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SPACE OPERATIONS CENTER

SYSTEM ANALYSIS

FINAL REPORT, VOLUME IV

SOC SYSTEM ANALYSIS REPORT (BOOK 2 OF 2)

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THE BOEING COMPANY

HAMILTON STANDARD Division of UNITED TECHNOLOGIES

D180-26495-4

SPACE OPERATIONS CENTER
SYSTEM ANALYSIS

Conducted for the NASA Johnson Space Center

Under Contract NAS9-16151

FINAL REPORT

VOLUME IV

(Book 2 of 2)

SYSTEM ANALYSIS REPORT

D180-26495-4

July , 1981

Gordon R Woodcock

Approved by
Gordon R. Woodcock,
SOC Study Manager

BOEING AEROSPACE COMPANY

P.O. BOX 3999

Seattle, Washington 98124

N82-20204 #

FOREWORD

The Space Operations Center System Analysis study (Contract NAS9-16151) was initiated in June of 1980 and completed in May of 1981. This was the equivalent of a NASA Phase A study. A separately funded Technology Assessment and Advancement Plan study was conducted in parallel with the System Analysis Study.

These studies were managed by the Lyndon B. Johnson Space Center. The Contracting Officers Representative and Study Technical Manager is Sam Nassiff. This study was conducted by The Boeing Aerospace Company, Large Space Systems Group with the Hamilton Standard Division of United Technologies as subcontractor. The Boeing study manager is Gordon R. Woodcock. The Hamilton Standard study manager is Harlan Brose.

This final report includes 8 documents:

D180-26495-1	Vol. I	-	Executive Summary
D180-26495-2	Vol. II	-	Requirements (NASA CR-160944)
D180-26495-3	Vol. III	-	SOC System Definition Report
D180-26495-4	Vol. IV	-	SOC System Analysis Report (2 volumes)
D180-26495-5	Vol. V	-	Data Book (Limited Distribution)
D180-26495-6		-	(Reserved)
D180-26495-7		-	Space Operations Center Technology Identification Support Study, Final Report
D180-26495-8		-	Final Briefing Book

For convenience to the reader, a complete listing of all of the known Space Operations Center documentation is included in the Reference section of each document. This includes NASA, Boeing, and Rockwell documentation.

KEY TEAM MEMBERS

Subject	JSC-Management Team	Contractor Team
SOC Technical Manager	S. H. Nassiff	G. R. Woodcock
SOC Technology Study Manager	R. Kennedy	R. L. Olson
System Design	S. H. Nassiff	G. R. Woodcock
Electrical Power	L. Murgia	S. W. Silverman
ECLSS/EVA	D. Thompson	K. H. Miller
		H. Brose (Ham Std)
		G. Rannenberg (Ham Std)
		R. Cushman (Ham Std)
Communications & Tracking	R. Dietz	F. M. Kim
Structures/Dynamic Control	R. Wren	R. M. Gates
Stab/Control	J. Bigham	J. H. Mason
Propulsion/Propellant Storage	D. Kendrick	G. R. Woodcock
Subsystem Interface	L. Monford	M. A. Stowe
		G. R. Woodcock
Programmatics	R. Kennedy	G. R. Woodcock
Software/Processing	E. Dalke	L. E. Silva
		G. L. Hadley
Config. Design/Docking & Berthing	J. Jones	J. J. Olson
		K. H. Miller
Health Maintenance Facility	D. Nachtwey	K. H. Miller
Crew Habitat	M. Dalton	K. H. Miller
Operations	B. M. Wolfer	K. H. Miller
Space Construction Facility	L. Jenkins	K. H. Miller
Flight Support Facility	H. Patterson	G. R. Woodcock
		K. H. Miller
Crew Operations	M. Dalton	K. H. Miller
Orbital Altitude	F. Garcia	G. R. Woodcock
Operations Concepts/Requirements	B. Wolfer	K. H. Miller
Transportation	B. Wolfer	K. H. Miller
		G. R. Woodcock

KEY TEAM MEMBERS (Continued)

Subject	JSC-Management Team	Contractor Team
Technology	R. Kennedy	E. A. Gustan
		R. L. Olson
Cost	W. H. Whittington	G. R. Woodcock
		T. Mancuso

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LIST OF ACRONYMS AND ABBREVIATIONS

AAP	Airlock Adapter Plate
AC	Alternating Current
ADM	Adaptive Delta Modulation
AM	Airlock Module
APC	Adaptive Predictive Coders
APSM	Automated Power Systems Management
ACS	Attitude Control System
ARS	Air Revitalization System
ASE	Airborn Support Equipment
BIT	Built in Test
BITE	Built in Test Equipment
CAMS	Continuous Atmosphere Monitoring System
C&D	Controls and Displays
C&W	Caution and Warning
CCA	Communications Carrier Assembly
CCC	Contaminant Control Cartrige
CEI	Critical End Item
CER	Cost Estimating Relationships
CF	Construction Facility
CMG	Control Moment Gyro
CMD	Command
CMDS	Commands
CO_2	Carbon Dioxide
CPU	Computer Processor Units
CRT	Cathode Ray Tube
dB	Decibles
DC	Direct Current
DCM	Display and Control Module
DDT&E	Design, Development, Test, and Evaluation
DOD, DoD	Department of Defense
DT	Docking Tunnel
DM	Docking Module
DMS	Data Management System
DSCS	Defense Satellite Communications System

LIST OF ACRONYMS AND ABBREVIATIONS (Cont.)

ECLSS	Environmental Control/Life Support System
EDC	Electrochemical Depolarized CO_2 Concentrator
EEH	EMU Electrical Harness
EIRP	Effective Isotropic Radiated Power
EMI	Electromagnetic Interference
EMU	Extravehicular Mobility Unit
EPS	Electrical Power System
EVA	Extravehicular Activity
EVC	EVA Communications System
EVVA	EVA Visor Assembly
FM	Flow Meter
FMEA	Failure Mode and Effects Analysis
ftc	Foot candles
FSF	Flight Support Facility
FSS	Fluid Storage System
GN&C	Guidance, Navigation and Control
GEO	Geosynchronous Earth Orbit
GHZ	Gigahertz
GPS	Global Positioning System
GSE	Ground Support Equipment
GSTDN	Ground Satellite Tracking and Data Network
GFE	Government Furnished Equipment
GTV	Ground Test Vehicle
HLL	High Level Language
HLLV	Heavy Lift Launch Vehicle
HM	Habitat Module
HMF	Health Maintenance Facility
HPA	Handling and Positioning Aide
HUT	Hard Upper Torso
H_z	Hertz (cycles per second)
ICD	Interface Control Document
IDB	Insert Drink Bag
IOC	Initial Operating Capability
IR	Infrared

LIST OF ACRONYMS AND ABBREVIATIONS (Cont.)

IVA	Intravehicular Activity
JSC	Johnson Space Center
KBPS	Kilo Bits Per Second
KM, Km	Kilometers
KSC	Kennedy Space Center
lbm	Pounds Mass
LCD	Liquid Crystal Display
LCVG	Liquid Cooling and Ventilation Garment
LED	Light Emitting Diode
LEO	Low Earth Orbit
LiOH	Lithium Hydroxide
LM	Logistics Module
LPC	Linear Predictive Coders
LRU	Lowest Replaceable Unit
LSS	Life Support System
LTA	Lower Torso Assembly
LV	Launch Vehicle
lx	Lumens
MBA	Multibeam Antenna
mbps	Megabits per second
MHz	Megahertz
MMU	Manned Maneuvering Unit
MM-Wave	Millimeter wave
MOTV	Manned Orbit Transfer Vehicle
MRWS	Manned Remote Work Station
MSFN	Manned Space Flight Network
N/A	Not Applicable
NBS	National Bureau of Standards
NSA	National Security Agency
N	Newton
NiCd	Nickel Cadmium
NiH_2	Nickel Hydrogen
Nm, nm	Nautical miles
N/m^2	Newtons per meter squared

LIST OF ACRONYMS AND ABBREVIATIONS (Cont.)

OBS	Operational Bioinstrumentation System
OCS	Onboard Checkout System
OMS	Orbital Maneuvering System
OTV	Orbital Transfer Vehicle
PCM	Pulse Code Modulation
PCM	Parametric Cost Model
PEP	Power Extension Package
PIDA	Payload Installation and Deployment Apparatus
P/L	Payload
PLSS	Portable Life Support System
PM	Power Module
ppm	Parts per Million
PRS	Personnel Rescue System
PSID	Pounds per Square Inch Differential
RCS	Reaction Control System
REM	Reoentgen Equivalent Man
RF	Radio Frequency
RFI	Radio Frequency Interference
RMS	Remote Manipulator System
RPM	Revolutions Per Minute
SAF	Systems Assembly Facility
SAWD	Solid Amine Water Desorbed
scfm	Standard Cubic Feet per Minute
SCS	Stability and Control System
SCU	Service and Cooling Umbilical
SEPS	Solar Electric Propulsion System
SF	Storage Facility
SM	Service Module
SOC	Space Operations Center
SOP	Secondary Oxygen Pack
SSA	Space Suit Assembly
SSP	Space Station Prototype
SSTS	Space Shuttle Transportation System
STAR	Shuttle Turnaround Analysis Report

LIST OF ACRONYMS AND ABBREVIATIONS (Cont.)

STDN	Spaceflight Tracking and Data Network
STE	Standard Test Equipment
TBD	To Be Determined
TDRSS	Tracking and Data Relay Satellite System
TFU	Theoretical First Unit
TGA	Trace Gas Analyzer
TIMES	Thermoelectric Integrated Membrane Evaporation System
TLM	Telemetry
TM	Telemetry
TT	Turntable/Tilttable
TV	Television
UCD	Urine Collection Device
VCD	Vapor Compression Distallation
VDC	Volts Direct Current
WBS	Work Breakdown Structure
WMS	Waste Management System

8.0 ORBIT ALTITUDE SELECTION ANALYSIS

8.0 ORBIT ALTITUDE SELECTION ANALYSIS

8.1 ATMOSPHERE MODEL

Four atmosphere models were used in deriving the orbit decay data, see Figure 8-1. The nominal model is the U.S. Standard Atmosphere, 1976. The other three models were generated via the quick-look density model in Appendix B of NASA document SP-8021, Models of Earth's Atmosphere (90 to 2500 km), for a latitude of 0^{o}. This model calculates an exospheric temperature. A data table is then used to obtain the log of the atmospheric density for the desired altitude(s) for the calculated temperature. The NASA Neutral and Short Time Maximum Models use values suggested for space shuttle studies. The NASA Neutral Model is a high solar activity model, with a value of 230 for the mean 10.7 cm solar flux and a geomagnetic index (A_p) of 20.3. The assumed local time is 0900 hours. The Short Time Maximum Model uses a 10.7 cm solar flux of 230, a geomagnetic index of 400, and a local time of 1400 hours. These conditions would occur only for a time of 12 to 36 hours during an extremely large magnetic storm. The Minimum Model uses figures of 0900 for the local time, 73.3 for the 10.7 cm solar flux, and 10.9 for the geomagnetic index. The solar flux and geomagnetic index figures are the 97.7 percentile figures for June 1987 from the Marshall Space Flight Center predictions.

The "NASA neutral" is considered to be the worst long-term or continuous case applicable to the 90-day resupply cycle. The short-time maximum will be used to establish thrust levels needed for control authority in all situations.

8.2 ORBIT DECAY TIME

Altitude selection is based on maintaining a minimum orbit decay time of 90 days if no orbit maintenance occurs.

The velocity was calculated using $V = \sqrt{\frac{M}{r}} \times 1000$ where M is the gravitational

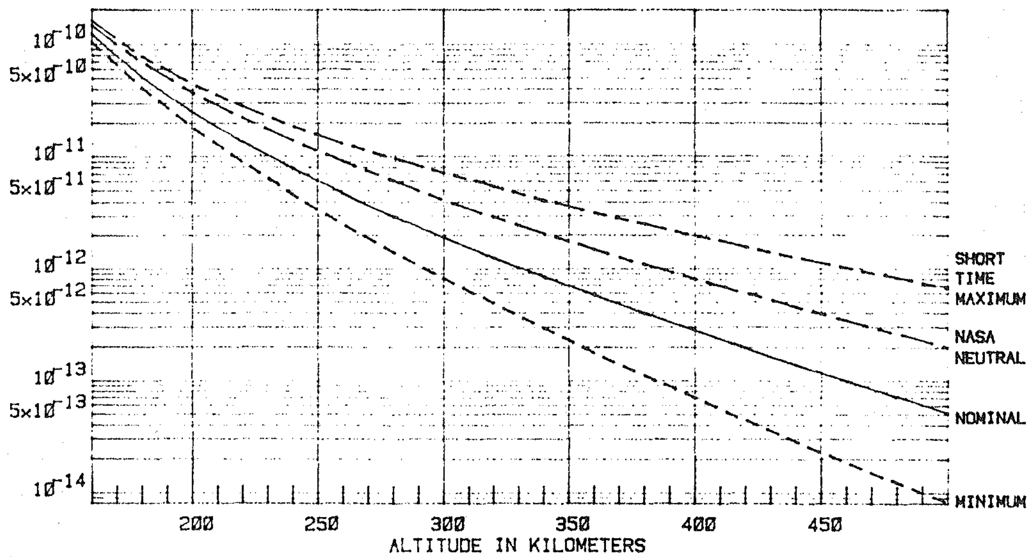


Figure 8-1. Atmosphere Density Models

coefficient, equal to 398,601.2 km^3/sec^2, r is the radius of the orbit, measured from the center of the Earth in kilometers, and the 1000 is a conversion factor to get the results in meters/sec.

Drag was calculated by $f = \frac{C_D A p V^2}{2}$, where C_D is the drag coefficient, A is the frontal area in meters, p is the atmospheric density in kg/m^3 and V is the velocity.

The decay rate was obtained from the formula $D = \frac{Mr\, C_D A p}{M} (8.64 \times 10^7)$, where C_D, A, p are as previously designated, M is the mass of the SOC in kg, and 8.64×10^7 is the conversion factor to get the results in km/day.

The decay time was calculated from $Q_A = Q_{A-1} + \left(\frac{.5}{D_{A-1}} + \frac{.5}{D_A}\right) H$ where D_x is the decay rate at altitude x and H is the difference of the two altitudes. The SOC characteristics used in this set of calculations are $C_D = 3.0$, $A = 300m^2$, $M = 100{,}000kg$, and $I_{sp} = 230$ sec.

Figure 8-2 shows the altitude requirement as a function of atmosphere model. The figure is based on a mean CdA of 1800 square meters and a SOC mass of 100 tonnes. Since the actual SOC mass will probably exceed this figure, the curve is slightly conservative.

Figure 8-3 shows atmospheric drag versus altitude for the four models and Figure 8-4 shows the orbit decay range. (Figure 8-2 was derived by numerical integration of Figure 8-4.)

8.3 PROPELLANT CONSUMPTION

Figure 8-5 shows the propellant consumption with the decay altitude limit superimposed. Propellant consumption was based on the use of monopropellant hydrazine at a specific impulse of 230 sec.

The propellant usage was equal to $\frac{86400f}{g\, I_{sp}}$ where f is the drag force in

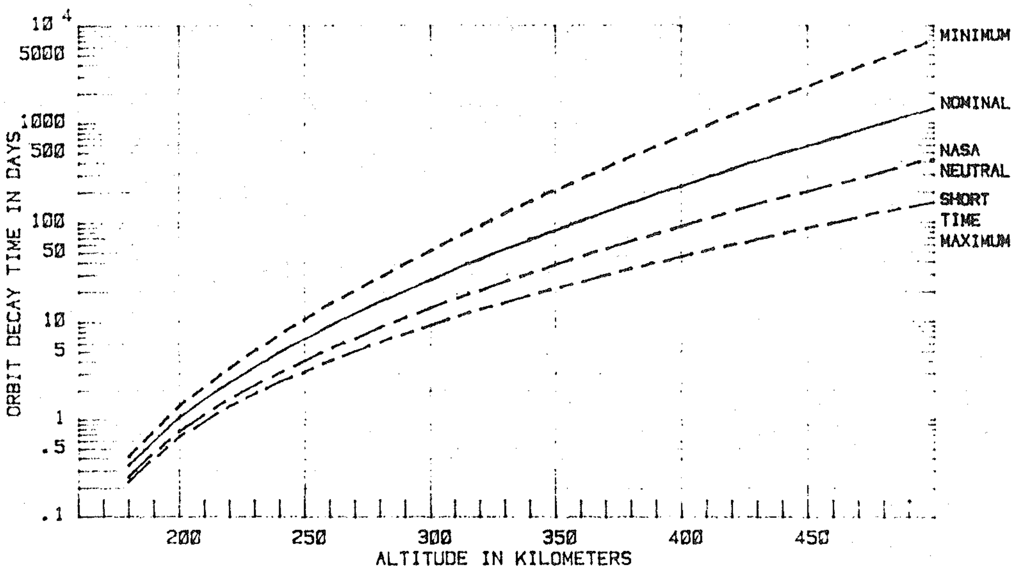


Figure 8-2. Orbit Decay Time vs. Altitude

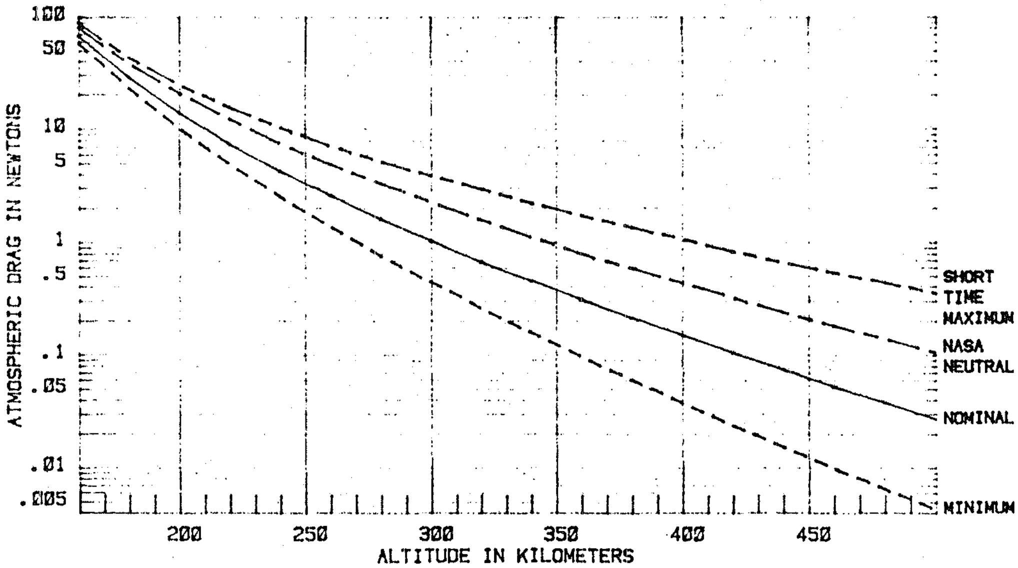


Figure 8-3. Atmospheric Drag vs. Altitude

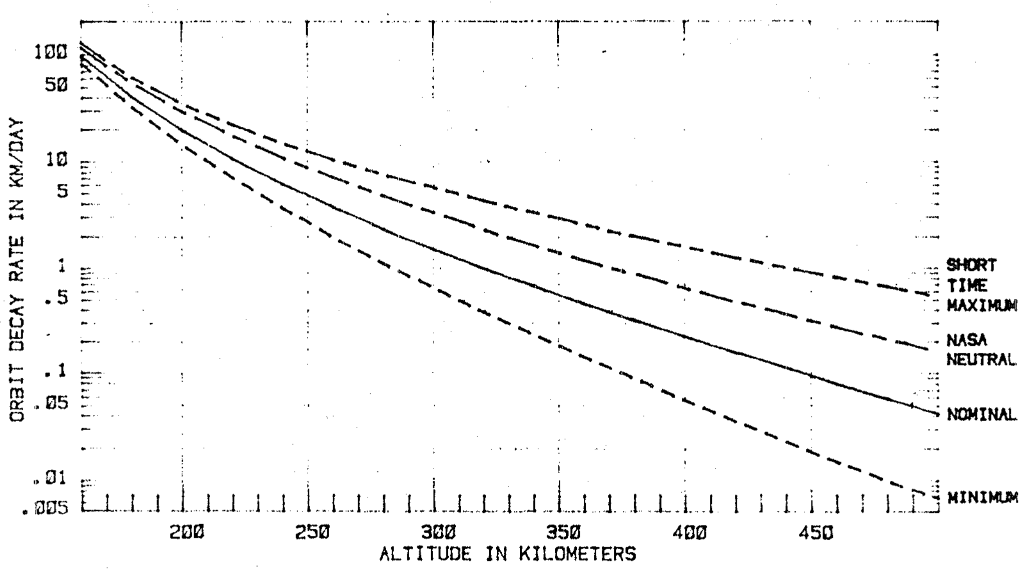


Figure 8-4. Orbit Decay Rate vs. Altitude

SOC-391

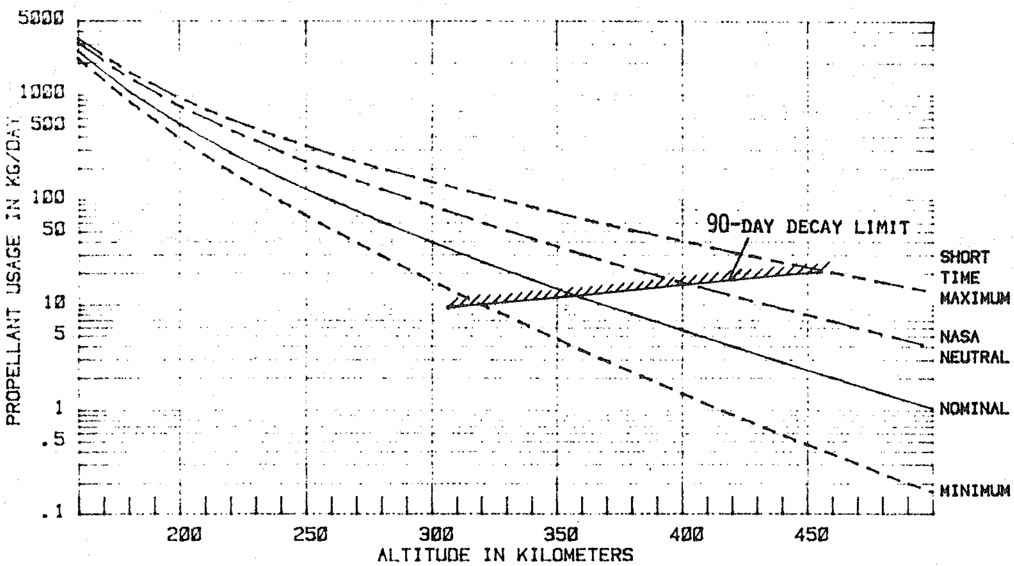


Figure 8-5. Propellant Usage vs. Altitude

newtons, g is the acceleration of gravity, 9.81 m/sec^2, I_{sp} is the specific impulse of the motors, and 86,400 is the conversion factor to get the results in kg/day.

A resupply requirement of 22 kg/day has been defined based on this curve, with 2kg/day added for atmosphere makeup (from hydrazine decomposition) and a 10% margin on the NASA neutral atmosphere point. The nominal resupply requirement will be somewhat less.

8.4 ORBIT ALTITUDE SELECTION

The selected orbit altitude is 370 km, as this is the altitude the shuttle can reach without OMS kits. This provides the maximum payload bay length capability. The mission model analyses show this to be extremely important. During periods of high solar activity, the altitude will be raised to 400 km. There are several operational options available to deal with the possibility of needing full shuttle payload bay when the SOC is above 370 km.

The resupply requirement has been set at 2200 kg for 100 days in sizing the logistics module. This requires only one ring of six 1.12 m (44") tanks on the logistics module.

9.0 ELECTRICAL POWER SYSTEM ANALYSIS

9.0 ELECTRICAL POWER SYSTEM ANALYSIS

9.1 INTRODUCTION

The bulk of the electrical power system analysis data is included in the SOC System Definition Report (Boeing - 19) under WBS 1.2.2.1.7 so it is not repeated herein. Section 9.2 gives the lower-level electrical load tables. Weight penalty calculations are found in the Data Book (Boeing - 21).

9.2 ELECTRICAL LOADS DATA

Table 9-1 gives the electrical load summary. The life support equipment loads were taken from Table B in WBS 1.2.1.1.13 in the SOC System Definition Report (Boeing-19). The other subsystem electrical loads are detailed in Table 19-2.

Table 9-1. Electrical Load Summary

LOAD	REFERENCE CONFIGURATION				INTERMITTENT POWER	
	SUNLIGHT		OCCULTED			
	AC	DC	AC	DC	AC	DC
• LIFE SUPPORT	7,119W	10,209W	6,565W	2,610W	3,850W	3,750W
• COMMUNICATIONS/TELEMETRY	–	9,370W	–	9,270W		
• DATA MANAGEMENT SYSTEM	–	1,000W	–	1,000W		
• PROPULSION SYSTEM (HEATERS)	–	200W	–	200W		
• THERMAL CONTROL SYSTEM	300W	2,000W	300W	2,000W		
• ATTITUDE CONTROL SYSTEM*	–	250W	–	250W		
• ELECTRICAL POWER SYSTEM						
LOADS	12,500W	4,500W	12,500W	4,500W		
BATTERY RECHARGE	–	29,900W	–	–		
TOTALS	19,919W	57,429W	19,365W	19,830W	3,850W	3,750W

* 1 KW STARTUP – 6 HR
(CAN BE SUPPORTED BY LOAD DIVERSITY)

SOC-1132

SYSTEM/COMPONENT	POWER, WATTS (EACH)	VOLTAGE		REG	DUTY CYCLE	LOAD EQUIPMENT	1/2 SOC		REFERENCE CONFIG		GROWTH CONFIG		REMARKS	LOCATION	DIST. FROM LOAD BUS	SM ALONE	SM/HM TENDED	REF. EMERGENCY	REF. DEGRADED
		DC	AC				SUNLIGHT W	OCCULT W	SUNLIGHT W	OCCULT W	SUNLIGHT W	OCCULT W							
COMMUNICATIONS/TELEMETRY																			
• KU BAND (TDRS)	50						50	50	50	50						–	–	–	50
• KU BAND (POWER AMPLIF.)	50						50	50	50	50						–	–	–	50
• S BAND STDN/TDRSS	200						200	200	200	200						–	–	200	200
• S BAND POWER AMPLIF.	250						250	250	250	250						–	–	250	250
• EVA SOC RCVR/XMTR	40						40	40	80	80						–	–	–	40
• EVA SOC RCVR/XMTR HEADSET	NEGL						–	–	–	–						–	–	–	–
• VOICE TERMINALS	60						60	60	120	120						–	60	60	120
• GPS RCVR/PROC.	100						100	100	100	100						100	100	–	100
• SURVEILLANCE RADAR	2,000						2,000	2,000	2,000	2,000						–	–	–	–
• CAUTION/WARNING SYSTEM	600						600	600	1,200	1,200						300	600	300	600
• CRT TERMINAL	1,000						1,000	1,000	1,000	1,000						–	1,000	100	500
• SIGNAL PROCESSING	500						500	500	1,000	1,000						250	500	250	500
• DIGITAL PROCESSING	500						500	500	1,000	1,000						250	500	250	500
• TV 15" RECEIVER	1,000						1,000	1,000	2,000	2,000						–	250	250	1,000
• TV CAMERA	60						60	60	120	120						–	60	60	120
• ANTENNA CONTROLS	200						200	100	200	100						100	100	100	100
									9,370	9,270						1,000	3,170	1,820	4,130
ELECTRICAL POWER SYSTEM																			
• LIGHTING			120		CONT	INCAND/FLOURESCENT	2,500	2,500	5,000	5,000						–	2,500	500	2,500
• DISTRIBUTION, SWITCHING		28			CONT		500	500	1,000	1,000						200	500	200	1,000
• COOKING (MICROWAVE)	250		120		50%		250	250	500	500						–	250	100	250
• BATTERY CHARGING - PORTABLE TOOLS		28			50%		500	500	1,000	1,000						–	250	–	500
• ENTERTAINMENT EQUIPMENT TV, STEREO, VIDEO PLAYERS/RECORDERS			120		CONT	ELECTRONICS	500	500	1,000	1,000						–	250	250	500
• APSM/DISPLAYS/CONTROLS		28			CONT	ELECTRONICS	1,000	1,000	2,000	2,000						200	2,000	250	2,000
• ALARMS		28			CONT	ELECTRONICS	250	250	500	500							250	250	500
• POWER TOOLS/MACHINERY			120		50%	MOTORS	2,500	2,500	5,000	5,000						–	2,500	–	2,500
• MEDICAL INSTRUMENTATION			120		25%	ELECTRONICS, MOTORS	500	500	1,000	1,000						–	500	500	500
• BATTERY RECHARGING - ENERGY STORAGE																–			
									17,000	17,000						400	8,000	2,000	10,250
DATA MANAGEMENT SYSTEM		28			CONT		1,000	1,000	1,000	1,000						200	500	500	1,000
PROPULSION SYSTEM																			
• HEATERS		28			50%	RESISTIVE HEATERS	100	100	200	200						100	100	200	200
THERMAL CONTROL SYSTEM																			
• HEATERS		28			50%	RESISTIVE HEATERS	1,000	1,000	2,000	2,000						500	1,000	–	2,000
• COOLING LOOPS			120		CONT	MOTORS	150	150	300	300						150	150	150	300
ATTITUDE CONTROL SYSTEM																			
• CONTROLS		28			CONT	ELECTRONICS, GYROS, ETC.	250	250	250	250			1 KW FOR 6 HR @ STARTUP ONLY			250	250	250	250

Table 9-2 SOC Subsystem Electrical Power Loads

10.0 ECLS AND EVA/IVA STUDIES

10.0 ECLS AND EVA/IVA STUDIES

10.1 VENTILATION CONCEPTS STUDIES

10.1.1 Forced Convection Level Required To Simulate Free Convection

Man's physiology is equipped to reject body heat and moisture without wind across his body, providing he is in the earth's one gravity atmosphere. The effect of gravity is to induce a quantity of convective heat transfer and air mass transfer, driven by the change of density occurring near the surface of the body. This convective force is not present at zero gravity, making necessary an artificially induced convective ventilation in order to simulate the free convection which is lost. This phenomenon has been evaluated and lived with in all previous spacecraft, and a fan induced average velocity of 25 feet/min has evolved as the accepted ventilation design value for spacecraft.

