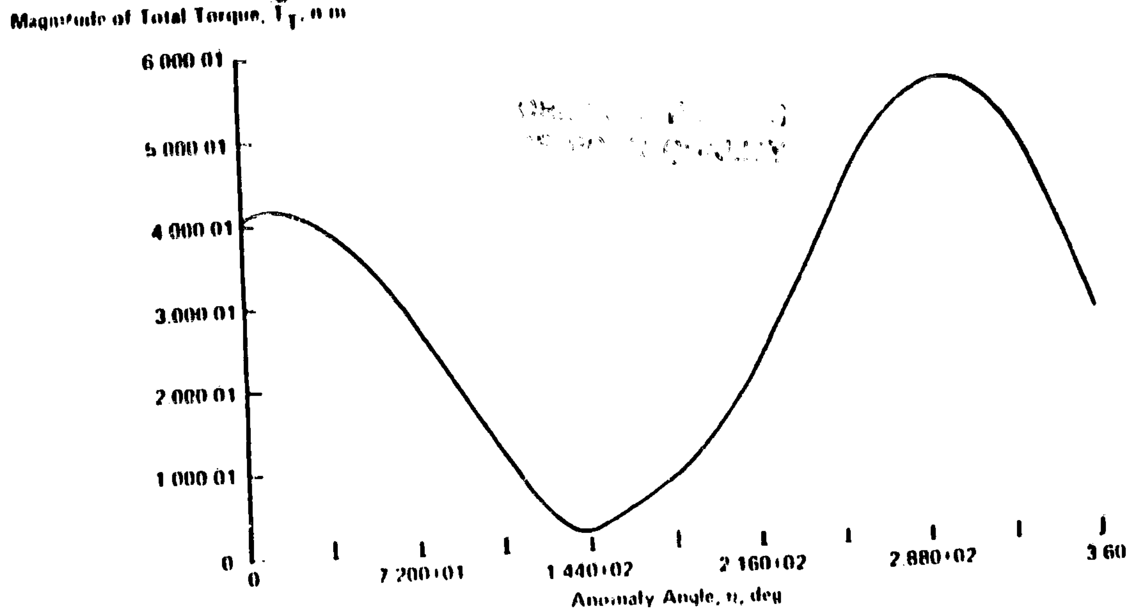

Figure 16 **Momentum Components Versus Anomaly Angle in Inertial Frame, β = 90 deg**

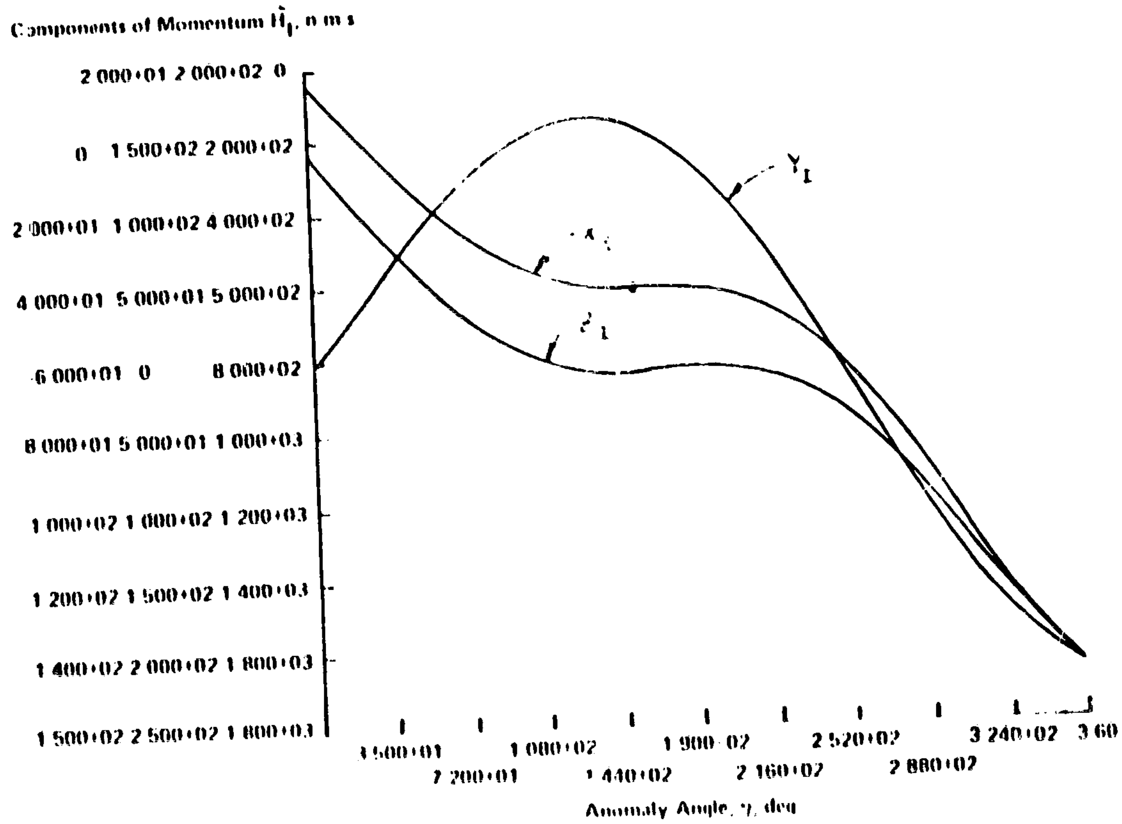

Figure 17 **Magnitude of Momentum Versus Anomaly Angle In Inertial Frame, β = 90 deg**

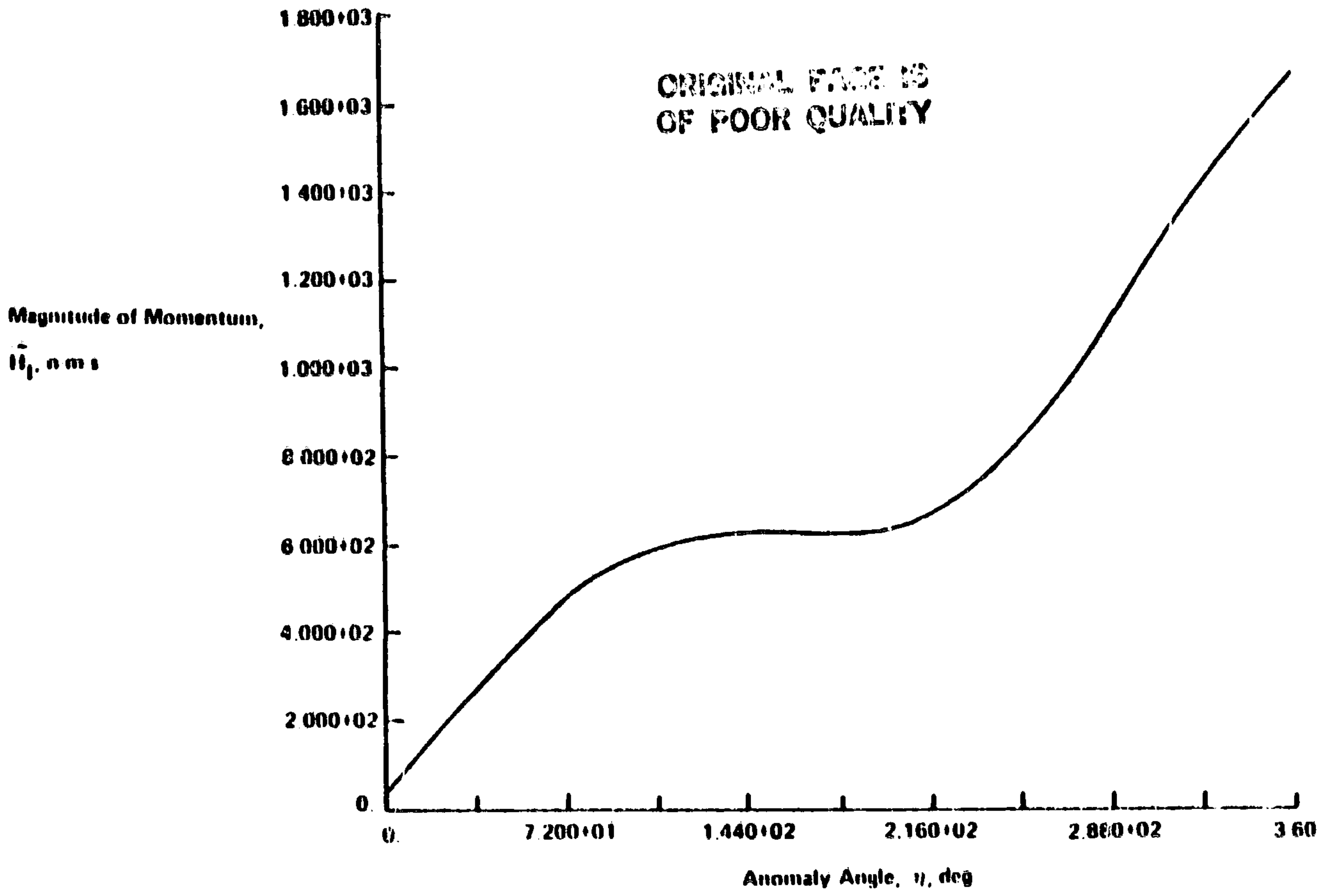

Figure 18

Control Torque Actuator Summary

Torque Actuator	Advantage	Disadvantage
Thrusters	No cross coupling with the vehicle motion to produce undesirable torques for which compensation must be made.	
Electric	High specific impulse, applicable to missions with a long lifetime.	Low-thrust, high-power requirements
Chemical	High thrust	Low specific impulse, not applicable to missions with a long lifetime; Tankage and weight problems

Figure 19

Control Torque Actuator Summary (concl)

Torque Actuator	Advantage	Disadvantage
Momentum exchange devices; Reaction wheels and CMG	Ideal when disturbances are cyclic with respect to an inertial reference frame and the secular component (bias torque) is small. Could reduce the size and number of thrusters required for the mission.	Cross coupling with the vehicle motion to produce undesirable torques that require compensation. Requires a desaturation control law scheme. Number required could be prohibitive in terms of size, weight, and power required.
Magnetic torquers	Used in conjunction with momentum exchange devices for desaturation or momentum management purposes.	Magnetic field of the Earth is time-variant and strongly altitude-dependent. At any instant, torque can be produced only along components normal to the local magnetic field vector. Practical limitations in power supply and coil size make the generation of large torques unfeasible.

Figure 20

Microthruster Candidates

System	Thrust, lb	I_{sp}, s
Chemical		
Inert gas	10^{-4} to 1.0	35 to 275
Vaporizing liquid	10^{-5} to 0.05	50 to 100
Subliming solids	10^{-4} to 10^{-2}	40 to 80
Hydrazine direct catalyst	0.05 to 1000	100 to 225
Bipropellant (storable)	0.05 to 10^{4}	170 to 320

Figure 21

Microthruster Candidates (concl)

System	Thrust, lb	I_{sp}, s
Electric		
Resistojet	0.01 to 5.0	175 to 860
Electrolysis	10^{-4} to 5.0	100 to 350
Pulsed plasma	10^{-6} to 10^{-3}	1000 to 5000
Ion (mercury)	10^{-3} to 0.5	2000 to 9000
Ion (noble gas)[a]	2.1×10^{-3}	5500 to 6400
MPD[a]	2.3×10^{-2} to 3.2×10^{-2}	2000 to 9000
Mass driver[a]	10^{-5} to 10	10^4 to 5×10^4

[a] Not flight-qualified

Figure 22

Environmental Forces

Beta Angle (deg)		Solar $N \times 10^{-3}$ X	Solar Y	Solar Z	Aerodynamic $N \times 10^{-3}$ X	Aerodynamic Y	Aerodynamic Z
0	Max	13.30	0	73.7	-5.86	0	0.43
	Min	-13.30	0	-36.8	-8.82	0	-0.52
45	Max	9.38	9.21	52.1	-5.86	0.11	0.07
	Min	-9.38	-9.21	-42.2	-7.77	-0.60	-0.29
90	Max	0	-13.0	0	-5.86	0.74	0
	Min	0	-13.0	0	-7.93	0	-0.25

Figure 23

Environmental Torques

Beta Angle (deg)		Solar $Nm \times 10^{-3}$ X	Solar Y	Solar Z	Aerodynamic $Nm \times 10^{-3}$ X	Aerodynamic Y	Aerodynamic Z	Gravity Gradient $Nm \times 10^{-3}$ X	Gravity Gradient Y	Gravity Gradient Z
0	Max	0	407.0	0	0	184.0	0	0	451.0	0
	Min	0	-407.0	0	0	117.0	0	0	520.0	0
45	Max	70.9	14.4	0	2.14	160.0	0	133.0	96.4	0
	Min	-70.9	-14.4	0	1.24	117.0	0	-163.0	-32.7	0
90	Max	-401.0	0	0	15.00	164.0	0	389.0	0	0
	Min	-401.0	0	0	0	116.0	0	162.0	-280.0	0

Figure 24

Propulsion System Comparison

Plasma Propulsion	Ion Propulsion
Electromagnetic forces	Electrostatic forces
Electrically neutral plasma accelerated	Charged ions accelerated
No neutralization required	Ions must be neutralized
Extremely short puffs of propellant are ejected (microseconds of flow time)	Steady flow of propellant is ejected (hours of flow time)
Instantaneous reaction forces are typically hundreds of pounds	Reaction forces inherently fractions of a pound

Figure 25

Mercury Ion and PPT Comparison

	Mercury Ion	PPT
Packaging	Very versatile[a]	Some size restriction[b]
Number of thrusters necessary	20	24
Total power requirement	4840 W	4080 W
Effects on structure	Possible degradation[c]	None
ACS requirements	Satisfied	Satisfied

[a] Modular components make many packaging schemes possible.

[b] Off-the-shelf PPTs are cylindrical, about 43.2 cm in diameter and 25.4-cm wide.

[c] Sufficient test data has not been acquired.

Figure 26 Thruster Location, Bottom View

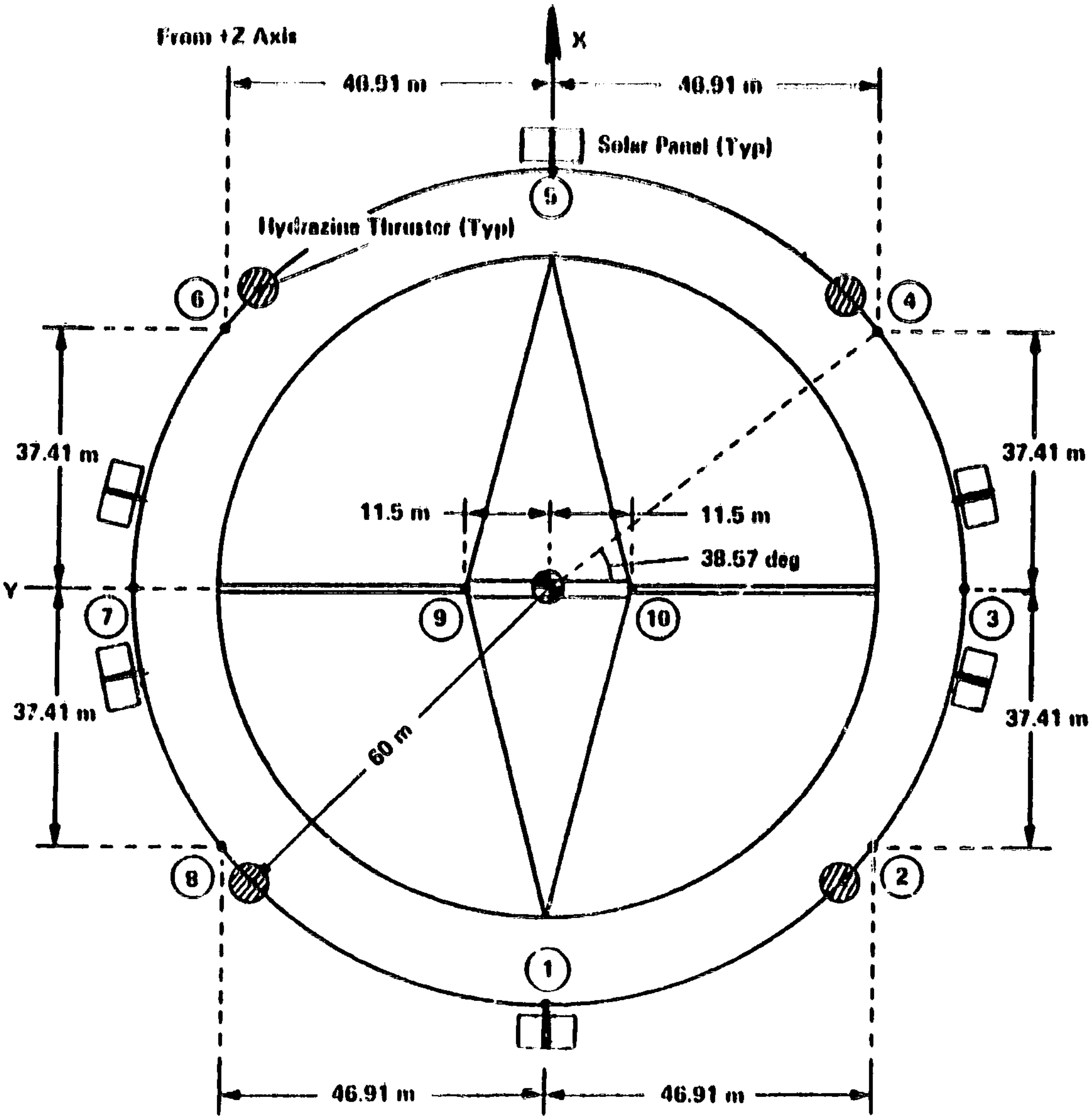

Figure 27 **Thruster Location, Side View**

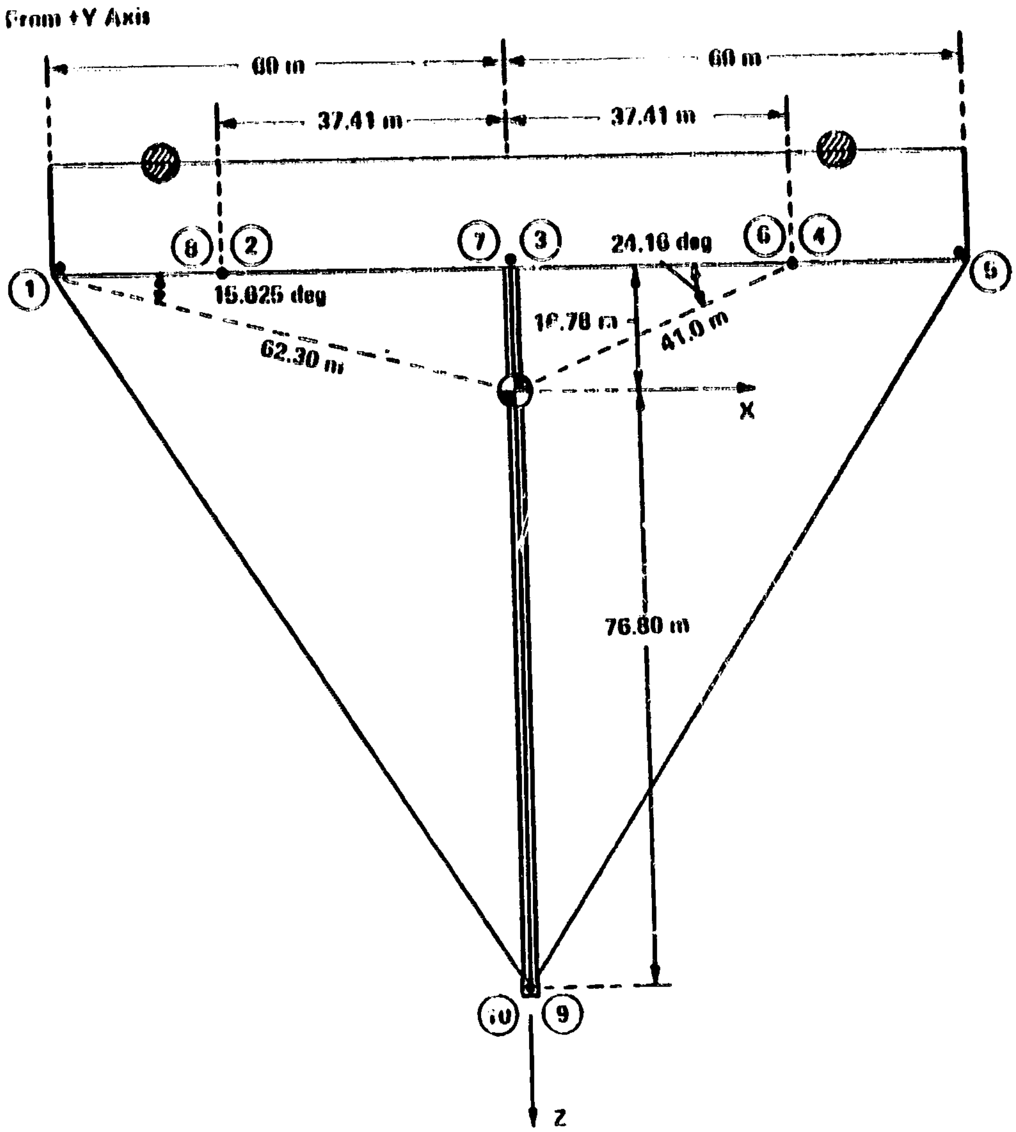

Figure 28 Thruster Location, Front View

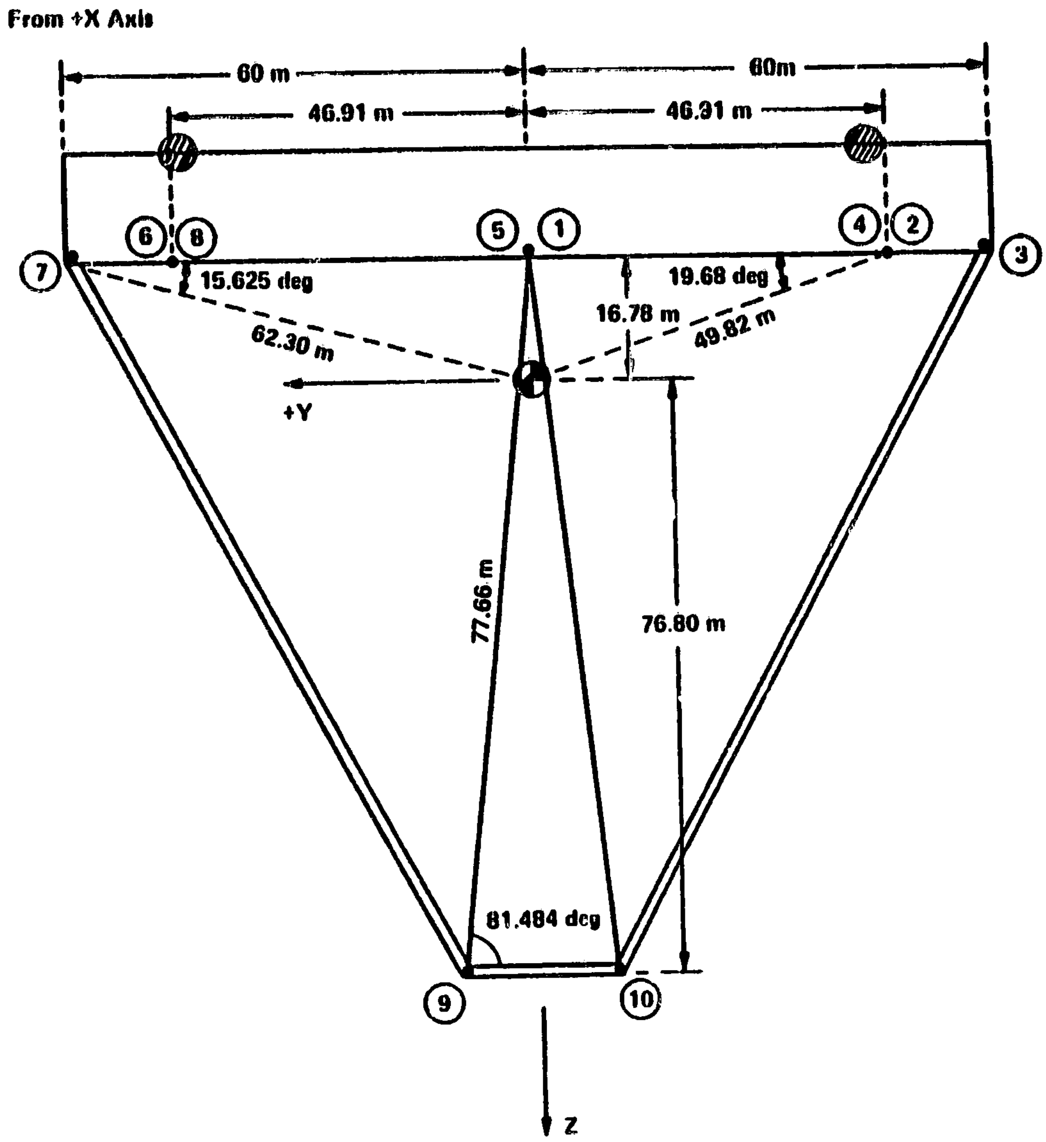

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Figure 29 **Time to Change Orbit Altitude, Δh = 50 km**

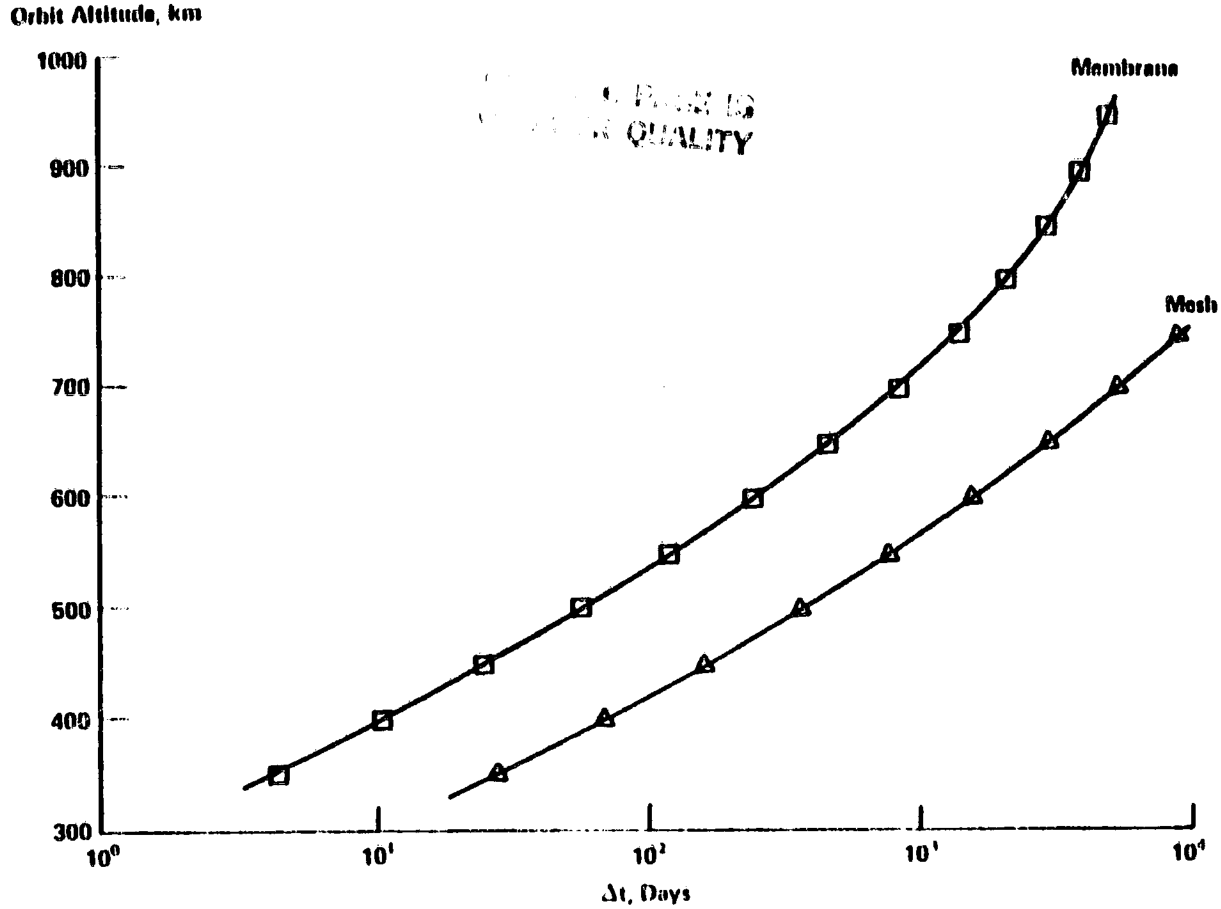


Figure 30 **Orbit Decay Parameters**

Parameter	Antenna surface: Membrane	Mesh
C_D	2.5	2.3
A_{ref}, m^2	1442	238
Mass, kg	6800	6800
μ, m/s^2	3.98418×10^{14}	3.98418×10^{14}
β, m^2/kg	0.257	0.04

Figure 31 Spacecraft Orbit for Inclination Angle and $\beta \neq 0$ deg

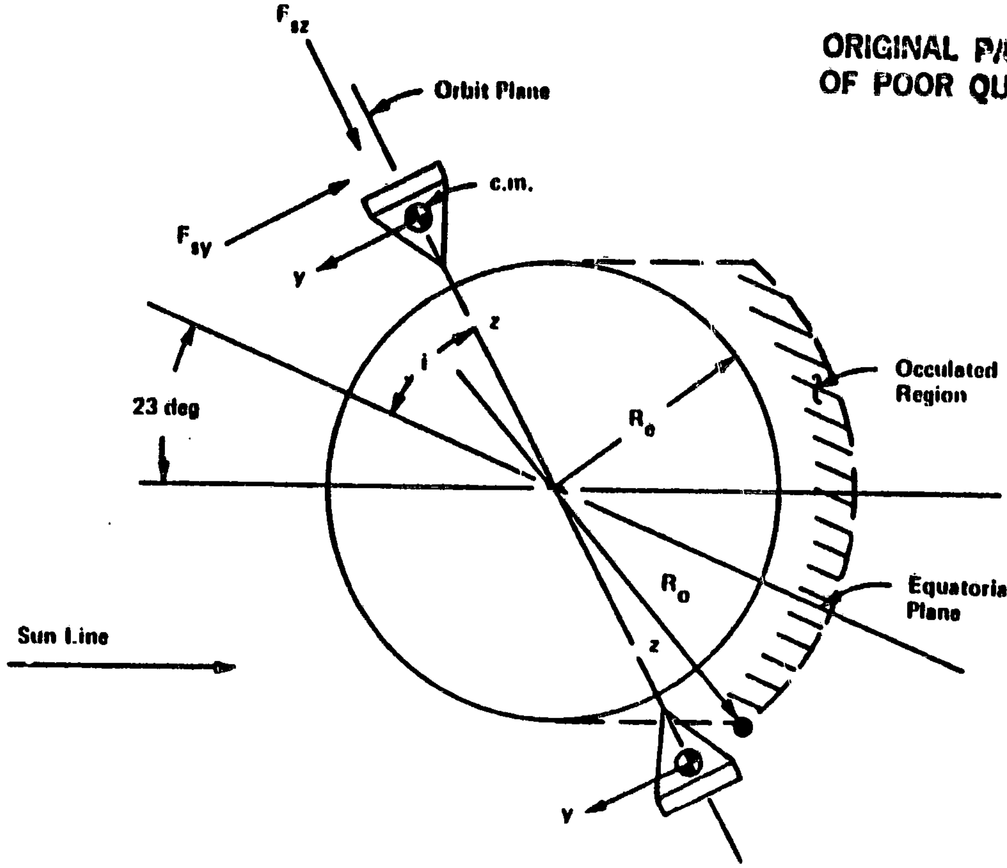
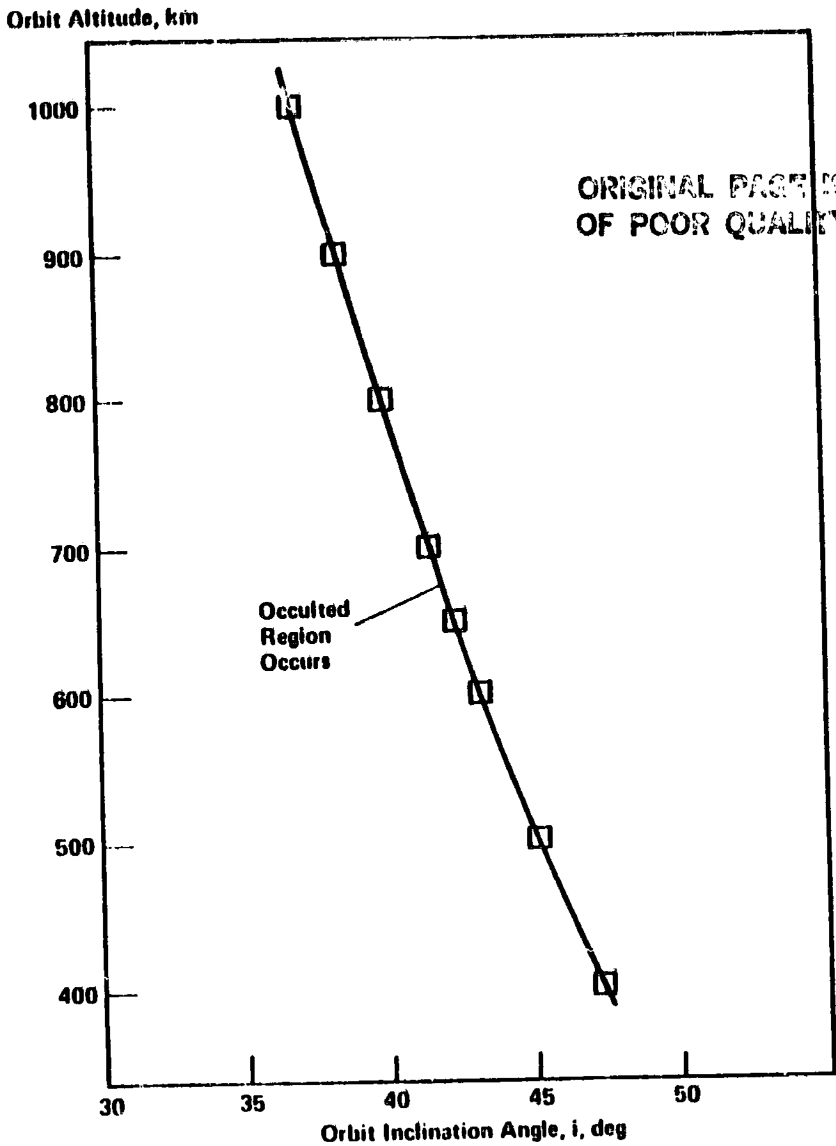


Figure 32 **Orbit Inclination Versus Orbit Altitude**



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N82 27375

CAPABILITIES IN LARGE SPACE SYSTEMS CONSTRUCTION

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- SYSTEMS STUDIES OVERVIEW
  - ORBITAL CONSTRUCTION DEMONSTRATION ARTICLE
  - SPACE CONSTRUCTION SYSTEM ANALYSIS
  - SOLAR POWER SATELLITE
  - SPACE OPERATIONS CENTER
- CAPABILITY DEVELOPMENT
  - SATELLITE SERVICES
  - HOLDING AND POSITIONING AID
  - SPACE CONSTRUCTION EXPERIMENT

ORBITAL CONSTRUCTION DEMONSTRATION ARTICLE

- DEFINED ORBITAL FACILITY MANNED FROM ORBITER IN A SORTIE MODE.
- SIGNIFICANT COST AND TIME TO ESTABLISH FACILITY.
- APPLICATION OF GENERAL CAPABILITY, I.E., FACTORY FLOOR CONCEPT TO SPECIFIC PROJECT CAN BE INEFFICIENT.

SPACE CONSTRUCTION SYSTEM ANALYSIS STUDY

- EVALUATED THE ORBITER AS A CONSTRUCTION BASE
- SELECTED CONSTRUCTION PROJECTS
  - COMMUNICATION TECHNOLOGY PLATFORM
  - SOLAR POWER SATELLITE TEST ARTICLE
- SYSTEM INSTALLATION DOMINATES CONSTRUCTION ACTIVITIES
- LINEAR CONFIGURATION SHOULD PASS WORK STATION
- FIXTURE COMPLEXITY ENHANCES PRODUCTION RATE
- FREE DRIFT MODE RECOMMENDED (FROM CONSTRUCTION POINT OF VIEW)

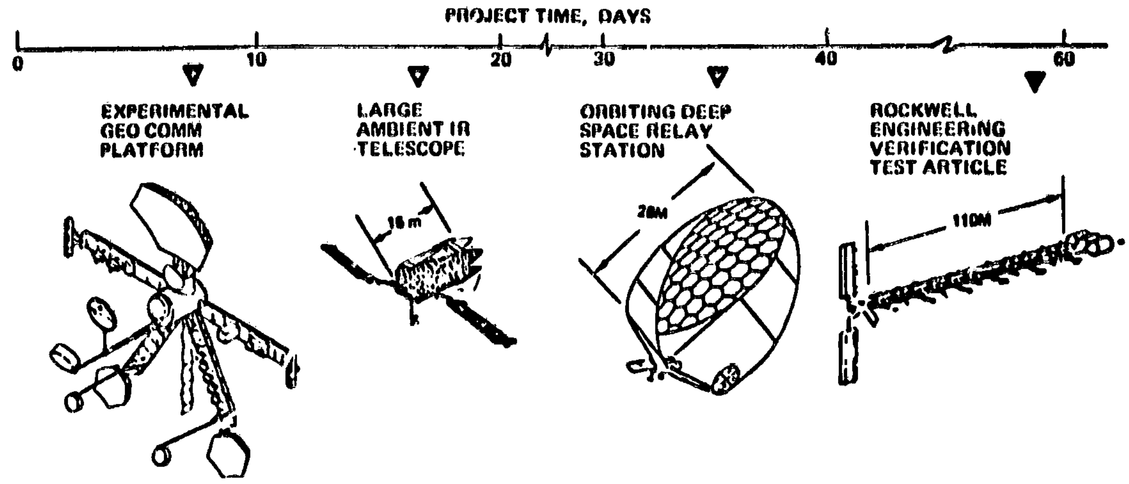
SOLAR POWER SATELLITE CONSTRUCTION

- DEFINED HIGHLY AUTOMATED CONSTRUCTION FACILITIES
- DEVELOPED CONSTRUCTION CONCEPTS
  - SPACE STRUCTURES (BEAM BUILDER APPLICATIONS)
  - SPACE MEMBRANES (SOLAR CELL BLANKETS)
  - MODULE INSTALLATION (RCS, ANTENNA ELEMENTS)

SPACE OPERATIONS CENTER

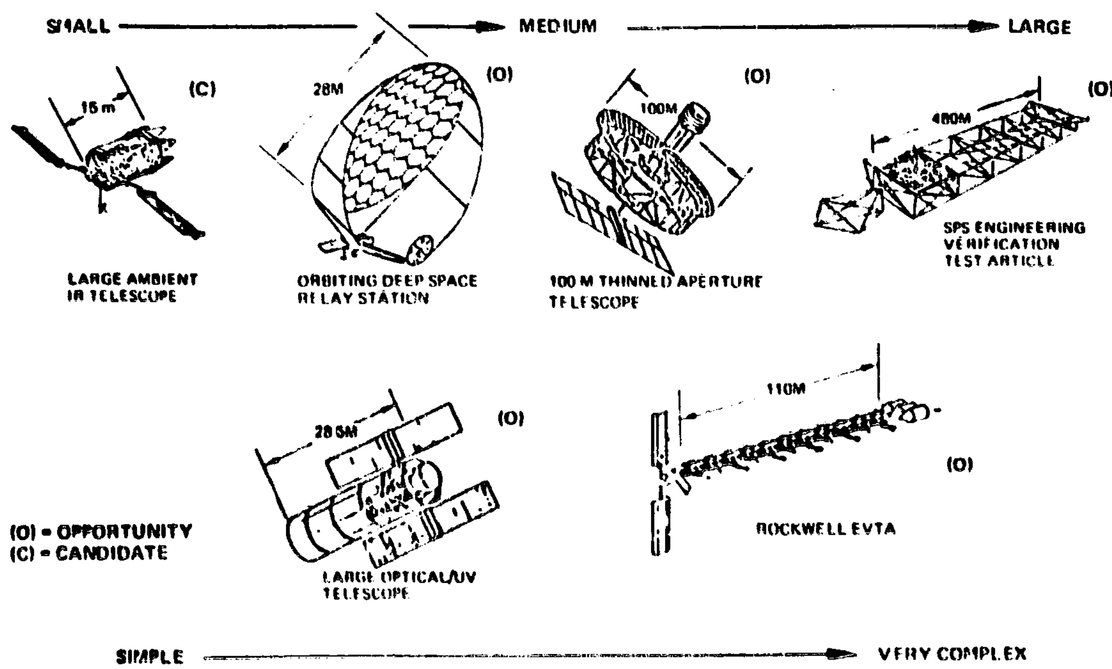
- DEFINED CONSTRUCTION FACILITY FOR SOC
- SOC OPERATIONS RELAX TIME CONSTRAINTS COMPARED TO ORBITER-BASED CONSTRUCTION
- DEVELOPED PIER CONCEPT WITH RECONFIGURATION CAPABILITY - ANALYZED COMMUNICATIONS PLATFORM, IR TELESCOPE, DEEP SPACE RELAY ANTENNA, AND ENGINEERING VERIFICATION TEST ARTICLE.

SOC CONSTRUCTION PROJECTS SUMMARY



	EXPERIMENTAL GEO COMM PLATFORM	LARGE AMBIENT IR TELESCOPE	ORBITING DEEP SPACE RELAY STATION	ROCKWELL ENGINEERING VERIFICATION TEST ARTICLE
	DEPLOY TEST MATE TO OTV LAUNCH	ASSEMBLE & DEPLOY TEST MATE TO OTV LAUNCH	ASSEMBLE & DEPLOY TEST MATE TO OTV LAUNCH	FABRICATE & DEPLOY ASSEMBLE TEST MATE TO OTV'S LAUNCH
NO. OF ORBITER FLTS	2	2	2	4
SOC CREW SIZE	4	8	9	8

Preliminary Set of Reference Construction Missions



SATELLITE SERVICES - NEAR ORBITER

OBJECTIVE:

THE DEVELOPMENT OF PRELIMINARY REQUIREMENTS AND DESIGN CONCEPTS FOR A SATELLITE SERVICES SYSTEM COVERING THE FOLLOWING CONSIDERATIONS:

- SATELLITE USER MARKET
- SERVICES NEEDED FOR SATELLITES
- SERVICE EQUIPMENT AND SERVICING MODES
- RESOURCES, IMPLEMENTATION, AND SCHEDULING OF A SATELLITE SERVICES PROGRAM

STATUS:

IMPLEMENTED STUDY CONTRACTS WITH GAC AND LMSC TO IDENTIFY SATELLITE SERVICE NEEDS DIRECTLY ASSOCIATED WITH THE ORBITER AND THOSE SERVICES THAT CAN BE PERFORMED WITHIN A FEW KILOMETERS OF THE ORBITER.

SATELLITE SERVICES ELEMENTS

- DEPLOYMENT
- OBSERVATION
- RETRIEVAL
- SUPPORT
- EARTH RETURN

SATELLITE SERVICES SYSTEM PLAN IS EVOLUTIONARY AND INCLUDES FOUR EQUIPMENT CATEGORIES

NEED DATE	1983	1984	1985	1986	1987	1988	1989

I. INHERENT EQUIPMENT

EQUIPMENT INHERENT WITH STS SYSTEM

II. GENERIC EQUIPMENT

EQUIPMENT WHICH INTEGRATES WITH THE INHERENT EQUIPMENT AND HAS GROWTH POTENTIAL

III. UNIQUE EQUIPMENT

EQUIPMENT UNIQUE TO SPECIAL MISSION REQUIREMENTS

IV. ADVANCED EQUIPMENT

EQUIPMENT POTENTIALLY NEEDED TO FULFILL FUTURE MISSION MODEL REQUIREMENTS

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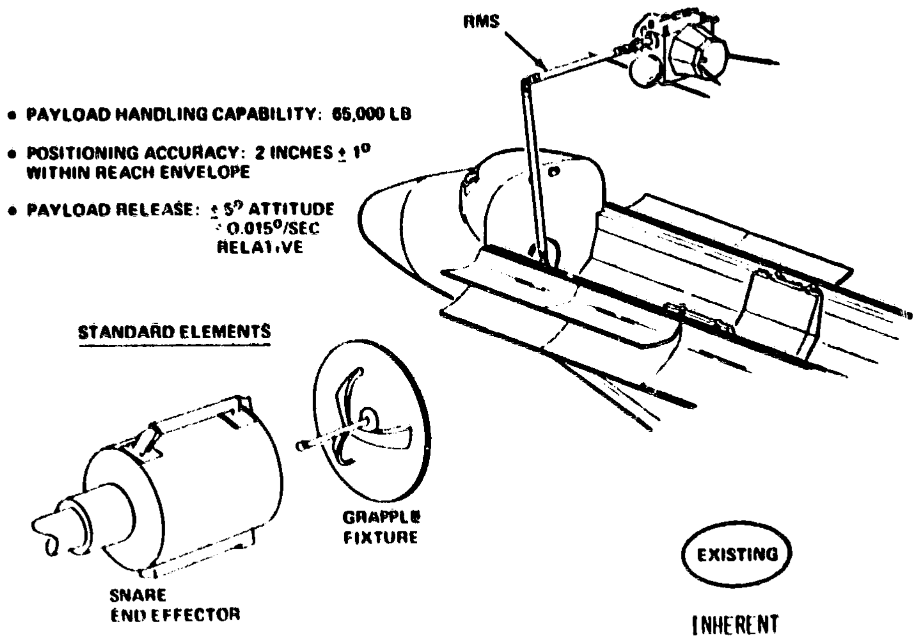
INHERENT EQUIPMENT

EQUIPMENT INHERENT WITH STS SYSTEM

INHERENT EQUIPMENT	SATELLITE SERVICE FUNCTION
PAYLOAD RETENTION SYSTEM - PRS	• ORBITER RETENTION AND RELEASE OF PAYLOADS
REMOTE MANIPULATOR SYSTEM - RMS	• DEPLOYMENT AND RETRIEVAL OF SATELLITES; ALSO FOR OBSERVATION VIA CCTV AND SUPPORT SERVICES.
EXTRAVEHICULAR MANEUVERING UNIT - EMU	• MANNED EVA CAPABILITY WITHIN THE CARGO BAY
MANNED MANEUVERING UNIT - MMU	• MANNED PROPULSIVE EVA CAPABILITY OUTSIDE THE CARGO BAY
ORBITER MANEUVERING SYSTEM KIT - OMS KIT	• ORBITER DELTA V CAPABILITY
AFT FLIGHT DECK - CONTROLS AND DISPLAYS	• CONTROL OF RMS, PRS AND OTHER REMOTE MECHANISMS FROM THE ORBITER AFT FLIGHT DECK.
EXTRAVEHICULAR MANEUVERING UNIT - TV	• CCTV DURING EVA
MODULAR EQUIPMENT STOWAGE ASSEMBLY - MESA	• EQUIPMENT STOWAGE WITHIN CARGO BAY
CLOSED-CIRCUIT TELEVISION - CCTV	• CCTV VIEWING OF CARGO BAY
ORBITER EXTERIOR LIGHTING	• LIGHTING OF CARGO BAY

REMOTE MANIPULATOR SYSTEM (RMS)



RMS DEPLOYMENT

Illustrated in the accompanying figure is the RMS deployment of dual Gravsat-A satellites. The cradle structure, which serves as the satellite retention system during Orbiter boost, is deployed from the payload bay, and the separation ΔV is provided by the cradle structure. The two satellites are separated from the cradle in the same orbit at a separation distance of 100 to 300 km.