10.1.2 Use Of Ventilation Direction To "Simulate" Gravity

Man's physiology and geometry are configured to reduce the likelihood of eye damage or choking from loose objects, such as something dropped while eating or dropped from the hands. The eyes and mouth are "up", and things normally fall the other way, "down". In a zero-gravity environment this characteristic of getting things to fall down may be partially simulated by utilizing a ventilation system which has its cabin airflow descend from ceiling to floor. This concept has been selected for SOC. However, it is not practical that the 25 feet/min ventilation velocity of the previous paragraph be entirely made up by this downward flow. Every air jet or anemostat is in effect the primary nozzle of an ejector which induces many multiples of secondary flow into its flow pattern. This in turn results in a circulation pattern where flow is concentrated in a downward direction under the anemostat, proceeds down toward the floor, and then circles back toward the anemostat and around again to rejoin the downward flow. The net flow is downward, but locally there is increased velocity in the down direction under the outlets, and in an up direction between the outlets. For SOC, the primary flow of the vent supply anemostats is sized by the flow required to pass through the heat rejection heat exchangers in order to maintain cabin temperature, as discussed in the next paragraph. If a downward velocity of 25 ft/min were incorporated, the power consumption of the ventilation system would be about 3 to 4 times the selected baseline power requirement.

10.1.3 Flow Required for Cabin Heat Rejection

There are two choices in selecting the quantity of airflow which is to be cooled in the heat rejection heat exchangers. One way is to use 40°F coolant fluid through the heat exchanger and calculate the airflow required to transfer the heat load. The 40°F value is selected as the lowest feasible temperature to avoid freezing in the coolant water to freon heat exchanger. In this case, there would be condensation in the cooling heat exchangers, due to the coolant being below the desired cabin air dew point. The moisture thus removed would be collected at each heat exchanger and pumped to the water processing system. The relative humidity of the air leaving these heat exchangers would be excessive, since the air would be essentially saturated. Also, when this method of selecting the airflow to be cooled in the heat rejection heat exchangers is utilized, the resulting airflow is too low for use as the primary anemostat flow to provide the required 25 ft/min local velocity in the cabin. Additional cabin air circulating fans would be necessary to raise the cabin air velocity to the required level. The above method of selecting airflow for the heat rejection heat exchangers was not selected because of the complexity of removing moisture at each heat exchanger, and the complexity of additional circulating cabin air fans.

The other way in which cabin airflow through the heat rejection heat exchangers can be selected was used in the SOC baseline system of this report. In this case, the coolant fluid is controlled to 55°F entering the heat exchanger, rather than 40°F as described in the previous paragraph. This prevents condensation of moisture present in the cabin air by keeping metal temperatures over the dew point, and eliminates the problems of separating, collecting, and pumping water, at each heat exchanger, as well as eliminating the possibility of fog generation at the cabin supply anemostats. It also provides adequate primary flow in the cabin supply anemostats to efficiently provide the desired cabin air velocity.

Both of the above methods of selecting cabin air heat rejection airflow were evaluated as part of the Space Station Prototype (SSP) program, and the second method was selected. There is no significant difference in requirements for SOC which would indicate that the selected concept for SSP should not be the preferred concept for SOC.

A schematic of this ventilation concept, as selected for SOC, is shown on Figure 10-1. Note on the figure that the principle of "downward" net airflow of Section 10.1.2 is accomplished by air supply anemostats in the ceiling, supplemented by local adjustable anemostats in accordance with the detailed floor plan (such floor plan details will become evident later in the SOC development). These supply anemostats are fed from a common plenum over the ceiling, which in turn is fed by the sum of flow leaving the temperature control heat exchangers plus the flow leaving the air revitalization packs. The term "air revitalization pack" is used here to describe the equipment group which includes the functions of removal of humidity, CO_2, odor, and trace contaminants. Details of this equipment group are summarized in WBS 1.2.1.1.13.2 in Boeing-19. The term "ventilation and temperature control" pack is used to describe the equipment group which includes ventilating airflow fans, particulate filtration, heat rejection heat exchangers, and appropriate sound suppression baffling, as described in WBS 1.2.1.1.13.1 in Boeing-19.

10.1.4 "Clean Air Supply Ducts vs. "Dirty" Air Return Ducts"

Another choice in design of the ventilation system is the way in which odor and moisture sources are handled. A supply of "fresh" air could be specially ducted to the toilet area, for example, or

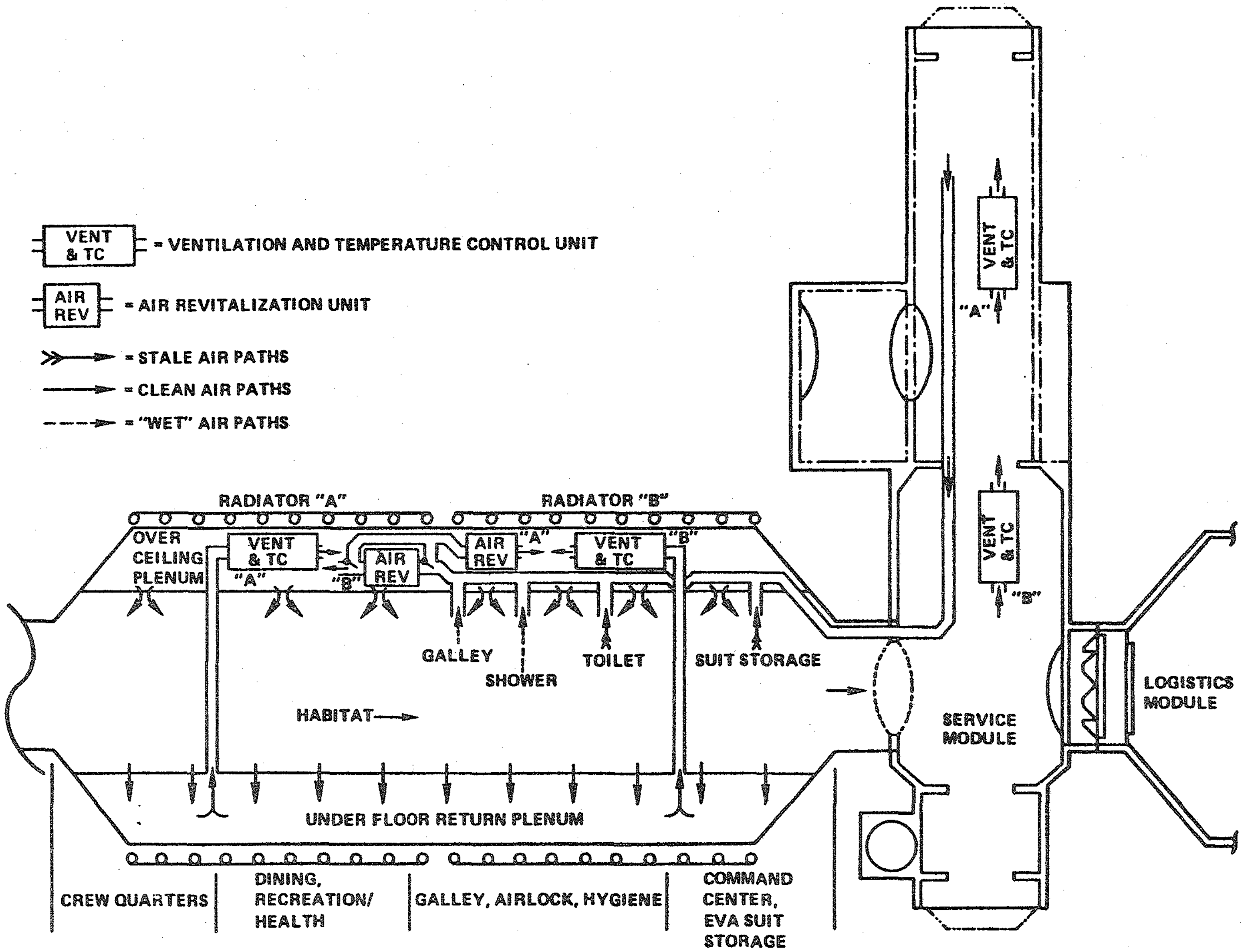


FIGURE 10-1
VENTILATION CONCEPT SCHEMATIC

a return duct carrying "dirty" air from the odor source could be utilized. The only reason for considering the "fresh" duct option is that a shorter, reduced volume duct could possibly be selected for baseline, in spite of its potentially larger duct volume. Most of the increased odor and moisture control duct volume will be in the overhead plenum on SOC, and it is presumed that SOC is a second generation spacecraft where such personal anemities as odor control should be considered. No trade-off on quality of living is possible, but the selected concept of controlling local odor and humidity sources appear to be common sense. In any case, the volume difference between these duct options is not a major matter, and furthermore an exact calculation of the difference in volume between the two concepts is impossible.

10.1.5 Ventilation During SOC Buildup

It is not currently envisioned that the SOC will be permanently inhabited until all baseline modules are in place. However, during the buildup sequence the crew may pressurize and enter the service module from the Shuttle. The service module will have its own power. Limited heat rejection will be provided by the battery and power conditioning equipment radiator. Humidity, and CO_2 control can be provided by using a snorkel line from the Shuttle air revitalization system. This capability (48 cfm) is a standard capability of the Shuttle for use with the Space Lab. Some thermal control is also provided by this air flow. Only one of the service module ventilation fans would be needed to provide minimum acceptable air mixing and air velocity for cooling.

After the first habitat module is in place the half SOC configuration will have an operational ECLS system. No air flow mixing between the Shuttle and the habitat is required. The Shuttle must remain attached to provide adequate safety in case the habitat must be evacuated.

10.1.6 Summary Description Of The Selected Ventilation Concepts

The selected ventilation concept was shown schematically on Figure 10-1. Local anemostat details will not become apparent until an actual design phase.

One major air recirculation path shown on Figure 10-1 consists of return grills in the floor leading to an under floor plenum, with two large vertical return ducts (approximately one square foot each) connecting the under floor plenum with the ventilation and temperature control packs in the overhead plenum. Each of these ducts supplies one "double" vent pack in the overhead plenum. These "double" packs each contain two independent temperature control systems as are described in detail in WBS 1.2.1.1.13.1 in Boeing-19. A half of one of these double ventilation and temperature control packs is also utilized independently at each end of the service module, as shown on Figure 10-1. In other words there are four ventilation half packs in each habitat module (in two double units), and two other half packs in each service module, making a total of six per half SOC or twelve per full SOC. This approach provides maximum commonality of hardware and improves reliability compared to the option where separate vent packs would be sized for the habitat module and the service module.

Another air circulation path shown on Figure 10-1 consists of the flow of dirty, odorous, or wet air taken from areas of contamination, and ducting to two air revitalization packs arranged in series in the overhead plenum. The suggested source areas shown on Figure 10-1 include the shower, toilet, suit storage area, and the remote end of the service module for contaminant control in the service module. Roughly 5 percent of the total habitat module supply airflow passes through the contaminant removal packs, so that it takes 76 minutes for them to pass an airflow equal to that of the entire cabin volume.

The total habitat module cabin supply airflow of 2440 CFM enters the cabin through anemostats placed several feet apart at intervals in the ceiling, and through adjustable anemostats placed as dictated by the final design. The velocity of these cabin supply airflow nozzles will generate an induced secondary airflow approximately 4.5 times the primary flow. The resultant total flow is adequate to provide an induced flow of 25 ft/min local velocity, of which the average up flow is 25 ft/min and the average down flow is 32 ft/min, resulting in a net down flow down of 7 ft/min. The induced flow performance of the primary airflow system selected for SOC is based on studies and development activities performed as part of the SSP program.

Figure 10-1 considers the baseline floor plan for the habitat module consisting of a single longitudinal floor. Other optional floor plans include "baloney slice" floors for at least a portion of the module. As far as the ventilation concept of this report is concerned, it is recommended that the basic concept resented be used regardless of the floor plan selected. Obviously the implementation of the ventilation system is easiest with a single longitudinal floor, and this is certainly one of the advantages of such a floor plan. If other factors should predominate, and a baloney slice floor plan is adopted, the ventilation duct system becomes more complex and difficult to visualize. The basic flow pattern from ceiling to floor should be preserved where feasible.

10.2 EMERGENCY PERFORMANCE LEVEL DEFINITIONS

A SOC requirement imposed by NASA in document NASA-6 directed a fail operational/fail safe design criteria. This criteria, with minor exception has been retained. This minor exception is that the "operational level" has been assumed a level which provide an acceptable performance for a 90 day period. In order to meet this safety requirement it is believed that no single failure of ECLS equipment shall force abandonment of a habitat module. This establishes the basic requirement for dual radiators, dual radiator transport loops, and dual atmosphere supply and processing. Each of these dual systems is not capable by itself to maintain the excellent environment referred to as the "operational" performance level, since the result would be an unnecessarily large vehicle penalty. Instead, it will take both of the dual systems operating together to produce the "operational" level. One of the dual systems operating will produce a "90 day acceptable" environment. By these steps of logic it is possible to design a system which is fail operational (acceptable)/fail safe and which imposes a minimum penalty to the vehicle. The performance level capability of the ECLS system is summarized on Table 10-1.

Referring to Table 10-1, note that the "operational" performance, which results without system failure and with the normal habitat module crew complement of 4, is what could be described as an excellent environment. This performance can be maintained in steady state for all mission activities of the 4 crew members. More than 4 crew members in that habitat module will not cause a noticeable reduction in the quality of the environment for relatively short duration activities such as meals or meetings. Neither will there be a noticeable reduction in environment quality for longer periods if the activity level is low, such as 8 men sleeping. However, more than 4 crew members in the habitat module continually, with full activity level, wash, shower, cooking, etc, can cause the performance level to approach the "90 day acceptable" level. The table shows that this same "90 day acceptable" performance level can be maintained with a 4 man complement after a worst single non-maintainable failure in that module. This capability enables the system to meet the fail operational criteria of not having to abandon the habitat module with a worst single failure.

If the failure such as a major fire or major breach of the cabin wall, has forced abandonment of a habitat module the entire crew will be in the remaining habitat. In this situation the reduced level of performance capability is still "90 day acceptable". As shown on Table 10-1, 8 man capability is provided for at least 300

TABLE 10-1

ECLS PERFORMANCE LEVEL REQUIREMENTS

Parameter	Units	"Operational"	"90 Acceptable"	"300 Hour Emergency"
CO_2 Partial Pressure	mmHg	3.8 max	7.6 max.	12 max
Temperature	°F	65-75	60-85	60-90
**Dew Point Temperature	°F	40-60	35-70	30-75
Ventilation	ft/min	15-40	10-100	5-200
Wash Water	lb/man day	40 min	20 min	0
***O_2 Partial Pressure	psia	2.6 or 3.1	2.4-3.8	2.3-3.9
Total Pressure	psia	10.0 or 14.7	10.0-14.7	10.0-14.7
Trace Contaminants	---	****24 hr. ind. st'd.	****8 hr. ind. st'd.	****8 hr. ind. st'd.
Maximun number of crew per habitat without failure in each habitat		4	8	12
Maximum number of crew per habitat with worst single non-maintainable failure in that habitat		NA*	4	8

*Acceptable level is adaquate to meet a "fail operational" reliability criteria.

**In no case shall relative humidities exceed the range of 25-75%.

***In no case shall the O_2 partial pressure exceed 26.9% or be below 2.3 psia.

****hr. ind. st'd. = hour industrial standard hours at the "300 hour emergency" level with a single worst non-maintainable failure after the crew has abandoned one habitat. It should be noted that a single habitat has a 12 man capability at the levels shown in the 300 hour emergency column without a time limit if no ECLS equipment has failed.

The design point used to size any particular ECLS subsystem is generally found in one of the columns and that subsystem will exceed the required performance of the other two columns. There also are subsystems which exceed the fail operational (acceptable)/fail safe design criteria because of redundance dictated by the requirement that no single non-maintainable failure shall cause evacuation of a habitat.

The performance levels listed on Table 10-1 were arrived at by establishing the lowest performance level that could be tolerated for the respective continuous 90 day and 300 hour day time periods.

The specific values are a result of years of study. Information from actual flight experience (Apollo, Skylab, Gemini, etc.) and space station study programs like the Space Station Prototype (SSP) have been used to define the values shown. These values also have been discussed and reviewed with NASA/JSC during the conduct of this SOC study.

10.3 CABIN PRESSURE ASSESSMENT

The baseline SOC cabin pressure for purposes of this study is 14.7 psia, but there are many factors which would favor the selection of a lower cabin pressure, as discussed in the following Section 10.3.1. However, selection of a lower cabin pressure has adverse effect on the size, weight, and power consumption of certain portions of the ECLS, as discussed in Section 10.3.2. An overall conclusion regarding the factors affecting cabin pressure selection is presented in Section 10.3.3.

10.3.1 Factors Influencing Selection Of Cabin Pressure

Hypoxia - One factor to be considered in selecting cabin pressure is avoidance of hypoxia, or reduced brain function, caused by low partial pressure of oxygen. Due to the fact that the majority of the earth's inhabitants live below 5,000 feet, the maximum continuous altitude at which body functions are measurably affected is not clear. A partial pressure of oxygen in the cabin corresponding to 4,000 feet or less would certainly be desirable, and is considered a requirement by NASA ("Medical Science Position on Space Cabin and Suit Atmospheres" Position paper by NASA JSC/SD, May 1980). Although an altitude as high as 8,000 feet equivalent oxygen level is considered to be an acceptable level for commercial aircraft pressurization.

Flammability - Another major factor to be considered in selecting cabin pressure is flammability of materials. The fire danger is related to the percent oxygen present in the cabin atmosphere. The normal sea level oxygen concentration is 21 percent, and certainly a concentration this low in the SOC cabin would be desirable from a flammability standpoint. Only one major material used in Shuttle, a silicon fiberglass line insulation, has failed to meet flammability tests at 35 percent O_2, and this material will be replaced in later Shuttle vehicles. The cabin pressure control tolerances of the current Shuttle result in a maximum normal oxygen concentration of 23.8 percent O_2. A caution and warning light is set on Shuttle to trip at the 25.9 percent level with a 26.9 percent O_2 absolute maximum level has been selected for SOC. These same levels are probably going to be inherited by SOC as the flammability requirement. The relation between flammability and cabin pressure is shown on Figure 10-2.

Eliminating Pre-breathe - The pre-breathe period required to prevent "bends" with the presently available 4.1 psi suit is approximately 4 hours pre-breathe with a 14.7 psia cabin, and approximately 2 hours pre-breathe with an 11 psia cabin. This lost time in pre-breathe can be eliminated by increasing suit

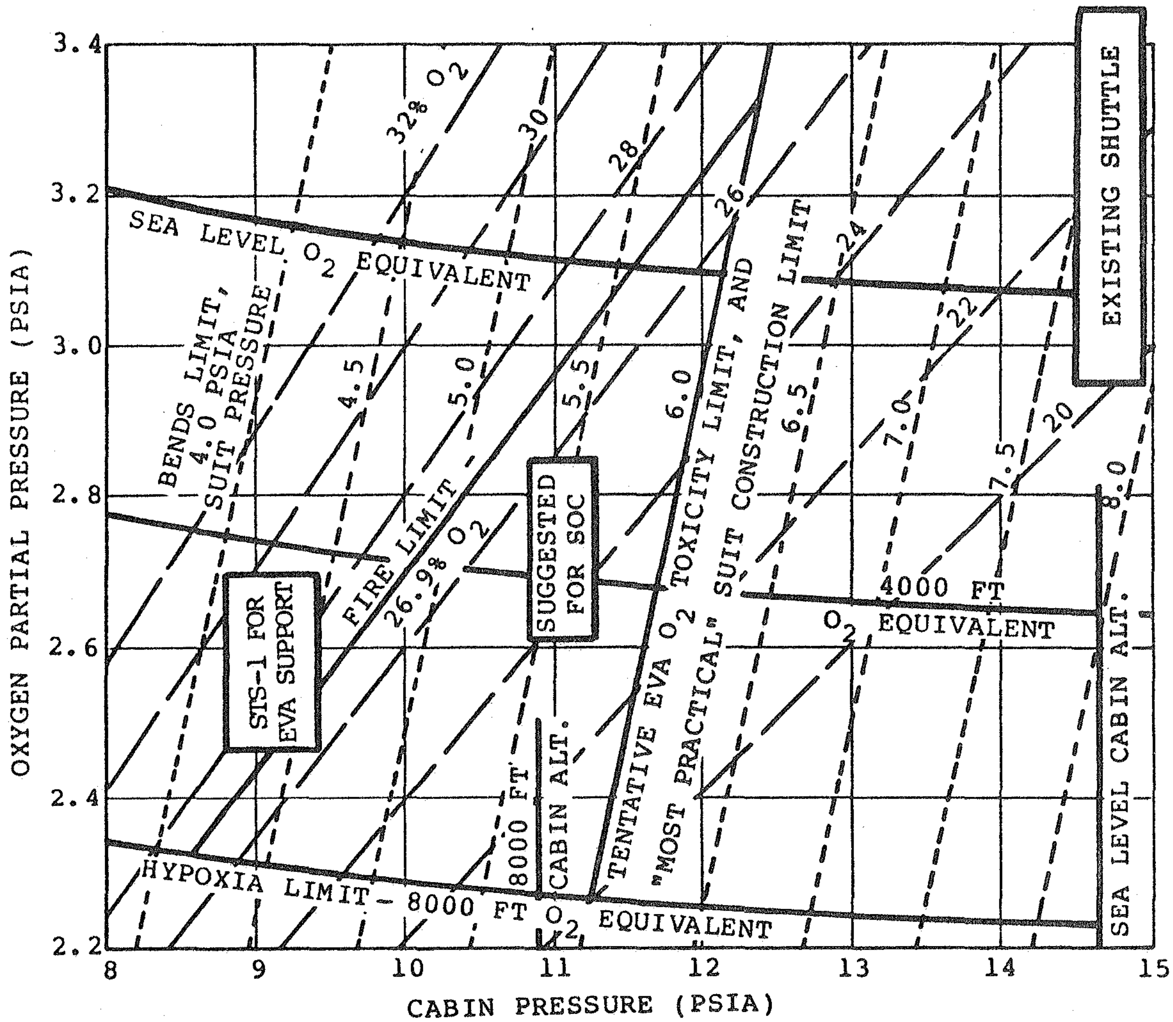


FIGURE 10-2

SOC CABIN AND SPACE SUIT
PRESSURE CONSIDERATIONS

pressure or decreasing cabin pressure. For SOC where several EVA's a day will be routine, eliminating pre-breathe is extremely desirable. The relationship between suit pressure to avoid pre-breathe and cabin pressure is also shown on Figure 10-2.

Oxygen Toxicity - The partial pressure of oxygen in a breathable atmosphere must be limited to avoid toxic effects. The crew is exposed continuously to the oxygen level in the module, as opposed to only eight hours per day exposure in the EVA suit. Because of this, the continuous oxygen cabin limit which can be tolerated in the cabin is lower than the short duration EVA oxygen limit.

The upper limit of oxygen partial pressure selected for Apollo and Skylab cabins was 5 psia, and in the case of Skylab this was for continuous use. There was some medical evidence of undesirable oxygen toxicity in these programs, as reported in the literature ("Extravehicular Crewman Work System Study Program", Final Report, Vol. II, Construction, July 1980, Contract NAS 9-15290 R. C. Wilde, Hamilton Standard). There has also been evidence of toxicity revealed in tests run since then, but there does not seem to be a real consensus on the degree of seriousness of these observed effects. An oxygen concentration as high as 4 psia O_2 partial pressure could probably be tolerated continuously in the SOC cabin, but this is a moot point because SOC will utilize a two gas atmosphere making this high a PPO_2 unnecessary, as shown on the left-hand vertical scale of Figure 10-2.

Oxygen toxicity during EVA is a different matter. First because EVA will occur for an individual crew member for a maximum of about 25 percent of his total in orbit time, and second because the atmosphere in the suit will in all likelihood be pure oxygen. There is some evidence that 8 psia pure oxygen pressure in the suit will result in unacceptable toxicity effects, as described in the literature (NADC-74241-40, "Physiological Responses to Intermittent Oxygen and Exercise Exposures", E. Hendler, NADC, Warminster, PA, 1974). For eight hours a day, a 4 psia level is generally accepted. The maximum allowable suit level of pure oxygen level for EVA therefore, probably lies between 4 and 8 psia but it is not a black or white matter, and considerable difference in tolerance between individuals undoubtedly occurs. A limit of 6 psia is logical since 4 psia is acceptable and 8 psia is not, but it is a tentative limit, not clearly defined. This tentative 6 psia suit pressure limit for pure oxygen is identified on Figure 10-2.

Weight of Stored Cabin Pressurization Gas - The leakage flow through any hole or leak in the vehicle pressure wall is directly proportional to cabin pressure. SOC cabin leakage is expected to be about 5 pounds of air a day. Another 5.3 pounds of air per day is expected to be lost in use of airlocks on an EVA day assuming pump down to 2 psia for a 14.7 psia cabin. This total air loss is made up by oxygen produced from wastewater by electrolysis, and by nitrogen obtained from the decomposition of 9.3 lb/EVA day of hydrazine. Capability for one complete repressurization utilizing stored high pressure gas weighs approximately 750 lb, plus tankage. The weight of the above varies as follows with cabin pressure:

Design Cabin Pressure	Resupply Hydrazine Required For Nitrogen Makeup Per 90 Days	Stored Repressurization Gas, Including Tankage	Resupply Water, Including Tankage Required For Oxygen Makeup Per 90 Days
14.7 psia	775	1321	270
11 psia	580	989	202
9 psia	474	809	166

Vehicle Mechanical Strength - Thickness of the SOC vehicle skin is dictated by the need for protection from meteorites and space junk. Reducing the vehicle cabin pressure would therefore not reduce skin weight.

Suit Considerations - The EVA suit is presently qualified for a nominal operating pressure in space of 4.1 psia. There is reason to believe, however, that this could be raised to 4.5 psia without significant difficulty. Beyond this, significant suit development would be required. Certainly a 6 psi suit would be less complex and more flexible than an 8 psi suit, and certainly it would cost less to develop in terms of time and money. Although this latter factor is not considered particularly significant in the overall evaluation, the increased safety and dexterity resulting from a 6 psi suit, rather than an 8 psi suit, could be particularly important on SOC where construction tasks and other functions of the vehicle place such emphasis on EVA capability.

A major consideration of an 8 psi suit used at 8 psi gage at sea level for training and development testing would require the standby use of a hyperbaric chamber for safety in the event of a suit pressurization failure. Rapid decompression may rupture lungs putting air into a pleural cavity. The lung may collapse and allow air into the blood. Decompression is essential to reduce bubble size to reduce danger of air embolisum. The highest suit pressure which does not require such a chamber for sea level safety is approximately 6 psi.

A 6 psi suit is identified on Figure 10-2 as a "most practical" upper limit for suit construction purposes, but this is a judgement call and not amendable to exact evaluation.

Adaptability of "Shelf Hardware" to Shuttle and SOC - There is an intangible benefit in utilizing a sea level cabin pressure in that it should reduce cost by making it easier to utilize commercial items already developed for earth use. This intangible benefit no doubt was a major factor in influencing Shuttle to be designed for a 14.7 psia cabin. Unfortunately, the real value of a 14.7 psia cabin in adapting shelf hardware is of less consequence than was hoped. Only air cooled electrical equipment items are effected by cabin pressure, and these are as much effected by the zero-gravity effect of space as they are by cabin pressure level. The lack of free convection cooling in space means that new fans will have to be added anyway to most items which were free convection cooled on earth. Once these fans are added, it is probably not a significant cost increment to select them for the appropriate cabin pressure. Figure 10-2 shows an 8,000 foot cabin pressure altitude, which is typical of airline practice, for reference, but it should be pointed out again that this altitude at zero-gravity poses entirely different equipment cooling problems.

Commonality With Shuttle Cabin Pressure - The rationale being used to select the final value of Shuttle cabin pressure becomes an important consideration in selecting SOC cabin pressure as well, since commonality between the two would be extremely desirable, if not essential. Shuttle cabin pressure selection is not yet firm, and in the event the final selected value for Shuttle differs from that considered in this report, this SOC cabin pressure assessment will require review and possible revision.

10.3.2 The Effect of Selected Cabin Pressure on ECLS System Components

The value of cabin pressure selected for design has many ramifications, as discussed in the previous section. One of these is the fact that a lower cabin pressure makes rejection of heat from the cabin air to the radiator coolant fluid more costly in terms of system size, complexity, and power consumption. This is because cabin air is the first stage coolant for rejecting most of the heat load generated in the cabin. This heat transfer is a function of air mass flow, not CFM, and therefore reduced air density increases the power needed to circulate the airflow required for heat transfer. This study considers a sea level cabin pressure as baseline. If this hardware were built and developed, and the cabin pressure were then reduced, the baseline heat rejection capability of the baseline ECLS would degrade as shown on Figure 10-3. This higher temperature may be undesirable so changes to the system may have to be made to accommodate lower cabin pressures. These changes need be made only in the components involved in the cabin air temperature control and ventilation functions, since the other ECLS systems components are unaffected.

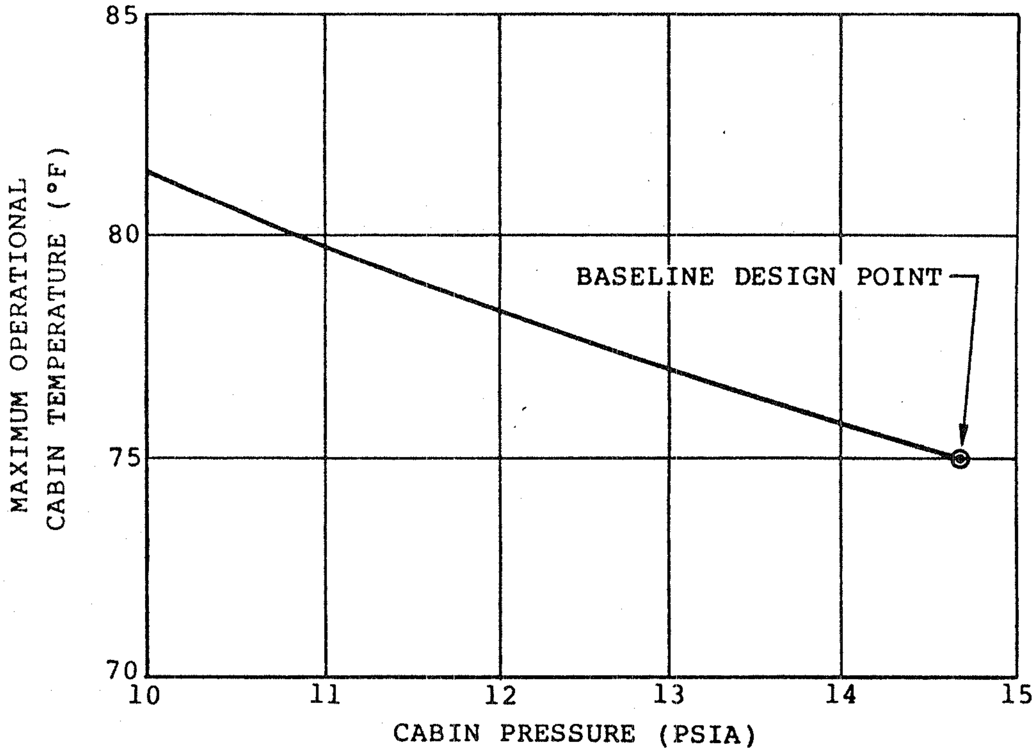


FIGURE 10-3

CABIN PRESSURE AFFECT
ON CABIN TEMPERATURE

The simplest change which can be made to the ECLS system to compensate for reduced cabin pressure would be to increase fan air handling capacity in order to maintain the design value of mass airflow, and accept the power and noise suppression penalty which would occur as a result. Figure 10-4 shows how fan power would increase to hold airflows, and therefore heat transfer, constant. Unfortunately, this solution of increasing fan size to accommodate a lower cabin design pressure would add significantly to the electric load demand of the ECLS. Figure 10-4 shows for example that an increase in power is required per full SOC from 3.6 kw to 6.5 kw, or a delta increase of 2.9 kw, in dropping cabin pressure from sea level to 11 psia. Battery weight needed to provide this 2.9 kw of power on the dark side would weigh 910 pounds. This weight does not include the weight of hydrazine which must be resupplied to keep an additional 2.9 kw of solar array in orbit.