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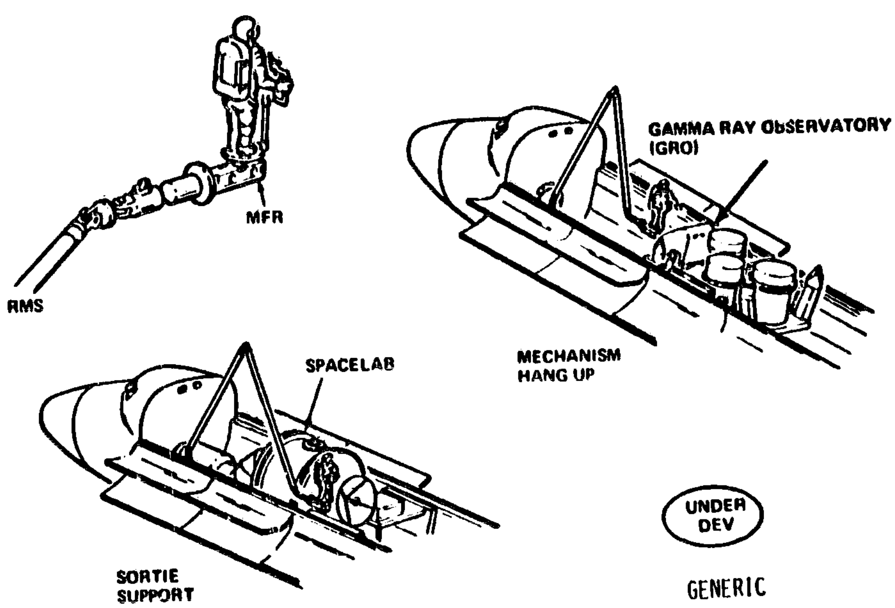
GENERIC EQUIPMENT

EQUIPMENT WHICH INTEGRATES WITH THE INHERENT EQUIPMENT AND HAS GROWTH POTENTIAL

GENERIC EQUIPMENT	SATELLITE SERVICE FUNCTION
MANIPULATOR FOOT RESTRAINT - MFT	• STABLE PLATFORM FOR MANNED ACTIVITY WITHIN OPERATING RANGE OF RMS
WORK RESTRAINT UNIT - WRU	• METHOD OF SATELLITE ATTACHMENT AND A STABLE WORK RESTRAINT DURING MMU ACTIVITY
MANEUVERABLE TELEVISION - MTV	• REMOTE SATELLITE (AND ORBITER) OBSERVATION CAPABILITY
HOLDING AND POSITIONING AID - HPA	• TEMPORARY HOLDING AND/OR POSITIONING OF A SPACE STRUCTURE WHILE WORK IS PERFORMED
FLUID TRANSFER	• CAPABILITY TO TRANSFER FLUIDS BETWEEN THE ORBITER AND SATELLITES
POWER TOOLS	• ENHANCED MANNED ACTIVITY DURING EVA

MANIPULATOR FOOT RESTRAINT (MFR)



GENERIC

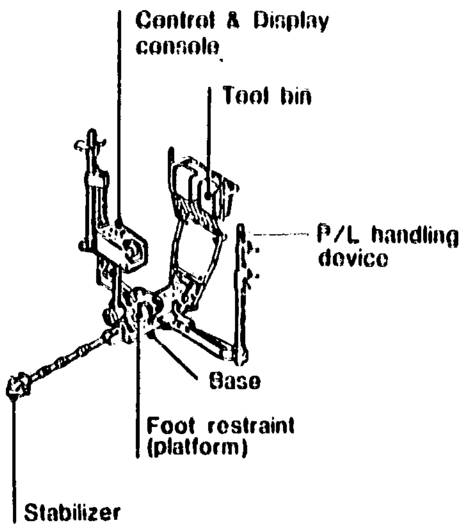
MANNED MANEUVERING UNIT/WORK RESTRAINT UNIT (MMU/WRU) ADAPTATIONS

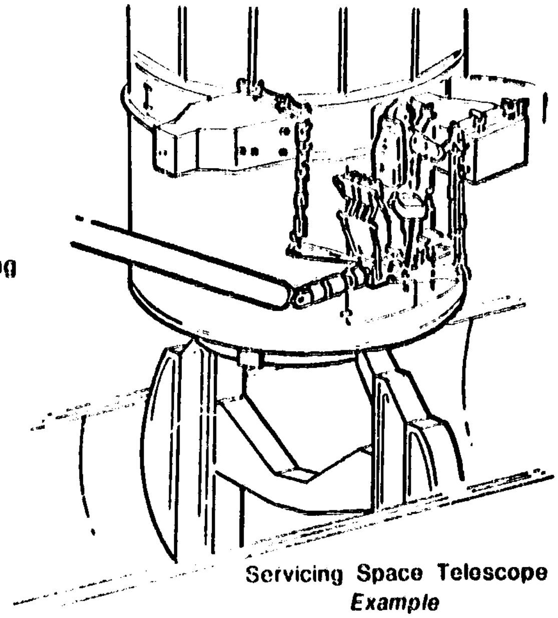
The chart illustrates three variations of an MMU/WRU adaptation that have been identified in this study. The Level 1 Operational Deployment Scenarios call for an MMU/WRU with a stabilizer to backup a satellite appendage hangup while in the process of being deployed by the RMS. This same adaptation also appears in other scenarios to back up mechanical hangup situations in the payload bay.

The MMU/WRU end effector adaptation has been shown to be applicable in all scenarios wherein the RMS is inoperative. The illustration shows its usage in removing a satellite from the payload bay in preparation for deployment.

Another MMU/WRU adaptation identified is with the unit adapted for a payload handling mode. Again, this could apply in an RMS inoperative situation during a servicing (revisit) mission.

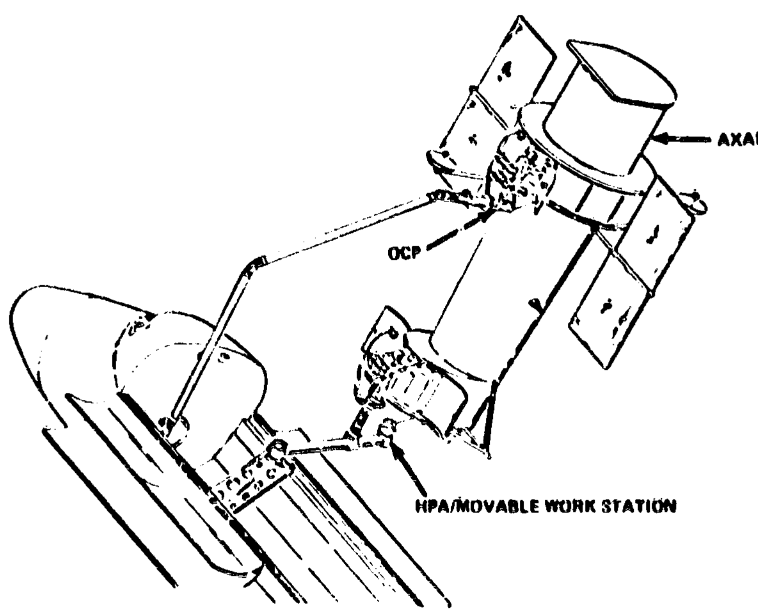
Open Cherry Picker *OCP*





Servicing Space Telescope
Example

**HANDLING AND POSITIONING
AID (HPA) AND OCP/RMS SERVICING**



INHERENT/GENERIC

EQUIPMENT STOWAGE

As identified in the Level 1 operational scenarios, a need exists for equipment stowage cannisters to transport components, replacement modules, instruments, and other equipment used in satellite servicing support, refurbishment, and reconfiguration. The accompanying chart shows three concepts that have been developed for positioning/deploying the cannisters during their use.

The side swing concept utilizes a PIDA mechanism to position the cannister out and away from the payload bay, thus leaving the area in and above the payload bay clear. The elevator concept utilizes a telescoping screw jack to raise the cannister to a position directly above the payload bay. The rotary concept utilizes an in-place carousel for accessing equipment and modules.

UNIQUE EQUIPMENT
EQUIPMENT UNIQUE TO SPECIAL MISSION REQUIREMENTS

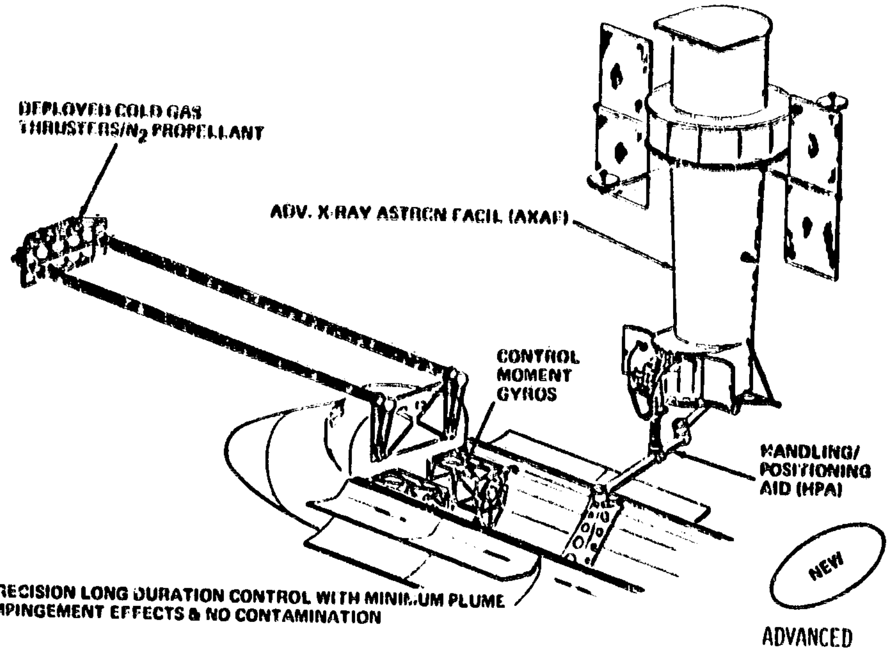
UNIQUE EQUIPMENT	SATELLITE SERVICE FUNCTION
HAND TOOLS	• ENHANCES MANNED ACTIVITY DURING EVA
EQUIPMENT STOWAGE	• PERMANENT AND TEMPORARY STOWAGE OF EQUIPMENT, SATELLITES, SPARE PARTS, TOOLS AND DEBRIS.
PAYLOAD HANDLING DEVICES	• CAPABILITY TO GRAPPLE AND HANDLE UNATTACHED PAYLOADS
RMS END EFFECTOR	• ENHANCES THE CAPABILITY OF THE RMS
TILT TABLE	• ORIENTATION OF PAYLOADS FOR DEPLOYMENT, BERTHING AND/OR SERVICING
SPIN TABLE	• CAPABILITY TO "SPIN-UP" SATELLITES PRIOR TO DEPLOYMENT
PAYLOAD INSTALLATION AND DEPLOYMENT AID	• DEPLOYMENT AND STOWAGE OF LARGE PAYLOADS WITH MINIMAL RISK OF DAMAGE TO THE ORBITER AND PAYLOAD

ADVANCED EQUIPMENT
EQUIPMENT POTENTIALLY NEEDED TO FULFILL FUTURE MISSION MODEL REQUIREMENTS

ADVANCED EQUIPMENT	SATELLITE SERVICE FUNCTION
TELEOPERATOR MANEUVERING SYSTEM	• PAYLOAD DELIVERY/RETRIEVAL TO/FROM SATELLITE OPERATIONAL ORBIT WHEN DIFFERENT FROM ORBITER ORBIT
NON-CONTAMINATING ATTITUDE CONTROL SYSTEM	• SERVICING RCS OF PLUME SENSITIVE SATELLITES
SUN SHIELD	• PROTECTION FOR SUN SENSITIVE PAYLOADS
ORBITAL STORAGE	• ENVIRONMENTAL PROTECTION FOR ON-ORBIT QUIESCENT "STORAGE" OF SATELLITES
OPTICAL ATTITUDE TRANSFER SYSTEM	• MEASURES PAYLOAD BAY DISTORTION RELATIVE TO THE INERTIAL MEASUREMENT UNIT (IUM) PLATFORM, HENCE TRANSFERRING ATTITUDE REFERENCE TO SATELLITES MORE ACCURATELY
LIGHTING ENHANCEMENT	• ENHANCES LIGHTING CAPABILITY
DEXTEROUS MANIPULATOR	• ENHANCES REMOTE "TELEOPERATOR" SERVICING CAPABILITY
DE-ORBIT PROPULSION PACKAGE	• CAPABILITY TO DE-ORBIT AND PROPEL EXPENDABLE SATELLITES TO EARTH

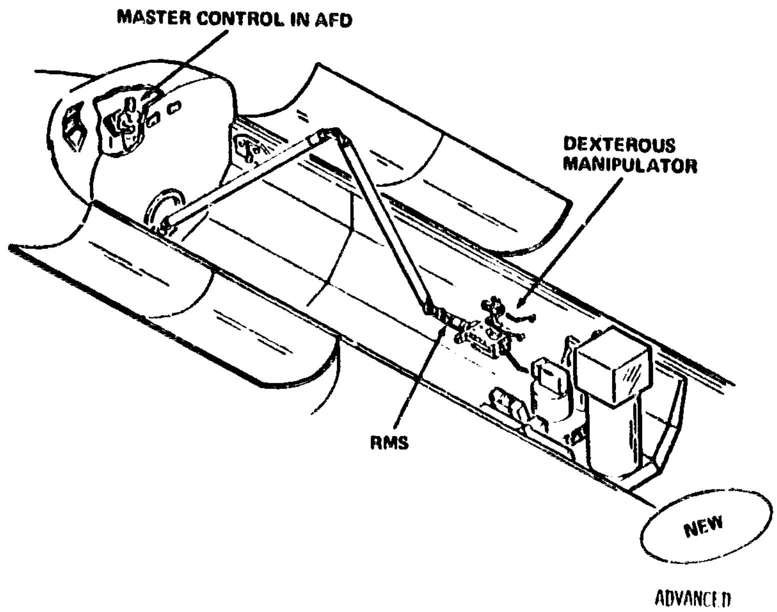
NON-CONTAMINATING ATTITUDE CONTROL SYSTEM

- SERVICING OF CONTAMINATION SENSITIVE SATELLITES



- PRECISION LONG DURATION CONTROL WITH MINIMUM PLUME IMPINGEMENT EFFECTS & NO CONTAMINATION

DEXTEROUS MANIPULATOR



DEXTEROUS MANIPULATOR - ADAPTATIONS TO HPA AND RMS

The accompanying chart illustrates adaptations of the dexterous manipulator to the HPA as well as RMS for servicing of large satellites.

HANDLING AND POSITIONING AID (HPA)

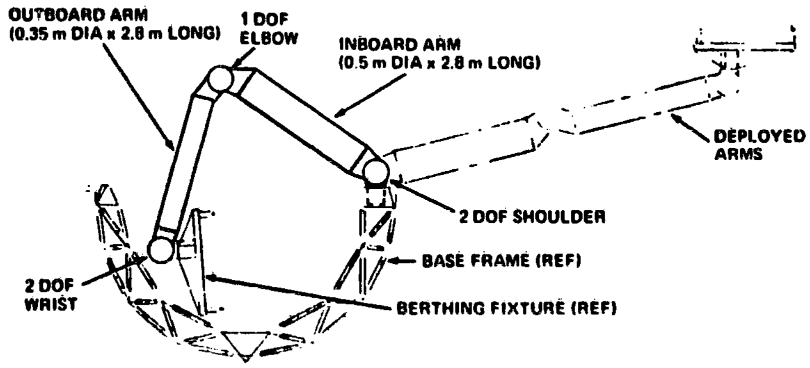
FUNCTIONS:

- PAYLOAD BERTHING
- MULTI-POSITIONING/ORIENTATION FOR ORBITER-BASED CONSTRUCTION AND SATELLITE SERVICING
- PROVIDES IMPROVED ACCESS AND OBSERVATION FOR PAYLOADS WITH COMPLEX GEOMETRIC CONFIGURATIONS
- COMPLEMENTS THE RMS

DEVELOPMENT (PATTERN SIMILAR TO RMS)

- GROUND TEST FUNCTIONAL SYSTEM (DTA)
- DEVELOP TECHNOLOGY AND SPECIFICATION FOR OPERATIONAL SYSTEM
- DEVELOP OPERATIONAL SYSTEM

HPA DEVELOPMENT TEST ARTICLE



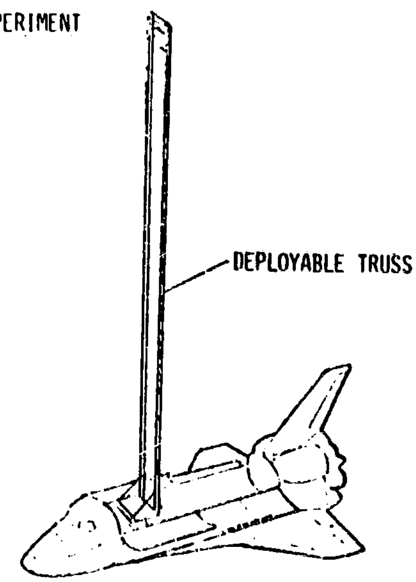
SPACE CONSTRUCTION EXPERIMENT

- ORBITER-ATTACHED LOW FREQUENCY STRUCTURE REPRESENTATIVE OF LARGE SPACE SYSTEM STRUCTURE
  - DEPLOYABLE
  - ANTENNA MAST SIZE
- CONTROL/STRUCTURE INTERACTION
  - ORBITER DIGITAL AUTOPILOT LIMITS
  - INCREMENTAL DEPLOYMENT, ORBITER-ATTACHED
- CONSTRUCTION OPERATIONS
  - RMS HANDLING (VISIBILITY, DEPLOYMENT AID)
  - EVA (INSTALL INSTRUMENTATION)
  - PROVIDE CORRELATION WITH GROUND SIMULATIONS (TIMELINE COMPARISON)

SPACE CONSTRUCTION EXPERIMENT

)BJECTIVES:

- DEVELOP AND DEMONSTRATE CONSTRUCTION CAPABILITIES
- EVALUATE DAP CONTROL WITH ATTACHED STRUCTURE
- OBTAIN DYNAMICS DATA ON LARGE DEPLOYABLE STRUCTURE
- EVALUATE ANTENNA FEED MAST TEST ARTICLE
- DEVELOP FLIGHT CONTROL TECHNOLOGY



SUMMARY OF JSC CONSTRUCTION RELATED STUDIES

STUDY	PRODUCT
• SOLAR POWER SATELLITE SYSTEMS STUDIES NAS9-15636	CONSTRUCTION BASE AND EQUIPMENT CONCEPTS, CONSTRUCTION OPERATIONS.
• ORBITAL CONSTRUCTION DEMONSTRATION STUDY NAS9-14916	FACTORY FLOOR FACILITY SUPPORTED BY ORBITER IN SORTIE MODE.
• ORBITAL CONSTRUCTION EQUIPMENT DEFINITION STUDY NAS9-15120	EQUIPMENT CONCEPTS TO PERFORM CONSTRUCTION FUNCTIONS I.E., OCP, SPACE CRANE, ETC.
• SPACE STATION SYSTEMS ANALYSIS NAS9-14958	FACILITY AND EQUIPMENT REQUIREMENTS FOR CONSTRUCTION IN LEO.
• SPACE CONSTRUCTION AUTOMATED FABRICATION EXPERIMENT DEFINITION STUDY NAS9-15310	BEAM BUILDER AND ASSEMBLY JIG FOR ORBITER BUILT PLATFORM.
• SPACE CONSTRUCTION SYSTEM ANALYSIS STUDY NAS9-15718	EQUIPMENT REQUIREMENTS AND OPERATIONS FOR ORBITER BUILT LARGE SYSTEMS. CONSTRUCTION EXPERIMENTS CONCEPTS. SHUTTLE CONSIDERATION DOCUMENT.
• DEPLOYABLE ORBITAL SERVICE PLATFORM CONCEPTUAL SYSTEM STUDY NAS9-15532	EQUIPMENT REQUIREMENTS FOR ASSEMBLY OF SMALL PLATFORM FROM ORBITER.
• SPACE PLATFORM ADVANCED TECHNOLOGY STUDY NAS9-16001	ASSEMBLY EQUIPMENT DESIGNS, BERTHING LATCH INTERFACE MECHANISM ARTICLE.
• SATELLITE SERVICES NAS9-16120 AND NAS9-16121	REQUIREMENTS FOR EQUIPMENT AND TECHNIQUES FOR SERVICING SATELLITES
• SPACE OPERATIONS CENTER SYSTEM ANALYSIS NAS9-16151	CONSTRUCTION CONCEPT DESCRIPTIONS AND FACILITY REQUIREMENTS, CONSTRUCTION TECHNOLOGY REQUIREMENTS.
• ACHIEVEABLE FLATNESS IN A LARGE MICROWAVE POWER ANTENNA NAS9-15423	EFFECT OF STRUCTURAL TOLERANCE VARIATIONS.
• COMPOSITE GEODETIC STRUCTURE FOR SPACE CONSTRUCTION NAS9-15678	DEVELOPMENT OF SPACE FABRICATED BEAM.

CONSTRUCTION SUPPORT EQUIPMENT

- REQUIREMENTS FOR GENERIC CONSTRUCTION SUPPORT FUNCTIONS HAVE BEEN IDENTIFIED
- DEVELOPMENT PLANNING IS PROCEEDING ON SOME EQUIPMENT
  - RMS - STS FLIGHT TEST
  - MMU - IN QUALIFICATION
  - MTV - GROUND TEST ARTICLE
  - PIDA - GROUND TEST ARTICLE
  - OCP - GROUND TEST ARTICLE
  - HPA - GROUND TEST ARTICLE DESIGN
  - BERTHING/DOCKING - LATCH MECHANISM TEST ARTICLE
- CONTINUED ANALYSIS OF CONSTRUCTION OPERATIONS MAY RESULT IN REQUIREMENTS FOR GENERIC EQUIPMENT AS WELL AS SPECIALIZED EQUIPMENT FOR SPECIFIC MISSIONS

CONCLUSIONS

- PRODUCTIVITY IS A FUNCTION OF FIXTURE COST AND COMPLEXITY.
- FOR CONSTRUCTION FROM THE ORBITER CARGO BAY, LONG, THIN STRUCTURES PASSING THROUGH THE WORK STATION ARE PREFERRED.
- SOME CONSIDERATIONS FOR ORBITER-BASED CONSTRUCTION ARE:
    - ORBITER ATTITUDE CONTROL - PLUME EFFECTS AND ACCELERATION
    - ORBITER STAY TIME - LIMITED
    - EVA - COST, SAFETY AND TRAINING TIME
- THE EXPECTED EVOLUTION OF CONSTRUCTION REQUIREMENTS IS:
    - DEPLOYABLES
    - DEPLOYABLES PLUS ASSEMBLY
    - FABRICATION - SORTIE
    - FABRICATION - SPACE-BASED
    - FABRICATION - SOC

  ADDITIONAL REQUIREMENTS FOR THE SOC MAY INITIATE THIS CAPABILITY EARLIER THAN CONSTRUCTION REQUIREMENTS DICTATE.

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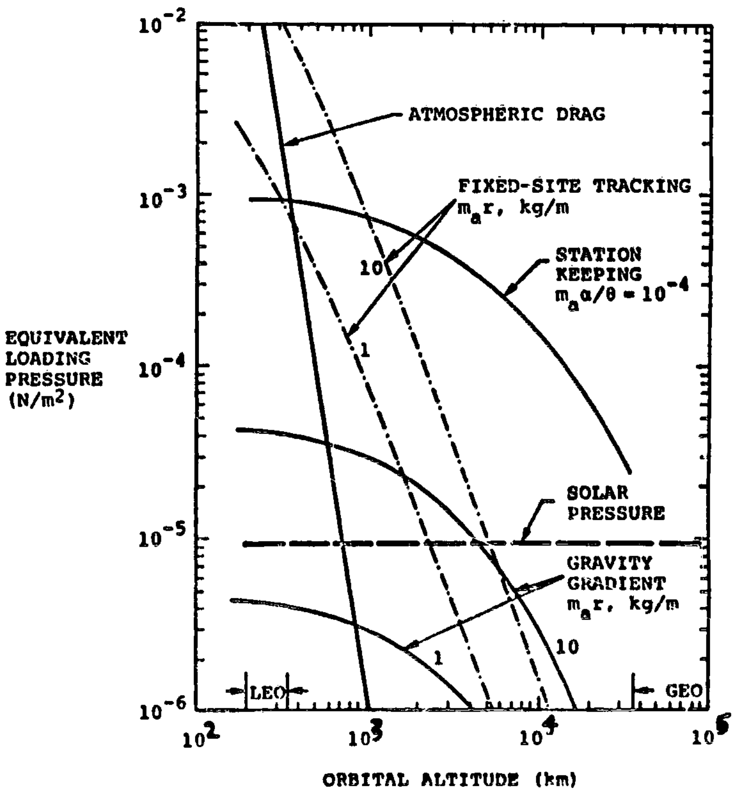
SOME INTERDISCIPLINARY TRADE-OFFS IN THE DESIGN OF LARGE SPACE STRUCTURES

John Hedgepeth

Astro Research Corporation

EQUIVALENT LOADING PRESSURE VERSUS ORBITAL ALTITUDE



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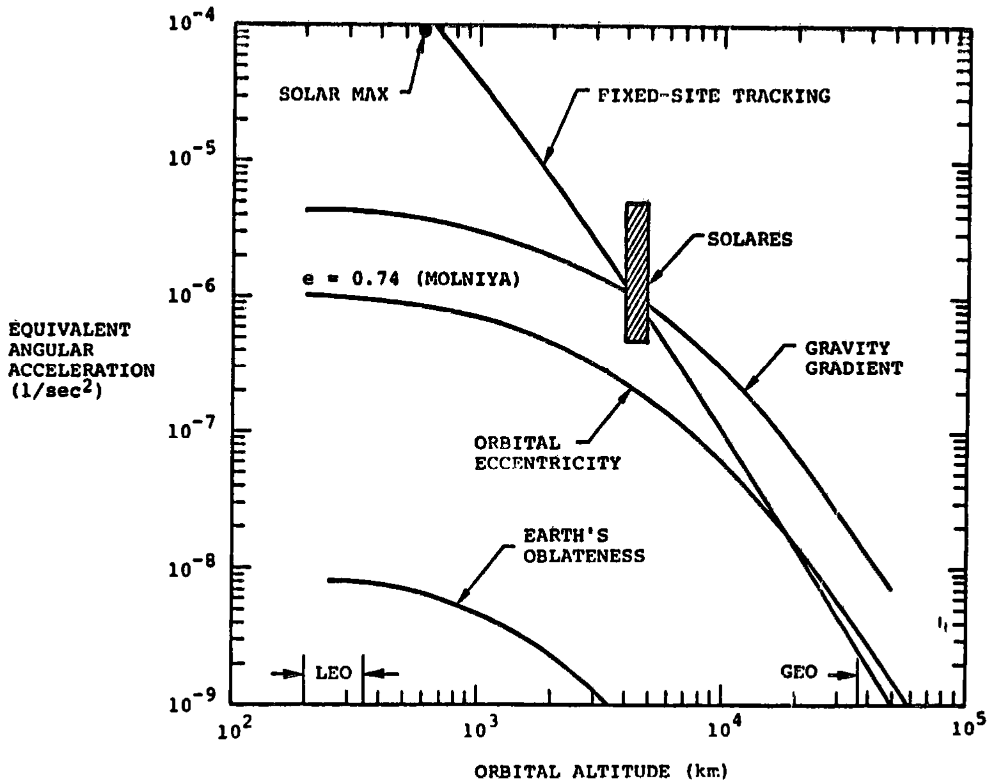
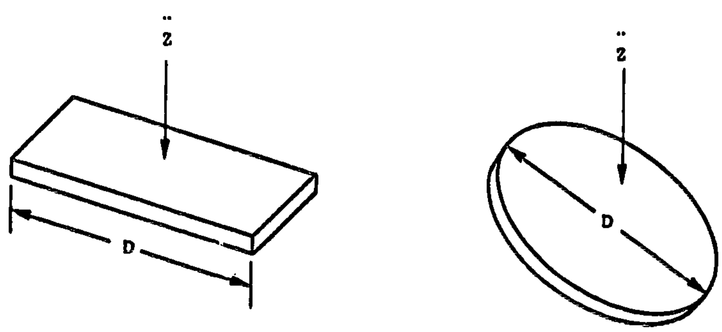
EQUIVALENT ANGULAR ACCELERATION VERSUS ORBITAL ALTITUDE



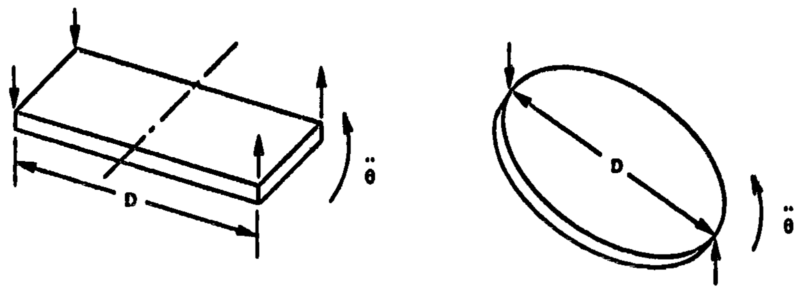
DEFORMATIONS DUE TO LATERAL ACCELERATION



$\Gamma_z = 6.1 \times 10^{-3}$ $\Gamma_z = 25.0 \times 10^{-3}$

$$w_{rms} = \Gamma_z \frac{\ddot{z}}{f_n^2}$$

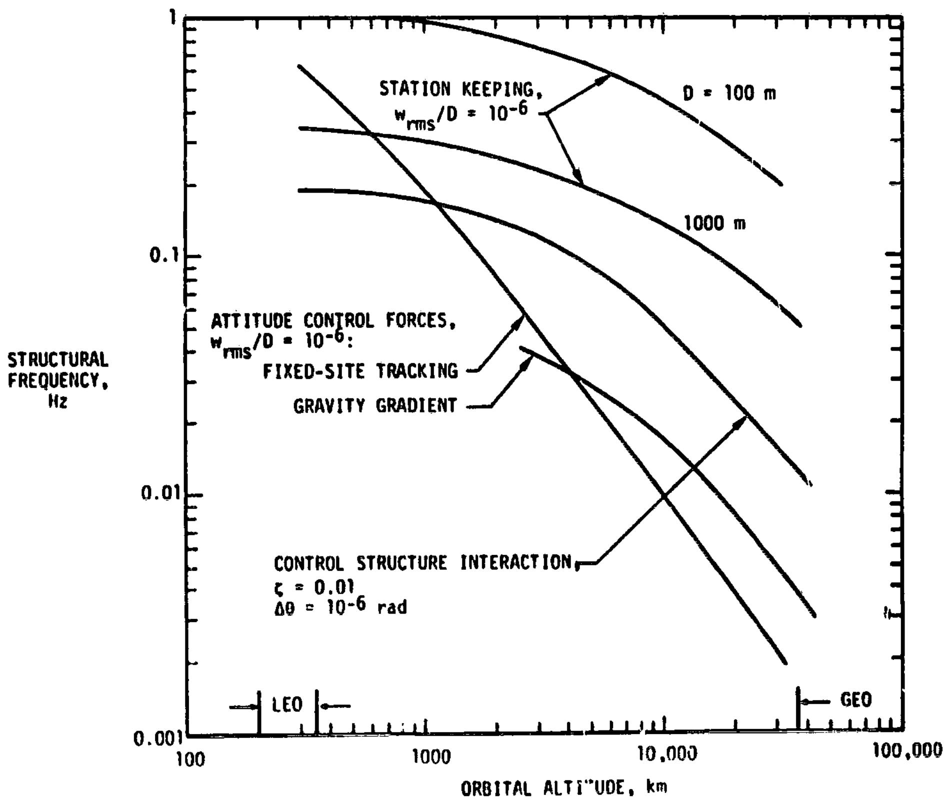
DEFORMATIONS DUE TO ATTITUDE CONTROL



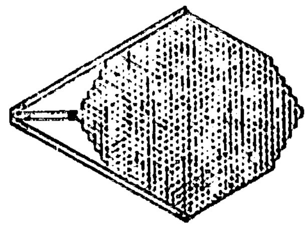
Γ_θ = 0.95 x 10^{-3} Γ_θ = 2.14 x 10^{-3}

$$\frac{w_{rms}}{D} = \Gamma_\theta \frac{\ddot{\theta}}{f_n^2}$$

STIFFNESS REQUIREMENTS

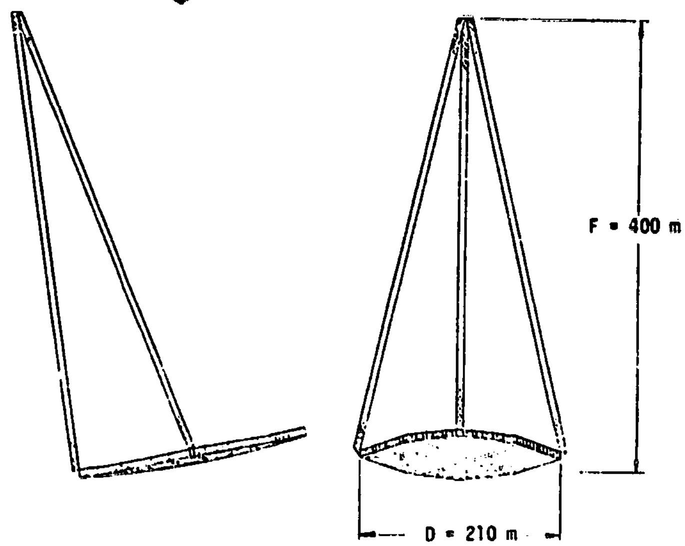


200-METER-DIAMETER DEPLOYABLE ANTENNA



CELL SIZE = 7 m
DEPTH = 7 m
STRUT DIAMETER = 40 mm
REFLECTOR MASS = 6500 kg
VIBRATION FREQUENCY (REFLECTOR ONLY) ≃ 1 Hz

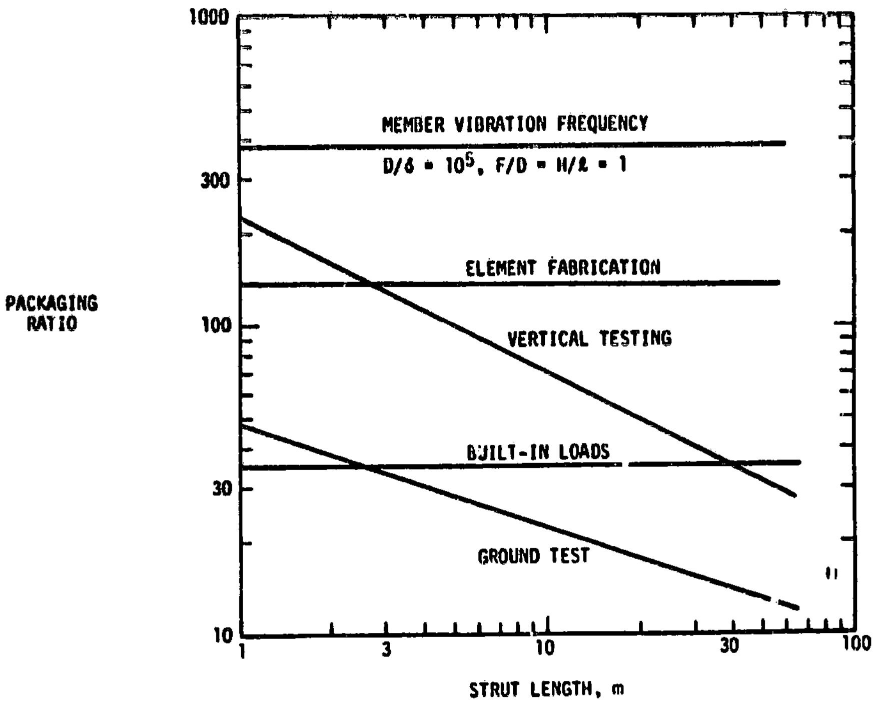
SURFACE ERROR ≃ 4 mm



ALLOWABLE MEMBER SLENDERNESS

$\ell/k_c <$	CASE
$\frac{573.9}{\ell^{1/3}}$	Horizontal testing. Gravity sag < $k_c/3$
$\frac{2700}{\ell^{1/2}}$	Vertical testing. Tension < 1/10 Euler load
1633	Fabrication. $\Delta\varepsilon = 10^{-6}$
414	Built-in loads < 1/10 Euler load, $\sigma_\varepsilon = 10^{-5}$
$2.8\,\frac{D^2}{H\ell}$	Member vibration frequency > 3 truss frequency, $m_p/m_2 = 2$, $k = 2$

LIMITATIONS ON PACKAGING RATIO



ACCURACY CONTROL
ACTIVE VERSUS PASSIVE

- Passive accuracy control is attractive and is used almost universally.
- Active control is used only primitively.
- Passive control is limited - <u>and</u> <u>limiting</u>!
- Many projected missions need active control.
- We need to understand how best to combine active with passive control.

ACTIVE CONTROL TASKS

- Sensing difficulty is directly dependent on the field of view and inversely dependent on the absolute accuracy desired.

- Computation becomes more lengthy as the ratio of largest to smallest quantity increases.

- Actuators are more complicated as the stroke increases relative to the required movement accuracy.

- Conclusion is that the expense of active control is dependent on its basic task of improving accuracy.

ASSUMED EXPENSE OF PRECISION

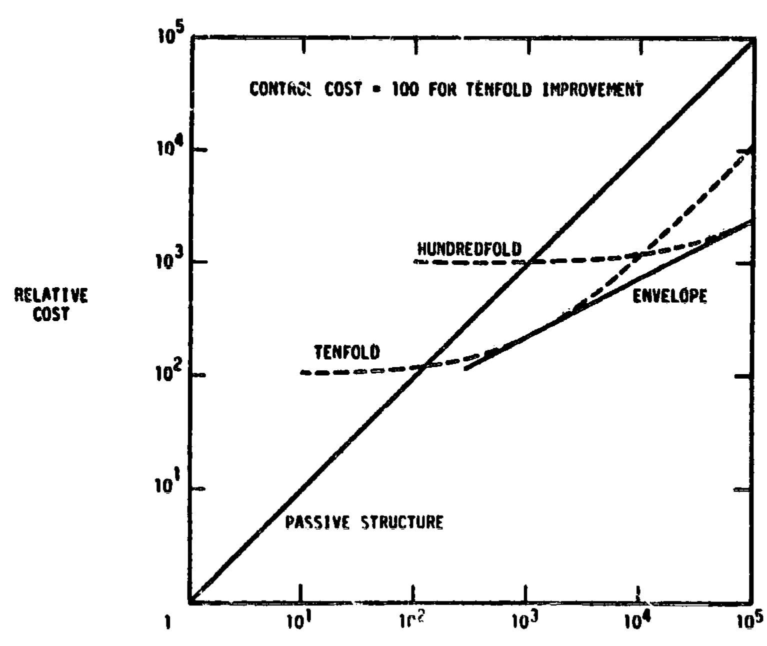
Structure:

$$\text{Effort} \sim \text{Required Accuracy}$$

Active Control:

$$\text{Effort} \sim \frac{\text{Required Accuracy}}{\text{Foundation Accuracy}}$$

ACTIVE ACCURACY CONTROL



CONCLUSIONS

- We should not look at active versus passive control. We should determine how passive accuracy can alleviate the active-control task.

- Attitude and orbit-keeping control forces will necessitate stiff structures.

- Ground-testing requirements may dictate package size.

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ACTIVE LARGE STRUCTURES

KETO SOOSAAR

LARGE SPACE STRUCTURES DIV.
C.S. DRAPER LABORATORY, INC.

BASIC MESSAGE OF THIS PRESENTATION

- AMBITIOUS MISSIONS LEAD TO LARGE SPACE STRUCTURES
- FOR SENSORS - POINTING AND ALIGNMENT TIGHTEN WITH SIZE
- STRUCTURAL FLEXIBILITY AND DISTURBANCES INCREASE WITH SIZE
- CONTROL BECOMES ONLY MEANS OF OBTAINING PERFORMANCE
- CONTROL BANDWIDTHS OVERLAP: TECHNOLOGY AND INTEGRATION PROBLEMS
- NEED TO WORRY ABOUT DEPLOYMENT PROBLEM
- THEORY, COMPONENT TECHNOLOGY, INTEGRATION TECHNOLOGY NEEDED

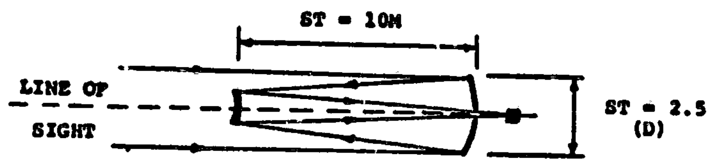
SCALING OF POINTING AND ALIGNMENT REQUIREMENTS

- CURRENT SPACE TELESCOPE AT 2.4M APERTURE

- REQUIREMENTS CHANGE IF APERTURE INCREASED TO 10M

- OTHER LARGER SYSTEMS ALSO CHANGE BUT SCALED BY WAVELENGTH (λ)

HIGH PERFORMANCE RADIATIVE SYSTEMS



λ = WAVELENGTH

D = DIAMETER OF OPTICS

WHEN DIFFRACTION LIMITED PERFORMANCE IS ACHIEVED

ANGULAR RESOLUTION = $\frac{1.22\,\lambda}{D}$

- LINE OF SIGHT STABILITY $\lesssim 0.2 \times \left(\frac{1.22\,\lambda}{D}\right)$ (STREHL RATIO-BASIS)

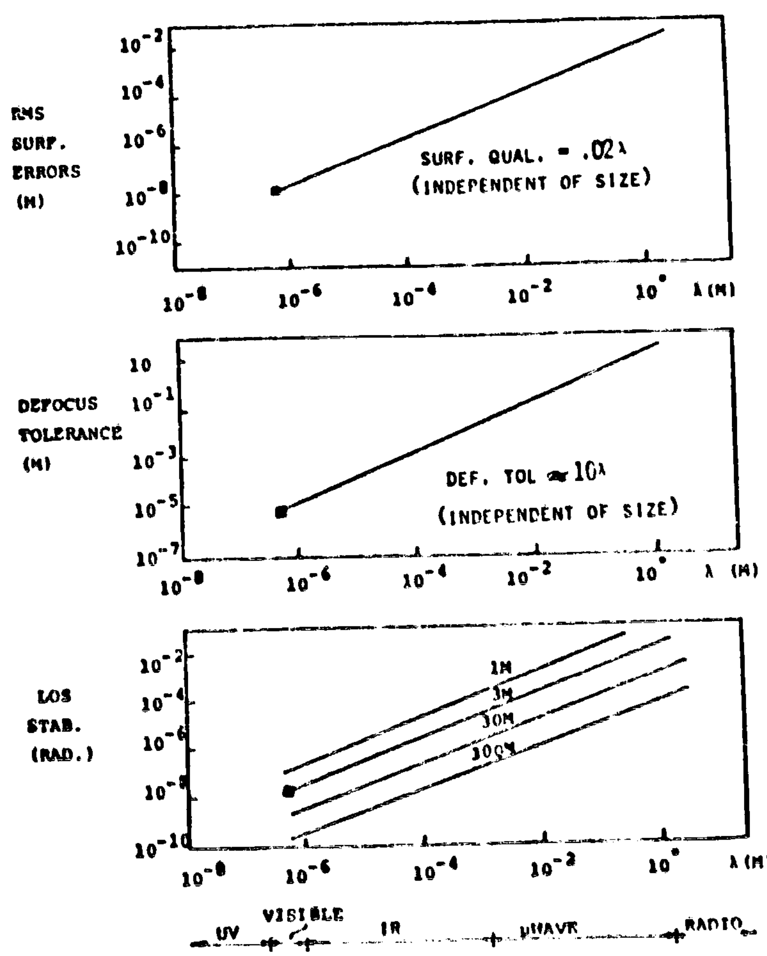
- MIRROR SURFACE ERRORS $\lesssim \lambda/50$

- DEFOCUS TOLERANCE $\approx 10\,\lambda$ (FOR DIFFRACTION-LIMITED R-C SYST DESIGNED AT D = 2.5, λ = .6μ)

PERFORMANCE REQUIREMENTS: LARGE OPTICS

PERFORMANCE CRITERION	NASA - ST	VLST
REFERENCE WAVELENGTH	λ = .6µM	λ = .6µM
REFLECTOR DIAMETER	2.5M	10M
SURFACE QUALITY	λ/50	λ/50
ANGULAR RESOLUTION	.3 x 10^{-6}RAD	.075 x 10^{-6}RAD
POINTING STABILITY	.06 x 10^{-6}RAD	.015 x 10^{-6}RAD
DEFOCUS TOLERANCE	[illegible]µM	10µM
OVER DISTANCE OF	10 M	40 M