An alternate approach would be to redesign all air handling components of the ECLS to maintain required airflow while holding the fan power increase to a minimum. This solution requires larger heat exchangers, filters, and distribution air ducting, as well as larger fans. The result of a family of such system designs is shown on Figure 10-5. Note on this Figure that the fan power delta is now only 1.9 KW in going from a sea level to 11 psia cabin. This is preferable to the 2.9 KW delta which results from changing only the fans, as shown on Figure 10-4. This lower power penalty is obtained by increasing the size of other air handling components in the system by 28 lb. and 1.7 ft^3.

10.3.3 Conclusions Regarding Selection Of Cabin Pressure

It is beyond the scope of this study to recommend the design value of cabin pressure which should ultimately be selected for SOC, but a "suggested" value is presented in this Section. As the preceding sections have pointed out, there are so many diverse factors to consider that the final selection is a difficult compromise. The following is a set of individual conclusions which may be reached concerning these factors:

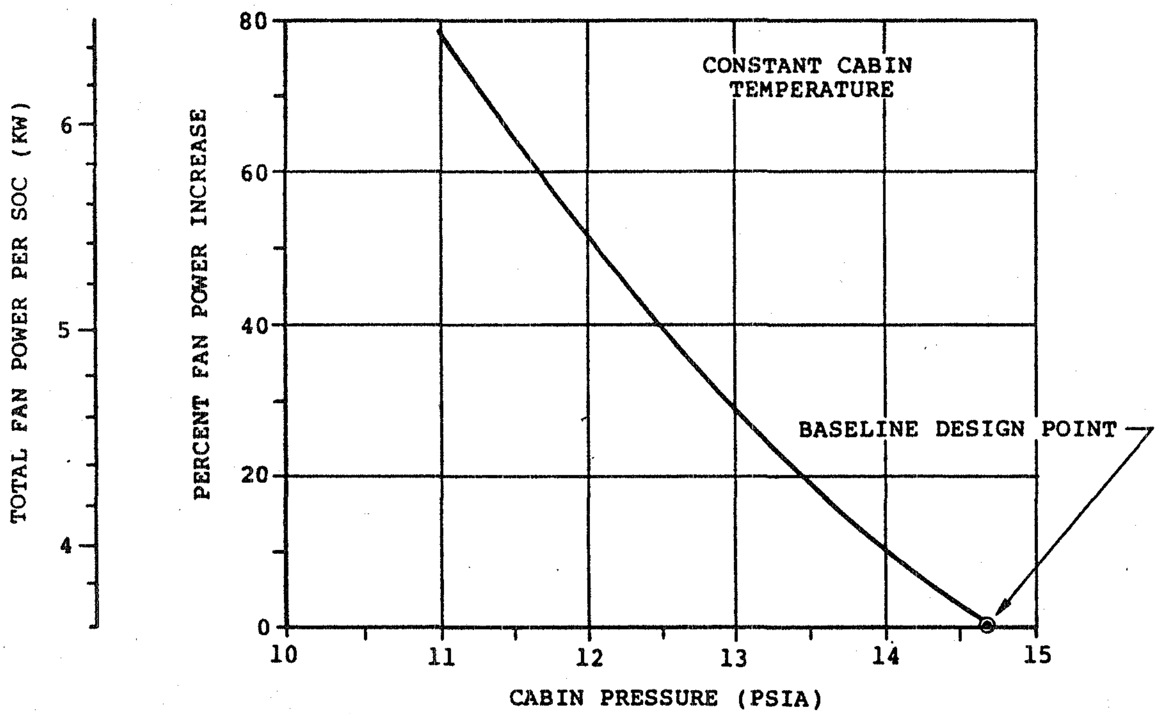


FIGURE 10-4

CABIN PRESSURE AFFECT ON FAN POWER

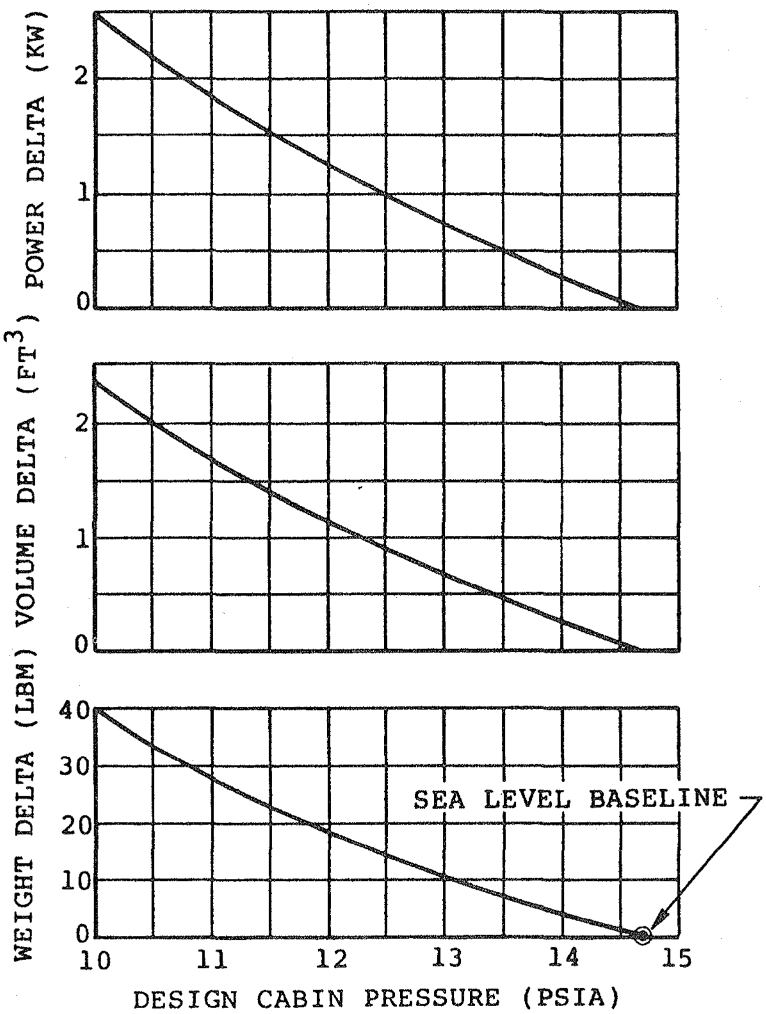


FIGURE 10-5

PENALTY ON REDUCED CABIN PRESSURE
(FAMILY OF OPTIMIZED SYSTEMS)

1. Referring to Figure 10-2 it can be seen that the logical cabin pressure for SOC lies within the boundaries of a triangle formed by the 26.9 percent oxygen Fire Limit on the left, the tentative O_2 Toxicity Limit and "Practical" Suit Limit on the right, and the 8,000 foot equivalent oxygen level at the bottom of the triangle. Existing Shuttle pressures are shown on the Figure for reference.

2. There is a preponderance of medical/health logic to favor selecting the SOC cabin toward the upper right portion of the triangular boundaries, mainly because man obviously works best near his ancestral sea level environment. The power, weight, and volume penalties of operating toward the upper right portion of the triangle, as opposed to operating toward the lower left portion, are not great. These penalties are about one percent of the total resources of SOC.

3. A normal cabin pressure error band of $\pm .2$ psi is recommended for SOC, based on this value being used on the current Shuttle.

4. The boundaries of the triangle call for tighter control on normal oxygen partial pressure level than is exercised on the existing Shuttle. A control of about $\pm .11$ psia oxygen partial pressure is recommended for SOC. This is the band used by the STS-1 for EVA support, shown on the Figure, and is held by manual control. Automatic control on SOC should be at least this accurate.

5. The box labeled "suggested for SOC" on Figure 10-2 is just that, a suggested compromise between the many diverse factors involved. Based on information available during this study, it is a logical, but not firm, selection. Use of the trade-off factors presented in Section 10.3 allows evaluation of the effect of other cabin pressure over the full range being considered.

10.4 REDUNDANCY PHILOSOPHY

The reliability of ECLS equipment to perform its intended function is improved by installing redundant equipment. Previous manned space programs relied on this principle of installed redundancy to provide a reliability adequate to achieve their mission objectives, and these early space programs were of a mission length which made achieving reliability goals by this method feasible. The SOC has a 10 to 20 year expected useful life requirement. Because of this long life requirement providing adequate reliability by installed redundancy is not feasible. Hardware designed for inorbit maintenance is mandatory to achieve the long SOC mission. Maintenance, however, does not delete the requirement for needing installed redundancy, but the amount of installed redundance for SOC ECLS hardware is dictated by different reasons than past manned space vehicles. The key reasons for installed redundancy on SOC are:

- A fail operational/fail safe design requirement
- No single ECLS failure shall cause abandonment of a habitat module
- No single ECLS failure shall require a Shuttle flight before the next planned flight.

The fail operational/fail safe requirement dictates the need to withstand two non-maintainable worst failures and still remain in a safe operating mode. A non-maintainable worst failure does not mean to imply that the ECLS system is not maintainable. All of the ECLS system can be maintained, however, some of the equipment such as main distribution plumbing, major wiring distribution bundles and equipment support structure, all which have a reliability of nearly "one" and would be expected to last for the life of the SOC, will be difficult to maintain and may require equipment and/or specialists to be supplied by the next Shuttle flight in order to conduct the maintenance. A non-maintainable failure also exists if the last spare has been used for equipment which is expected to be maintained. This first ground rule needs specified mission time periods to be meaningful. The fail operational time period is defined as 90 days, which is the normal SOC resupply period. The fail safe time period is defined to be 300 hours, which would allow for an emergency rescue by Shuttle.

The requirement that no single ECLS failure shall cause abandonment of a habitation module makes it clear that abandoning a habitat due to a single worst non-maintainable failure is unacceptable. The requirement that no single failure shall require a Shuttle flight before the next planned flight is self-explanatory. As a result of the above, all critical ECLS functions must be redundant.

The implementation of this redundancy philosophy requires the installation of dual heat transport and rejection loops, dual air revitalization subsystems, and dual water processing modules in each habitat SOC.

There are instances, however, where dual redundancy per habitat was not followed. For example, there are four cabin ventilation and thermal control packages installed in each habitat module. The sizing of this equipment into a larger number of smaller modules was determined by the desire for commonality with the thermal control units used in the service module. The service module units need to handle only about one half the heat load and ventilation flow as compared to the habitat module requirements.

Only one O_2 generation and one hydrazine decomposition subsystems were installed in each service module, which in turn supports one habitable module. Each, however, are double sized to be capable of servicing the full SOC. Intercabin plumbing between modules permits either of these subsystems to maintain the pressure and atmosphere composition control in both habitat modules and both service modules. Fail safe operation following two non-repairable failures is provided by a 300 hour stored gaseous supply.

Some of the equipment categorized under health and hygiene are not considered as critical functions. The backup capability provided by inflight maintenance and alternate operation modes will provide a reliability level commensurate with the mission requirements. Therefore, one washing machine, one shower and one dishwasher are considered adequate.

A complete listing of SOC ECLS system packages including the location of the packages, and the number of packages installed in each location is provided in Table 10-2.

This redundancy philosophy which has evolved for SOC will provide a comfortable environment in each habitat for 4 man crew with short term capability for an 8 man crew when all of the ECLS equipment is operational. After the first worst failure (before maintenance is performed), the ECLS equipment will provide an acceptable environment for a 4 man crew. Even with the first worst non-maintainable failure the ECLS system will support an 8 man crew for a 300 hour emergency period at degraded levels if the other habitat must be evacuated. The capability of the system to satisfy various emergency levels is presented in Section 10.2.

10.5 MAINTENANCE CONCEPT DEFINITION

The ECLS system, with its many pumps, fans, valves, instruments, controllers, etc., must be designed to be in-orbit maintainable in order to achieve the 10-20 year useful life required for the SOC.

The hardware in the ECLS system will be designed for maximum life but all dynamic hardware, like that mentioned above, will have an unsatisfactory probability of failure and replacement will be required. In-orbit maintenance places specific design requirements on the hardware. It also must be assumed that maintenance must be performed by a SOC crew man who does not have the detailed training that a factory technician would have and further must be performed with the general purpose tools available in the SOC tool kit. The general requirement for the ECLS system to be in-orbit maintainable and the assumptions regarding crew training and tool availability dictate that the ECLS system be designed so that:

- Fault isolation to the Lowest Replacement Level(LRU) hardware item be generally automated (some interaction with the crew man to provide yes/no information to the fault isolation process is acceptable).
- The LRU hardware item be adequately accessible.

TABLE 10-2

ECLS EQUIPMENT PACKAGES AND LOCATION

ECLS FUNCTION	MAJOR EQUIPMENT	Number of Packages and Location						Total SOC ECLS Packages
		HAB 1	HAB 2	SM 1	SM 2	LM	DM	
Cabin Ventilation and Thermal Control	Ventilation Fans	4	4	2	2	—	2	} 12 PACK'S + 2 FANS
	Air Cooling Heat Exchangers	4	4	2	2	—	—	
	Cold Plates	10	10	26	26	—	—	40 N.S. + 32 STRUCT.
Air Revitalization	Dehumidification	2	2	—	—	—	—	4
	CO_2 Removal	2	2	—	—	—	—	4
	Catalytic Oxidizer	2	2	—	—	—	—	4
	CO_2 Reduction	—	—	1	1	—	—	2
	Odor Control	2	2	—	—	—	—	4
	Atmosphere Monitoring	1	1	—	—	—	—	2
Heat Transport and Rejection	Radiators	2	2	—	—	—	—	4
	Freon Coolant Pump Packages	2	2	1	1	—	—	6
	Freon To Water Heat Exchangers	2	2	—	—	—	—	4
	Water Coolant Pump Packages	2	2	—	—	—	—	4
Atmospheric Supply	O_2 Generation	—	—	1	1	—	—	2
	Hydrazine Decomposition-N_2 Supply	—	—	1	1	—	—	2
	Hydrazine Storage	—	—	—	—	5	—	5
	Emergency O_2 Storage	—	—	1	—	—	—	1
	Emergency N_2 Storage	—	—	2	—	—	—	2
Water Processing And Management	Evaporation Purification Units	2	2	—	—	—	—	4
	Water Quality Monitoring	1	1	—	—	—	—	2
	Wastewater Storage	3	3	—	—	—	—	6
	Potable Water Storage	3	3	—	—	—	—	6
	Emergency Water Storage /EVA Water	4	—	—	—	22	—	26
Health And Hygiene	Waste Collection And Storage	1	1	—	—	—	—	2
	Emergency Waste Collection	1	1	—	—	—	—	2
	Hot Water Supply	1	1	—	—	—	—	2
	Cold Water Supply	1	1	—	—	—	—	2
	Shower	1	—	—	—	—	—	1
	Hand Wash	1	1	—	—	—	—	2
	Clothes Washer/Dryer	—	1	—	—	—	—	1
	Trash Compactor	1	1	—	—	—	—	2
	Food Refrigerator	1	1	—	—	—	—	2
	Food Freezer	1	—	—	—	1	—	1
	Oven	1	1	—	—	—	—	2
	Dishwasher	1	—	—	—	—	—	1
EVA/IVA Support	Suits And Backpacks	3	3	—	—	—	—	6
	Recharge Stations	1	1	—	—	—	—	2
	Air Lock Support	1	1	—	—	—	—	2
	Emergency Escape System	3	2	—	—	—	—	5
System Control	ECLS Central Control/Display	1	1	—	—	—	—	2
	Portable Maintenance Control/Display	1	1	—	—	—	—	2

- LRU attachment fittings, plumbing connections, wire connectors, etc. be generally standardized to minimize training and tool requirements and captivated to avoid loss.
- Drain and refill of systems be avoided for LRU replacement.

The definition of the LRU requires optimization studies. Previous ECLS systems studies (i.e., the Space Station Prototype program) indicated that the component level (pump, fan, valve, instrument, controller, etc.) was the proper maintenance level. The component level probably allows for the maximum use of commonality. Common components used in a variety of installations will significantly reduce spares inventory requirements, logistics requirements and crew training. The disadvantage is that common components will be slightly oversized or undersized for some installations.

With minor exception maintenance below the component level is not practical because of excessive crew training, spares inventory and special tool requirements. A higher level LRU than components may have some merit. Benefits may be realized in reduced weight abundance of installed equipment, on-board spares and resupplied spares. Crew time and fault isolation instrumentation and software will also be reduced. At this point a component level LRU is being assumed, however, during the SOC Phase B effort when more detailed subsystem schematics and hardware sizing information is available, an optimization study should be conducted to determine the appropriate LRU level and the degree to which hardware commonality is incorporated. Figures 10-6 and 10-7 show graphically how an optimum LRU selection and a commonality decision might be made.

In addition to determining an optimum LRU and commonality levels, as discussed above, certain goals must be set in order to have a comprehensive maintenance philosophy. The following goals, some of which were stated earlier, have been established to minimize the impact of maintenance on the SOC ECLS system:

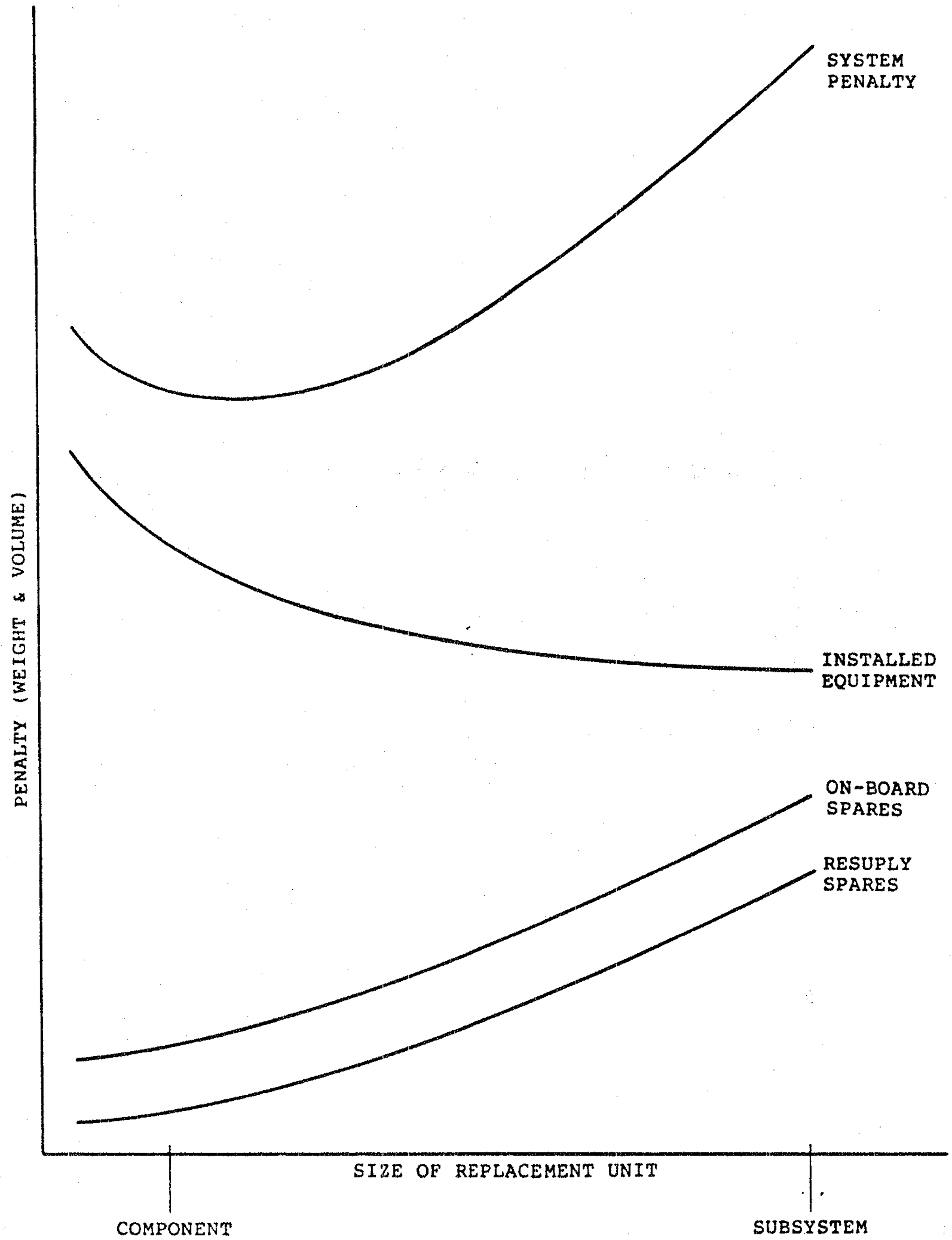


FIGURE 10-6

LOWEST REPLACEABLE UNIT OPTIMIZATION

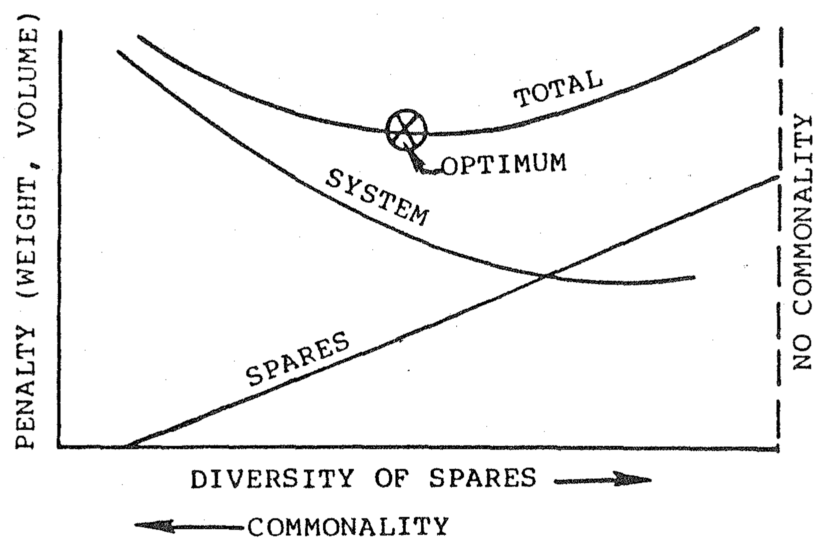


FIGURE 10-7

LOWEST REPLACEABLE UNIT COMMONALITY OPTIMIZATION

- No maintenance task shall exceed 4 hours.
- Unscheduled maintenance shall not exceed 8 man hours per month.
- Scheduled maintenance shall not exceed 40 man hours per month.
- Maintenance shall be accomplished with SOC tool kit TBD.
- Maintenance skill level shall be within SOC technician capability.
- Spares commonality shall be selected to provide minimum penalty.
- Equipment shall be designed to avoid fluid loss and air inclusion during maintenance.

For ease of maintenance, life support equipment will be located behind removable ceiling and floor panels. The equipment will be located so that replaceable component and/or modular units are generally no more than a single layer deep with adequate perimeter access providing up to five side access for maintenance. The replaceable modular units are packaged within a panel cavity so that plenum airflow is not disturbed during a maintenance procedure.

Modular units should generally be configured as a group of components forming a logic group. On-board repair of components is not ordinarily considered feasible. However, consideration should be given to possible emergency repairs below the component level by using standard (common) parts.

Malfunctions shall be isolatable to the logic group level and, as a goal, to the LRU level. Automatic detection of the malfunction or degradation shall be provided for critical functions. Where the fault isolation cannot be narrowed to a specific LRU in the allowable down time, replacement shall be made to all suspect LRU's.

Sufficient spares shall be stored on-board to sustain the system performance within the resupply period, and reliability allotments. Cannibalization of other systems can be considered if it is possible to be done within allowable maintenance time period.

The components in the SOC system which require replacement fall into several combinations of the following categories: low pressure, high pressure, hazardous liquid, non-hazardous liquid, hazardous gas, and non-hazardous gas.

<u>Gas-line components</u>. In general, gas-line component problems are not as severe as liquid-line problems. Some gas lines do, however, contain hazardous gases (such as hydrogen) or contaminated gases (such as commode outlet gas). Lines carrying hazardous gases have provisions for either purging or evacuating prior to component removal. Thereafter, maintenance considerations are the same as for any other gas-line and are described in the following paragraphs.

<u>High-pressure gases</u>. High-pressure lines are defined generally as those containing pressures which exceed 5 psig, while low-pressure lines are defined generally as those containing pressures below 5 psig. Vacuum lines should be considered and maintained in the same manner as high-pressure lines.

The high-pressure gas-lines shall contain bypass lines and shutoff and depressurization valves which are properly located to allow for depressurization and maintenance without interrupting other critical system functions. Once depressurized, the lines may be opened at the component fittings to allow component replacement.

<u>Low-pressure gases</u>. Components in the low-pressure lines will be connected to the duct through the use of flexible hoses, beaded tubes, and flanges or Marman-type couplings (or both). The Marman type flanges shall use of dovetailed grooves to captivate any seals which are used, and they should be coupled together with a quick release clamp. Where the system must remain operating, caps shall be provided to close off the open ports.

Liquid-line components. When considering the liquid subsystems and the requirement that the ECLS must operate in a zero-g environment, two important aspects of liquid-line component maintenance became apparent: (a) prevention of liquid loss and, (b) prevention of gas ingestion. Prevention of liquid loss precludes a recharge operation with the attendant need for replacement liquid and minimizes spillage which could induce other component failures, introduce possible contamination, and require varied, complex, and time consuming cleanup operations. Prevention of gas ingestion eliminates the need for evacuation or bleeding and reduces the performance requirement of gas separation equipment.

One of the maintenance methods considered for liquid-line components was the subsystem "drain and fill" approach. Although this approach would permit the use of a greater number of "off-the-shelf" or standard components, the design constraint resulting from the possibility of gas entrapment could jeopardize the system integration effort. This approach would also require many drain ports and complex servicing equipment.

Requirements for draining, complex servicing equipment, and clean-up problems are undesirable. Therefore, the subsystem shutdown and drainage approach has been eliminated as a normal liquid-line maintenance approach. The selected basic approach to maintenance of the ECLS liquid-line subsystems is that it can be accomplished without requiring draining or complete subsystem deactivation. Several concepts are being developed which allow for replacement of components and valves without system drainage and also allow for bypassing failed line sections.

Non-maintainable Items

Failure of an item which is not normally maintainable is considered an exceptional event. Repair of these items may require disruption of normal activities since the repair effort may be of long duration, requiring more than one crew member, and, in extreme, the evacuation of the module may be required. The first worst case failure of this equipment will cause a degraded performance condition to exist which is acceptable for 90 day operation.

Items which cannot be maintained by the use of on-board spare parts are required to have a reliability which approaches one. This equipment includes frames, main distribution fluid lines and main distribution electrical harnesses.

Non-maintainable item should be protected so that maintenance of adjacent components does not pose the possibility of damage to the non-maintainable item. In the case of fluid lines and harnesses, these should be located behind panels and the harnesses should be routed in conduit. Where redundant fluid lines and harnesses are required, they should be routed in separate compartments and should be located as far apart from one another as possible.

Non-maintainable items should be designed with conservative safety factors. Major structural members should contain redundancy where possible. The non-maintainable items should be designed so that a failure can be easily detected.

In order to minimize the number of random failures, low reliability and life limited items should be replaced before the expected failure occurs. The tradeoff is between crew convenience and cost. The minimum cost approach is to let all equipment operate until it wears out. The maximum cost approach is to replace equipment at the minimum expected life. A third approach is to monitor equipment and predict equipment failure based on trend analysis. If the predictions are fairly accurate, this third method would be close to the minimum cost, while providing flexibility for scheduling maintenance. The third approach should be considered whenever it can be accomplished without the addition of instrumentation. It should also be considered when the equipment is costly and there is a large spread between the minimum and maximum expected life.

10.6 EVALUATION OF SELECTABLE CABIN TEMPERATURE, AND CABIN TEMPERATURE BAND EXPANSION

There are several facets to a discussion of variations in cabin temperature. First, there is the matter of control band variation, that is to say the cabin temperature band over which the cabin temperature controls may be set by the crew. The set temperature is not related to the capacity of the ECLS system to always provide this selected temperature. For example, the thermostat used by the crew to select cabin temperature could be settlable at any temperature between 65°F and 80°F. When in the full cold or full hot mode, any cabin temperature within this band would be maintained within the capacity of the ECLS to do so. Cabin cooling capability, as discussed in this section, refers only to the full cold capability of the ECLS to satisfy the operating condition in question. The degree to which cooling capacity should be provided, to ensure that the lowest cabin temperature selectable by the crew will be achieved, has weight, volume, and power impact on the ECLS system. On the other hand, the ability to heat the cabin has no penalty since the vehicle is so well insulated, and since there is always a significant electrical heat load to be dissipated. It is the cooling capacity required that sizes the thermal control packages, and the remainder of the heat rejection system as well.