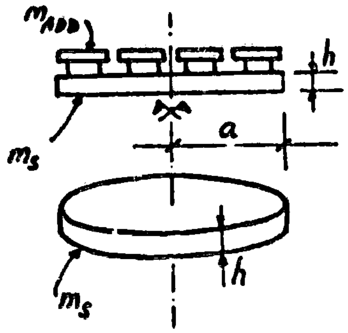
DIFFRACTION - LIMITED PERFORMANCE REQUIREMENTS



STRUCTURAL AND DYNAMIC EFFECTS

- WITH INCREASED SIZE NATURAL VIBRATION FREQUENCIES DECREASE
- VERY MANY VIBRATIONAL MODES IN DISTURBANCE AND CONTROL BANDWIDTHS
- RESPONSE INCREASES RAPIDLY WITH DECREASING FREQUENCY
- ACCURACY OF STRUCTURAL MODELLING TOOLS STRESSED
- DAMPING GENERALLY LOW IN PRECISION STRUCTURES (<0.5%)

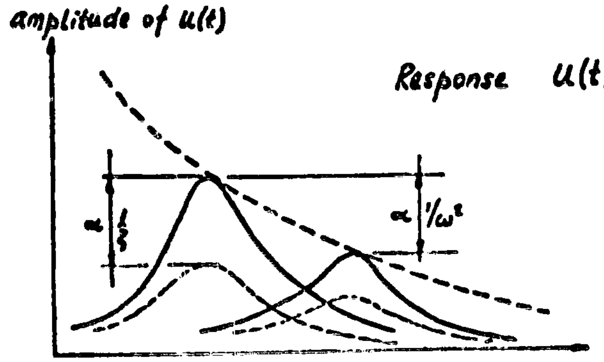
STRUCTURAL SCALING FACTORS



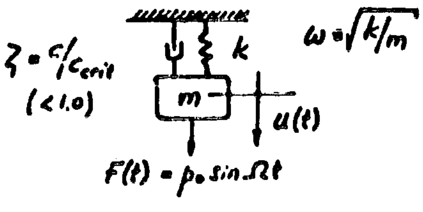
$$\omega = \frac{5.25}{a^2}\sqrt{\frac{E h^3}{12(1-\nu^2)}\cdot\frac{1}{(m_s + m_{ADD})}} \quad [rad/s]$$

Material

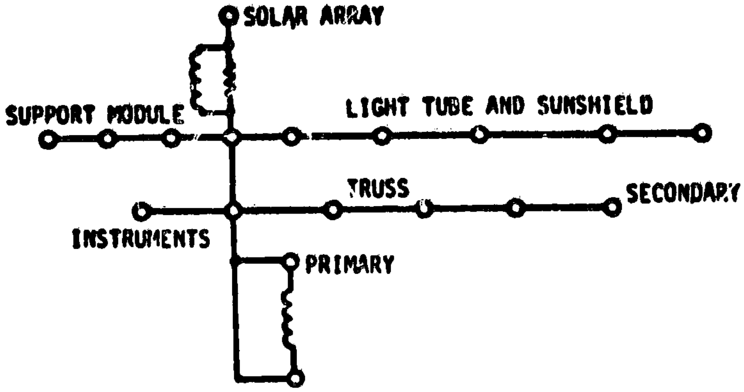
	E	*Young's modulus*
	ν	*Poisson's ratio*
	ρ	*Density*



Response $$u(t) = \frac{p_0}{m\omega^2}\cdot\frac{\sin(\Omega t-\theta)}{\sqrt{\left[1-\left(\frac{\Omega}{\omega}\right)^2\right]^2+\left(2\zeta\frac{\Omega}{\omega}\right)^2}}$$

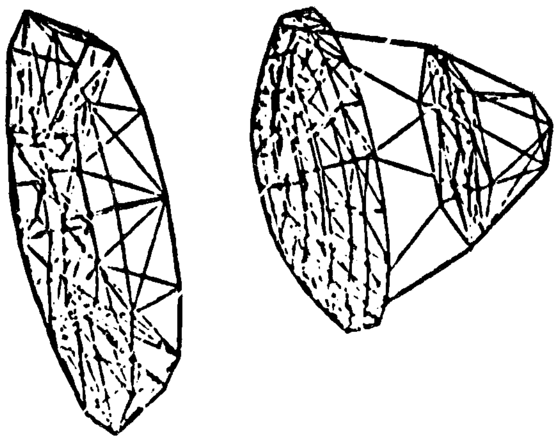


FLEXIBLE STRUCTURE MODELING



ST MODEL

- STICK MODEL
- 17 NODES
- 14 MASSES



DETAIL REQUIRED FOR BASIC STRUCTURE MODELING OF LARGE OPTICS CONCEPT

- 3-DIMENSIONAL MODEL
- 222 NODES
- 71 MASSES

TYPICAL MODE IDENTIFICATIONS

LSS VIBRATION MODES

MODE DESCRIPTION

MODE NUMBER	FREQ. (HZ)	SOLAR PANEL	SUNSHADE	METERING TRUSS	LARGE MIRROR
7	.57	X3 BNDG.			
8	.60	X1 BNDG.			
9	.63	X3 BNDG.			
10	.69	X1 BNDG.			
11	1.26	X5 TORS.			
12	1.27	X5 TORS.			
13	2.68		X1 BNDG.		
14	3.24	X5 TORS.			
15	3.25	X5 TORS.			BNDG I/P
16	3.25	X5 TORS.			BNDG I/P
17	3.26				BNDG O/P
18	3.51		X1 BNDG.	X1 BNDG.	BNDG O/P
19	3.98	X3 BNDG.	X2 BNDG.		
20	4.07	X3 BNDG.	X2 BNDG.		
21	4.16	X1 BNDG.			BNDG I/P
22	4.17	X1 BNDG.			BNDG I/P
23	4.24	X1 BNDG.			
24	4.32	X1 BNDG.			
25	5.05		X2 BNDG.	BNDG. TORS.	BNDG O/P
26	5.14		X2 BNDG.	BNDG. TORS.	
27	5.23	X3 BNDG.	AXIAL	AXIAL	BNDG O/P
28	5.65	X1 BNDG.	X2 BNDG.	TORSION	BNDG I/P
29	6.85	X1 BNDG.			
30	7.11		X1 BNDG.	AXIAL	
31	7.89		X2 BNDG.	X2 BNDG.	BNDG I/P
32	7.97		X1 BNDG.	X1 BNDG.	BNDG I/P
33	9.28			AXIAL	BNDG I/P
170	99.78				

SPACE TELESCOPE VIBRATION MODES

FREQ. (HZ)	MAIN DESCRIPTION
.2	SOLAR PANEL
6.2	LIGHT TUBE &
9.0	SUNSHADE FLEX
15.0	PRIMARY/ROCK.
20.0	PRIMARY/S/C.
26.0	TRUSS
45.0	PRIMARY/BNDG.
55.0	

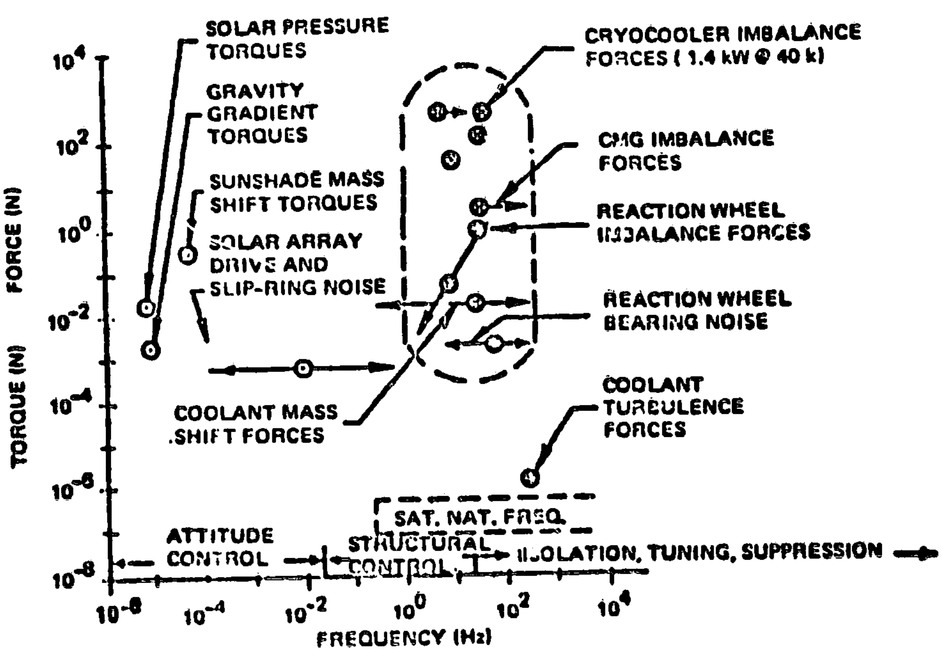
CONCLUSION

UNLIKE SPACE TELESCOPE LSS LOWER MODES (3.26 HZ) AFFECT OPTICAL QUALITY

DISTURBANCE ALSO SCALE WITH SIZE

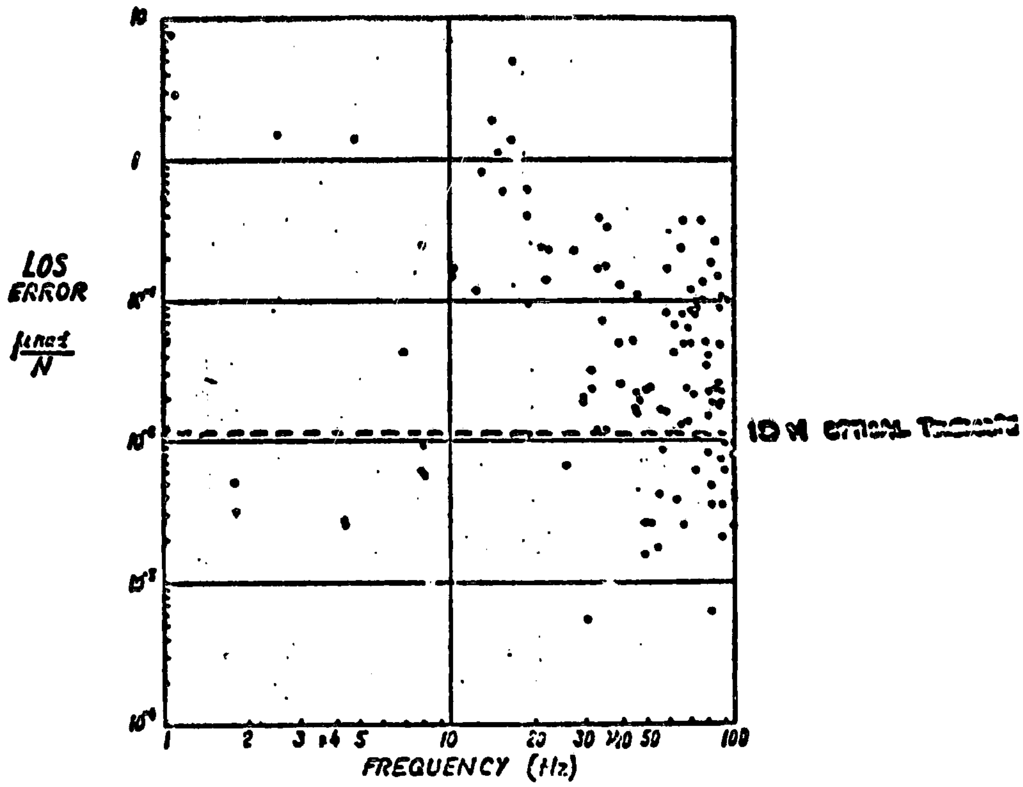
- **ENVIRONMENTAL** - RADIATION PRESSURE AND WIND
  GRAVITY GRADIENT
  ATMOSPHERIC DRAG
  THERMAL
  ETC.
- **ON-BOARD** - CRYOCOOLERS
  HIGH-ENERGY LASER DEVICES
  CMG/REACTION WHEEL IMBALANCES
  THRUSTERS
  ETC.
- **MISSION-RELATED** - RAPID MANEUVERS TO RETARGET
  SCANNING PROCEDURES

DISTURBANCE FACTORS



OTHER SIGNIFICANT DISTURBANCES:
SUNSHADE TRACK MISALIGNMENT
SUNSHADE DRIVE BEARING NOISE
SHUTTER ACTIVATION
MOMENTUM DUMPING THRUSTERS
STATIONKEEPING THRUSTERS

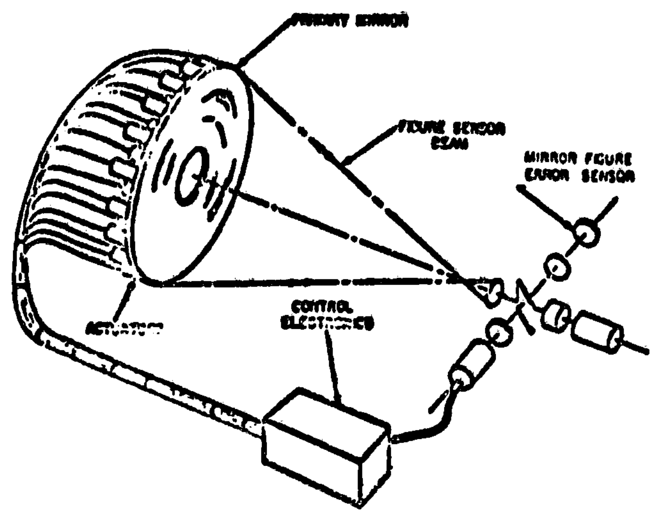
LOS ERROR RESONANCE PEAKS



TYPES OF CONTROL IN LARGE SATELLITES

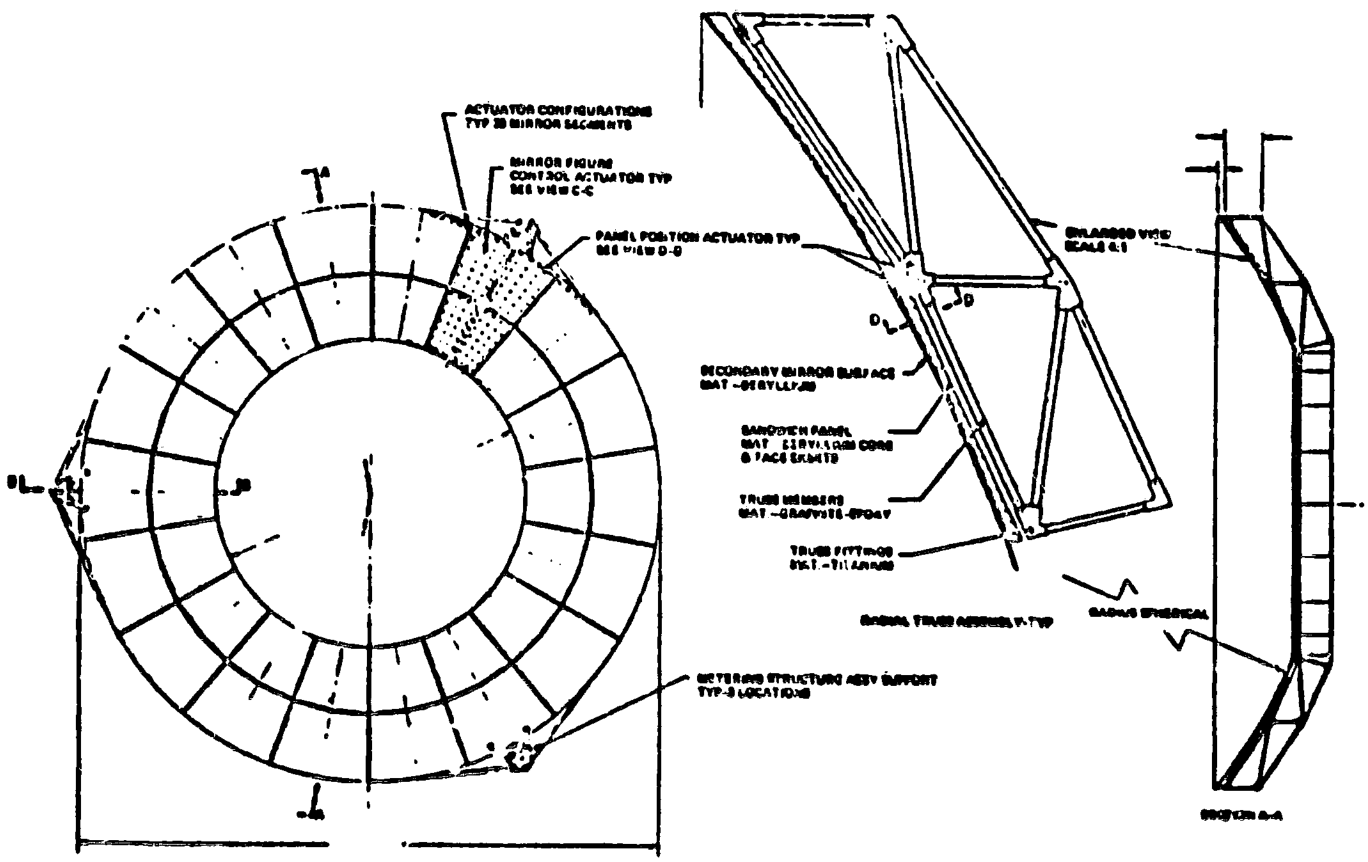
- THERMAL - SYSTEM NEEDS TO BE KEPT AT
  - UNIFORM TEMPERATURE
  - BELOW (OR ABOVE) SPECIFIC TEMPERATURE
- FIGURE/SURFACE CONTROL - DEFORMATIONS KEPT BELOW REQUIRED LEVEL
  - DISTRIBUTED MULTI-DEGREE OF FREEDOM CONTROL
  - SCALE OF CONTROL USUALLY ONLY PERMITS QUASI-STATIC CORRECTIONS
- STRUCTURAL CONTROL - SUPPRESSION OF VIBRATIONAL RESPONSE THROUGH CONTROL MEANS:
  - DISTRIBUTED DYNAMIC MULTI-DEGREE OF FREEDOM CONTROL
  - VERY PRECISE BUT RELATIVE ALIGNMENT GENERALLY
- ATTITUDE CONTROL - MAINTENANCE OF SATELLITE POINTING DIRECTION
  - PRESENCE OF FLIXIBILITY IN CONTROL BANDWIDTH

FIGURE/SURFACE CONTROL



- DISTRIBUTED PARAMETER CONTROL FLEXIBLY COUPLED
- TECHNOLOGY DEVELOPED IN EARLY 1970's FOR STATIC CORRECTION - NASA/DOD
- HIGH-BANDWIDTH OPTICS DEVELOPED IN LASER PROGRAMS
- CORRECTION FOR MIRROR DYNAMICS LARGELY UNEXPLORED
- WIDE-FIELD DYNAMIC SENSORS STILL IMMATURE

SECONDARY MIRROR ASSEMBLY



VIBRATION CONTROL OF STRUCTURES

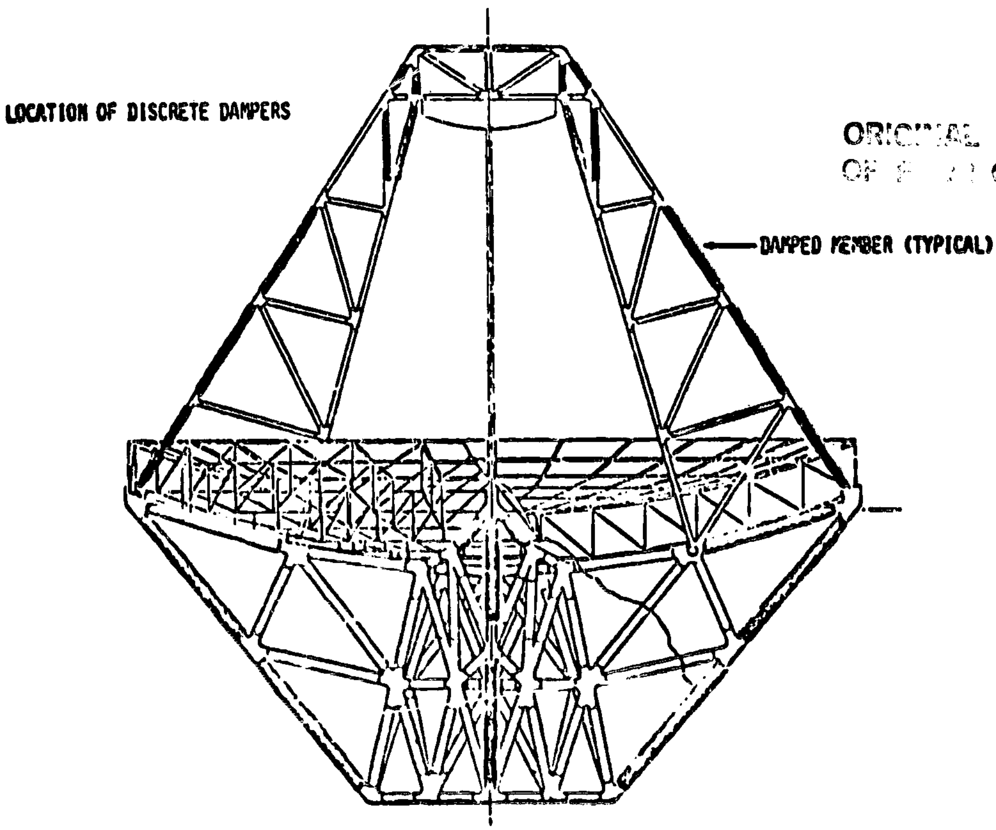
- RIGID-BODY EFFECTS SMALL COMPARED TO FLEXIBLE JITTER
- FLEXIBLE JITTER INCLUDES BOTH APPENDAGES AND PAYLOAD
- ACOSS (ACTIVE CONTROL OF SPACE STRUCTURES) - DARPA PROGRAM.
  - TO AUGMENT INHERENT STRUCTURAL DAMPING THROUGH CONTROL MEANS
  - TO OBTAIN 5 TO 25% OF CRITICAL DAMPING
  - APPROACHES
    - VISCOELASTIC STRUCTURAL MATERIAL
    - PASSIVE AND ACTIVE SOURCE ISOLATION
    - DISTRIBUTED PASSIVE DAMPERS
    - MODAL DAMPING - PASSIVE AND ACTIVE
    - MODERN CONTROL APPROACHES
      - DISTRIBUTED OUTPUT FEEDBACK
      - MODERN MODAL CONTROL
      - ADAPTIVE CONTROLLERS

VIBRATION CONTROL OF STRUCTURES: CURRENT APPROACHES

- CONSTANT GAIN - OUTPUT FEEDBACK
  - SIMPLEST CONTROL APPROACH
  - ELIMINATES ON-LINE STATE ESTIMATORS
  - MODEL ERROR SENSITIVITY LOW
  - STABILITY THEOREMS AVAILABLE FOR VELOCITY OUTPUT FEEDBACK
- FINITE-DIMENSIONAL COMPENSATORS
  - MODERN MODAL CONTROL
    - REQUIRE ON-LINE ESTIMATORS
    - CONTROL AND OBSERVATION SPILLOVER CAN CAUSE INSTABILITIES
    - THEORY FOR CONTROLLING FINITE DIMENSIONAL SYSTEMS IMMATURE
  - ADAPTIVE CONTROLLERS
    - ON-LINE OR OFF-LINE SYSTEM IDENTIFICATION REQUIRED
    - TIME-DEPENDENT CONTROL GAINS
    - THEORY IMMATURE