This SOC study considers as its baseline a system capable of providing the following "full cold" cabin cooling capacity:

Cabin Heat Load	Crew Per Hab. Mod.	ECLS Operating Mode	No. Temp. Control Units Failed Per Hab. Mod.	Hab. Mod. Crew Per Operating Temp. Control Unit	Maximum "Full Cold" Cabin Temp.	Maximum Cabin Dew Point
Maximum	4	Operational	0	1	75	60
Maximum	4	90 Day Degraded	2	2	85	70
Maximum	8	90 Day Degraded	0	2	85	70
Maximum	8	14 Day Emergency	2	4	90	75

In practice, the thermal system cannot be sized to simultaneously "just meet" all four of the above operating modes. In this case, the operational mode will be just met, and the cooling capacity requirement of the other three will be somewhat exceeded. The baseline system of this study, in the operational mode, will provide sufficient full cold cooling capacity at the maximum heat load case for a cabin temperature of 75°F. The question arises; what would be the effect on the ECLS system if this cooling capacity requirement were increased to drop this maximum operational cabin temperature to 70°F, and 65°F respectively.

Preliminary designs of a family of different systems would exhibit the power, weight and volume penalties shown on Figure 10-8. Note that the penalties in going from 70°F to 65°F are much more severe than going from 75°F to 70°F. The reason for this is that there is an infinite penalty in going to a 55°F cabin because the ability to hold cabin relative humidity of 75% becomes impossible below a cabin temperature of 55°F. This is because the coolant water and therefore the dew point is limited to 40°F to prevent icing in the coolant water to freon heat exchanger.

The penalty study results shown in Figure 10-8 indicate that there is relatively little penalty to change the present 75°F selected SOC baseline cabin temperature specification to require a full cold operational maximum temperature of 70°F. This figure shows that the increased comfort which would result from a 70°F cabin at high work rates would cost approximately 35 lb, 2.2 ft^3 and 400 watts. A further decrease in full cold cabin temperature capability to 65°F appears to have an unacceptable penalty.

This study does not recommend a reduction in max load cabin temperature capability from the baseline 75°F value to 70°F, but the tradeoff study suggests that the increased penalty of this 5°F reduction is small enough to make such a change in the specification reasonable if it is desired for additional comfort for high work loads.

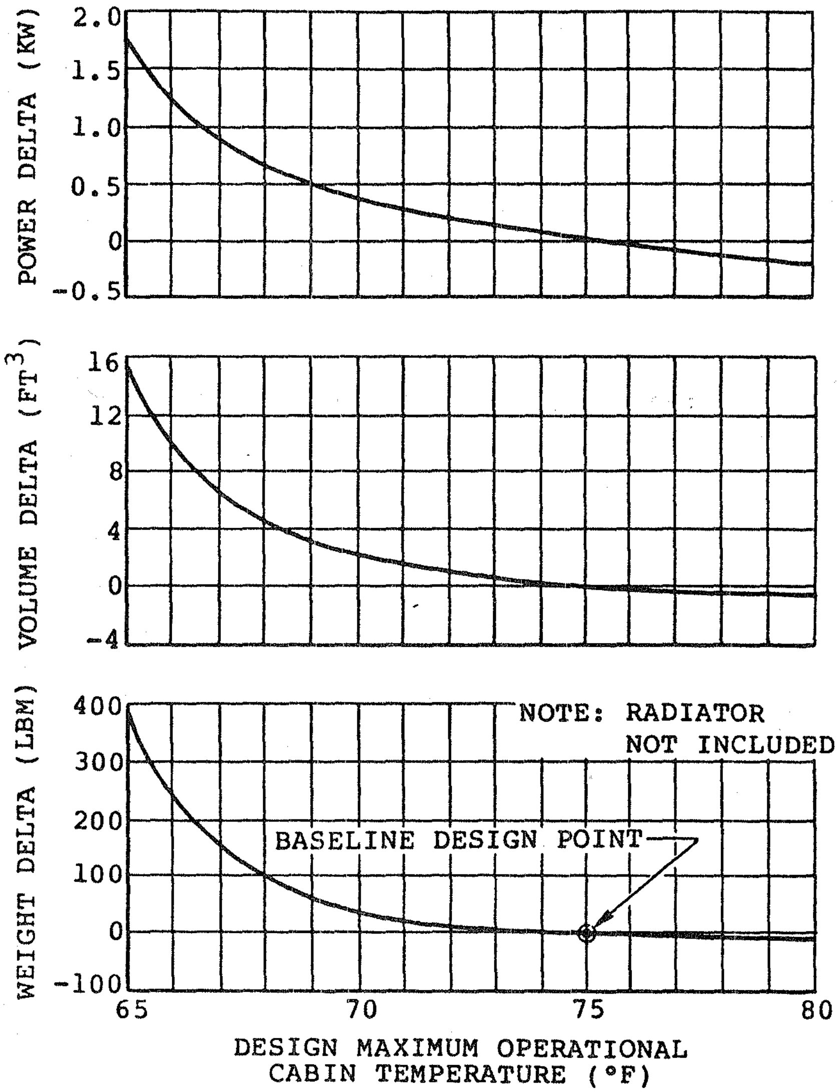


FIGURE 10-8

SELECTABLE CABIN TEMPERATURE PENALTY

Most building air conditioning systems include a thermostat settable in the range of 65°F to 80°F, regardless of the capacity of the buildings system to satisfy this range under conditions of maximum load. It is suggested that the SOC cabin temperature thermostat likewise have a thermostat settable range from 65°F to 80°F.

10.7 ECLS SUBSYSTEM SELECTION RATIONALE

At the initiation of Boeing's SOC Systems Analysis effort, an ECLS reference baseline was defined by NASA in NASA-3 and NASA-6. The ECLS system defined in Boeing-19 differs from the NASA reference baseline in some areas. A rationale for the most significant deviations is provided in this section. The differences discussed are:

- CO_2 Removal Concept Selection
- Wastewater Processing Concept Selection

10.7.1 CO_2 Removal Subsystem Concept Selection

Hamilton Standard has been evaluating, by analysis and test, spacecraft CO_2 removal concepts for more than 20 years. Both expendable and regenerable techniques have been evaluated. The concepts receiving considerable attention were:

- Lithium hydroxide
- Lithium peroxide
- Solid amines
- Molecular sieves
- Electrochemical
- Molten carbonate

With the exception of Skylab, all U.S. spacecraft to date have used expendable lithium hydroxide. Skylab used a regenerable molecular sieve CO_2 removal and dump system. Spacecraft such as the planned SOC will require a regenerable CO_2 removal system which will supply pure CO_2 at atmospheric pressure for a CO_2 reduction process in order to save the oxygen. Two concepts have evolved which meet these SOC requirements: solid amine and electrochemical. Both of these concepts have been thoroughly tested as prototype hardware under both NASA contracts and contractor R&D funds demonstrating their capability to perform the CO_2 removal and concentration functions. For the SOC study activity the Solid Amine Water (Atmospheric Pressure Steam) Desorbed (SAWD) has been selected by Hamilton Standard. The reasons for this selection are that SAWD as compared to the Electrochemical Depolarized CO_2 Concentrator (EDC) when integrated into the SOC life support system:

1. Has less system weight, volume and power penalty.
2. Is not dependent on an emergency LiOH backup system.
3. Does not require over-sizing the electrolysis subsystem.
4. Is less dependent on other subsystems for operation (cascading failures).
5. Allows the ECLS to be designed without introducing H_2 into habitat module or requiring H_2 lines passing through bulkheads.
6. Is free from potential caustic carryover.
7. Is tolerant to the cabin humidity range without preconditioning the air.
8. Can be exposed to a depressurized cabin without requiring shut-off valves or other precautionary action.
9. Uses power on the light side of the orbit without a significant sizing penalty.
10. Is less expensive hardware.

In order to evaluate statement number 1, it is necessary to provide a design specification for sizing the SOC CO_2 removal unit. It is believed that as a result of the SOC fail operational (90 day degraded)/fail safe (14 day further degraded) requirement the SAWD CO_2 removal units must be sized for 8 men CO_2 output at 12 mmHg partial pressure. Four units sized to this criteria would be installed in the SOC, two per habitat, to meet safety requirements. The EDC CO_2 removal unit can be sized for the same case as the SAWD supported by four 4 man (+ EDC O_2 and H_2) electrolysis units operated on the sun side only. In this case, EDC would shut down on the dark side of the orbit. As another option, the EDC can be sized to run continuously with continuously running electrolysis units. In order to meet safety requirements, two EDC units and two electrolysis units would be required per habitability module in both of these cases.

Another approach using an EDC unit which would also meet the safety criteria is to incorporate a backup LiOH system for the 14 day emergency case. This approach allows the EDC to be designed for 4 men at 7.6 mmHg CO_2 partial pressure (the 90 day degraded operation level). Again two units per habitat are required. Only one electrolysis unit per habitat is required in this case (O_2 can be supplied to either habitat from either service module electrolysis unit).

Still another approach using an EDC unit which satisfies the safety criteria is to use one 8 man (+EDC O_2 and H_2) electrolysis unit in each service module and in case of a double failure of the electrolysis systems backup O_2 would be added to the 14 day emergency O_2 supplies and H_2 would be provided from the hydrazine decomposition unit. In this case, no LiOH backup is required. This case would require more hydrazine decomposition than needed to provide cabin nitrogen. Therefore, during the emergency, the hydrazine decomposition unit would have to be configured to dump some nitrogen overboard (before it is mixed with cabin air) in order to provide adequate H_2 for EDC.

In all of the above design scenarios it is assumed that a first failure could place the 8 man SOC crew in one habitat. The 90 day degraded operational levels are acceptable for this situation. A second failure could then place the crew in the 14 day emergency level. It is also believed that an ECLS equipment failure, as a first failure, must allow at least 4 men to exist at the 90 day degraded levels without leaving the habitat.

Considering the above discussion, Figures 10-9, 10-10, 10-11, 10-12 and 10-13 provide a pictoral description of each design scenario. The comparative weight, volume and power for the design scenarios are shown in Tables 10-3, 10-4, 10-5 and 10-6. The most favorable EDC situation would appear to be that shown in Table 10-4, however, weight and volume penalties associated with the extra batteries required to run the EDC and electrolysis units continuously will be large. Specific penalty values for continuous vs. sun side only power are not firm at this time, however, preliminary power weight penalties from Boeing were used and the weight for using dark side power is indicated in the Table. Even without final power penalty numbers it is clear that the SAWD is the best choice for a SOC CO_2 removal system. Sizing data for the EDC was taken from Life Systems Incorporated report no. LSI ER-319-24. Sizing data for electrolysis was taken from the RLSE program report "Thermal Control and Life Support Subsystems", March 1977, prepared under Contract NAS 9-14782.

Statement numbers 2 and 3 have been discussed above. Also from the above discussion, statement number 4 has been partially discussed. The SAWD is dependent on electrical power to operate and on the cabin humidity control. The principal use of power is to generate steam for amine desorption during the light side of the orbit. This steam is evaporated from the amine and condensed in the humidity control unit during CO_2 adsorption.

The EDC is also dependent on electrical power and on cabin humidity control. However, since the principal electrical power is supplied to the electrolysis system in order to provide oxygen and hydrogen, the EDC is also dependent on electrolysis. The EDC generates water vapor and sensible heat from the oxygen and hydrogen reaction and is, therefore, dependent on the cabin humidity control unit. The EDC must also have a closely controlled inlet air humidity. This control is required in order to keep the EDC electrolyte from over-drying or flooding. Humidity

DESIGN POINT

2 - 8 MAN, CYCLIC ELECTROLYSIS UNITS
4 - 8 MAN, 12 mmHG, SAWD UNITS

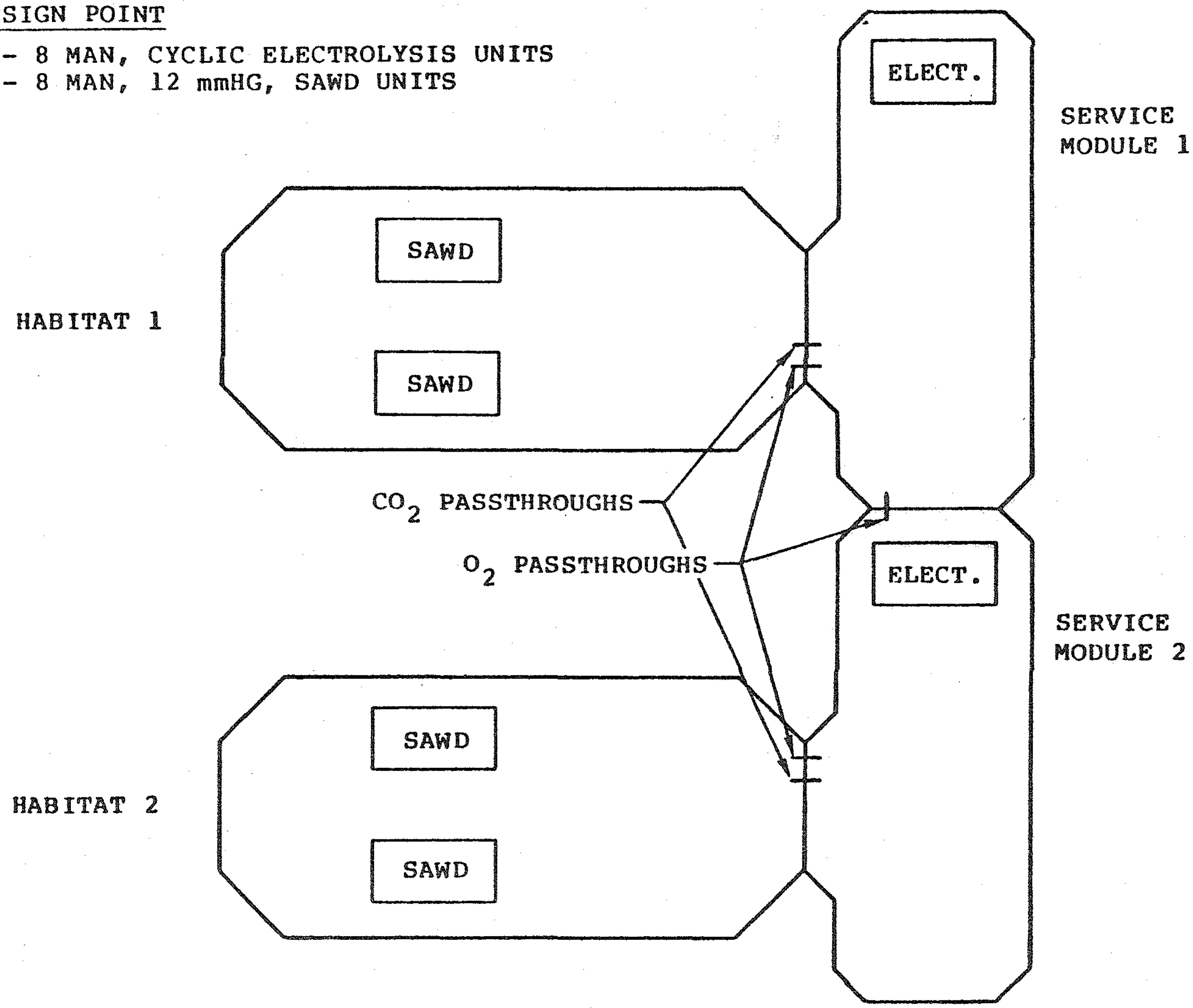


FIGURE 10-9
SAWD INSTALLATION

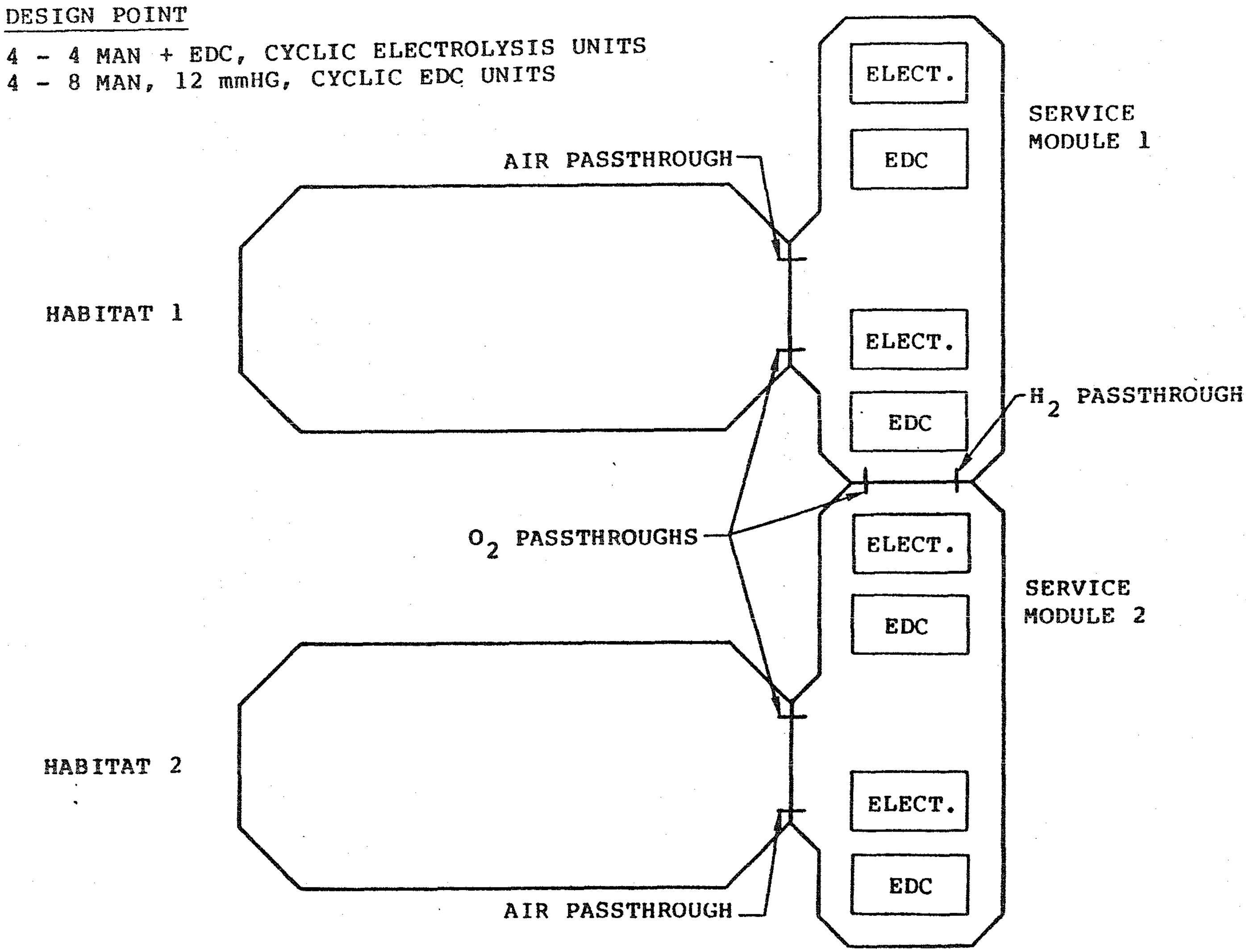


FIGURE 10-10
EDC INSLALLATION "A"

DESIGN POINT

4 - 4 MAN + EDC, CONTINUOUS ELECTROLYSIS UNITS
4 - 8 MAN, 12 mmHG, CONTINUOUS EDC UNITS

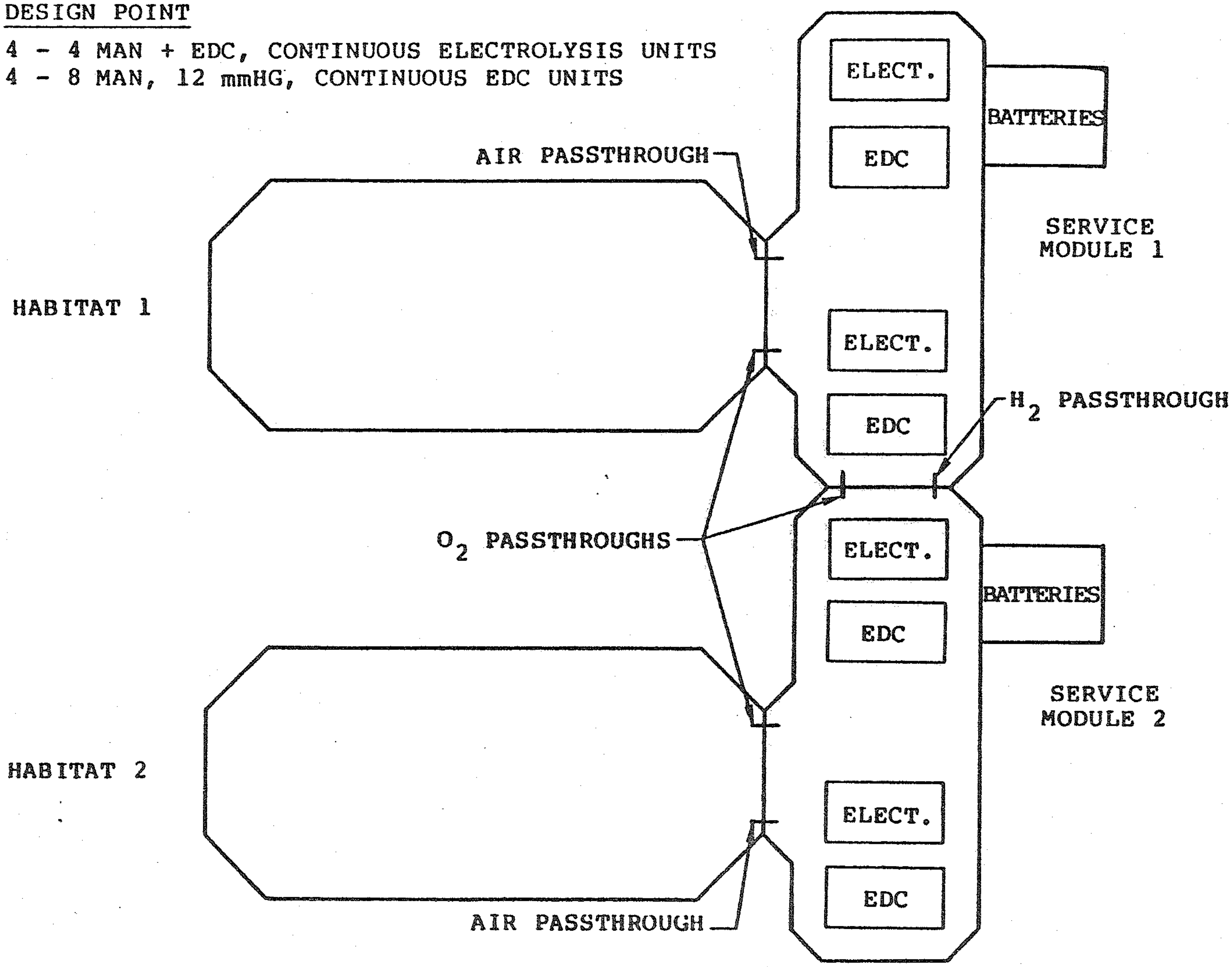


FIGURE 10-11
EDC INSTALLATION "B"

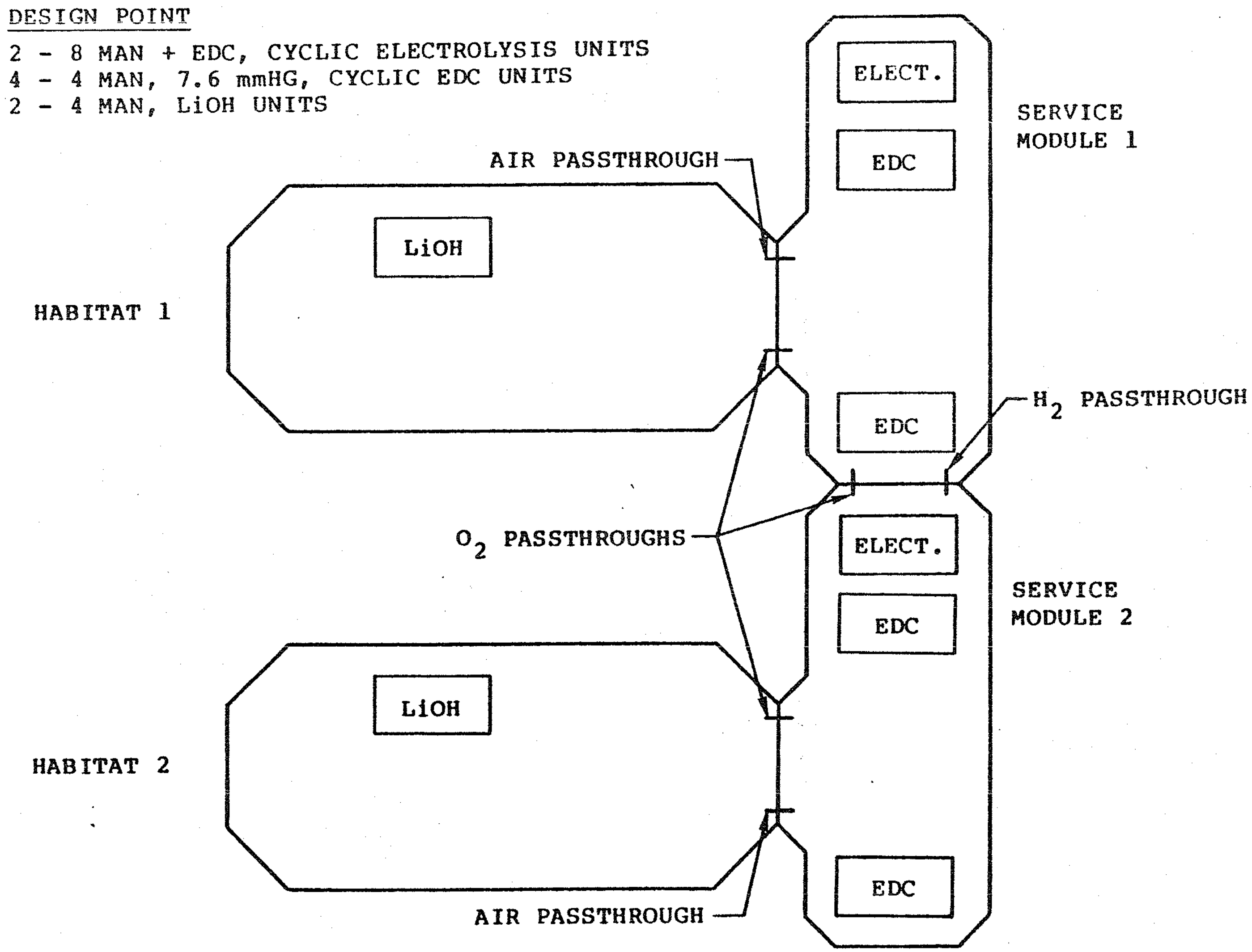


FIGURE 10-12
EDC INSTALLATION "C"

DESIGN POINT

2 - 8 MAN + EDC, CYCLIC ELECTROLYSIS UNITS
4 - 8 MAN, 12 mmHg, EDC UNITS
N_2H_4 DECOMPOSITION AND EMERGENCY OXYGEN STORAGE SIZED TO RUN EDC IN 14 DAY EMERGENCY

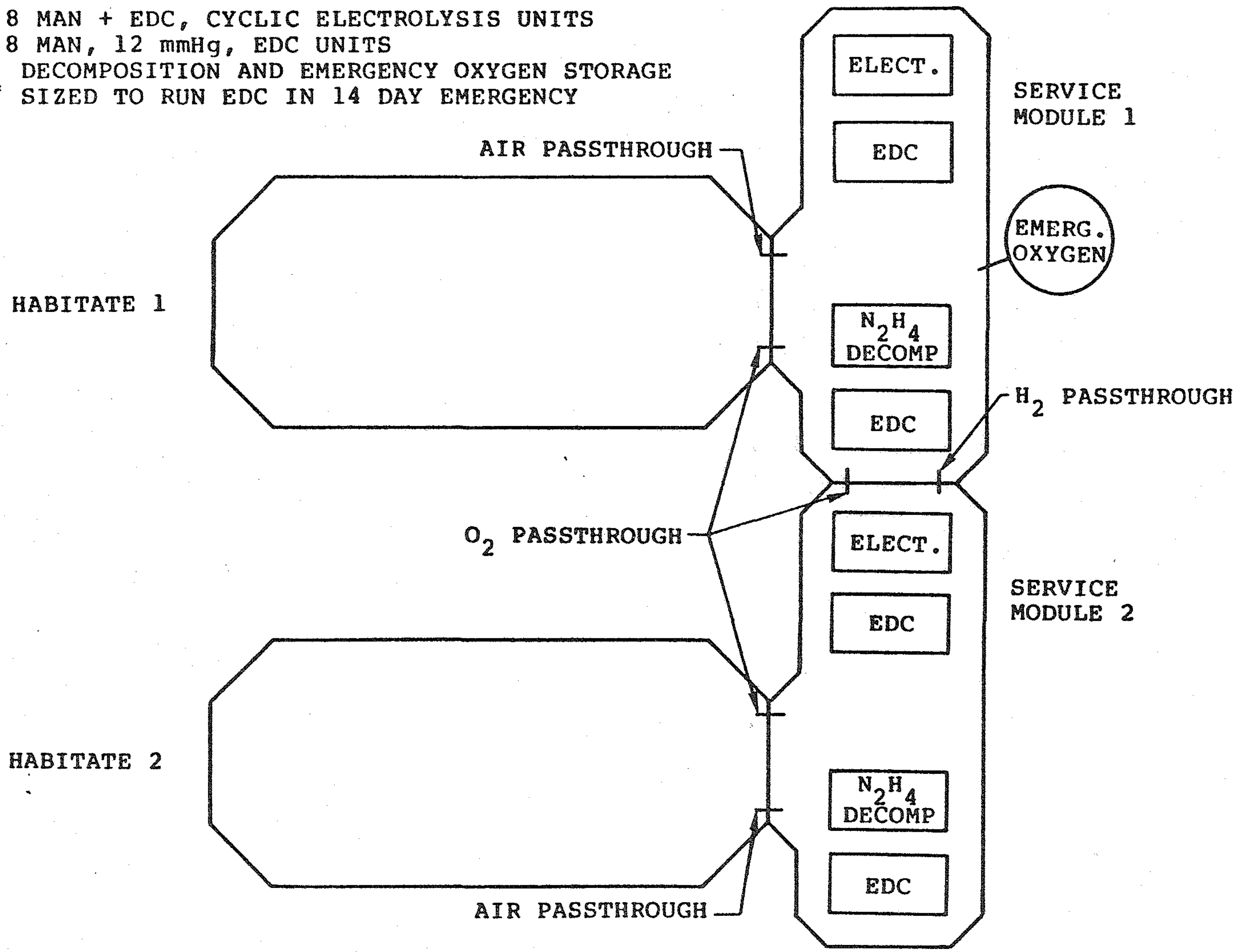


FIGURE 10-13

EDC INSTALLATION "D"

TABLE 10-3

SOC CO_2 REMOVAL SUBSYSTEM TRADE
(CYCLIC OPERATION - NO LiOH BACKUP)