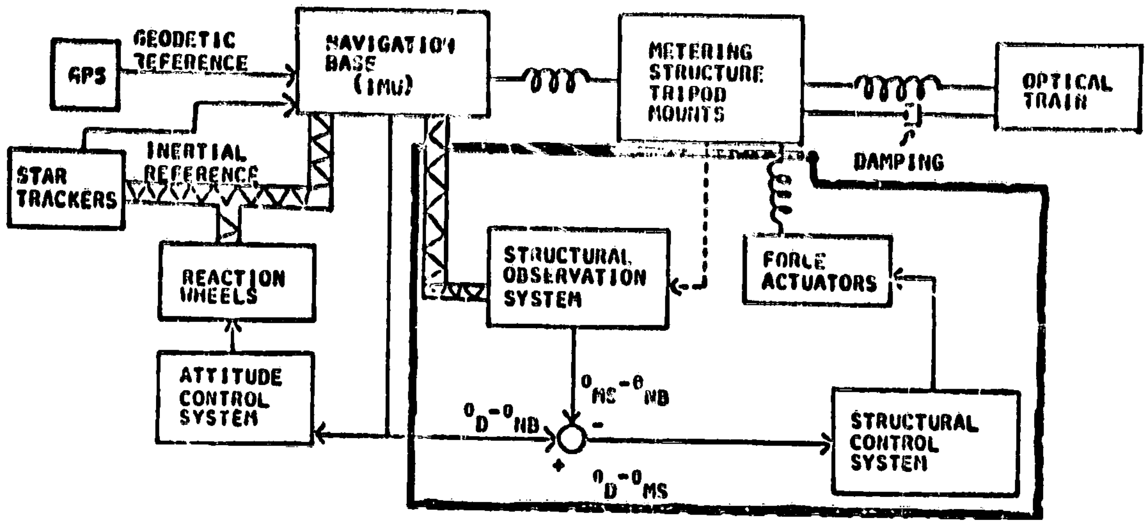
VIBRATION CONTROL OF STRUCTURES: PRIMARY IMPLEMENTATION ISSUES

- FINITE DIMENSIONAL CONTROLLER
  - INSTABILITIES ARISING FROM TRUNCATED OR UNMODELLED MODES
  - SENSITIVITY TO PARAMETER VARIATIONS
- CONTROLLER INTERACTIONS
  - ATTITUDE/VIBRATION/FIGURE CONTROL
  - EXTERNAL/ON-BOARD DISTURBANCES
- SENSOR-ACTUATOR REQUIREMENTS
  - SUBMICRON RESOLUTION, HIGH BANDWIDTH
  - SENSOR/ACTUATOR DYNAMICS
- SYSTEM IDENTIFICATION
  - FREQUENCIES AND MODESHAPES, TRANSFER FUNCTION
- DIRECT DIGITAL DESIGN
- SENSOR-ACTUATOR PLACEMENT
- FAULT TOLERANT, RELIABLE FLIGHT HARDWARE



OF POOR QUALITY

ATTITUDE/STRUCTURE CONTROL SYSTEM



CONNECTIONS:

- ——— INFORMATION FLOW
- ······· OPTICAL
- FLEXURAL
- STIFF STRUCTURE

ATTITUDES:

- θ_D = DESIRED
- θ_{NB} = NAVIGATION BASE
- θ_{MS} = METERING STRUCTURE AT TRIPOD MOUNTS

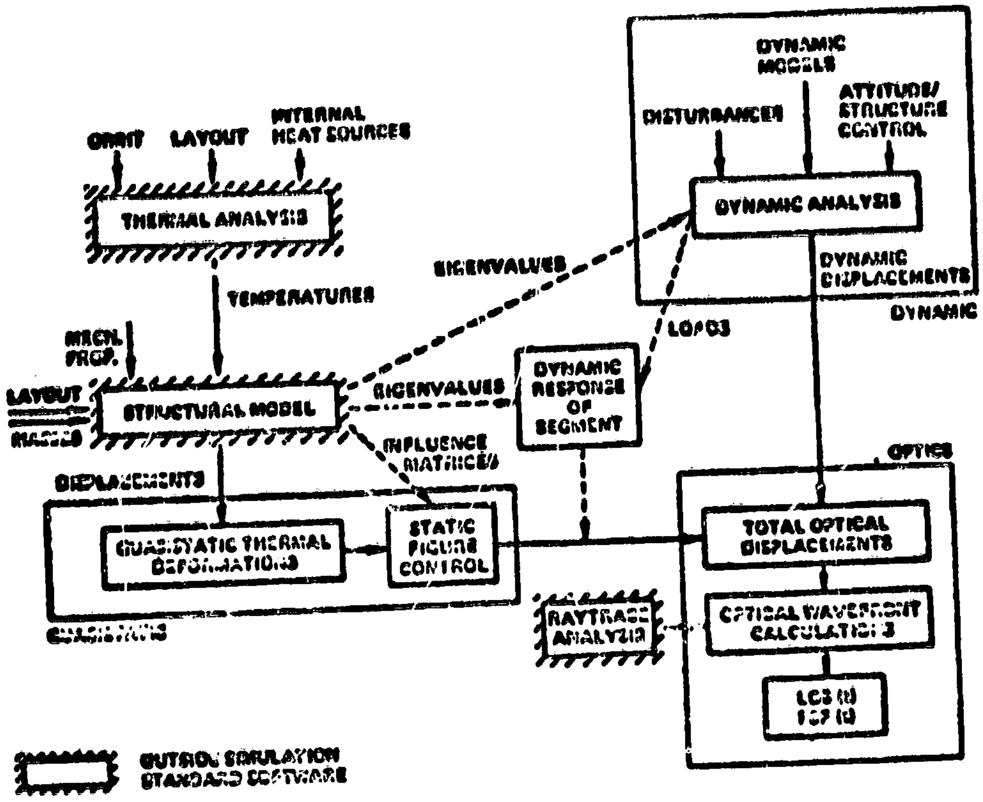
BANDWIDTHS:

IMU	10 Hz
ATTITUDE CONTROL	0.01 Hz
STRUCTURE CONTROL	5 Hz

UNPRECEDENTED INTERACTION BETWEEN CONTROLLERS

- SENSOR PERFORMANCE UNOBTAINABLE WITHOUT CONTROL
- INTERACTION BETWEEN VARIOUS CONTROL SYSTEMS:
  - THERMAL - FIGURE - VIBRATION - ATTITUDE
- SYSTEMS PERFORMANCE NOT MEASURABLE FROM SUBSYSTEM ANALYSIS
- SYSTEM-LEVEL SIMULATION NECESSARY
- ILS^3 - INTEGRATED LARGE SPACE SENSOR SIMULATION
  - HAS BEEN USEFUL IN SORTING OUT SOME PROBLEMS
  - PROVIDES GUIDANCE FOR FUTURE SYSTEMS INTEGRATION PROBLEMS

**DRAPER INTEGRATED LARGE SPACE STRUCTURES SIMULATION (ILS^3)
BLOCK DIAGRAM**



DEPLOYMENT DYNAMICS, CONTROL AND SIMULATION

- DEPLOYMENT (UN-PACKAGING) WILL BE PART OF MANY LSS SYSTEMS
- HISTORICALLY ONE OF MOST TROUBLESOME AREAS OF SATELLITE DESIGN
- MODELLING OF DEPLOYMENT
  - DIFFICULT MULTI-BODY PROBLEM DYNAMICALLY
  - LARGE DISPLACEMENT STRUCTURAL BEHAVIOR
  - STABILITY OF MAJOR CONCERN
  - "DISCOS" NEEDS TO BE EXTENDED/SUPPLEMENTED
- GENERAL-PURPOSE DESIGN/ANALYSIS TOOLS NECESSARY HERE
- DEVELOPMENT OF HINGES, LATCHES AND MECHANISMS OF GREAT RELIABILITY
- POSSIBLE FLIGHT CONTROLLER IMPLEMENTATION

TECHNOLOGY DEVELOPMENT NEEDS. I

STRUCTURES

- IMPROVED ACCURACY OF EIGENVALUE ANALYZERS AT HIGHER MODES
- IMPROVED CHARACTERIZATION OF MATERIAL PROPERTIES ESPECIALLY DAMPING
- DESIGN METHODOLOGY TO MINIMIZE STRUCTURAL RESPONSE (OPTIMIZATION)
- DESIGN METHODOLOGY TO APPLY DAMPING TO SPECIFIED MODES
- DAMPING METHODOLOGY FOR HIGHLY DAMPED STRUCTURES WITH REAL MATERIALS
- DEVELOPMENT OF STRUCTURAL SYSTEM IDENTIFICATION TOOLS OF HIGH ORDER

DISTURBANCES

- IMPROVED CHARACTERIZATION OF NATURAL DISTURBANCES
- IDENTIFICATION OF ON-BOARD SOURCES AND THEIR REDUCTION AT SOURCE

TECHNOLOGY DEVELOPMENT NEEDS: II

- DYNAMICS: RIGID AND FLEXIBLE
  - IMPROVEMENT OF MODELLING APPROACHES FOR MULTIPLE RIGID BODIES
  - DEVELOPMENT OF CONSISTENT APPROACHES FOR MULTIPLE RIGID/FLEX BODIES
  - IMPROVEMENT OF MODELLING NON-LINEAR DYNAMICS AND STRUCTURES
  - CONSISTENT AND RELIABLE TRUNCATION APPROACHES FOR LARGE ORDER SYSTEMS
    - CONTROL DESIGN AND VERIFICATION
    - FLIGHT CODE IMPLEMENTATION
- VERIFICATION METHODOLOGY
  - IF, STRUCTURES CAN BE TESTED IN 1G - NEED NEW GENERATION OF FACILITIES
  - IF NOT: MUST DEVELOP HIGH-FIDELITY SIMULATION TOOLS
    - COUPLED STRUCTURES, DYNAMICS AND CONTROL
    - CONTROL TO INCLUDE FIGURE, VIBRATION AND ATTITUDE SYSTEMS
    - SIMULATIONS MUST INTERFACE WITH OPTICS OR RADAR CODES

TECHNOLOGY DEVELOPMENT NEEDS: III

- UNFURLING AND DEPLOYMENT - DYNAMICS AND CONTROL
  - RELIABLE DYNAMIC MODELLING TOOLS TO PREDICT DEPLOYMENT PROCESSES
  - DEVELOPMENT OF CONTROL APPROACHES SUITABLE FOR DEPLOYMENT/SPACE CONSTRUCTION
  - DEVELOPMENT OF HINGES, LATCHES AND MECHANISMS WITH HIGH RELIABILITY
  - DESIGNS OF SYSTEMS FOR GRACEFUL DEGRADATION (UNMANNED ALTITUDES)
  - FLIGHT CONTROLLER IMPLEMENTATION
- STRUCTURAL CONTROL TECHNOLOGY
  - DESIGN METHODOLOGY INSENSITIVE TO TRUNCATION AND PARAMETER VARIATIONS
  - INTERACTION WITH ATTITUDE AND FIGURE CONTROLLERS
  - SUBMICRON, MACRORADIAN HIGH BANDWIDTH SENSORS AND ACTUATORS THAT MAY DISTRIBUTED
  - DEVELOPMENT OF SYSTEM IDENTIFICATION METHODOLOGIES
  - DIRECT DIGITAL DESIGN AND SENSOR-ACTUATOR PLACEMENT
  - FAULT TOLERANT FLIGHT HARDWARE
  - FLIGHT COMPUTER IMPLEMENTATION

TECHNOLOGY DEVELOPMENT NEEDS: IV

- ATTITUDE CONTROL
  - IMPROVED DYNAMIC MODELS WITH HIGH ORDER APPENDAGE AND PAYLOAD FLEXIBILITY
  - REFERENCE SYSTEM TECHNOLOGY
    - LOW DRIFT
    - LOW SCALE FACTOR ERROR
    - LOW JITTER
  - WITH LARGE FLEXIBLE PAYLOADS
    - COORDINATE TRANSFER BETWEEN SENSORS AND PAYLOAD AXES CRITICAL WITH FLEXIBILITY
    - CMG'S AND RW'S MUST HAVE HIGH CONTROL AUTHORITY BUT LOW NOISE
    - DISTRIBUTED SENSING/ACTUATION MUST BE CONSIDERED
    - CONTROLLED JET PROFILES ARE DESIRABLE
    - MINIMUM SETTLING TIME APPROACHES FOR RETARGETING - PROFILED/CONTROLLED SLEW
    - FAULT TOLERANT FLIGHT HARDWARE
    - FLIGHT COMPUTER IMPLEMENTATION

THRUSTER CONTROL FOR LSS ATTITUDE MANEUVERS

- HIGH TORQUE REQUIREMENTS - EXCEEDS EXISTING CMG'S AND RW'S
  - LARGER CMG'S HAVE LARGER NOISE PROBLEMS
  - MANY DISTRIBUTED SMALL CMG'S - WEIGHT PENALTY
- THRUSTERS MAY BE MOST ADVANTAGEOUS
  - LIMITED NUMBER OF MANEUVERS IN SYSTEM OPERATIONAL LIFETIME
- PROBLEMS WITH THRUSTERS
  - DISCONTINUITIES IN CONTROL PROFILES - EXCITATION
  - EXPENDABLE FUELS ALTER PLANT DESCRIPTION
  - SOME RW/CMG ASSIST AT END OF MANUEVER STILL DESIRABLE
- WORK BY PROF. VAN DER WELDE OF MIT DIRECTLY APPLICABLE
  - NEEDS REFINEMENT AND EXPERIMENTAL VERIFICATION

TECHNOLOGY DEVELOPMENT NEEDS: V

PACE TRANSPORTATION SYSTEM ISSUES

- FLEXIBLE ATTACHED SATELLITE EFFECTS ON SHUTTLE AUTOPILOT: STABILITY, FUEL CONSUMPTION
- SHUTTLE EFFECTS ON FLEXIBLE PAYLOAD: OVERSTRESS WHEN JETS COME ON
- TRANSPORT TO HIGHER ORBITS
  - UNFURLED CONFIGURATION: LOW "G" TOLERANCE, SLOW THROUGH VAN ALLENS
  - STOWED CONFIGURATION: UNMANNED DEPLOYMENT AT HIGH ORBITS

- TECHNOLOGY VERIFICATION IN SPACE
  - PROBABLY MOST IMPORTANT SINCE THE PERFORMANCE MANY SYSTEMS ARE NOT VERIFIABLE IN THE PRESENCE OF GRAVITY AND ATMPOSPHERE

SUMMARY

- OPPORTUNITIES WITH STS LEAD TO LARGE PAYLOADS
- GEOMETRICAL TOLERANCES TIGHTEN WITH INCREASED SIZE
- FLEXIBILITY INCREASES WITH SIZE
- DISTURBANCES INCREASE WITH SIZE
- OPEN-LOOP PERFORMANCE GREATLY INADEQUATE
- CONTROL NECESSARY TO ACHIEVE AND MAINTAIN PERFORMANCE
- NEW CONTROL THEORIES, SENSORS, ACTUATORS MUST BE DEVELOPED
- UNPRECEDENTED INTERACTION BETWEEN VARIOUS CONTROLLERS
- MUCH WORK NECESSARY IN UNDERSTANDING INTEGRATION ISSUES

Power Systems Integration

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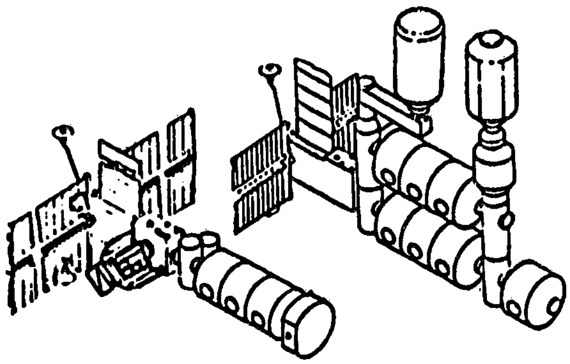
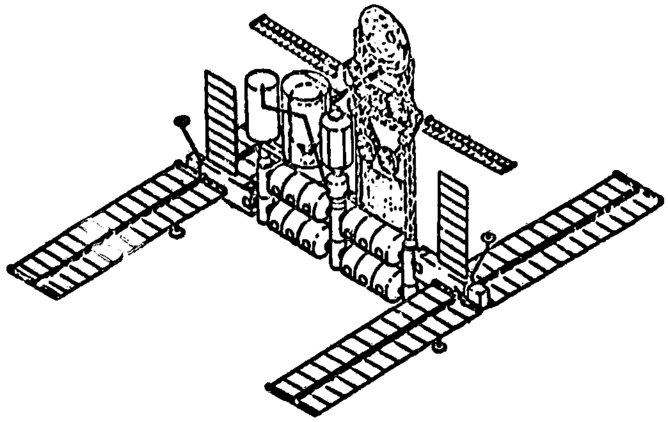
George C. Marshall Space Flight Center
Marshall Space Flight Center, AL 35812

POWER SYSTEM INTERACTIONS

- ATTITUDE CONTROL/DYNAMICS
- ENERGY STORAGE/HEAT REJECTION
- SHADOWING
- SOLAR ARRAY REDUCTION OF THERMAL RADIATOR VIEW FACTOR
- VOLTAGE/POWER LEVELS VS. DISTRIBUTION DISTANCES
- PLASMA INTERACTIONS
- LAUNCH PACKAGING CONSTRAINTS/ON-ORBIT DEPLOYMENT
- MAINTENANCE, LOWEST REPLACEABLE UNIT SIZE/WEIGHT
- RCS PLUME IMPINGEMENT



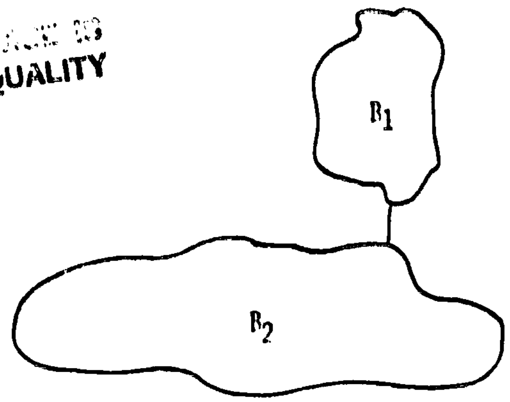


LARGE FLEXIBLE BODY CONTROL

SOLAR ARRAY IS A TYPICAL EXAMPLE

- **Problem**
  - In past with more rigid body vehicles (Saturn, Skylab, HEAO) control frequencies were much lower than vehicle vibrational frequencies; i.e., (f_c, f_o, f_1,....f_n) where $f_c << f_o$
  - For current program like Space Telescope (ST), vehicle vibrational frequencies are spaced with a sufficient control band gap between the fundamental frequency and the higher order modes (f_o, f_c, f_1 .... f_n) where $f_o << f_c << f_1$
  - For future large flexible space structures the overall vehicle configuration will vary (during construction and buildup) and will have low modal fundamentals (0.1 Hz and lower) and densely packed higher modes; i.e. (f_o, f_1, f_2 .... f_i, f_{i+1}.... f_n) where $f_i - f_{i+1} < \Delta$, " Δ " is small.
- **Solution**
  - Find a satisfactory way to imbed the control frequency within nested modes; i.e. (f_o, f_1. .... f_i, f_c, f_{i+1}... f_n) with on-orbit ability to vary position of f_c as configuration and vibrational modes are shifted.
- **Approaches to Solution**
  - Develop control theories, ground verify to extent practical, demonstrate in space
  - Disturbance isolation control (DIC) theory - MSFC in-house
  - Multi-level control (one of several multi-variable or multiple-loop control techniques) - Bendix
  - Other

DEVELOPING MODELING TOOLS FOR FLEXIBLE BODIES
(HONEYWELL: GENERAL PURPOSE DYNAMICS ANALYSIS PROGRAM)



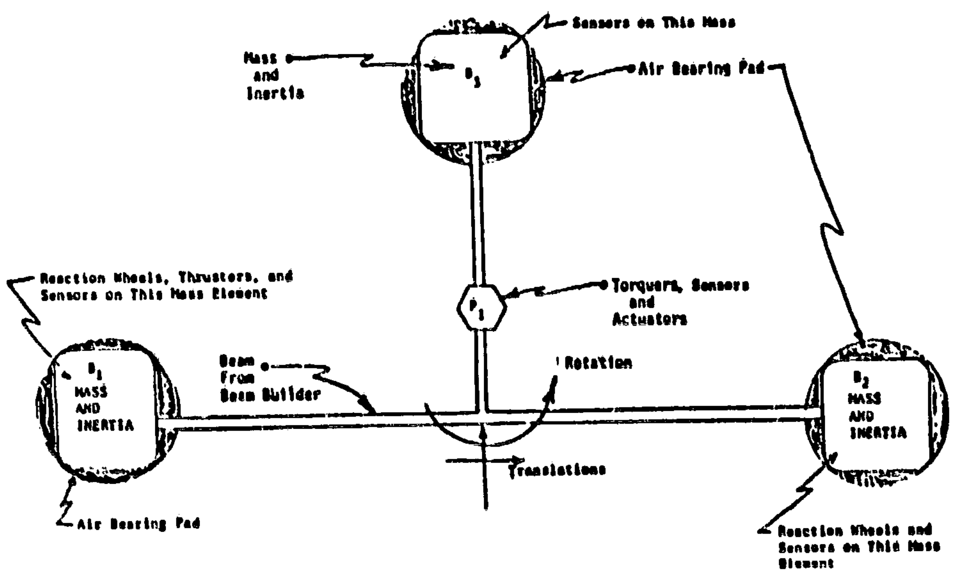
FIND VIBRATIONAL CHARACTERISTICS OF B_1

FIND VIBRATION CHARACTERISTICS OF B_2

THIS MODELING TOOL ALLOWS FOR CONNECTING OF B_1 AND B_2 IN SUCH A WAY TO ASCERTAIN VIBRATIONAL CHARACTERISTICS OF TOTAL CONFIGURATION. IT ALLOWS LARGE MOTIONS AT CONNECTIONS POINTS. ALSO PROVIDE LINK FOR CONTROL MARGIN. IT IS COMPUTATIONAL EFFICIENT RELATIVE TO AMOUNT OF VIBRATIONAL DATA NECESSARY TO OBTAIN TOTAL STRUCTURAL CHARACTERISTICS.

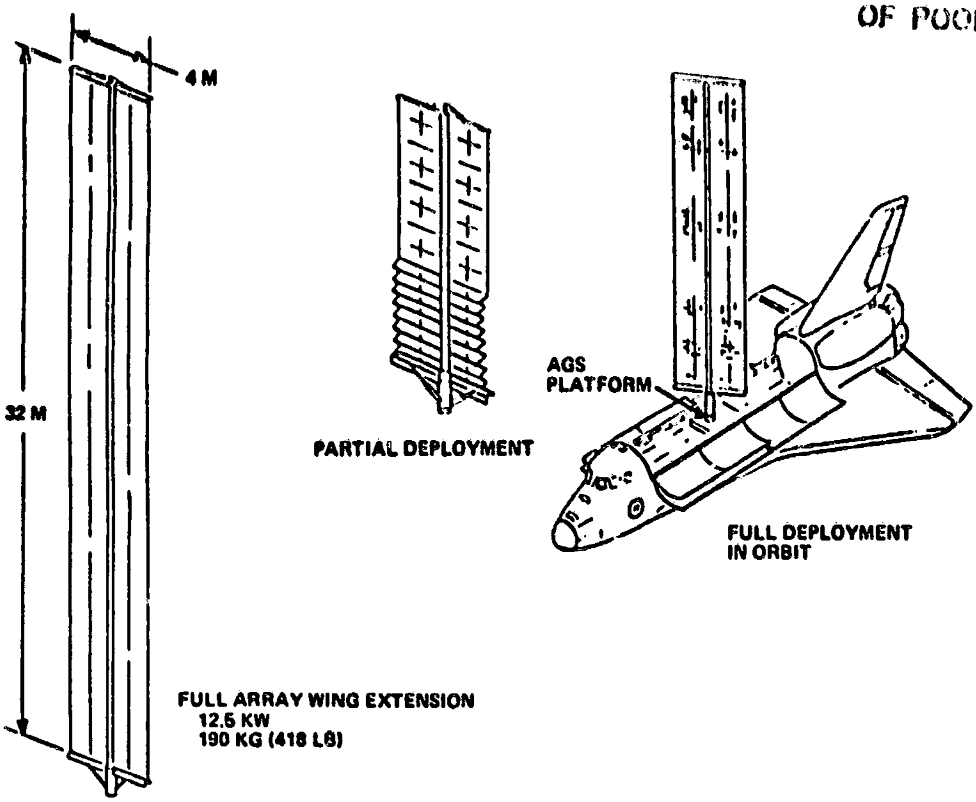
CONTROL TECHNIQUE VERIFICATION

OBJECTIVE: TO DESIGN AND COST A GROUND TEST FACILITY AND AN EXPERIMENT THAT COULD VERIFY CONTROL SYSTEM CONCEPTS THAT ARE BEING CONSIDERED FOR FUTURE LARGE SPACE STRUCTURE APPLICATIONS. THE EXPERIMENT SHOULD HAVE SUFFICIENT FIDELITY TO REASONABLY ENSURE SUCCESSFUL ON-ORBIT OPERATION.

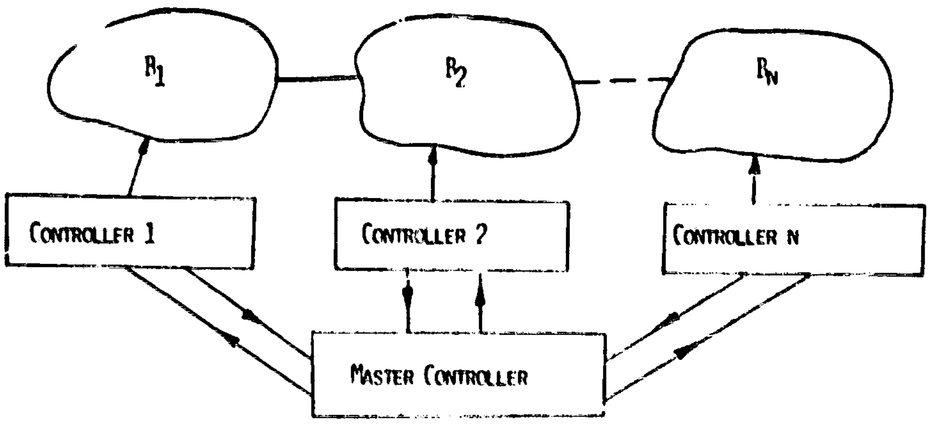


OUTLOOK: CONTROL SYNTHESIS TOOLS HAVE BEEN DEVELOPED TO CONTROL THE TOTAL CONFIGURATION IN THE FIGURE FROM EITHER BODY B_1 OR BODY B_2 OR FROM BOTH B_1 AND B_2. USING A RECENTLY DEVELOPED DISTURBANCE ISOLATION CONTROL (DIC) THEORY, ONE CAN ISOLATE BODY B_3 FROM THE DISTURBANCES AND THE DYNAMICS OF BOTH B_1 AND B_2 BY AN APPROPRIATE TORQUE COMMAND AT POINT P_1. A PROCUREMENT ACTION IS CURRENTLY UNDERWAY TO VERIFY THE CONTROL SYNTHESIS TOOLS AND THE DIC THEORY. THE CONTRACT SHOULD START BEFORE THE END OF FY-81.

2ND SEP SOLAR ARRAY FLIGHT EXPERIMENT
FLEXIBLE STRUCTURE DISTURBANCE ISOLATION CONTROL TEST

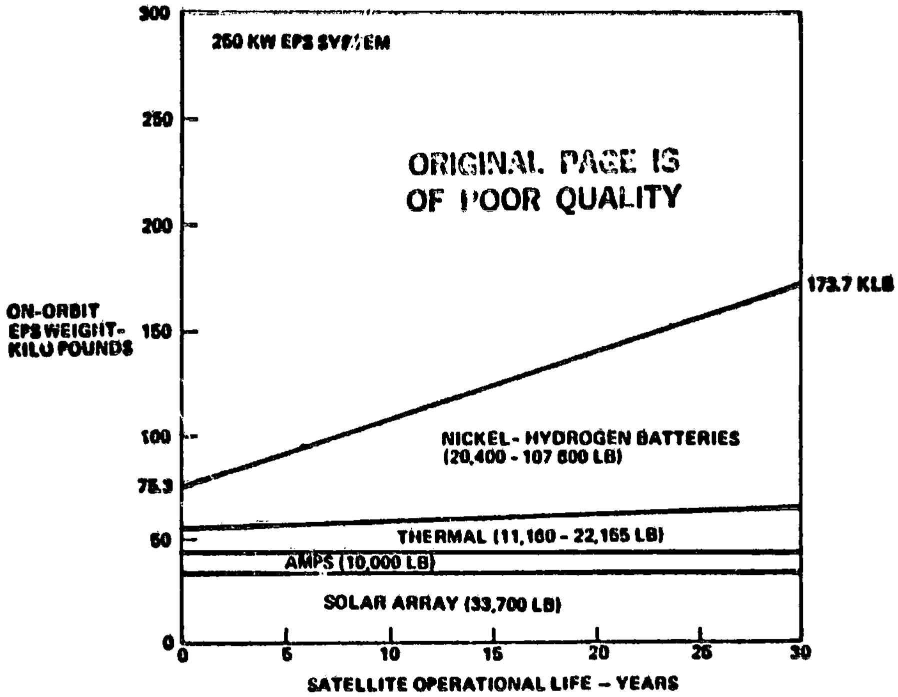


BENDIX MODULAR CONTROL
MULTILEVEL CONTROL

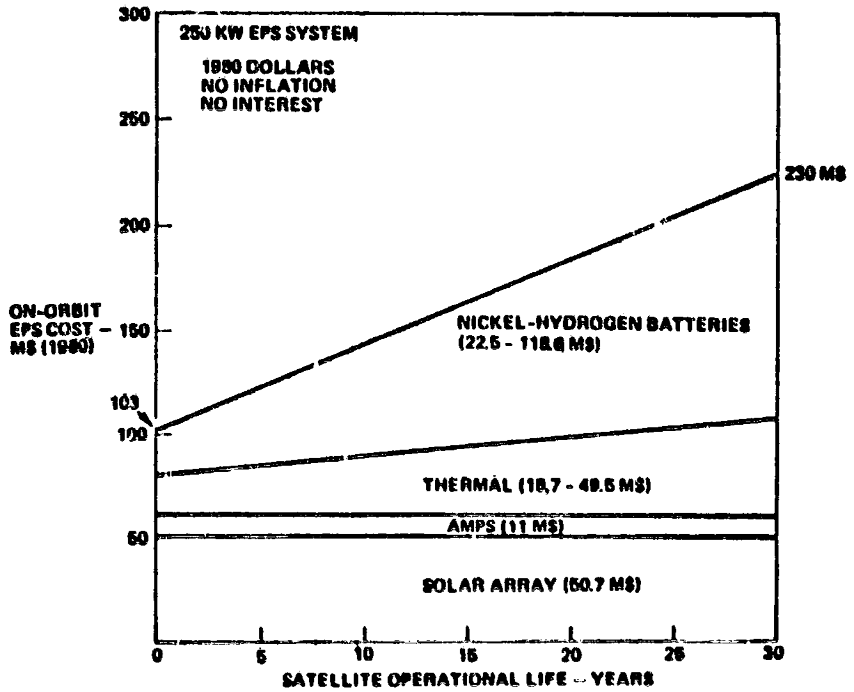


SPACECRAFT CONFIGURATION IS COMPOSED OF MANY LOOSELY COUPLED BODIES. OVERALL CONTROL SYSTEM IS DIVIDED INTO A HIERARCHY OF GOAL-SEEKING SUBSYSTEMS OR DECISION PROBLEMS.

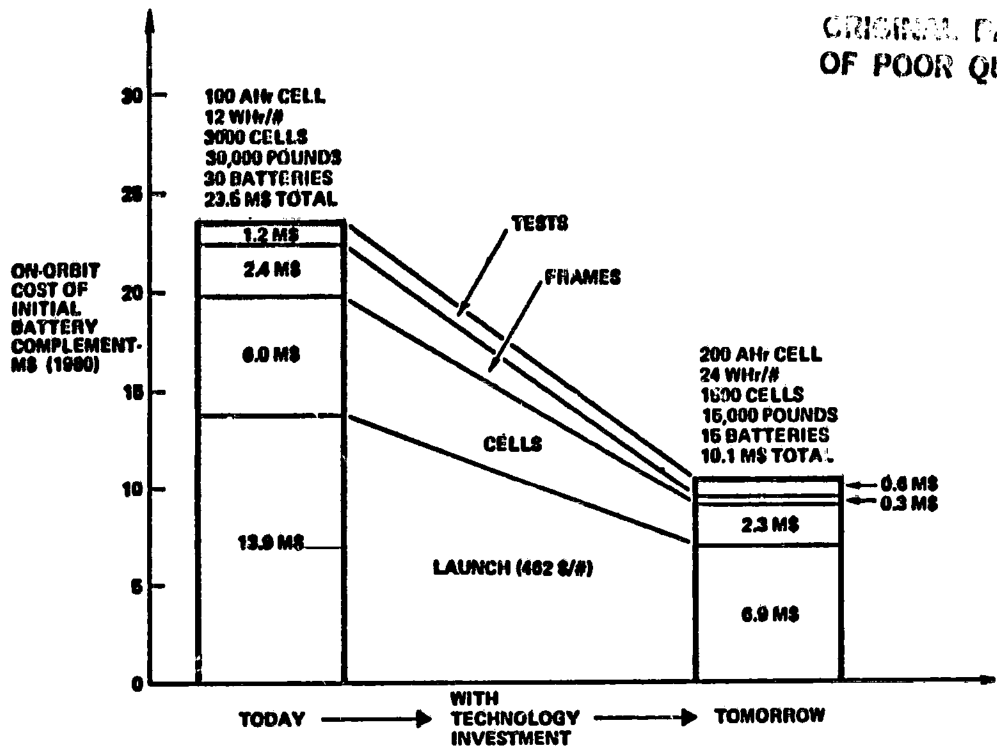
BATTERY DOMINATES LIFE-CYCLE WEIGHT



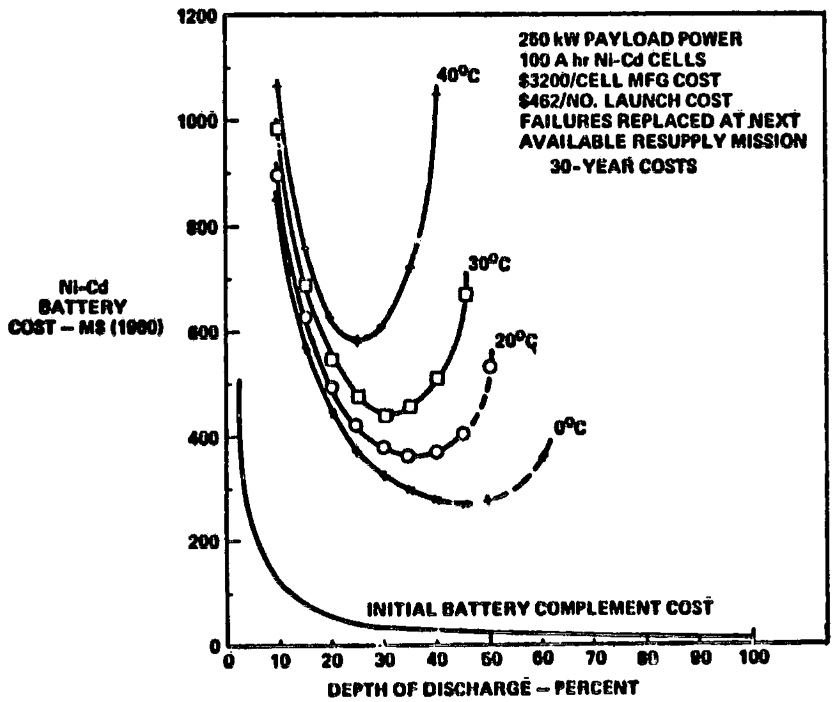
BATTERIES DOMINATE LIFE-CYCLE COST



LAUNCH COST KEEPS NI-Cd COST HIGH



40 - 50% DOD MINIMIZES BATTERY COST WITH 0°C TEMPERATURE



VARIATION OF RADIATOR SPECIFIC MASS WITH INLET TEMPERATURE

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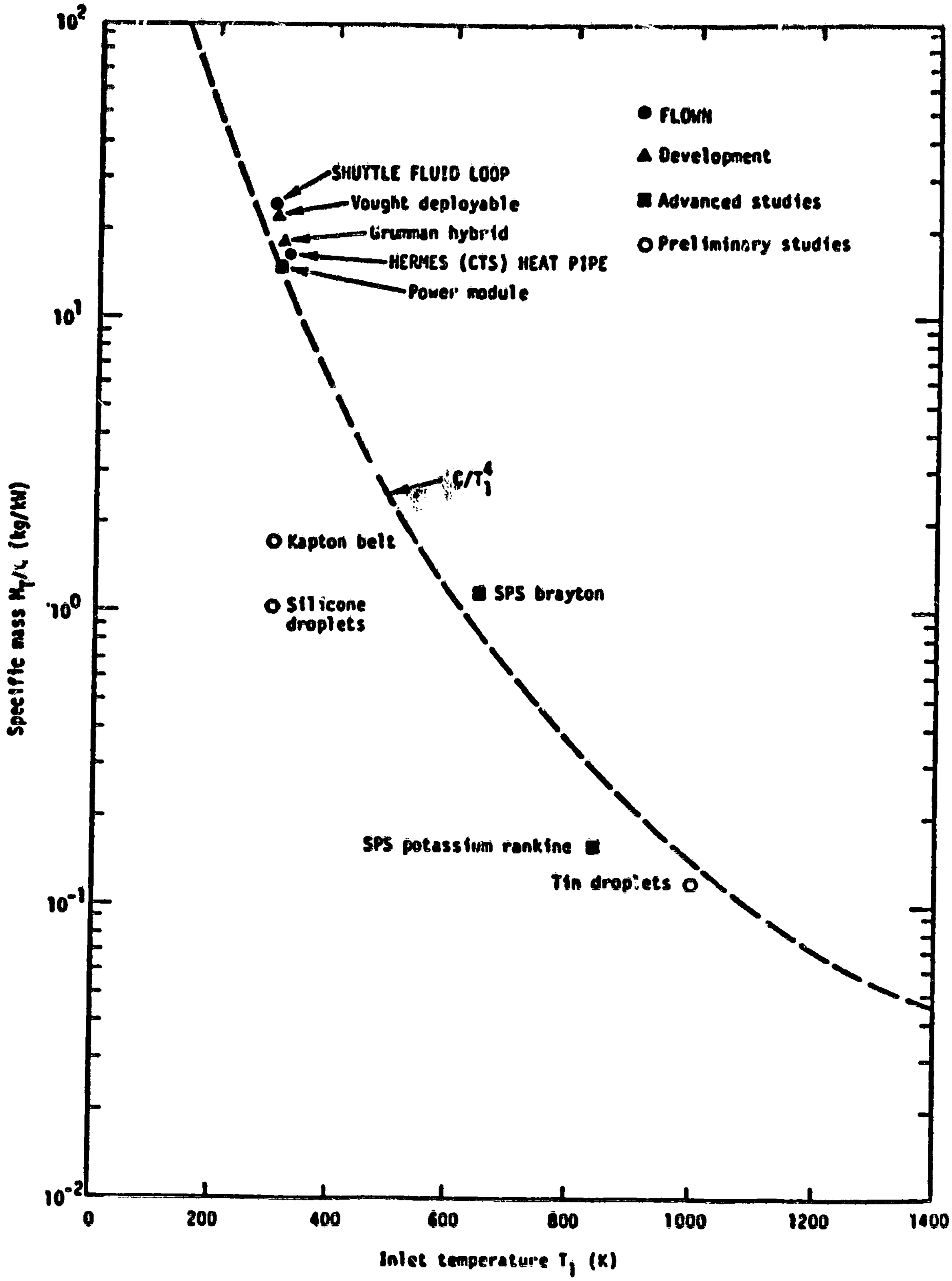
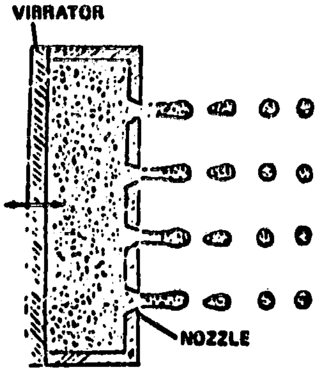
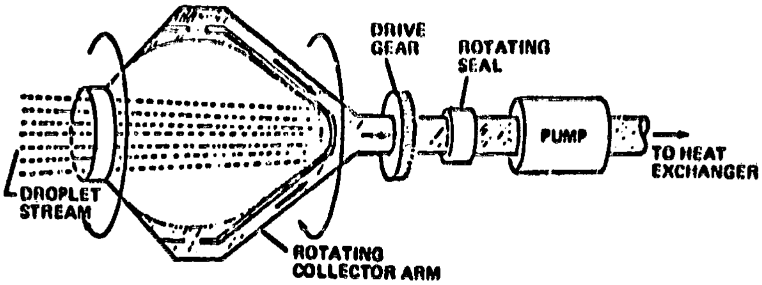


LIQUID METAL DROPLET GENERATOR



DROPLET COLLECTOR



LONG TERM TECHNOLOGY GOALS

- LOWER MANUFACTURING COST OF ENERGY STORAGE
- LIGHTER ENERGY STORAGE
- LONGER LIFE ENERGY STORAGE
- LOW COST SOLAR ARRAY
- LOW COST POWER PROCESSING FOR PAYLOADS
- LONGER LIFE COOLANT PUMPS
- LIGHTWEIGHT, LOW-COST RADIATOR
- LARGER CAPACITY BATTERY CELLS

OF POOR QUALITY

HOT SPOT ANALYSIS

NATURE OF PROBLEM: LARGE REVERSE VOLTAGE BIAS AND POWER DISSIPATION IN ONE OR MORE SOLAR CELLS ON AN ELECTRICAL MODULE.

CAUSES :
- o SHADOWED CELLS
- o BROKEN CELLS
- o CELL MISMATCH

AFFECTS :
- o CELL SHORTING REDUCING POWER IN A GIVEN CELL SUBMODULE
- o CELL SOLDER MELT CAUSING AN OPEN CIRCUIT IN A GIVEN CELL SUB MODULE

SOLUTIONS :
- o VARY SOLAR ARRAY CIRCUIT CONFIGURATION (NOT SUFFICIENT IN ALL CASES)
- o INCORPORATE CURRENT BYPASS DIODES IN SOLAR ARRAY DESIGN

COMPLETED TASKS

- DEVELOPED TEST CIRCUIT PROCEDURES, AND SOFTWARE FOR REVERSE BIAS TESTING OF SOLAR CELLS.
- SUBJECTED TEST SAMPLES TO NUMEROUS REVERSE-FORWARD BIASED CYCLES.
- DERIVED DIODE CHARACTERISTICS FOR REVERSE BIASED CELLS FROM FORWARD-REVERSE I-V CURVES FOR THREE TEMPERATURE CASES.
- CONSIDERED EFFECTS OF THE REVERSE I-V CURVE DUE TO THE CELL BEING 50% SHADOWED UNDER ILLUMINATION.
- ATTEMPTED CONTROLLED DEGRADATION OF THE REVERSE BIASED REGION.
- CONDUCTED ULTIMATE DESTRUCTIVE TESTING OF REVERSE BIAS CELLS TO DETERMINE FAILURE MODES, EXTENT OF DAMAGE, AND MAX POWER DISSIPATION
- DATA SUPPORTED THE ANALYSIS OF HOT SPOT EFFECTS DUE TO SHADOWING OF THE ARRAY.

POWER DISSIPATION 2 X 4 'S

SINGLE PANEL ANALYSIS RESULTS

NO. CELLS SHADED PER STRING	CROSS STRAPPING	% POWER LOSS	P_0 WATTS
½	NONE	37.4	66.4
1	NONE	78.3	19.2
1½	NONE	78.3	19.2
2½	NONE	93.9	3.1
3½	NONE	97.4	1.2
4½	NONE	97.4	.7
½	11	38.1	21.9
1	11	78.8	3.3
1½	11	78.8	3.0
2½	11	94.1	.4
3½	11	97.5	.1
4½	11		
½	22	35.9	18.6
1	22	77.2	3.1
1½	22	77.2	3.1
2½	22	92.9	.4
3½	22	96.5	.1
4½	22	98.2	.1

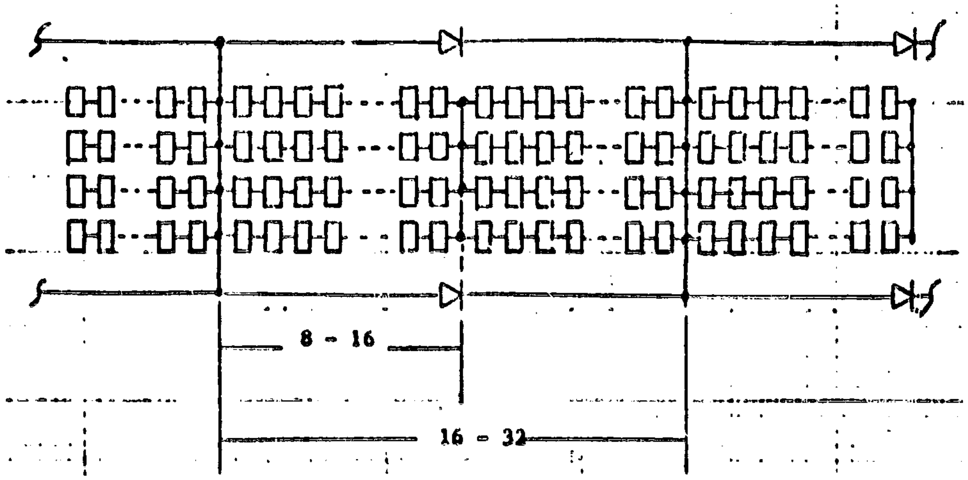
BY-PASS DIODE CONFIGURATION

SOLAR ARRAY PROTECTION OPTIMIZATION

2 x 4 CELL EXAMPLE

CROSS STRAPPING 8 - 16

DIODES 16 - 32



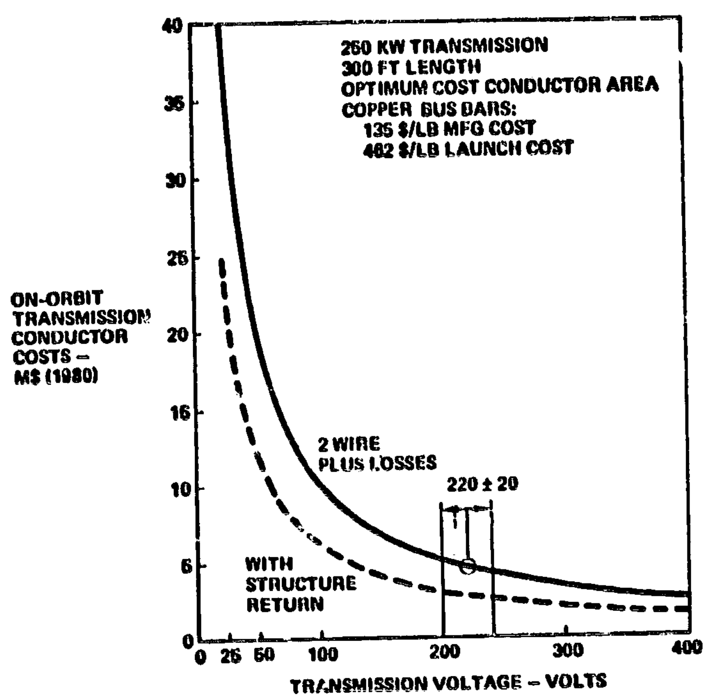
CONCLUSION FOR S/A CELL SHADOWING

ALL PANELS WITH MORE THAN 3½ CELLS/STRING SHADOWED ARE EFFECTIVELY TURNED OFF. NO HEAT/POWER DISSIPATION RESULTS.