4-8 MAN, 12MMHG CO_2, SAWD UNITS 4-8 MAN, 12MMHG CO_2, EDC UNITS 2-8 MAN ELECTROLYSIS--SAWD 4-4 MAN ELECTROLYSIS--EDC	EDC (CYCLIC ELECTROLYSIS OPERATION)	SAWD (CYCLIC ELECTROLYSIS OPERATION)
WEIGHT (LB.)	500 EDC + 570 ELECTROLYSIS 1070	526
VOLUME (FT^3)	16 EDC + 37 ELECTROLYSIS 53	29
POWER (WATTS)		
8 MEN, 12MMHG (ONE CO_2 UNIT OP.)	- 196 EDC +1500 ELECTROLYSIS 1304	1215
8 MEN, 3.8MMHG (FOUR CO_2 UNITS OP.)	- 70 EDC +1750 ELECTROLYSIS 1680	1396

TABLE 10-4

SOC CO_2 REMOVAL SUBSYSTEM TRADE
(CONTINUOUS OPERATION, CYCLIC SAWD OPERATION - NO LiOH BACKUP)

4-8 MAN, 12MMHG CO_2, SAWD UNITS 4-8 MAN, 12MMHG CO_2, EDC UNITS 2-8 MAN ELECTROLYSIS--SAWD 4-4 MAN ELECTROLYSIS--EDC	EDC (CONTINUOUS ELECTROLYSIS OPERATION)	SAWD (CYCLIC ELECTROLYSIS OPERATION)
WEIGHT (LB.)	408 EDC + 432 ELECTROLYSIS +1474 BATTERIES 2314	526
VOLUME (FT^3)	14 EDC + 26 ELECTROLYSIS 40	29
POWER (WATTS)		
8 MEN, 12MMHG (ONE UNIT OP.)	- 162 EDC +1000 ELECTROLYSIS 838	1215
8 MEN, 3.8MMHG (FOUR UNITS OP.)	+ 34 EDC +1100 ELECTROLYSIS 1134	1396

TABLE 10-5

SOC CO_2 REMOVAL SUBSYSTEM TRADE
(CYCLIC OPERATION - LiOH BACKUP FOR EDC)

4-8 MAN, 12MMHG CO_2, SAWD UNITS 4-4 MAN, 7.6MMHG CO_2, EDC UNITS 2-8 MAN ELECTROLYSIS--SAWD 2-8 MAN ELECTROLYSIS--EDC 2-4 MAN, LIOH UNITS--EDC	EDC (CYCLIC ELECTROLYSIS OPERATION)	SAWD (CYCLIC ELECTROLYSIS OPERATION)
WEIGHT (LB.)	416 EDC + 116 ELECTROLYSIS + 728 LIOH 1260	526
VOLUME (FT^3)	14 EDC + 9 ELECTROLYSIS + 30 LIOH 53	29
POWER (WATTS)		
8 MEN, 12MMHG (ONE CO_2 UNIT OP.)	0 EDC 0 ELECTROLYSIS + 45 LIOH 45	1215
8 MEN, 7.6MMHG (TWO CO_2 UNIT OP.)	- 36 EDC +1550 ELECTROLYSIS + 0 LIOH 1514	1275
8 MEN, 3.8 MMHG (FOUR CO_2 UNIT OP.)	- 70 EDC +1750 ELECTROLYSIS + 0 LIOH 1680	1396

TABLE 10-6

SOC CO_2 REMOVAL SUBSYSTEM TRADE
(CYCLIC OPERATION - N_2H_2 FOR EDC BACKUP H_2)

4-8 MAN, 12MMHG CO_2, SAWD UNITS 4-8 MAN, 12MMHG CO_2, EDC UNITS 2-8 MAN ELECTROLYSIS--SAWD 2-8 MAN ELECTROLYSIS--EDC EDC-12MMHG-USES STORED O_2 AND H_2 FROM N_2H_4 DECOMPOSITION	EDC (CYCLIC ELECTROLYSIS OPERATION)	SAWD (CYCLIC ELECTROLYSIS OPERATION)
WEIGHT (LB.)	500 EDC + 116 ELECTROLYSIS + 172 O_2 + TANKAGE 788	526
VOLUME (FT^3)	16.0 EDC + 9.0 ELECTROLYSIS + 12.2 O_2 + TANKAGE 37.2	29
POWER (WATTS) 8 MEN, 12MMHG (ONE CO_2 UNIT OP.)	- 196 EDC + 0 ELECTROLYSIS - 196	1215
8 MEN, 3.8MMHG (FOUR CO_2 UNITS OP.)	- 70 EDC +1750 ELECTROLYSIS 1680	1396

control can be done by placing the EDC downstream of the cabin humidity control unit or building air humidity conditioning into the EDC. The SAWD is tolerant to the cabin humidity range. This is done by designing proper airflow through the unit.

It is desirable to design the SOC ECLS without the need for hydrogen lines in the habitat or the need to pass hydrogen lines through docking or berthing ports. The SAWD easily complies with statement 5 because it is not dependent on a hydrogen feedline. It would be difficult to design the SOC ECLS to meet this desire with an EDC system. Since the EDC would probably want to be located downstream of the humidity control unit, it would be necessary to locate the EDC in the habitat or move the humidity control units into the service module. With built-in humidity control the EDC could be located in the service module. For safety reasons it is desirable to keep critical life support functions in the habitat. Oxygen and nitrogen supply is obviously critical, however, the electrolysis systems are cross-linked so that either system can provide oxygen for the complete two habitat SOC. Oxygen and nitrogen are also stored as high pressure gas for a 14 day emergency. For these reasons and to keep hydrogen out of the habitat, the electrolysis and nitrogen generation systems were installed in the service module.

In reference to statement 6, because the EDC electrolyte is a caustic liquid which expands and contracts in volume dependent on the air inlet humidity, it may be possible to get some electrolyte carried out of the EDC. The solid amine is not a liquid and is retained between filter screens in the SAWD canisters.

Statement number 7 has been discussed above. Statement 8 is met by SAWD because it is completely compatible with vacuum exposure. After long term vacuum exposure, the amine will be dry; however, normal steam desorb and CO_2 adsorb cycling, will regain bed moisture equilibrium. Exposure of the EDC electrolyte to vaccum will first freeze it and then dry it. This may cause irreversible damage.

With regard to statement number 9, the SAWD can, with only a small penalty in amine quantity, be operated so that the steam desorption power is used in the light side of the orbit only. Adsorption of CO_2 is done whenever air is flowing through the canister. The airflow power is low and adsorption would continue during the orbit dark side.

Since the electrolysis system is such a large power consumer, it should be operated on the light side of the orbit only. As a result, the EDC would shut down during the dark and must be sized larger because of this off period. An alternate may be to install a hydrogen accumulator for dark side operation, but this is contrary to safety desires. It is possible with the present SOC ECLS definition to avoid any significant hydrogen accumulation such that any hydrogen line can be broken without causing the cabin hydrogen partial pressure to exceed safety limits.

In comment to statement 10 electrochemical systems are probably more expensive to fabricate than a system using canisters filled with inexpensive solid amine. Increasing the electrolysis system size would also add to the cost of an EDC approach.

10.7.2 Wastewater Processing Concept Selection

The SOC crew will generate a calculated 393.6 lb per day of wastewater which must be processed to return it to a reusable state preferably to potable quality. The SOC ECLS baseline defined by NASA in documents NASA-3 and NASA-6 showed a vapor compression distillation (VCD) system for processing urine, a hyper-filtration system for processing wash water and a multi-filtration system for processing humidity condensate and water from CO_2 reduction. If all these processes are employed by the SOC water management system, keeping the water streams separate, and providing wastewater holding tanks and various degrees of good water holding tanks, it is evident that the SOC water management system will be very complex and difficult to manage. The redundancy considerations for a reliability requirement of fail operational, fail safe makes the system design even more complex.

An assumption of the above described water management system is that humidity condensate is potable with a multi-filtration treatment. There are several concerns about this assumption. The humidity condensate will be contaminated by the floating particulate and aerosols in the zero-gravity cabin. It is not practical to filter the condenser airflow to a low enough level to eliminate this contamination. Bacteria and virus will contaminate the condenser. The condenser is an important trace contaminant removal system; for example, the condenser removes all ammonia from the cabin air. Evaporated cabin water and other liquids (vomit, urine from animal experiments, etc.) will be condensed by the condenser. For all these reasons it is believed that water, which is planned for astronaut consumption, should be processed by a closed loop phase change process with the contaminated feed water chemically pretreated to kill virus and bacteria. The condensed product water should also be treated to ensure that sterility is maintained. The two concepts which meet these requirements are the VCD and the Thermoelectric Integrated Membrane Evaporation System (TIMES).

The use of either VCD or TIMES to process all of the SOC wastewater has considerable merit. Not only will there be a significant reduction in complexity, but total weight and volume will be less and power will be only slightly higher as shown in Table 10-7. A desirable operational situation results from using one technique for processing all the SOC wastewater. Four units are planned to process all SOC wastewaters. Since the amount of wash water used can be varied if one of the processing units fail (by wearing clothing longer, taking less showers, etc.), added reliability for the critical potable water generation function results without adding large overcapacity to the system. Only one operating process unit is required to provide all the potable water needed for crew survival. At this point in SOC ECLS studies, it is not necessary to make a selection between VCD and TIMES for processing wastewater because both systems integrate into the SOC ECLS in the same way. Because of the importance of reprocessing wastewater for the SOC, it is desirable to continue research and development work on both concepts.

TABLE 10-7

WATER PROCESSING CONCEPT COMPARISON

Selected System	INSTALLED WT	INSTALLED VOL	POWER	RESUPPLY WT	RESUPPLY VOL
Waste Storage Tanks (6)	303	40	20	9.1	1.2
Potable Storage Tanks (6)	303	40	20	9.1	1.2
Evaporation Units (4 at 4.1 lb/hr rate)	1660	148	1440	246.8	11.7
Water Quality Monitor (2)	120	8	80	3.6	0.2
	2386	236	1560	268.6	14.3

Original Baseline	INSTALLED WT	INSTALLED VOL	POWER	RESUPPLY WT	RESUPPLY VOL
Waste Storage Tanks (8)	404	53	40	12.1	1.6
Potable Storage Tanks (6)	303	40	20	9.1	1.2
Reprocessed Wash Water Storage Tanks (4)	202	27	20	6.0	0.8
Evaporation Units (2 at 1.47 #/hr rate)	513	42	258	76.3	3.6
Multi-filtration Units (4 at 0.88 #/hr rate)	400	10	60	350.0	7.5
Hyperfiltration Units (2 at 6.58 #/hr rate)	860	74	875	1250.0	115.0
Water Quality Monitor (4)	240	16	160	7.2	0.4
	2922	262	1433	1710.7	129.2

11.0 COMMUNICATIONS AND TRACKING ANALYSES

11.0 COMMUNICATION AND TRACKING ANALYSES

11.1 INTRODUCTION

The Space Operations Center communications analyses emphasized definition of a reference system that could meet the defined requirements. Link analyses were conducted in support of these studies and are reported herein. Also, tracking requirements were estimated for the traffic control radar, and a brief review was conducted of SOC RFI considerations and millimeter-wave communications concepts.

11.2 S-BAND LINK ANALYSIS

11.2.1 S-Band Orbiter-to-SOC Link Analysis (Forward Link)

The Orbiter-to-SOC link analysis is shown in Table 11-1. It assumes the use of the Orbiter's payload interrogator and a compatible, STDN mode SOC transponder. The analysis shows that the link is capable of handling the 2 kbps orbiter data. In order to enable voice communications, the orbiter is required to convert voice to 16 (or 32) kbps data stream. Further, the orbiter must raise its EIRP by approximately 10 dB to 40 dBm.

11.2.2 S-Band SOC-to-MOTV RF Link (Forward Link)

The TDRSS mode, SOC-to-MOTV link analysis is shown in Table 11-2. The example assumes that SOC is the interrogator and the MOTV the transponder. The SOC's EIRP and the propagation range is identical to that used in the SOC-to-Orbiter link analysis. If the sensitivity of the MOTV is equal or greater than that of the transponder, the acquisition and detection of 50 kbps data can be readily accomplished. The data rate is sufficient to handle digital voice and command data.

SOC-349

PARAMETER	NOMINAL VALUES	SOURCE
TYPE LINK	STDN	
FREQUENCY	2092	STDN
RANGE	600 KM	SOC REQ.
ORBITER EIRP	30 DBM	*
L_θ (SOC ANT. POINTING LOSS)	0.5 DB	ESTIMATE
L_P (POLARIZATION LOSS)	0.5 DB	"
L_S (SPACE LOSS)	155.4 DB	CALC.
G_R (SOC ANT. GAIN)	18 DB	HORN
L_C (RECEIVE CKT LOSS)	3 DB	ESTIMATE
REVD PWR (INTO TRANSPONDER)	-111.4 DBM	CALC.
SIGNAL ACQUISITION		
REQUIRED SIGNAL POWER	-120 DBM	MOTOROLA TRANSPONDER (MOST SENSITIVE MODE)
MARGIN	8.6 DB	CALC.
COMMAND PERFORMANCE		
REQUIRED SIGNAL POWER (2 KBPS DATA)	-119.4 DBM	MOTOROLA TRANSPONDER
MARGIN	8.0 DB	CALC.

* JSC 07700 VOL. NO. 14

Table 11-1. S-Band Orbiter-to-SOC Link (Forward Link)

SOC-381

PARAMETER	NOMINAL VALUES	SOURCE
TYPE LINK	TDRSS	STDN 101.2
FREQUENCY (2025-2120 MHz)	2100 MHz	
RANGE	600 Km	EXAMPLE
RF POWER	43 dBm	20 WATT Tx
G_T (SOC ANT. Tx GAIN)	18 dB	HORN
L_C (TRANSMIT CKT LOSS)	3dB	ESTIMATE
L_O (SOC ANT. POINTING LOSS)	0.5 dB	ESTIMATE
L_P (POLARIZATION LOSS)	0.5 dB	ESTIMATE
L_S (SPACE LOSS)	154.5 dB	CALC.
TOTAL RECEIVED POWER	-97.5 dBm	CALC.
SIGNAL ACQUISITION		
REQUIRED SIGNAL POWER	-135.5 dBm	MOTOROLA TRANSPONDER
MARGIN	38 dB	CALC.
DATA PERFORMANCE		
REQUIRED SIGNAL POWER @ 10^{-5} BER 50 KBPS ($-155.6 + 10 \log R_B$)	-108.6 dBm	MOTOROLA TRANSPONDER
MARGIN	11.1 dB	CALC.

Table 11-2. S-Band SOC-to-MOTV RF Link (Forward Link)

11.2.3 S-Band MOTV-to-SOC RF Link (Return Link)

The TDRSS mode MOTV-to-SOC return link analysis, shown in Table 11-3, assumes a 20 watt RF amplifier and a 6 dB conical log spiral antenna for MOTV. The digitized voice and telemetry data is modulated onto the I and Q channels, respectively, in accordance with one of the TDRSS formats.

The analysis shows that the link can be closed with adequate (13 dB) margin.

11.3 KU-BAND LINK ANALYSIS

11.3.1 Ku-Band, TDRSS Forward Link Analysis

The TDRSS forward link analysis is shown in Table 11-4, utilizing the TDRS K-Band Special Access (KSA) mode. The SOC transponder sensitivity is indicative of an "off-the-shelf" type of transponder. The antenna and circuit noise temperature have been assumed to be 290^{0}K. Circuit loss is assumed and the margin is a bare 3 dB. The preamp must be located close to the antenna to minimize any circuit loss. Under the assumed conditions, the required antenna diameters for several data rates are given. At the maximum data rate of 25 Mbps (forward link TV, plus data and voice), the antenna diameter is 18.4 ft.

A 6 dB margin requires that the antenna diameter be increased to 26 ft for the same receiver system noise temperature. Further study is suggested to detail circuit losses and to investigate the feasibility of reducing the system noise temperature.

11.3.2 Ku-Band, TDRSS Return Link Analysis

The TDRSS return link analysis is shown in Table 11-5. The relay satellite is assumed to be in the K-Band Special Access (KSA) mode. The modulation format is consistent with the TDRSS standard as defined in STDN No. 101.2. An initial division of power between the I and the Q channels is 4:1 reflecting the higher data rate in the I channel relative to the Q (though not in the same ratio).

SOC-382

PARAMETERS	NOMINAL VALUES	SOURCE
TYPE LINK	TDRSS	
FREQUENCY (2200 - 2300 MHz)	2250	STDN 101.2
RANGE	600 Km	EXAMPLE
RF POWER	43dBm	D180-26090-4*
G_T (TRANSMIT ANT. GAIN)	6dB	D180-26090-4*
(.8dB OFF BORESIGHT)		
L_{CT} (TRANSMIT CKT LOSS)	3dB	ESTIMATE
L_P (POLARIZATION LOSS)	0.5 dB	ESTIMATE
L_S (SPACE LOSS)	154.7 dB	CALC.
G_R (SOC ANT. GAIN)	18 dB	HORN
L_O (SOC ANT. POINTING LOSS)	0.5 dB	ESTIMATE
L_{CR} (RECEIVE CKT LOSS)	3.0 dB	ESTIMATE
RECEIVED RF PWR (BY RCVR)	-94.7 dBm	CALC.
DATA FORMAT	DGI, MODE1	STDN 101.2
I CHANNEL: 32 KP KBPS VOICE		
Q CHANNEL: 32 KBPS TLM		
REQUIRED E_b/N_o	10.5 dB	10^{-5} BER (COHERENT DET.)
(DETECTION LOSS)	2.5 dB	ESTIMATE
T_E (EFFECTIVE SYSTEM NOISE TEMP)	960^o K	(T_A = 290^o K , NF = 5.2 dB)
N_o (NOISE DENSITY)	-168.8 dBm/Hz	CALC.
REQUIRED RF POWER	-107.7 dBm	CALC.
MARGIN	13 dB	CALC,

* FINAL REPORT' OTV CONCEPT DEFINITION STUDY

Table 11-3. S-Band MOTV-to-SOC RF Link (Return Link)

SOC-342

PARAMETERS	NOMINAL VALUES	SOURCE
FREQUENCY	13.775 GHz	STDN 101.2
NF	6.5 DB	TRW TDRS TRANSP.
EB/NO (ENERGY/BIT, @10^{-5} BER	10.5 DB	MOTOROLA TRANSP.
(TRANSPONDER DETECTION LOSS)	2.5 DB	MOTOROLA TRANSP.
RANGE (200 KM ALT. SOC)	46,000 KM	GEO-TO-LEO
L_Θ (ANT. POINTING LOSS)	0.5 DB	EST.
L_P (POLARIZATION LOSS)	0.5 DB	EST.
L_S (SPACE LOSS)	208.5 DB	CALC.
T_A (ANTENNA NOISE TEMP)	290^oK	ASSUMPTION
T_E (EFFECTIVE SYSTEM NOISE TEMP)	1295^oK	CALC.
TRACKING THRESHOLD	-135.5 DBM	MOTOROLA TRANSP.
M (MARGIN)	3 DB	ASSUMPTION
PT/PC (TOTAL PWR TO PWR IN MD)	0.5 DB	STDN 101.2
ADR (ACHIEVEABLE DATA RATE @ 10^{-5} BER)	50.2 + (9.9-10.5) + G/T = 49.6 + G/T	STDN 101.2 PLUS DELTA E_B/N_O
REQUIRED ANTENNA GAIN/DIAMETER		
a CASE 1, VOICE PLUS HIGH DATA RATE (1 MBPS TOTAL)	41.5 DB/3.7 FT	CALC.
b CASE 2: VOICE PLUS LOW DATA RATE (20 KBPS TOTAL)	24.5 DB/0.3 FT	CALC.
c CASE 3: VOICE, HIGH DATA RATE, TV (25 MBPS TOTAL)	55.5 DB/18.4 FT.	CALC.
d CASE 4: TRACKING	5 DB/CLS	CALC.

Table 11-4. Ku Band (KSA) TDRSS Forward Link Summary

SOC-343

PARAMETERS	NOMINAL VALUES	SOURCE
FREQUENCY	15 GHZ	STDN 101.2
SIGNAL FORMAT	DG-2	STDN 101.2
RANGE	46,000 KM	LEO-TO-GEO
BER (BIT ERROR RATE)	10^{-5}	
L_Θ (SOC ANT. POINTING LOSS)	0.5 DB	ESTIMATE
L_P (POLARIZATION LOSS)	0.5 DB	ESTIMATE
M (MARGIN)	3 DB	ASSUMPTION
L_C (CONSTRAINT LOSS)	0.5 DB	ESTIMATE
P_T/P_I (4 TO 1)	1.3 DB	STDN 101.2
ADR (ACHIEVEABLE DATA RATE, @ 10^{-5} BER – NO CODING)	$25.1 + EIRP + LC + L_\theta) + \Delta E_b/N_o = 22.2 + EIRP$	STDN 101.2 AND L_C, L_θ
REQUIRED EIRP (G_T, P_T)		
a CASE 1: I CHANNEL (50 MBPS) Q CHANNEL (VOICE, DATA, 1 MBPS)	54.8 DBW (56.2 DB, 0.7 WATTS)	18.4 FT DISH (WORST CASE I CHANNEL)
b CASE 2: I CHANNEL (1 MBPS) Q CHANNEL (VOICE - 16 KBPS)	37.8 DBW (56.2 DB, 0.014 W)	18.4 FT DISH (WORST CASE I CHANNEL)
c CASE 3: TRACKING REQUIREMENT	30 DBW (56.2 DB, 6 MW)	18.4 FT DISH STDN 101.2
d CASE 4: CASE 1 WITH 3.7 FT ANTENNA	54.8 DBW (42.3 DB, 17.8 WATTS)	3.7 FT DISH (WORST CASE I CHANNEL)
e CASE 5: CASE 3 WITH 3.7 FT ANTENNA	30 DBW (42.3 DB, 0.6 WATTS)	3.7 FT DISH STDN 101.2

Table 11-5. Ku Band (KSA) TDRSS Return Link Summary

The required EIRPs and the RF transmitter powers at the antenna interface are shown for the two antenna sizes tentatively established by the forward link analysis. It is noted (case 1) that the transmission of 50 Mbps TV plus 1 Mbps of voice/data requires 0.7 watt of RF power with the 18.4 ft dish antenna. The actual transmitter power required should be increased by any circuit losses. The same data rate with the 3.7 ft dish antenna is shown to require 17.8 watts of RF power at the antenna interface.

11.4 DSCS III LINK ANALYSIS

11.4.1 DSCS III as a Relay Satellite

Each of the several DSCS III antennas are permanently dedicated to RF reception or transmission unlike the TDRS whose antennas are capable of full duplex operation. Another difference is in the frequency band. DSCS III utilizes X-Band and UHF as opposed to TDRS's S and Ku-Bands. DSCS III can relay wideband (TV) data in half duplex mode, i.e., TV data can be relayed to SOC or from SOC from/to a ground terminal as shown in Figure 11-1. Low data rate voice/data may be communicated in full duplex. In order to permit full duplex high data rate (TV), another radiating steerable dish antenna is required.

The half duplex TV concept utilizes the antennas as follows:

(1) TV in Forward Link (X-Band)

- DSCS III receives TV/Voice/CMDS signals from ground station via one of two receive horn antennas. DSCS III relays SOC's low data rate voice/TLM to ground station via one of two transmit horns.

- DSCS III relays TV/Voice/CMDS to SOC with the mechanically steered dish antenna. It receives voice/TLM data from SOC with the receive 61-beam MBA (Multi-Beam) antenna.

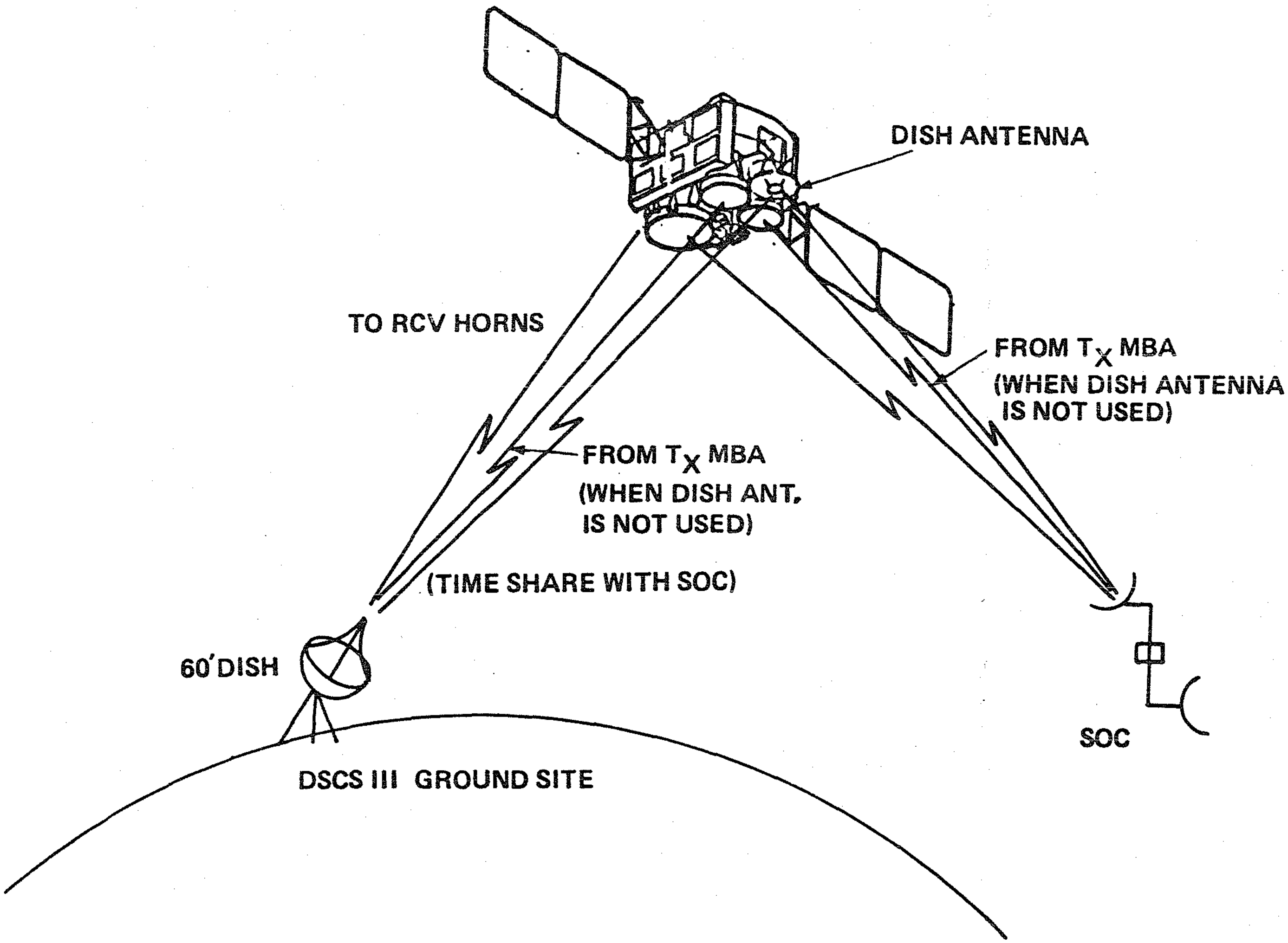


Figure 11-1. DSCS III Communications Antenna Plan

(2) TV in Return Link (X-Band)

- o DSCS III receives TV/Voice/TLM from SOC via the 61-beam receive MBA antenna. It relays the data to ground via the steerable dish antenna, which is now aimed at the ground terminal.

- o DSCS III transmits Voice/CMDS to SOC with the transmitting, 19-beam MBA antenna.

11.4.2 DSCS III-to-SOC Link Analysis (TV Forward Link)

The analysis for the TV/Voice/CMD link from the DSCS III relay satellite to SOC is shown in Table 11-6. The receiving circuit loss is generally included in the system noise temperature, but it is separated here to call attention to its value. The link analysis makes the reasonable assumption that the signal quality being relayed by DSCS III is flawless.

The analysis shows that a 50 ft diameter antenna is required to close the link. Note that if the same margin and circuit loss conditions are assumed as with TDRSS, i.e., 3 dB margin and 0 dB circuit loss, the required antenna diameter will be 25 ft.

11.4.3 SOC-to-DSCS III Link Analysis (TV Return Link)

The TV/Voice/TLM return link, SOC-to-DSCS III analysis is shown in Table 11-7. A 250-watt transmitter power is assumed along with a 3 dB circuit loss and a 50 ft diameter antenna, determined from the forward link. Also assumed is a 50 Mbps digitized TV, voice and telemetry data rate together with the use of the MBA narrow-coverage receive-only DSCS III antenna. The signal-to-noise ratio in the data bandwidth of 50 MHz is shown to be 20.7 dB. The signal is translated and transmitted to the ground station at another frequency in X-band. The analysis for the DSCS III-to-Ground link is shown in a separate table.

SOC-399

PARAMETERS	NOMINAL VALUES	SOURCE
TYPE DSCS III ANTENNA	DISH	
FREQUENCY	7365 MHz	CHANNEL 2 (OR 1)
RANGE	46,000 Km	GEO-TO-LEO
EIRP	74 dBm	DSCS DISH, 40W TX
L_P (POLARIZATION LOSS)	0.5 dB	ESTIMATE
L_S (SPACE LOSS)	203.1 dB	CALC.
L_C (RECEIVE CKT LOSS)	3 dB	ESTIMATE
M (MARGIN REQUIRED)	6 dB	ESTIMATE
T_a (ANTENNA NOISE TEMP)	290^oK	ESTIMATE
NF	6 dB	ESTIMATE
T_e (RECEIVE SYSTEM NOISE T)	1154^oK	CALC.
REQUIRED E_b/NO	10.5 dB	COHERENT DETECTION
δ(DETECTION LOSS)	2.5 dB	ESTIMATE
B (DATA RATE)	25 Mbps	TV, VOICE, DATA
N (NOISE POWER)	-94 dBm	KT_eB
REQUIRED RF POWER	-81 dBm	$N + E_b/NO + \delta$
RECEIVED RF POWER	$-139.6 + G_r$ dBm	CALC.
G_r (REQUIRED ANT. GAIN)	58.6 dB	CALC.
REQUIRED ANT. DIA.	49.5 FT.	CALC.
	50 FT.	