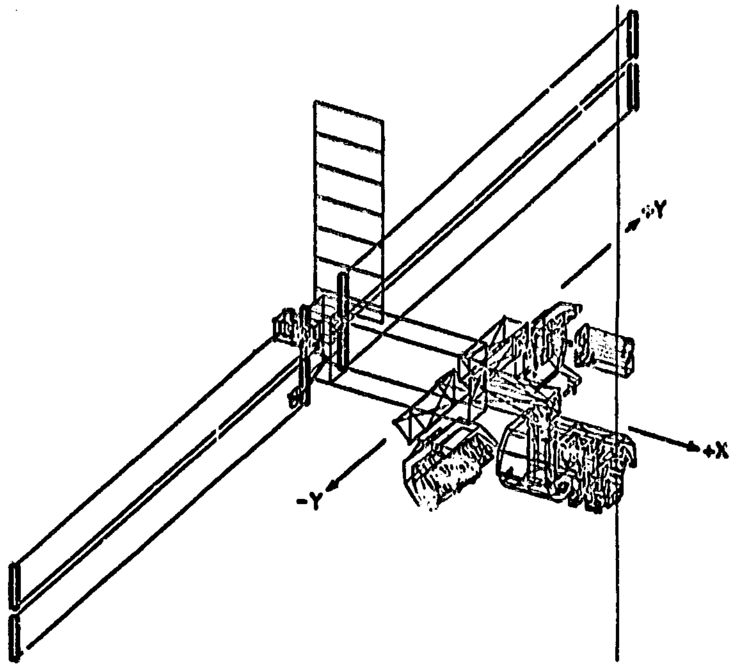
ALL PANELS WITH 2½ CELLS SHADOWED OR LESS RESULTS IN HIGH POWER DISSIPATION MODES. THE BEST CROSS-STRAPPING CONFIGURATION CAN MINIMIZE THESE POWERS FOR ONLY SPECIALIZED CASES.

BY PASS DIODES USED TO PROTECT AGAINST HIGH POWER DISSIPATION ALSO MAXIMIZES POWER AVAILABILITY AND SOLAR ARRAY RELIABILITY.

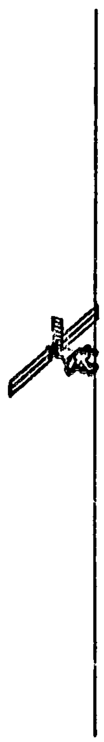
TRANSMISSION COSTS LOW ABOVE 100 VOLTS



STO STUDY
INITIAL CONCEPT



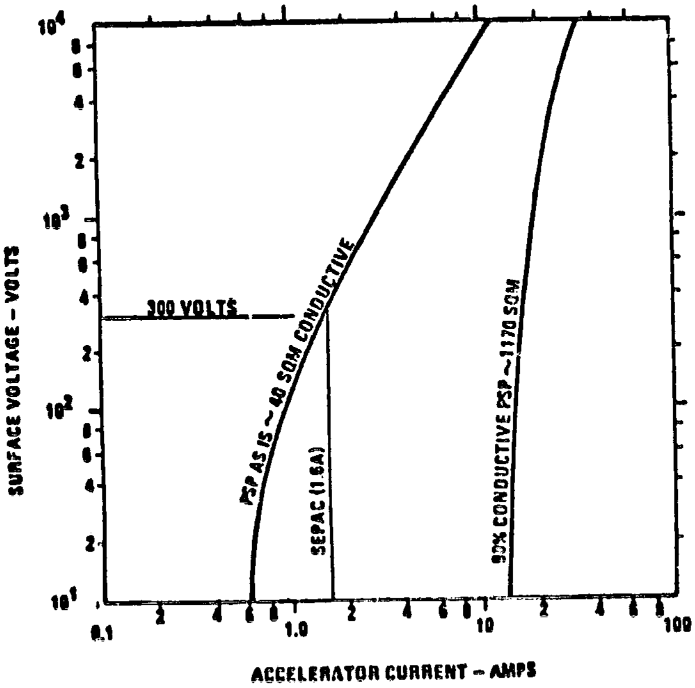
STO STUDY
INITIAL CONCEPT WITH
WISP ANTENNA DEPLOYED



ELECTROMAGNETIC COMPATIBILITY FOR POWER SYSTEM PLATFORM (PSP)

- TYPICAL SCIENCE PLATFORM CHARACTERISTICS
  - SEPS TECHNOLOGY SOLAR ARRAY
  - 12 KW FOR PAYLOADS
  - 235 N. MI, 57° INCLINATION
  - EXPERIMENTS MOUNTED ON PALLETS
  - PALLETS CHANGED OUT AT 6 MONTH INTERVALS
- BASELINE EMC SPECIFICATIONS
  - TAILORED MIL-STD-461B
  - MIL-B-5087
- SCIENCE REQUIREMENTS OUTSIDE OF ABOVE SPECIFICATIONS
  - ELECTROSTATIC DISCHARGE (ESD) CONTROL
  - E-FIELD EMISSIONS FROM 300 METER DIPOLE ABOVE 300 V/M
  - LOW FREQUENCY SUSCEPTIBILITY TO 1 HZ (300 HZ RCVR)
  - HIGH FREQUENCY SPECIFICATIONS TO 100 gHZ
  - H-FIELDS FROM SUPERMAG OF 40 GAUSS AT 10 FT

VOLTAGE BUILDUP FROM ELECTRON EMITTING PSP



BASIS

SPACECRAFT MUST ATTRACT CURRENT FROM BACKGROUND PLASMA EQUAL TO THE CURRENT BEING EMITTED

TRADITIONAL ANALYSIS

MORE CURRENT EMITTED THAN CAN BE COLLECTED

∴ SPACECRAFT VOLTAGE BUILDUP

LATER OBSERVATIONS

PLASMA RETURN CURRENT MUCH GREATER THAN EXPECTED

- GLOW DISCHARGE ?
- CURRENT ENHANCEMENT ALONG BEAM PATH?

DESIGN IMPLICATIONS

(97% OF PSP SURFACES ARE DIELECTRICS)

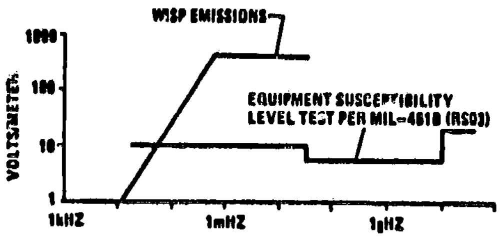
THIN CONDUCTIVE SURFACE COATINGS AND HIGH VOLTAGE SPECIFICATIONS

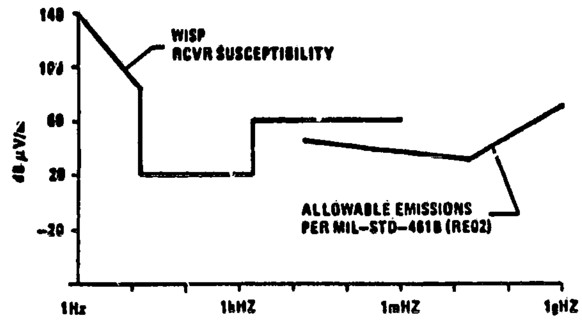
OR

THICKER COATINGS AND MODERATE VOLTAGE SPECIFICATION

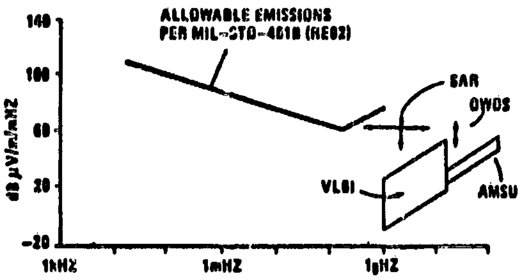
NEW EMC CONSIDERATIONS FOR PSP

LOW FREQUENCY LIMITS



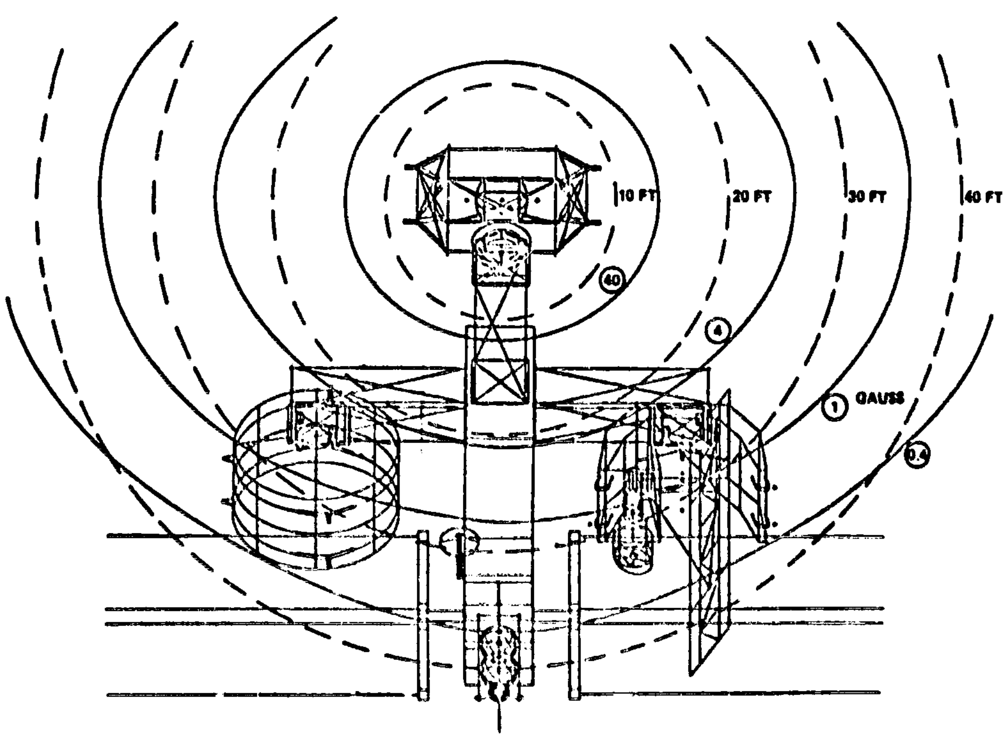


HIGH FREQUENCY LIMITS



WISP: WAVES IN SPACE PLASMA
VLBI: VERY LONG BASELINE INTERFEROMETRY
AMSU: ADVANCED MICROWAVE SOUNDING UINT
SAR: SYNTHETIC APERTURE RADAR
OWDS: OCEAN WAVE DIRECTIONAL SPECTROMETER

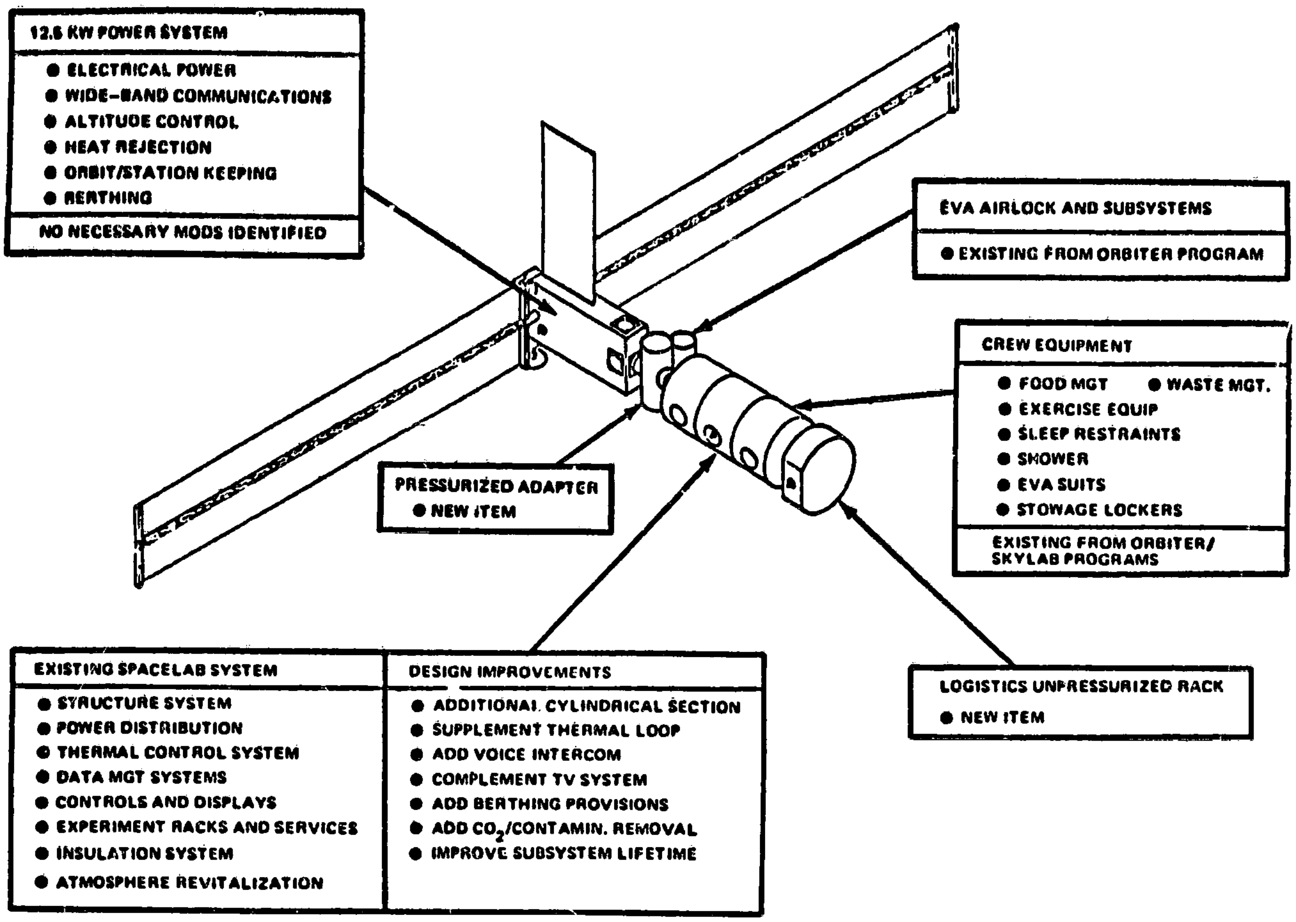
SUPER MAG FIELD PROPAGATION



ELECTROMAGNETIC COMPATIBILITY
FOR POWER SYSTEM PLATFORM

- Spacecraft ESD Design Considerations
  - Conductive coatings for solar array backside, thermal blankets, mirrors, antennas
  - Protective devices for ESD
  - Analytical tools needed to select which surfaces require ESD measures
  - Grounding, interconnection techniques
- Spacecraft EMI Design Considerations
  - ESD characterization and specification
  - Low frequency near field emission analysis and test
  - Common sync frequency for power converters
  - Reduced power source ripple and reduced allowable conducted black box emissions
  - Eliminate exposed current carrying conductors, e.g., , solar cell solder contacts
  - Increased equipment and cable shielding
  - Impact of Supermag is TBD
  - Review magnetic field specifications (emissions and susceptibility)
- Technology Requirements
  - Specifications: working group?
  - Analysis and ground test
  - Hardware: materials, protective coatings, interconnect methods
  - Flight tests:
    - a) Spacelab investigations with SEPAC and PICPAB
    - b) Solar array demonstration flight
      - Plasma interactions
      - Charging from accelerators
      - Thruster interactions
      - Electrical discharges at high voltage
      - Field measurements to guide specification preparation

SCIENCE AND APPLICATIONS MANNED SPACE PLATFORM



POWER SYSTEM PLATFORMS
DEGREE OF SYSTEM INTERACTION

TRADE ELEMENTS	STRUCTURE TYPE	CONTROL SUBSYS.	POWER SUBSYSTEM	THERMAL MGMT. SUBSYSTEM	DYNAMICS	TRANSPORTABILITY	ASSEMBLY	REBOOST	AUTOMATION OPTIONS	TOTAL SCORE	PERCENT
STRUCTURE TYPE	X	3	–	–	3	3	2	2	1	14	52%
CONTROL SUBSYSTEM		X	1	1	3	–	1	3	3	15	56%
POWER SUBSYSTEM			X	3	2	2	2	–	3	13	48%
THERMAL MGMT. S. S.				X	–	–	2	–	3	9	33%
DYNAMICS					X	2	3	3	–	16	59%
TRANSPORTABILITY						X	3	1	1	12	44%
ASSEMBLY							X	3	2	18	67%
REBOOST								X	–	12	44%
AUTOMATION OPTIONS									X	13	48%

3 HIGH INTERACTION
2 MODERATE
1 LOW
– NIL
X REQUIRED SUBSYS. DISCIPLINE TRADES

PSP COST ESTIMATE SUMMARY
MILLIONS OF 1980 DOLLARS

	% OF TOTAL
ELEC. POWER	26.4
THERMAL CONTROL	8.2
STRUCT. & MECH.	9.6
COMMUN. & DATA HANDL.	18.1
ATTITUDE CONTROL	7.5
REBOOST	2.6
FLIGHT SUPPORT EQUIPT.	1.4
LAUNCH & MISSION SUPPORT	2.5
GROUND SUPPORT EQUIPT.	5.6
TEST & VERIFICATION	3.2
SYSTEM ENG. & INT.	8.3
PROGRAM MGT.	6.6
TOTAL	100.0

N82 27379

"SYSTEM DESIGN AND INTEGRATION" PANEL WORKSHOP SUMMARY

C. Carl

Jet Propulsion Laboratory
4800 Oak Grove Drive
Pasadena, CA 91103

SYSTEM DESIGN AND INTEGRATION PANEL

- CRITERIA + PROCESS
- PRODUCTS
  - PRIORITIES IN PROPULSION TECHNOLOGY PROGRAM
  - KEY OBSERVATIONS ON SYSTEM DESIGN

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PRIORITY CRITERIA

- SUPPORTS/ENABLES MISSIONS THAT WILL SELL
- GENERIC BETTER THAN MISSION - SPECIFIC; DOD + NASA USE TOPS!
- NEAR TERM NEED ($\leq$ 10 YRS) MORE IMPORTANT THAN FAR TERM (10 - 20 YRS) NEED (CONSISTENT WITH PAYOFF)
- PRODUCES TANGIBLE PRODUCT/PAYOFF

PROCESS SIFT

ALL INTEGRATION PROBLEMS

↓

TECHNOLOGY PROBLEMS

↓

GENERIC PROPULSION TECHNOLOGY

↓

PRIORITY

ALL INTEGRATION PROBLEMS

	GENERIC PROPULSION	TECHNOLOGY
1. COMPLEX CONTROL OF THRUST IMPULSE VS. THROTTLEABLE PROFILE?	•	•
2. DISTRIBUTED CONTROL	-	•
3. MODULAR PROPULSION (DISTIBUTED)	-	-
4. MASS CONSTRAINED DESIGN TO GEO	-	-
5. DEPLOYMENT RISK OF COMPLEX LSS	-	•
6. ANTENNA FEED DESIGN/INTEGRATION	-	•
7. FLEX STRUCT/SMART CONTROL VS. STIFF STRUCT/DUMB CONTROL	-	•
8. IN-FLIGHT MECHANICAL ALIGNMENT	-	-
9. PROPULSION CONTAMINATION	•	•
10. SOLAR ARRAY/FOV/TC SHADOWING	-	-
11. LEO LONG LIFE DESIGN (SOLAR P/DRAG)	•	•

		GENERIC PROPULSION	TECHNOLOGY
12.	SHUTTLE EVA/NO EVA, RMS, SERVICE	-	-
13.	SHUTTLE RELATCH/SAFING (REFUL/STOW) FOR RETURN	-	-
14.	GROUND PERFORMANCE TESTING	-	•
15.	SHUTTLE PLUME (+) ENVIRONMENT	-	-
16.	STRUCTURE FOR CG = CP	-	-
17.	FAULT TOLERANCE OF ACTIVE CONTROL	-	•
18.	DEPLOYABLE SYSTEM UTILITIES (POWER, TC FWID, ETC.)	-	•
19.	ATTITUDE CONTROL DURING RELEASE/DEPLOYMENT	-	•
20.	OTV BEFORE DEPLOY?	-	•
21.	END OF LIFE: DISPOSAL	•	•
22.	BERTHING/DOCKING/MATE	-	-
23.	ASSEMBLY/SERVICING/MAINT.	-	•
24.	STANDARD HI VOLT POWER PROCESS.	-	•
25.	LSS DAMAGE TOLERANCE	-	•

PROPULSION TECHNOLOGY PRIORITIES

NEAR TERM NEED ($\leq$10 YRS)

1. PROPULSION CONTAMINATION
   - CHARACTERIZATION, EMC FOR ELECTRIC, ENVIRONMENTAL IMPACT. CHARGE CONTROL, CLEAN RESUPPLY.
2. LONG LIFE, HIGH DUTY CYCLE, HIGH I_{sp}, AUXILIARY THRUSTERS.
3. DEVELOP LINEAR, WIDE DYNAMIC RANGE THROTTEABLE THRUSTER.

FAR TERM NEED (10 - 20 YRS)

4. EP $> I_{sp} >$ CHEMICAL FOR GEO TRANSIT
5. MEGAWATT ELECTRIC PROPULSION

KEY OBSERVATIONS

- NEAR TERM MISSIONS REQUIRE LARGER SHUTTLE COMPATIBLE OTV TO GEO THAN PRESENTLY PLANNED.

- SYSTEM WORK, EMPHASIZING INTEGRATED PROP/STRUCTURE/THERMAL/CONTROLS, MUST BE DONE TO SET TECHNOLOGY REQUIREMENTS, EG

    - DEPLOYMENT AT LEO
    - DEVELOP INTEGRATED STRUCTURE CHARACTERIZATION
    - END OF LIFE DISPOSAL
    - MULTI PURPOSE PROPULSION