Table 11-6. DSCS III-to-SOC Link (TV Forward Link)

SOC-390

PARAMETERS	NOMINAL VALUES	SOURCE
TYPE DSCS III ANTENNA	MBA	
FREQUENCY	8187.5 MHz	CHANNEL 3 RECEIVE
RANGE	46000 Km	GEO-TO-LEO
P_T (TRANSMIT POWER)	54 dBm	250 W TX
SOC ANTENNA GAIN	59.6 dB	(50 FT. DISH)
L_P (POLARIZATION LOSS)	0.5 dB	ESTIMATE
L_S (SPACE LOSS)	204.0 dB	CALC.
L_C (TRANS. CKT LOSS)	3.0 dB	ESTIMATE
M (REQUIRED MARGIN)	6.0 dB	ESTIMATE
P_R (POWER RECEIVED BY DSCS)	-99.9 dBm	CALC. (0 dB ANT.)
B (DATA RATE)	77 dB-Hz	50 Mbps
G/T	-1 dB/K^o	MBA NARROW COVERAGE
k (BOLTZMANN'S CONSTANT)	-198.6 dBm/k^o	
SNR (SIGNAL-TO-NOISE-RATIO)	20.7 dB	P_R + G/T –k –B

Table 11-7. SOC-to-DSCS III Link (TV Return Link)

11.4.4 DSCS III-to-Ground Link (TV Return Link)

The TV/Voice/TLM data received from SOC is translated in frequency and transmitted to the ground station. A previous table shows that the signal to be relayed has a signal-to-noise ratio of 20.7 dB. The analysis, see Table 11-8, shows that the translated signal plus noise-to-ground receiver noise is 6.7 dB, which is unacceptable. By increasing the DSCS III EIRP by 10 dB, through the use of the dish antenna on a time-shared-basis, the S+N to N ration is 16.7 dB and the signal to total noise ratio is 15.2 dB. This is more than adequate to achieve a bit error rate of 10^{-5}.

11.5 SURVEILLANCE RADAR REQUIREMENTS

A surveillance radar is included in the SOC tracking and communications subsystem. The function of this radar is to monitor the location and approach or departure of space vehicles in the general vicinity of SOC. Such vehicles include shuttle orbiters, OTVs, co-orbiting spacecraft, and SOC-based systems with free-flight capability such as teleoperators.

Early in the present study, the feasibility of using the surveillance radar to detect approaching space debris and give collision-avoidance warning was investigated. It was concluded that the power and gain requirements to enable adequate collision warning were entirely impractical. Instead, the SOC pressure hulls were made thick enough that a penetration due to a debris collision will be extremely unlikely.

Vehicle approach and departure paths were analyzed in order to estimate radar viewing requirements. It is a design goal to provide a radar system with four-pi viewing capability. This is approximated in the reference configuration by locating millimeter-wave radar antennas near the service module docking ports. One dish views the forward hemisphere; the other views the aft hemisphere. Presence of a shuttle docked to either of these ports will block viewing, but the shuttle may be docked to one of the docking tunnel ports, "tail away", as shown in Figure 11-2, to minimize blockage. This docking position also minimizes:

SOC-389

PARAMETERS	NOMINAL VALUES	SOURCE
TYPE DSCS III ANTENNA	MBA	
FREQUENCY	7462.5 MHz	CHANNEL 3
RANGE	46000 Km	GEO-TO-GROUND
EIRP	64 dBm	NARROW COVERAGE, 10 WATT
L_P (POLARIZATION LOSS)	0.5 dB	ESTIMATE
L_S (SPACE LOSS)	203.2 dB	CALC.
L_C (RECEIVE CKT LOSS)	3.0 dB	ESTIMATE
M (MARGIN REQUIRED)	6.0 dB	ESTIMATE
B (DATA) RATE	77 dB-Hz	50 Mbps
G_R (ANTENNA GAIN)	60.4 dB	60 FT. DISH
P_R (POWER RECEIVED)	-88.3 dBm	CALC.
T_C (SYSTEM NOISE TEMP)	417°K	ESTIMATE (2 dB NF)
NOISE POWER	-95.0 dBm	CALC.
SNR (TRANSLATED (S+N) TO RCVR N)	6.7 dB	CALC.
DSCS III DISH ANTENNA		
(S+N)/N WITH DISH ANTENNA	16.7 dB	EIRP = 44 dB
SNR (SIGNAL-TO-NOISE RATIO) RATIO)	15.2 dB	
REQUIRED SNR	13.0 dB	(Eb/NO = 10.5 dB, δ= 25 dB)
EXCESS MARGIN (OVER 6 DB)	2.2 dB	CALC.

Table 11-8. DSCS III-to-Ground Link (TV Return Link)

SOC-602

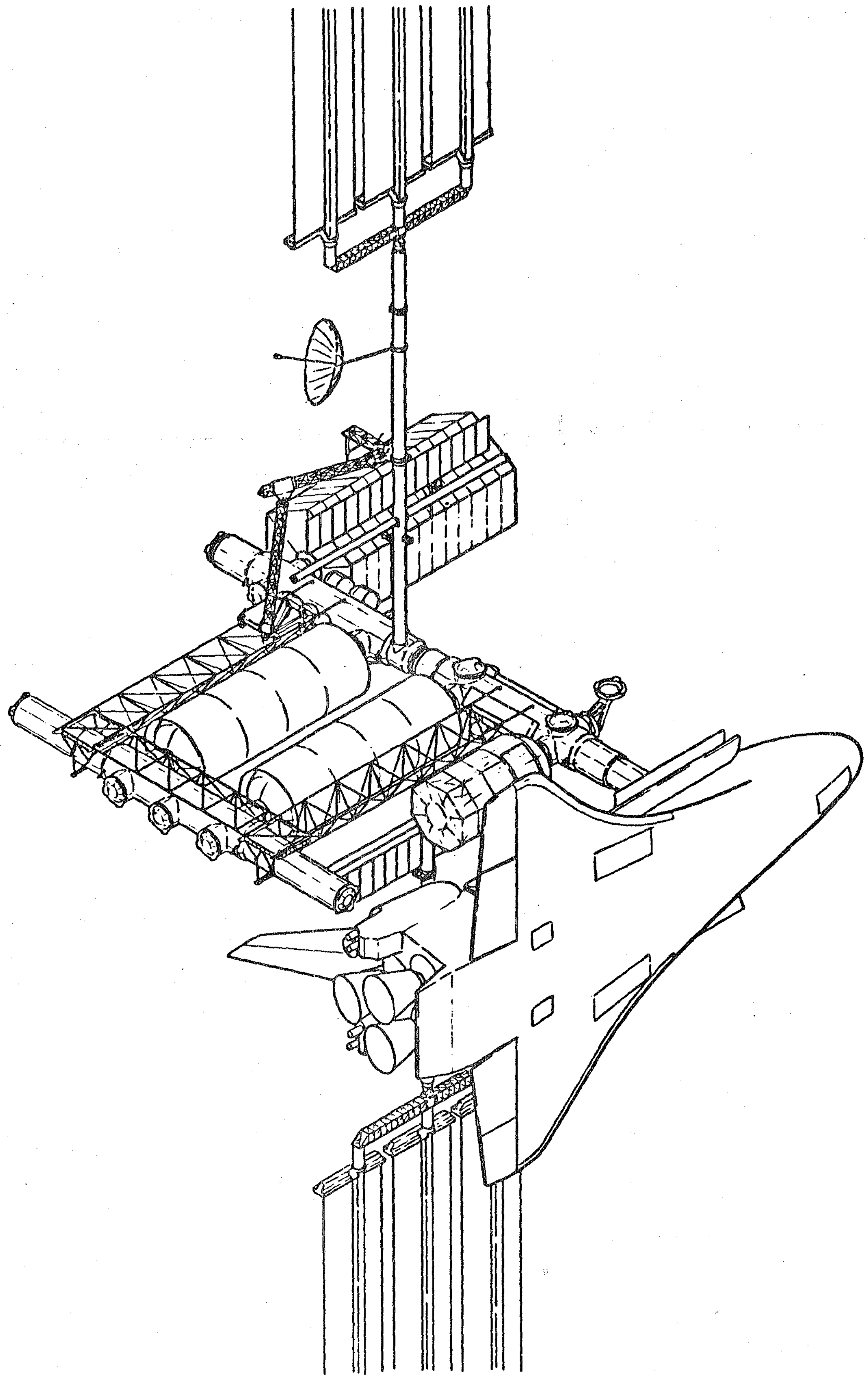


Figure 11-2. Antenna Coverage is Minimized When the Orbiter is Docked at Any of the Four Docking Ports in the Orientation Shown Here (Tail Toward the Earth)

(1) plume impingement on the SOC from shuttle thrusters, and (2) attitude offsets required to null gravity gradients.

Representative shuttle and OTV approach paths are shown in Figure 11-3. As may be seen, the shuttle is expected to approach from below and behind the SOC, passing below, and terminating in front, while the OTV normally will approach from above and in front. Co-orbiting satellites will normally approach from in front of or behind the SOC.

A preliminary estimate of radar requirements is as follows:

(1) Provide sufficient power and gain to skin-track the shuttle and OTVs at up to 300 km range. Provide range and range-rate information as well as direction.

(2) Provide sufficient power and gain to skin-track a co-orbiting satellite at up to 100 km range. The cross-section may be assumed as 1000 square cm. Provide range and range-rate information as well as direction.

(3) Provide radar installations at locations normally able to scan the volumes shown in Figure 11-3. Under extraordinary circumstances, such as presence of a large construction artifact that blocks radar view, operational work-arounds (e.g. attitude slewing) can be used as required to provide essential surveillance. Presence of a shuttle orbiter should not interfere with surveillance capability.

(4) Provide a scan rate such that space vehicles within the field of view and within range will be acquired within one minute without prior knowledge of presence or location. Search scan shall be maintained while an approaching or departing vehicle is being tracked.

(5) Provide position updates on tracked vehicles every ten seconds and every second for any vehicle within 10 km of the SOC.

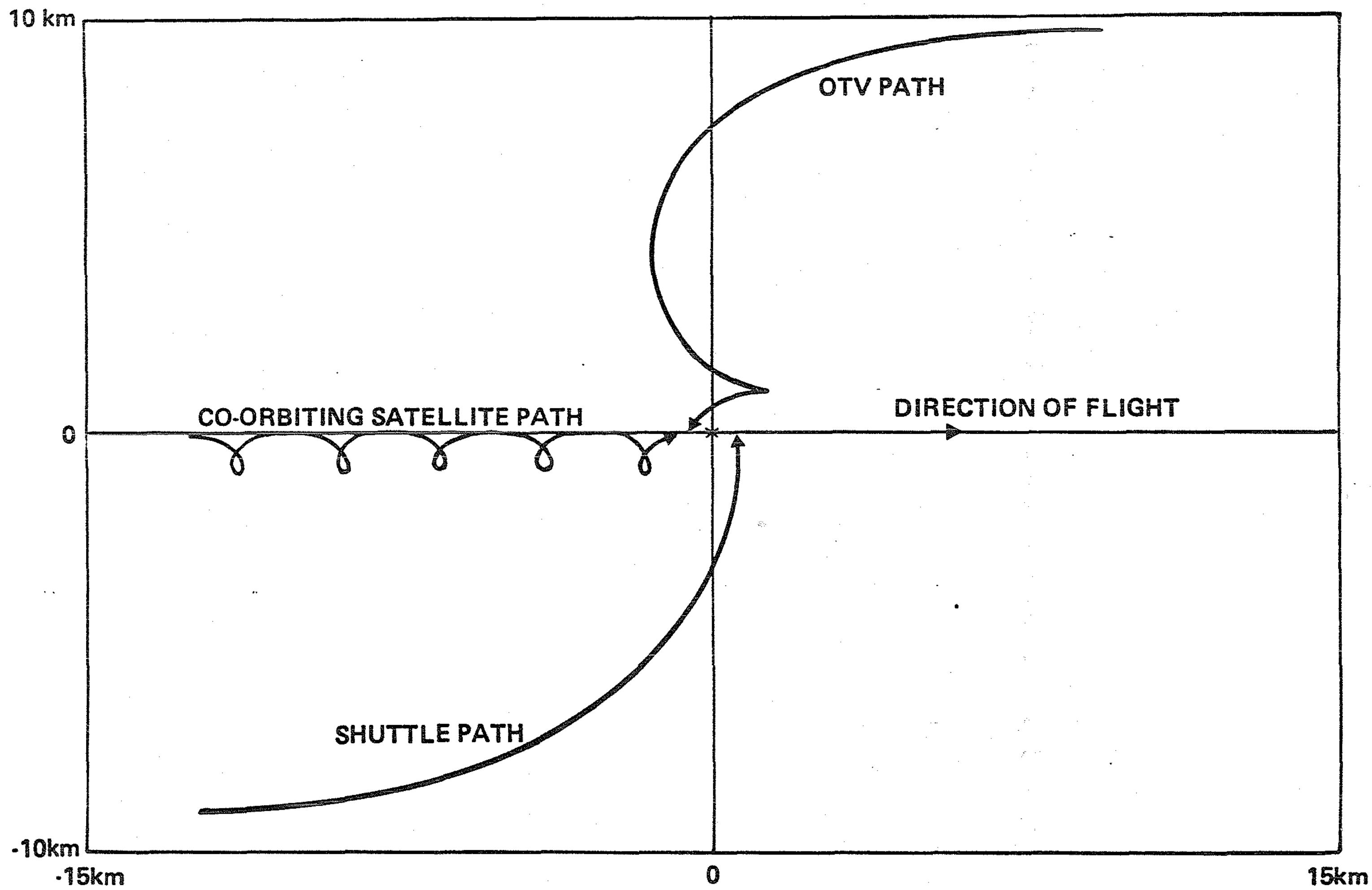


Figure 11-3. Approach Paths Relative to SOC
(Note: Will be replaced with computer-generated trajectories)

(6) Provide automatic annunciation at the main controls and display stations whenever a space vehicle is acquired. Whenever the positional data for an approaching vehicle indicates it will approach within 10 km of the SOC, activate an appropriate caution and warning annunciation unless the presence of the vehicle has been acknowledged by the communications operator.

11.6 RATIONALE FOR MILLIMETER WAIVE COMMUNICATIONS

The present NASA frequency standards at S and Ku-bands have serious limitations for SOC communications in terms of bandwidth, frequency assignment availability and RFI. A case is presented for shifting the operating frequency of SOC's intersatellite communications to certain segments of the mm-wave spectrum.

11.6.1 Deficiencies of Current Frequencies

A summary of NASA's current frequency usage plan is shown in Table 11-9. It embodies the time-honored S-band GSTDN and the soon-to-be available S/Ku-band TDRSS concepts. Frequency plans involving intersatellite communications, exclusive of relay satellites, are not specifically addressed, but use the S-band frequency segments shown under GSTDN. Satellite communications via the "bent pipe" relay is scheduled to operate in Ku-band and S-band as shown in the table. The S-band options include the Special Access (SA) and the Multiple Access (MA) modes.

The total S-band allocation is only 90 and 100 MHz in the forward and return direction respectively. Furthermore, the frequency assignments must be shared with the military space programs. Therefore, it is questionable that SOC can obtain several duplex S-band channels to support the several simultaneous SOC-User satellite/free flyers communications. In addition, the projected growth requirements for intersatellite communications may well call for the use of the total S-band allocation for a single duplex link.

The bandwidth allotted for the TDRSS relay link at Ku-band is not deemed to be a major deficiency. The return link capacity (300 mbps) is adequate for the initial and growth versions of SOC. The forward link, however, is restricted to

RF Link	TDRSS			GSTDN (S-Band)	Authorization
	Ku-Band	S-Band(SA)	S-Band(MA)		
Gd-Relay Sat.					
Uplink(Forward)	14.6-15.25				1979 WARC
Downlink(Return)	13.4-14.05				(13.4-14.2,14.5-15.35)
Relay Sat.-Satellite					
Forward	13.775				1979 WARC
		2025-2120	2106.4		NTIA
Return	15.0034				1979 WARC
		2200-2300	2287.5		NTIA
Gd-Satellite					
Uplink				2025-2120	NTIA
Downlink				2200-2300	NTIA
Deep Space (Gd-Satellite)				2290-2300	1979 WARC

Table 11-9. Satellite Frequency Plan

25 mbps. This will force the use of less powerful error correction algorithms than that presumed in Figure 11-4, and will preclude the growth to NTSC Color TV.

Besides the inadequacy of bandwidth, particularly at S-band, the allocated spectrum is subject to severe RFI caused by terrestrial emitters from the international community. This is a particularly severe limitation to the TDRSS Relay Satellite since the satellite receivers are exposed to ground emitters from large areas.

11.6.2 Potential Frequency Bands

For the SOC-User satellite and the SOC-Relay satellite links, it appears reasonable to utilize the high atmospheric absorption frequencies shown in Figure 11-4 since competition with terrestrial users are minimized along with the potential for RFI. The 1979 WARC has assigned frequencies for intersatellite communications, as shown in Table 11-10.

From the atmospheric attentuation point of view, the frequency bands 59-64 GHz and above are viable. From the near-term hardware state-of-the-art considerations, the upper band should be limited to 116-134 GHz. Hence, the potential MM-Wave frequency segments are 59-64 and/or 116-134 GHz.

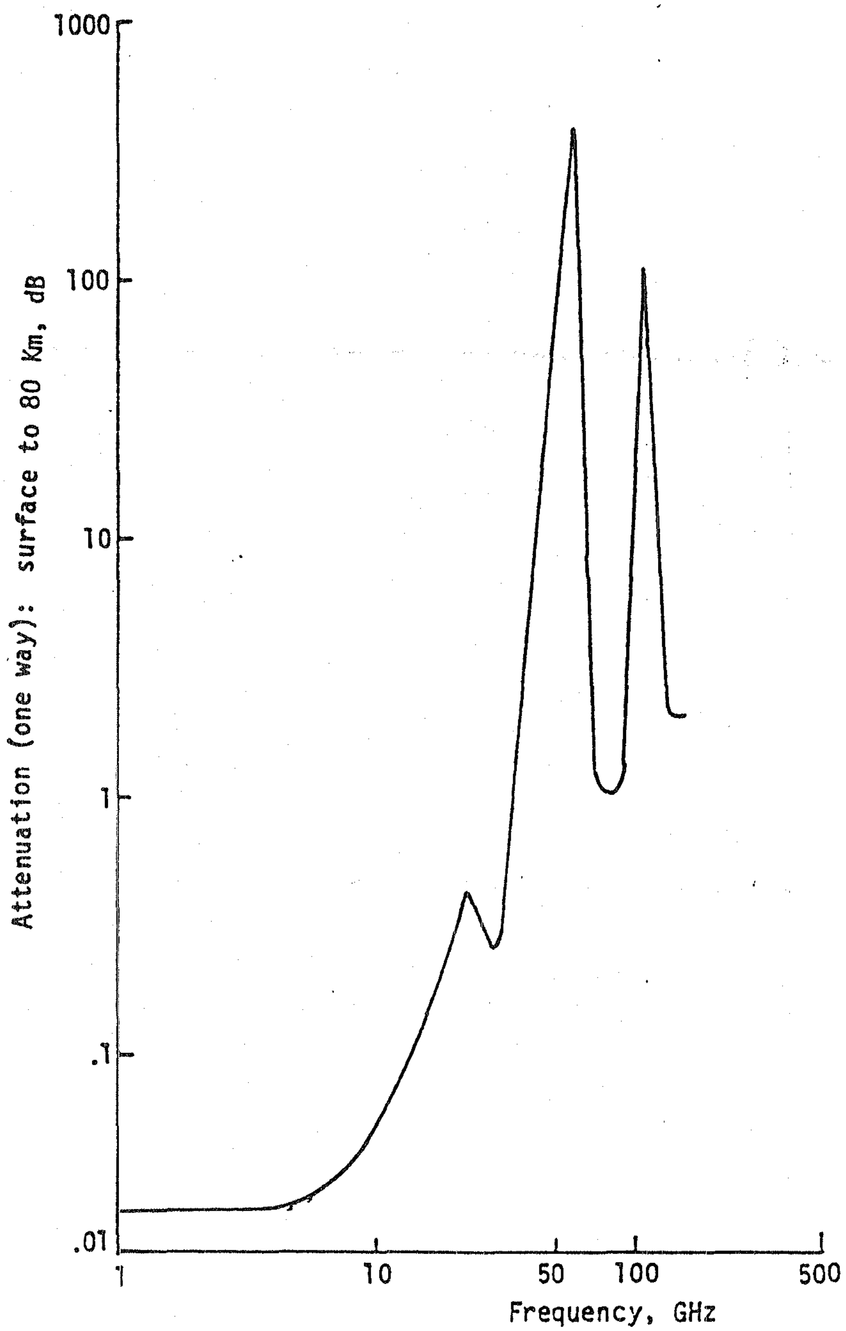


Figure 11-4. Atmospheric Attenuation vs. Frequency at Zenith

Frequency		Bandwidth (GHz)
Band	Allocation(GHz)	
K (18-26.5)	22.55 - 23.55	1
Ka(26.5-40)	32 - 33	1
U (40-60)	54.25 - 58.2	3.95
V (50-75)	59 - 64	5
F (90-140)	116 - 134	18
G (170-220)	170 - 182	12
G (170-220)	185 - 190	5

Table 11-10. 1979 WARC Intersatellite Service Allocations

12.0 STRUCTURAL DYNAMICS ANALYSIS

12.0 STRUCTURAL DYNAMICS ANALYSIS

12.1 PROBLEM

Structural flexibility must be considered in the design of the SOC and its control system. Structural dynamics contributes to the determination of structural loads and produces a destabilizing influence on the control system. With large flexible solar arrays, the first resonant frequency of the SOC will be lower than 0.1 Hz and there will be many modes with frequencies below 1.0 Hz. These resonances must be identified for use in control system design and to determine the effects of structures/control interaction.

12.2 OBJECTIVE

The objective of this study is to determine the structural dynamic characteristics of the SOC at various stages of its build-up. These configurations range from a minimum capability configuration, through the baseline configuration to an extended capability configuration.

A second objective is to determine the dynamic response of the baseline SOC configuration to a transient forcing function representative of the transient produced by the docking of the Orbiter.

12.3 APPROACH

The dynamic analysis was performed using finite element models created using NASTRAN. With the exception of the solar arrays, solar array booms, construction facility piers, and the Rockwell Engineering Test and Verification Platform (ETVP), all modules were modeled as rigid bodies. Module and interface flexibility was lumped at the interfaces between modules. Details of the finite element representations are presented in Section 12.4.1.

Mode shapes and frequencies were calculated for five SOC configurations which range from a half-SOC configuration to an extended mission configuration. The modules which form each of the five configurations are shown in Table 12-1. Section 12.4.2 discusses the dynamic characteristics calculated for each of the five configurations.

The finite element model of configuration 2 was used to find the dynamic response of the SOC to a transient input representing the disturbance resulting from Orbiter docking. A discussion of the assumptions used and the results from this analysis is presented in Section 12.4.3.

12.4 DYNAMIC ANALYSIS

The five configurations for which dynamic characteristics were calculated are:

Configuration 1: SOC baseline
Configuration 2: SOC baseline plus Orbiter and OTV
Configuration 3: SOC baseline plus Orbiter, 2 propellant tanks, 2 OTV's, ETVP, storage facility and extended construction facility piers
Configuration 4: Half-SOC
Configuration 5: Half-SOC plus Orbiter

and are shown in Figures 12-1 through 12-4.

Details of the analyses to determine the dynamic characteristics of each of the SOC configurations and the dynamic response of configuration 2 to a docking transient are given in the following sections.

12.4.1 MODELING

Finite element stick models of each of the SOC component modules were developed using the NASTRAN structural analyzer. Flexible representations of the solar array assemblies, construction facility piers and the ETVP were developed while rigid body representations of the each of the other modules were used. Preliminary estimates for the mass and inertia of SOC modules were used in the analysis. These data are tabulated in Table 12-2. Module masses for configurations 4 and 5 were modified as indicated to reflect updated mass estimates. Mass data for the ETVP were obtained from Reference 1.

The finite element models created for each module are shown in Figure 12-5. NASTRAN rigid elements (RBE's) were used to model the rigid SOC modules with the mass concentrated at the mass center. The flexibility of each module was "lumped" at the interface between modules. A representative module stiffness was determined for each module and added (in series) to a stiffness which was calculated to represent the docking tunnels and interfaces. NASTRAN CELAS elements were used to model these stiffnesses.

The flexible modules (solar arrays, construction facility piers, and ETVP were modeled using NASTRAN CBAR elements. Each of the two solar array assemblies consists of a solar array boom with attached RCS arms, a solar array support truss and three deployable solar arrays. The outboard ends of the three solar arrays are tied together to prevent interference during dynamic events. The solar array mass was concentrated at 10 places on each array assembly (3 degrees of freedom each) as shown in Figure 12-5(d). Stiffness of each solar array was calculated to give a primary bending frequency of approximately 0.04 Hz. Preliminary sizing of the solar array booms and construction

Table 12-1: Configuration Summary

	MODULE	CONFIGURATION 1	2	3	4	5
SM-1	Service Module #1	√	√	√	√	√
SA-1	Solar array #1	√	√	√	√	√
SM-2	Service Module #2	√	√	√		
SA-2	Solar array #2	√	√	√		
HM-1	Habitat Module #1	√	√	√	√	√
HM-2	Habitat Module #2	√	√	√		
DM	Docking Module	√	√	√		
CFP	Construction Facility Piers	√	√	√[2]		
HG-1	OTV Hangar #1	√	√	√		
HG-2	OTV Hangar #2	√	√	√		
LM	Logistics Module	√	√	√		
ORB	Orbiter (full)		√	√		√
OTV-1	OTV-1 (full)		√[1]	√[3]		
OTV-2	OTV-2 (full)			√[3]		
PT-1	Propellant Tank #1 (full)			√		
PT-2	Propellant Tank #2 (full)			√		
ETVP	ETVP			√		
SF	Storage Facility			√		
LP	Logistics Pallet				√	√
OA	Modified Orbiter Airlock				√	√

[1] inside Hangar #1
[2] with extension
[3] attached to ETVP

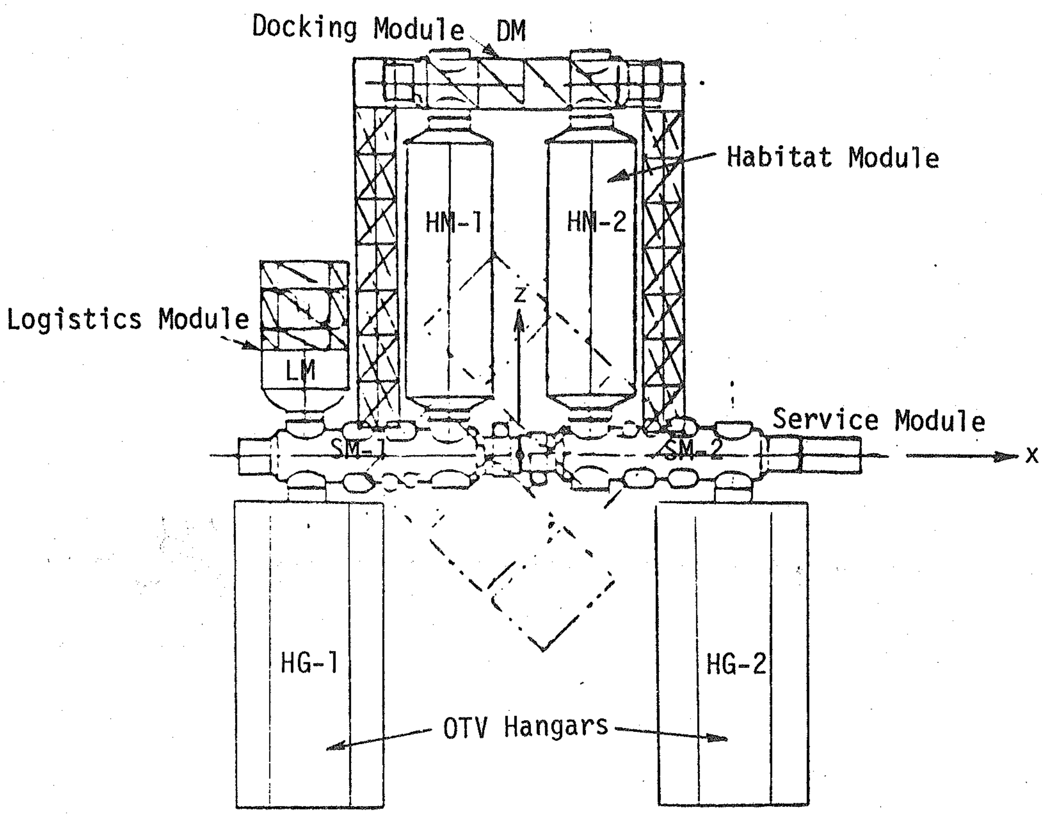


Figure 12-1: SOC Configuration 1

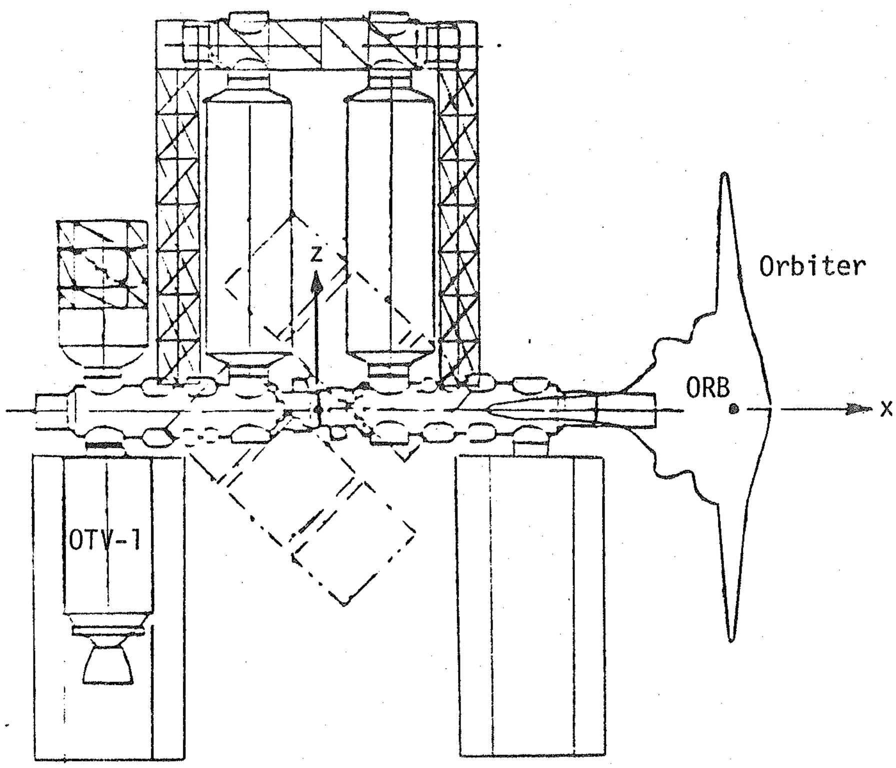


Figure 12-2: SOC Configuration 2

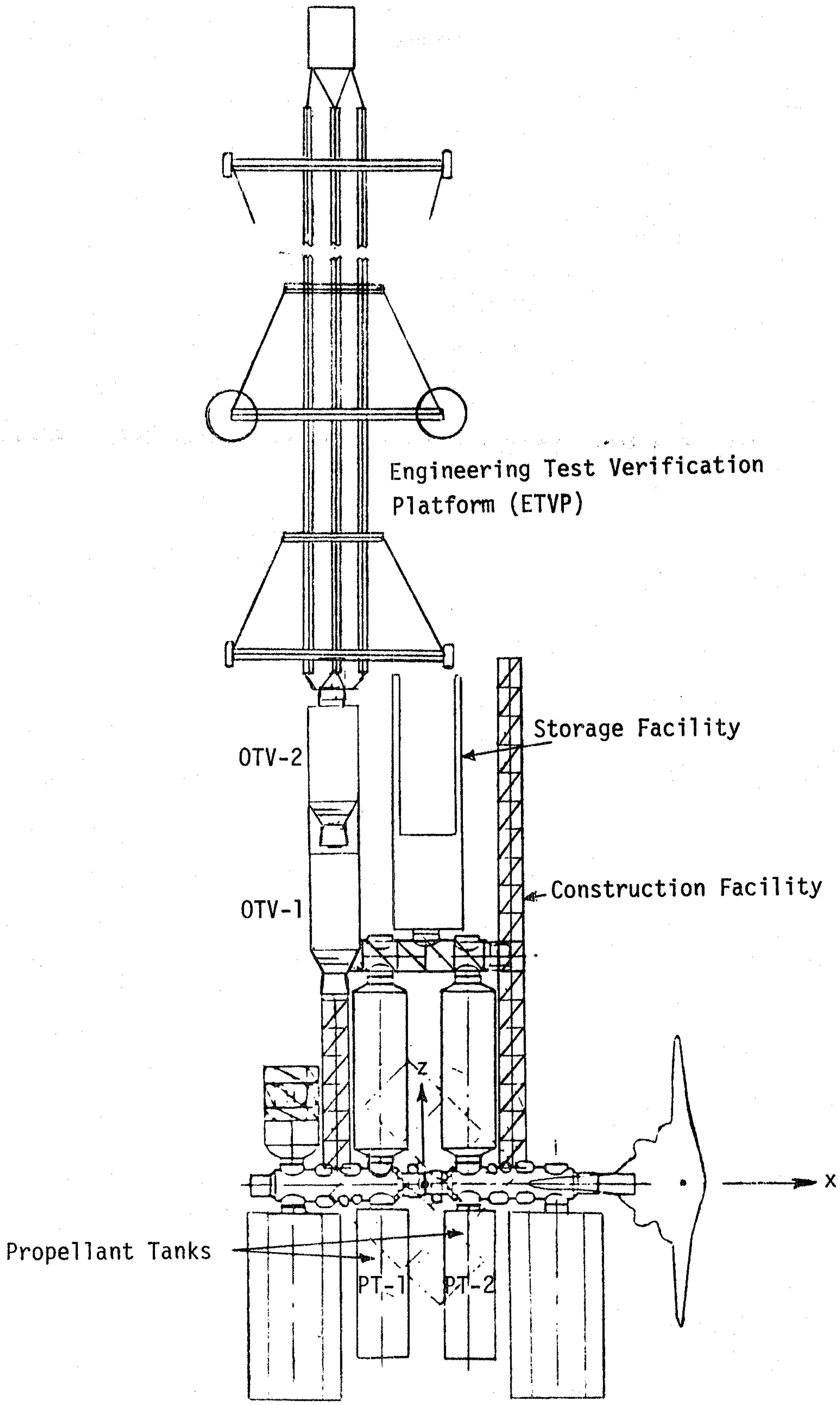


Figure 12-3: SOC Configuration 3

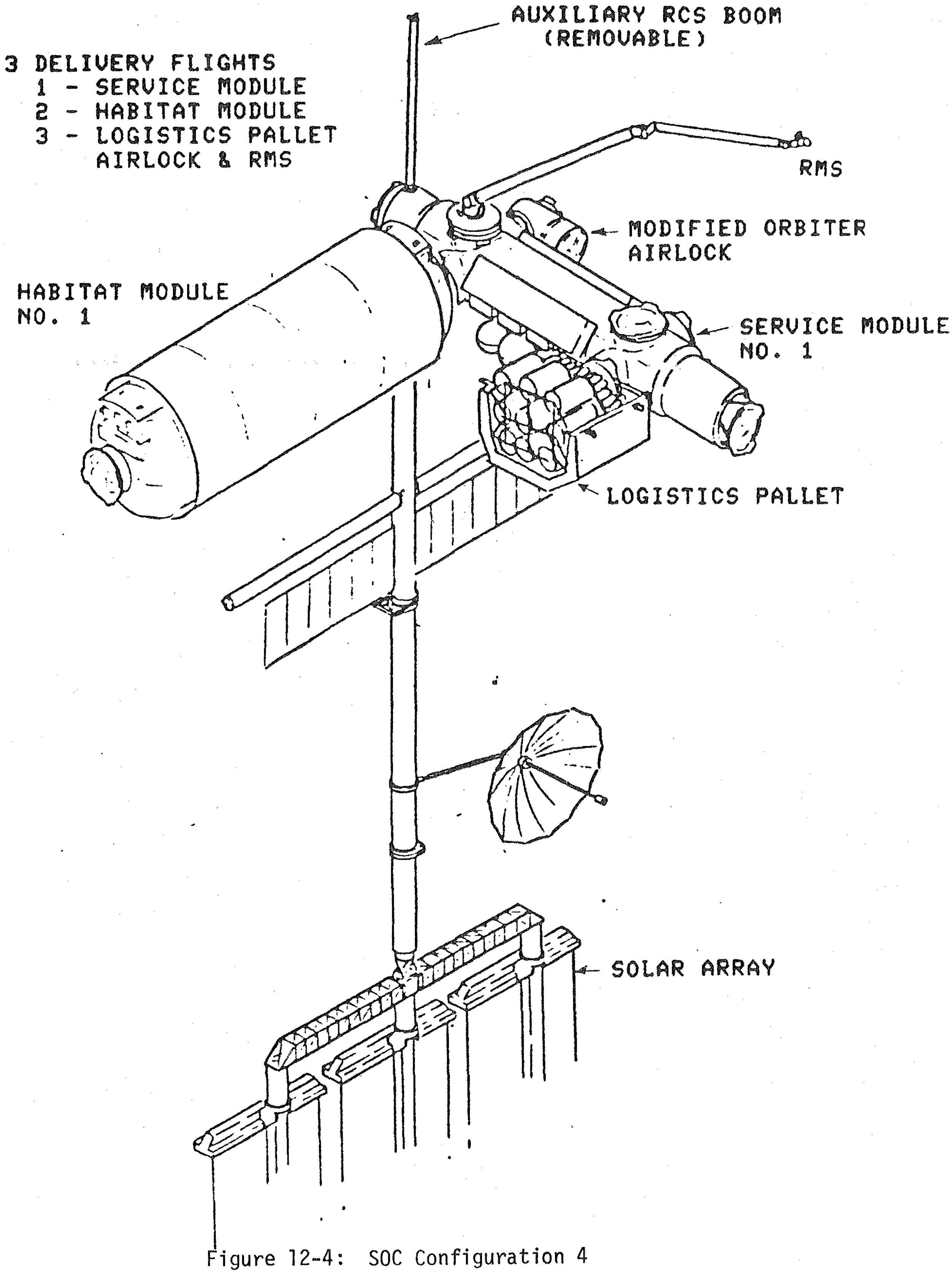
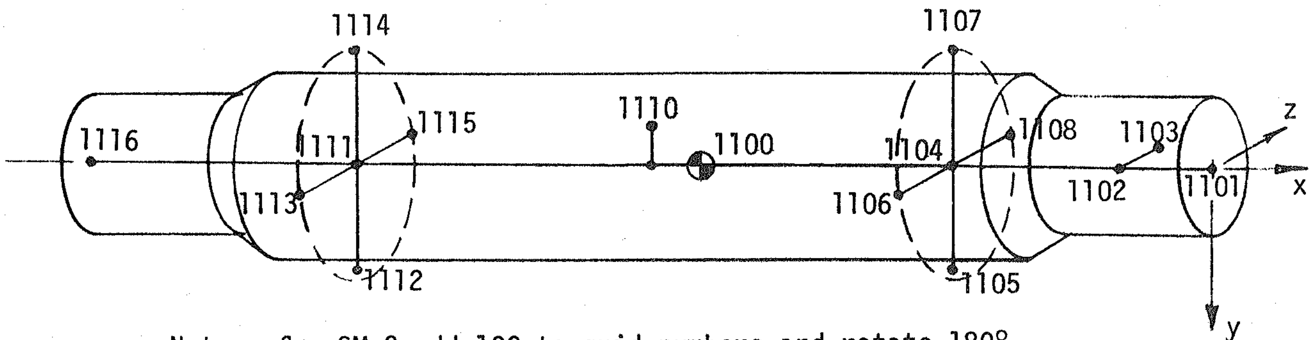


Figure 12-4: SOC Configuration 4

Table 12-2: SOC Module Mass Properties

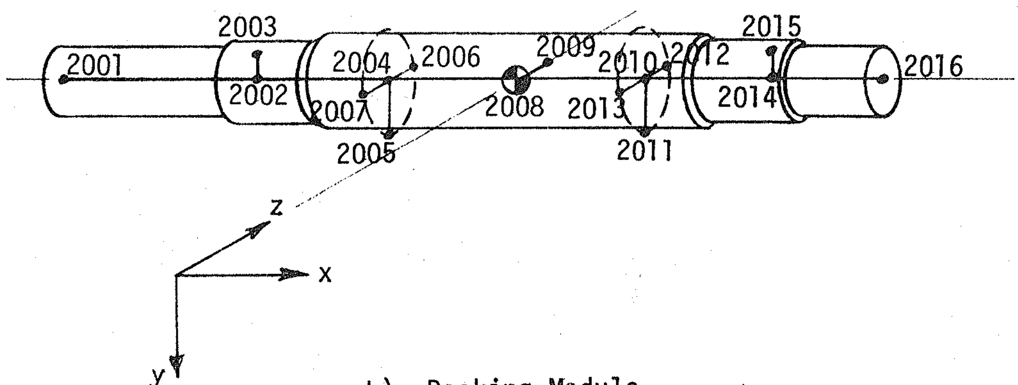
Module	Mass (kg)	Center of Mass (m) x	y	z	Moments of Inertia (kg-m) Ixx	Iyy	Izz
SM-1	14650.	-6.3	0.	0.	20100.	326700.	326700.
SM-1	19175.	-7.6	0.	0.	26100.	377400.	377400.
SA-1	1481.	distributed					
SM-2	14650.	6.3	0.	0.	20100.	326700.	326700.
SA-2	1481.	distributed					
HM-1	24110.	-3.5	0.	10.54	534800.	534800.	61000.
HM-1	18690.	-3.5	0.	7.551	413000.	413000.	44100.
HM-2	24110.	3.5	0.	10.54	534800.	534800.	61000.
DM	9100.	0.	0.	18.12	12500.	233500.	233500.
HG-1	6800.	-10.5	0.	-9.575	207400.	207400.	124600.
HG-2	6800.	10.5	0.	-9.575	207400.	207400.	124600.
LM	8800.	-10.5	0.	5.575	58400.	58400.	23000.
ORB	97670.	18.19	-12.11	0.	7.786E6	1.072E6	7.493E6
OTV	38650.	-7.0	-10.5	-8.66	695200.	695200.	97000.
OTV-1	38650.	-7.0	-4.36	15.89	695200.	695200.	97000.
OTV-2	38650.	-7.0	-4.36	30.06	695200.	695200.	97000.
PT-1	52000.	-3.5	0.	-9.075	1.04E6	1.04E6	1.305E5
PT-2	52000.	3.5	0.	-9.075	1.04E6	1.04E6	1.305E5
ETVP	38806.	distributed					
SF	2275.	0.	0.	26.12	180900.	193500.	13100.
LP	2800.	-10.5	0.41	2.58	2600.	3900.	4000.
OA	227.	added to SM mass					

* SOC coordinate system
** In SOC coordinate directions, about module mass centers
*** Modified for Configurations 4 and 5

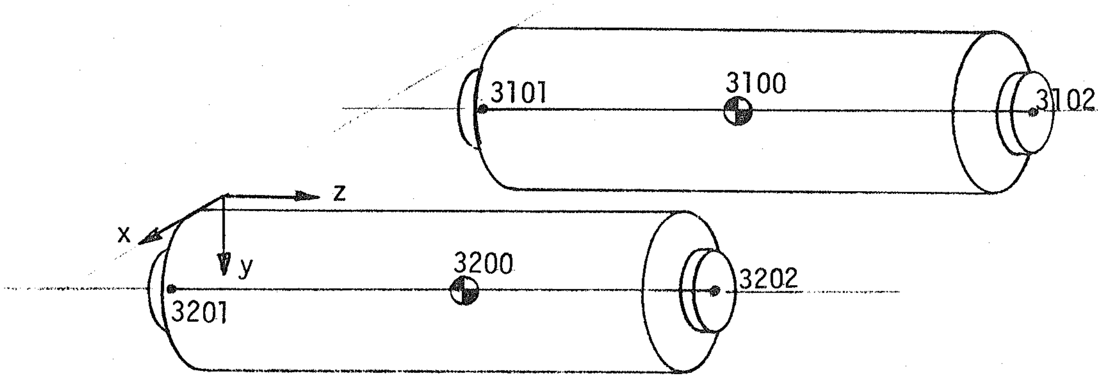


Note: for SM-2 add 100 to grid numbers and rotate 180° about y-axis

a) Service Module, SM-1

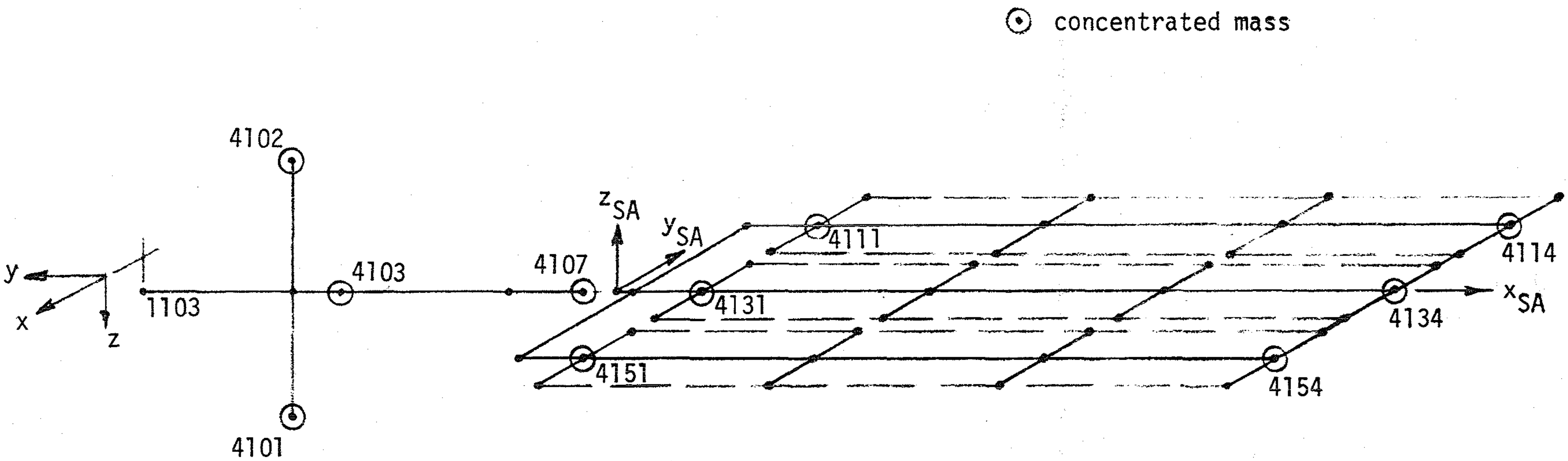


b) Docking Module



c) Habitat Modules

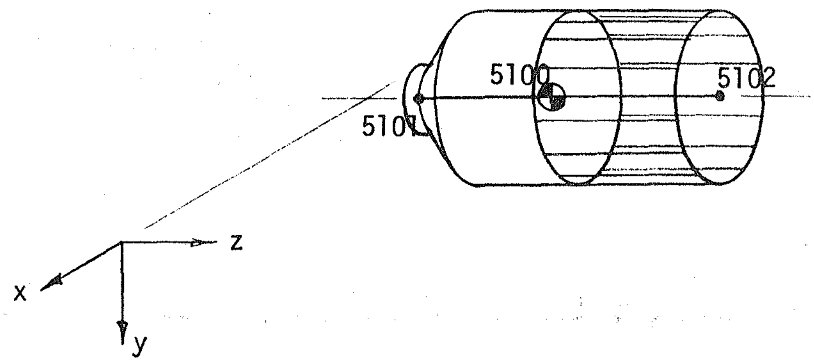
Figure 12-5: NASTRAN Finite Element Models



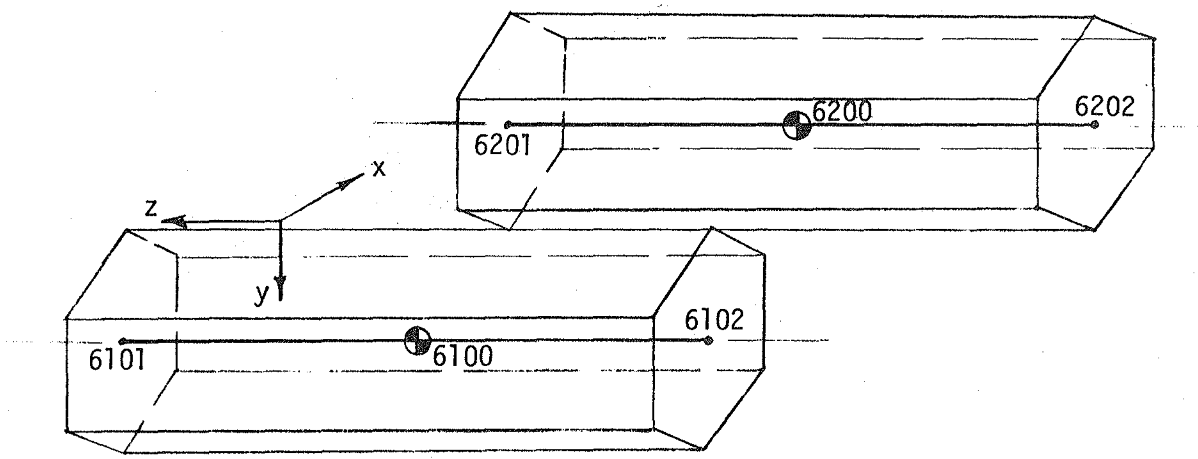
Note: for SA-2 add 100 to grid point no's and rotate 180 degrees about x axis

d) Solar Array, SA-1

Figure 12-5: NASTRAN Finite Element Models (cont'd)

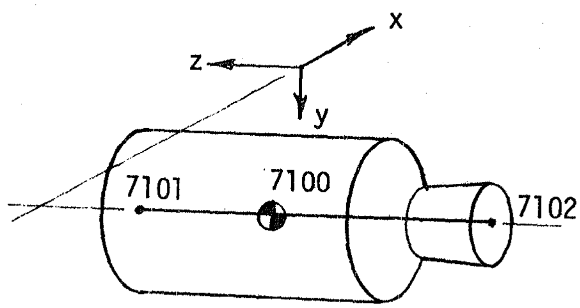


e) Logistics Module

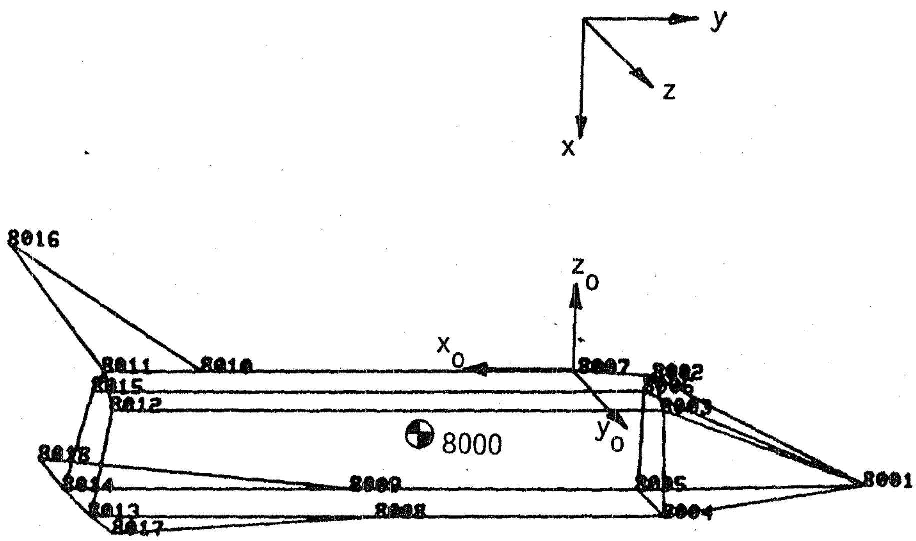


f) OTV Hangars

Figure 12-5: NASTRAN Finite Element Models (cont'd)

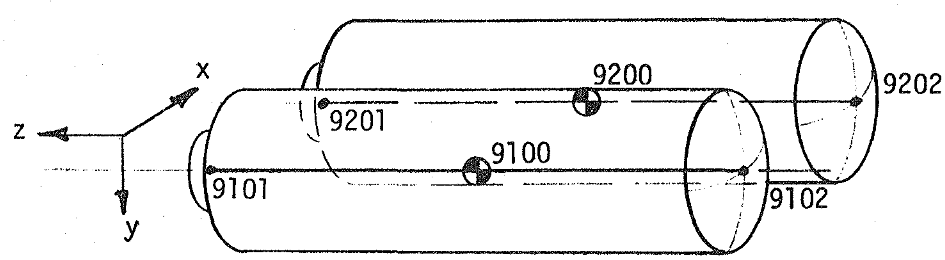


g) Orbital Transfer Vehicle (OTV)

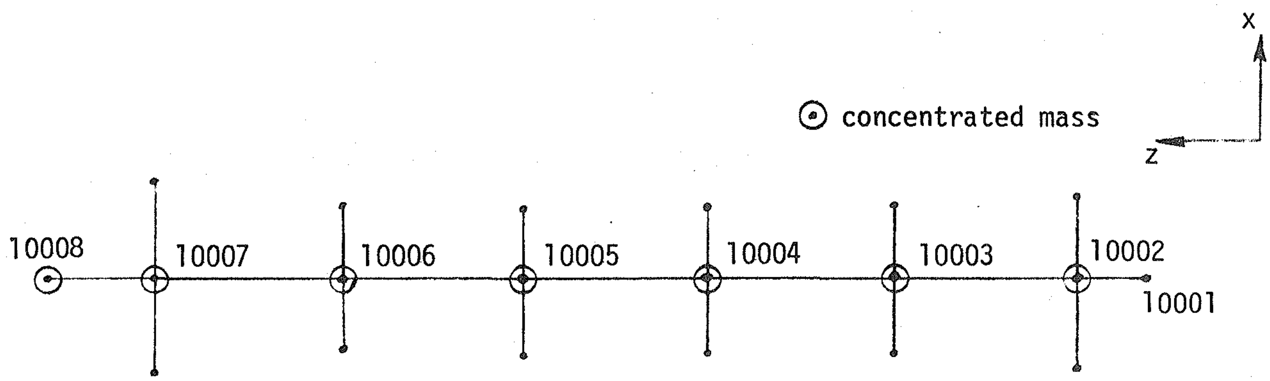


h) Orbiter

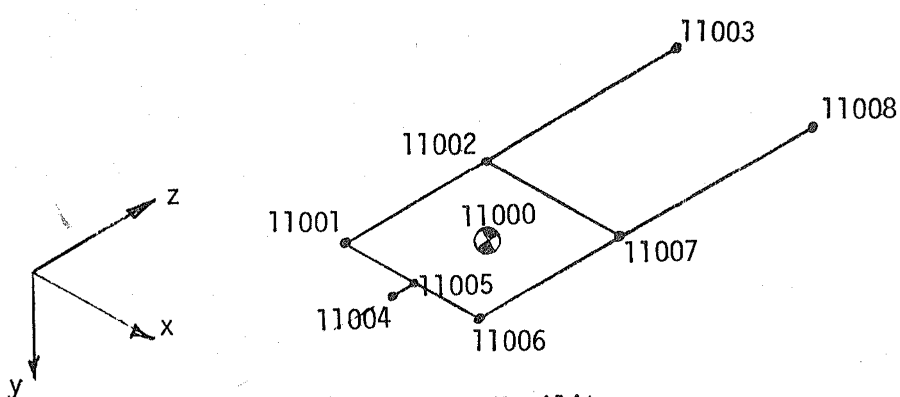
Figure 12-5: NASTRAN Finite Element Models (cont'd)



i) Propellant Tanks



j) Engineering Test Verification Platform (ETVP)



k) Storage Facility

Figure 12-5: NASTRAN Finite Element Models (cont'd)

facility piers were used to define their stiffnesses. Reference 1 data were used to calculate equivalent stiffnesses for the ETVP. The mass of the ETVP was lumped at 7 locations along its centerline (6 degrees of freedom each).

Figure 12-6 shows the undeformed stick model representations of configurations 1 through 3 used in the dynamic analysis. The models for configurations 4 and 5 are shown in Figure 12-7.

12.4.2 DYNAMIC CHARACTERISTICS

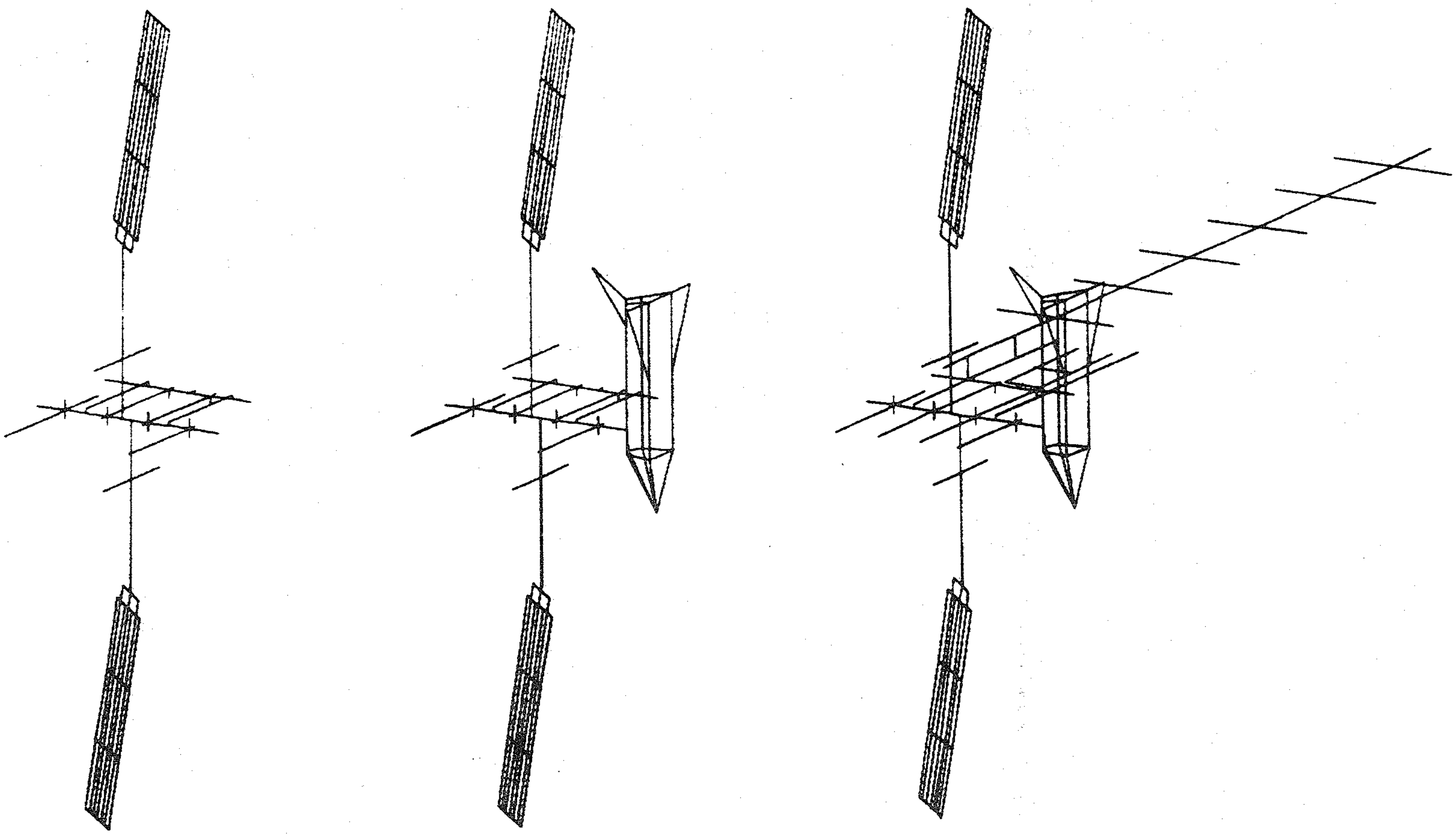
The dynamic characteristics of the five configurations were calculated using the Givens eigenvalue method in NASTRAN. The number of dynamic degrees of freedom for each of the configurations is tabulated below:

Configuration	DOF
1	102
2	114
3	168
4	45
5	51

The first 50 mode shapes were calculated and plotted for configurations 1 through 3. Modal frequencies vs mode number are shown in Figure 12-8 for these configurations. The first 6 modes are rigid body modes with zero frequencies and are not plotted. The first mode frequency for configurations 1 and 2 is associated with the first bending mode of the solar arrays (.042 Hz). For configuration 3, the first mode frequency results from torsional motion of the ETVP (.008 Hz). The frequencies of the SOC modules begin at a frequency of approximately 5.6 Hz for configuration 1 and 1.2 Hz for configurations 2 and 3 (Orbiter attached). Plots of the 50 mode shapes for configurations 1 through 3 are shown in the data book (Boeing-21).

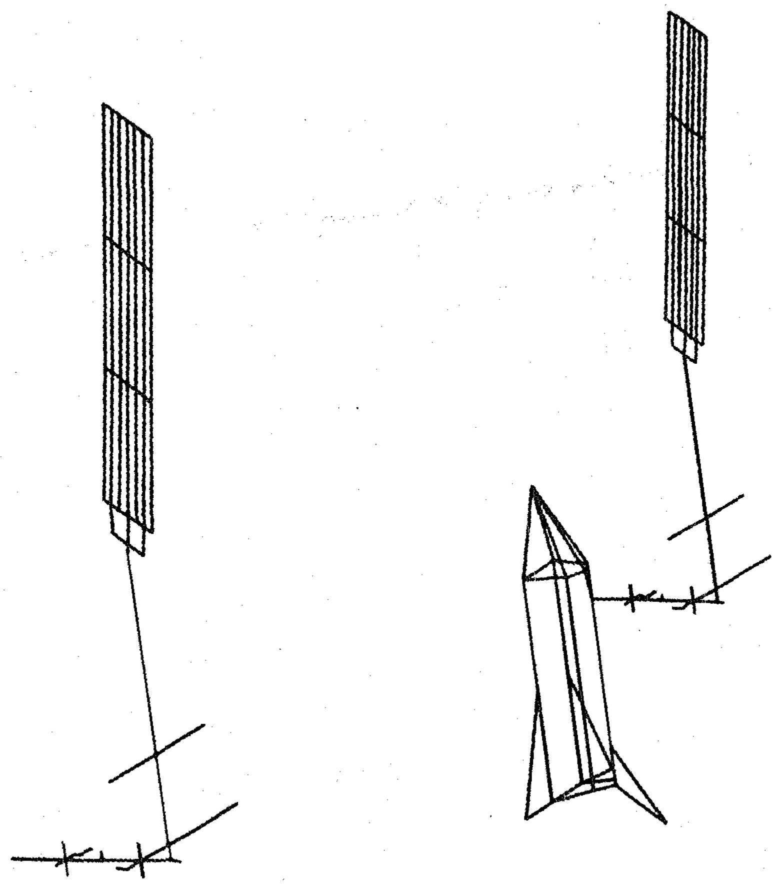
It is observed that there are many solar array mode frequencies present below the first frequency of the module assembly. For configuration 1, for example, there are 35 modal frequencies below the first SOC module frequency. With a more detailed finite element representation of the solar array assemblies, there will be many more low frequency modes.

The first 30 mode shapes were calculated and plotted for the two half-SOC configurations (configurations 4 and 5). These modal frequencies are compared in Figure 12-9. Again, the first mode frequencies are associated with the solar array bending modes. SOC module frequencies begin at a frequency of approximately 11.9 Hz for configuration 4 and at approximately 1.9 Hz for configuration 5. The first 30 mode shapes for these two configurations are shown in the data book (Boeing-21).



Configuration 1   Configuration 2   Configuration 3

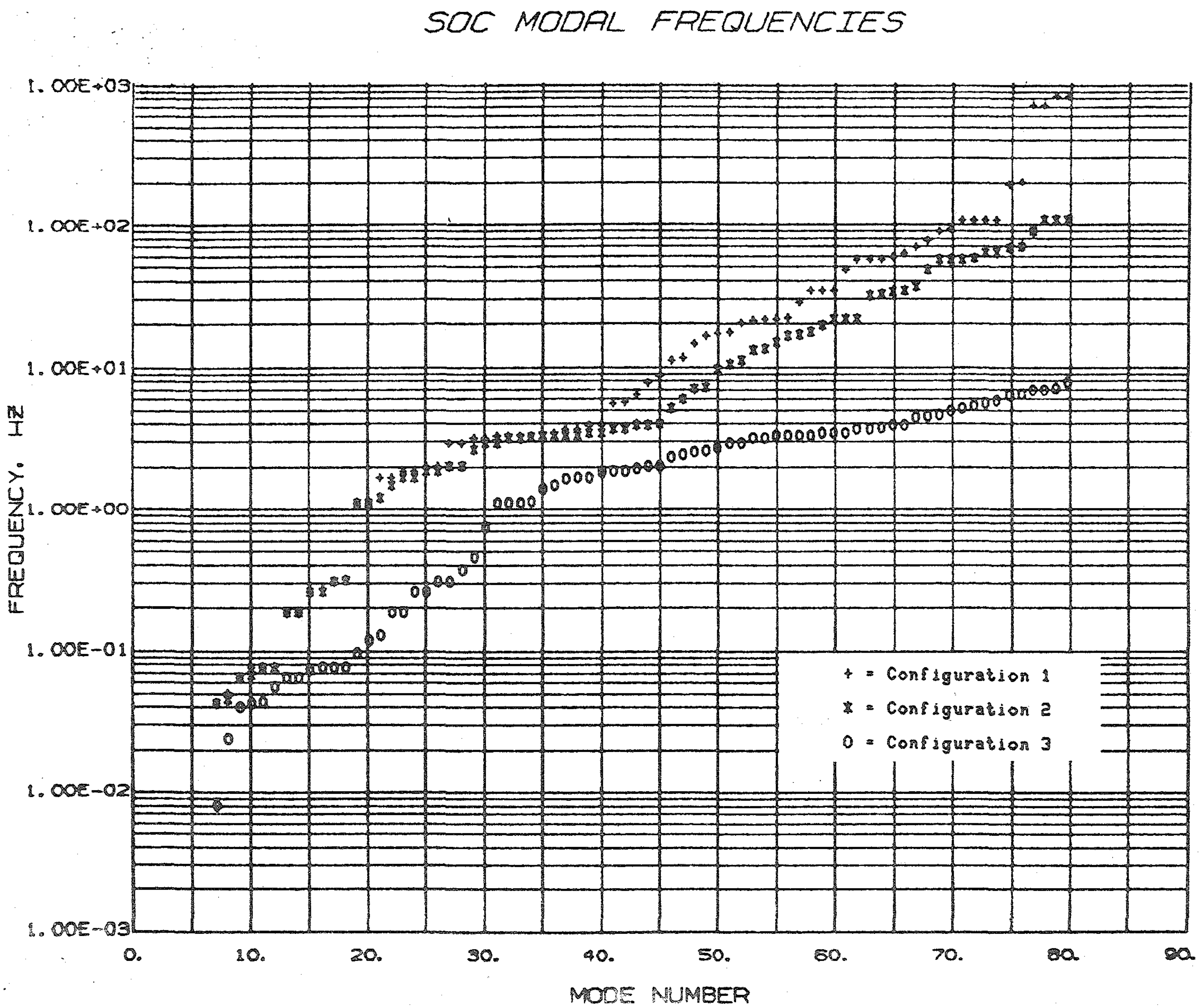
Figure 12-6: NASTRAN Stick Models-- Configurations 1-3



Configuration 4

Configuration 5

Figure 12-7: NASTRAN Stick Models, Configurations 4 and 5



c

Figure 12-8: Modal Frequency Comparison - Configurations 1-3

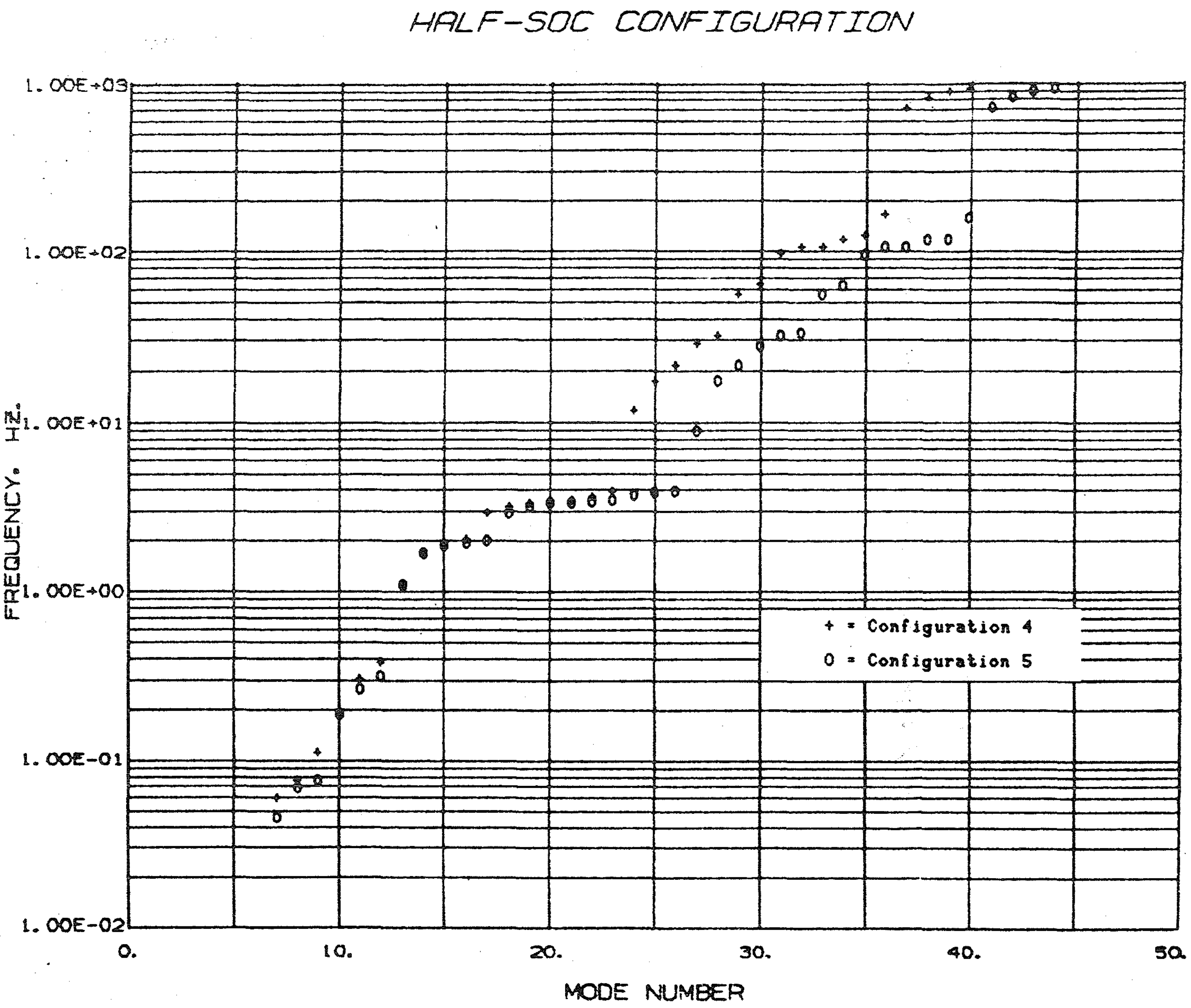


Figure 12-9: Modal Frequency Comparison - Configurations 4 and 5

12.4.3 TRANSIENT DYNAMIC RESPONSE

The NASTRAN model of configuration 2 was used to perform a transient dynamic response analysis to determine the dynamics caused by Orbiter docking. Two docking conditions were analyzed: soft docking and hard docking. Soft docking occurs when the Orbiter contacts the SOC docking mechanism in its extended position. In this position, the docking mechanism is supported by soft springs and dampers to cushion the impact. Hard docking refers to the transient which occurs when the docking mechanism is retracted with the Orbiter attached and is latched solidly against the docking tunnel to provide a pressure seal to permit the transfer of personnel and material through the tunnel into the SOC.

An assumed force history shown in Figure 12-10 was applied to the Orbiter center of mass in the x-direction to impart an initial velocity.

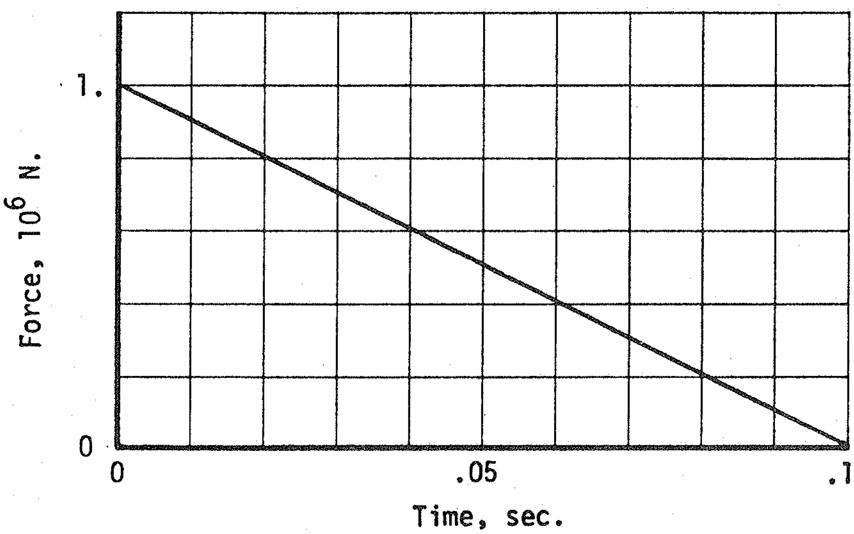


Figure 12-10 : Orbiter Docking Input Pulse

The magnitude of the force was arbitrarily chosen such that the Orbiter experienced an acceleration of approximately 1.0g. The docking mechanism spring constant (axial) was chosen such that the velocity imparted to the Orbiter was approximately 1 ft/sec (0.3 m/sec). The other Orbiter/SOC interface stiffnesses were scaled accordingly. The resulting modal frequencies are compared with the hard docked configuration in Figure 12-11. The reduced interface stiffness causes the first mode frequency (mode 7) to be reduced to 0.02 Hz and is assiciated with the Orbiter pitch rotation combined with solar array bending.

The transient analyses of soft and hard docking used the same forcing function and the SOC was uncontrolled during docking. Since the rigid body motions would mask the flexible responses (velocities and displacements), only flexible modes were used in the NASTRAN modal transient analysis. All flexible modes below 5.0 Hz were used in the soft docking analysis and all flexible modes below 10.0 Hz were used in the hard docking analysis. A modal damping ratio of 0.02 was used for all modes and damping elements (CDAMP) were included at the SOC/Orbiter interface in the soft docking simulation.

The soft docking responses of the SOC components are shown in Figures 12-12 through 12-14. The flexible displacement of a solar array tip is in excess of 3.0 meters as shown in Figure 12-12. Figure 12-13 shows that the initial velocity of the Orbiter docking interface in the axial direction is approximately 0.23 m/sec (0.75 ft/sec). Linear accelerations of the major SOC modules are plotted in Figure 12-14. These accelerations are the flexible component of acceleration and do not include the rigid body accelerations. Peak accelerations of 0.25 m/sec (0.026 g) occur on SOC modules and 0.3 m/sec (0.031 g) on the Orbiter center of mass during soft docking.

The dynamic responses of the SOC modules to the hard docking transient are shown in Figures 12-15 through 12-17. The maximum flexible diplacement of the solar array tip (Figure 12-15) is seen to be approximately 1.3 meters at the first flexible mode frequency (0.042 Hz). Orbiter docking interface axial and rotational velocities are shown in Figure 12-16. The flexible accelerations of SOC components are shown in Figures 12-17. Considerably higher frequency responses occur during hard docking due to the higher stiffness of the docking interface. Peak acceleration occuring on SOC modules is approximately 1.75 m/sec (1.8 g) while the peak flexible acceleration of the solar array tip is approximately 4.5 m/sec (0.46 g). The maximum acceleration of the Orbiter center of mass is approximately 1.1 m/sec (0.11 g) in the x-direction.

To determine the rigid body response to the docking transient, a separate analysis was performed using the rigid body mass matrix generated by NASTRAN. Peak accelerations and residual velocities are tabulated in Table 12-3.

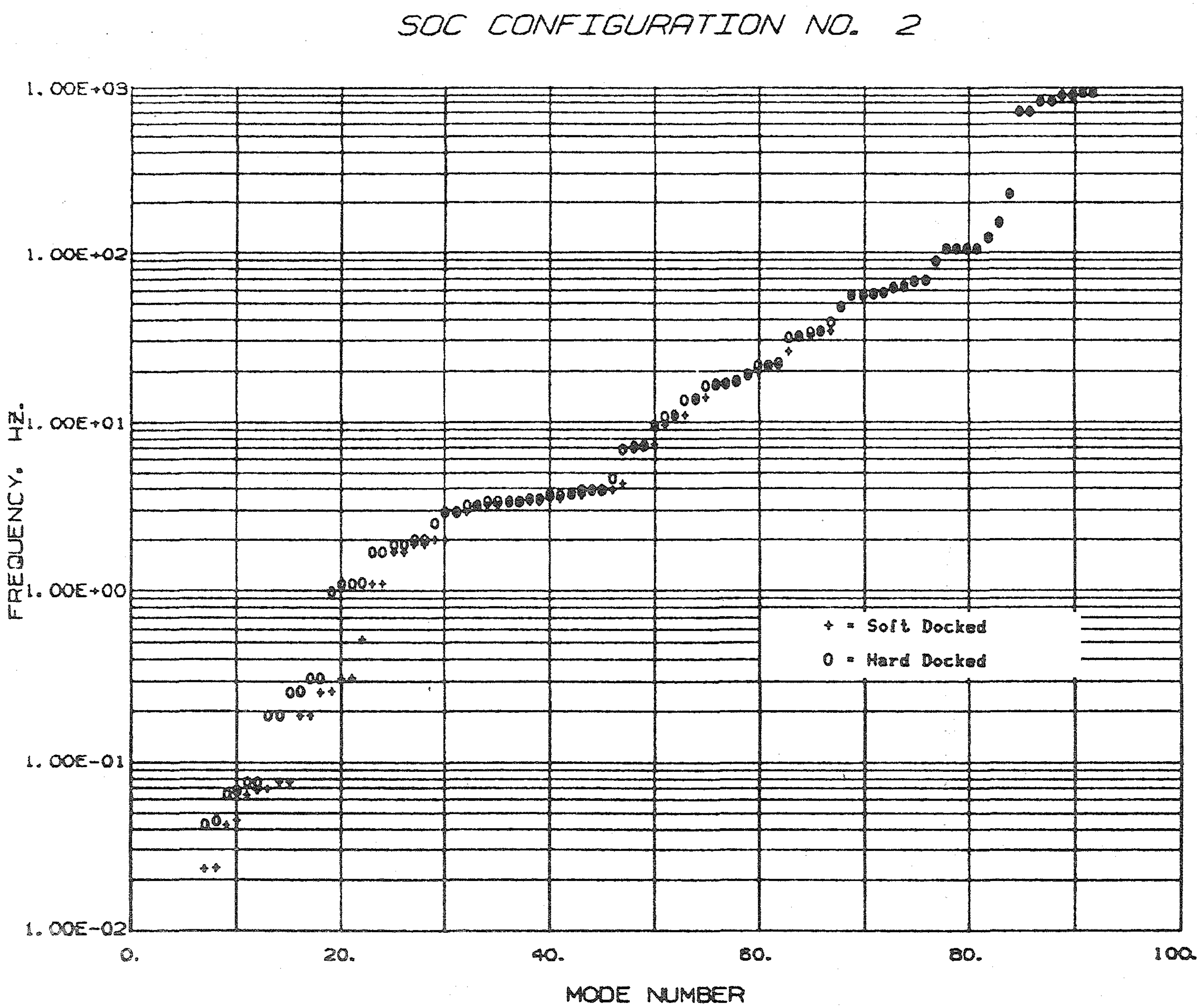


Figure 12-11: Comparison of Soft and Hard Docked Modal Frequency

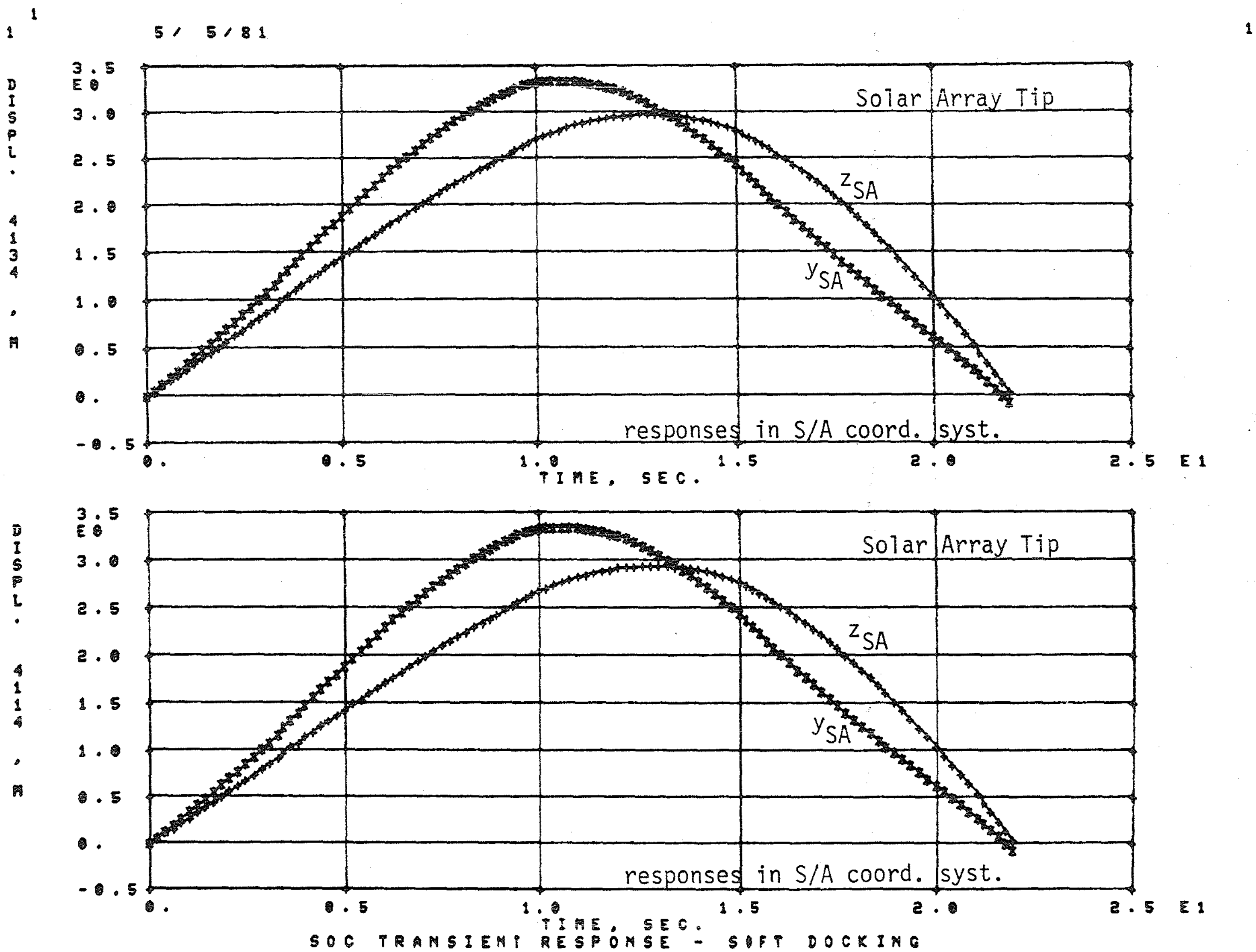


Figure 12-12: Solar Array Tip Displacements - Soft Docking

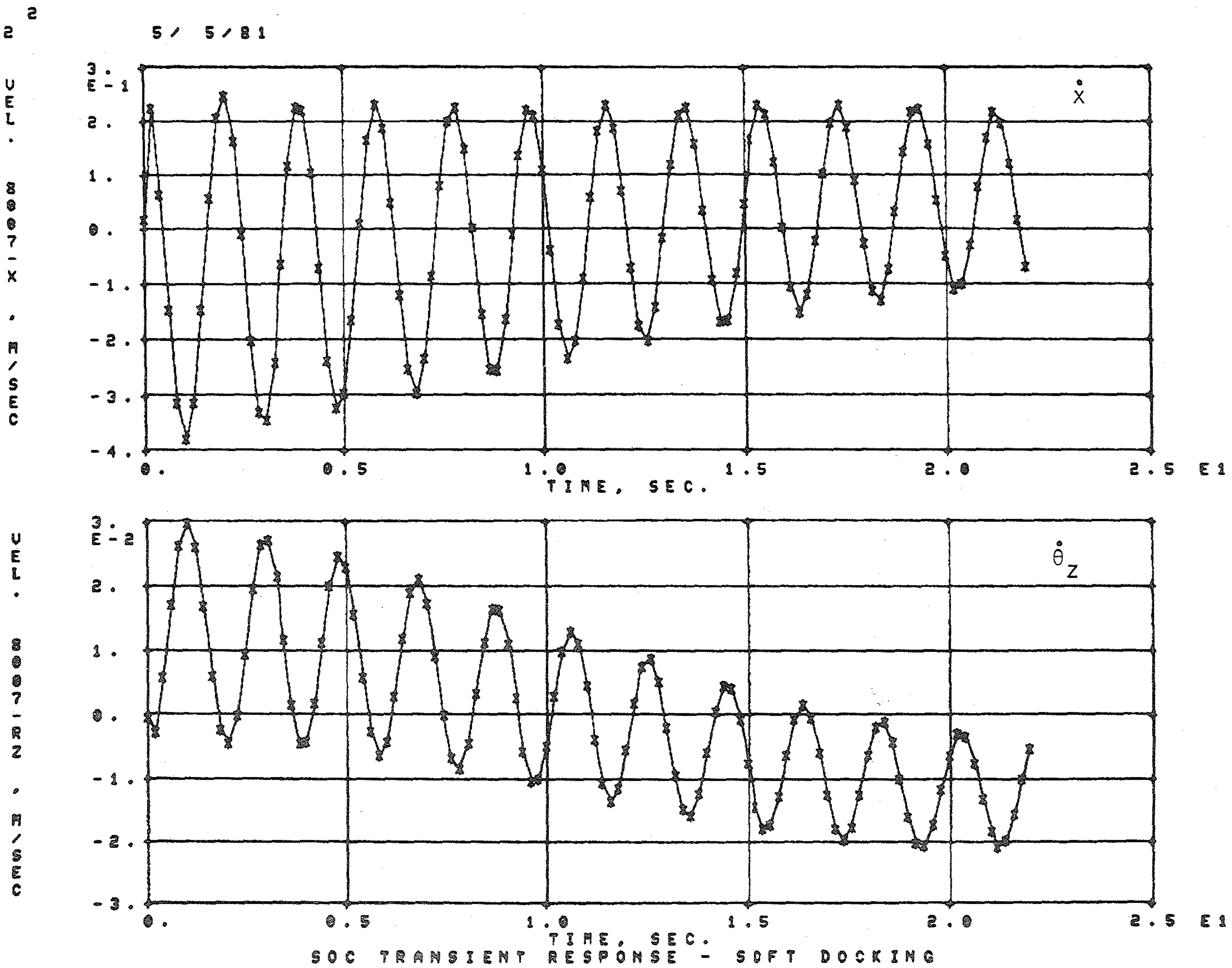


Figure 12-13: Orbiter Docking Interface Velocity - Soft Docking

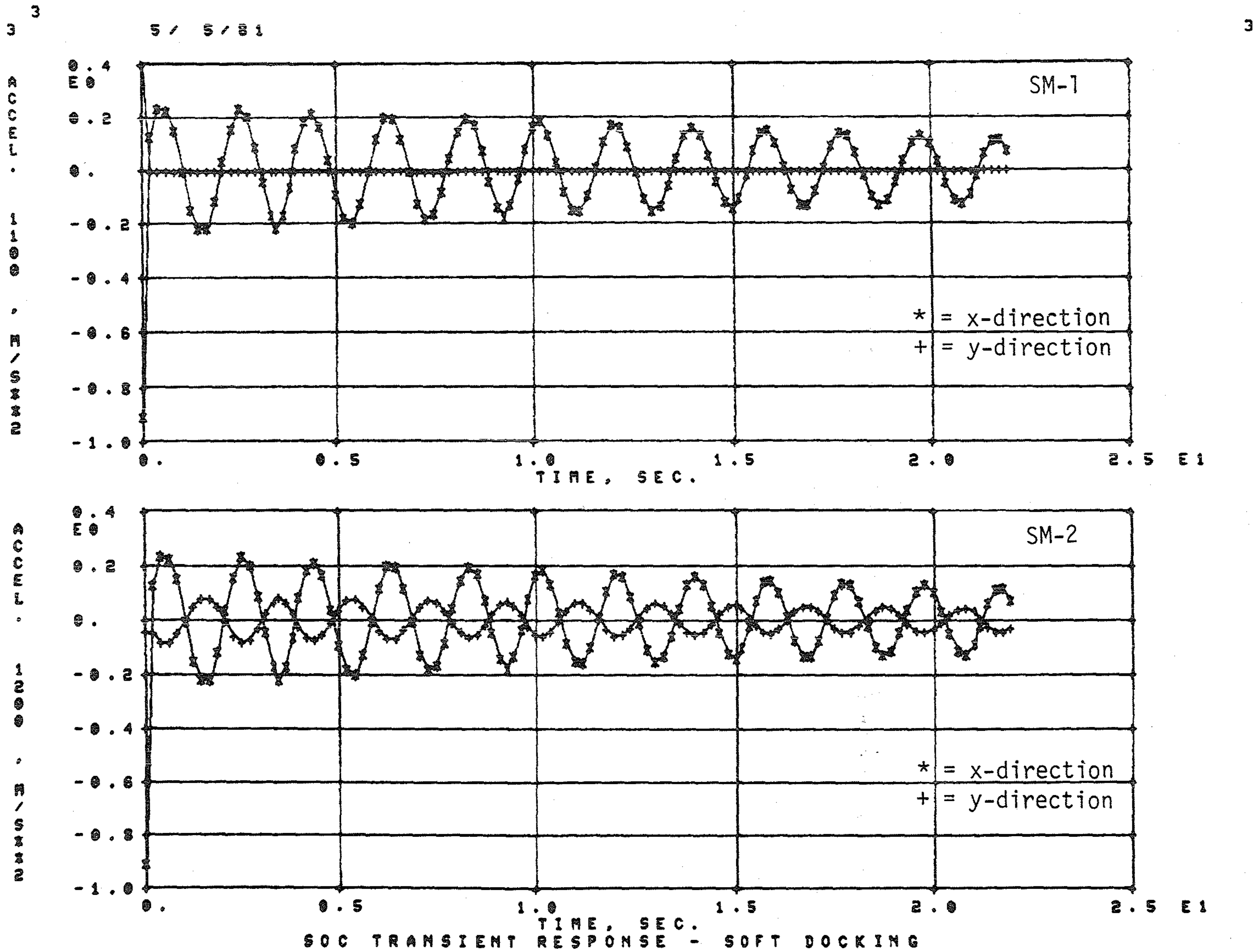


Figure 12-14: SOC Module Accelerations - Soft Docking

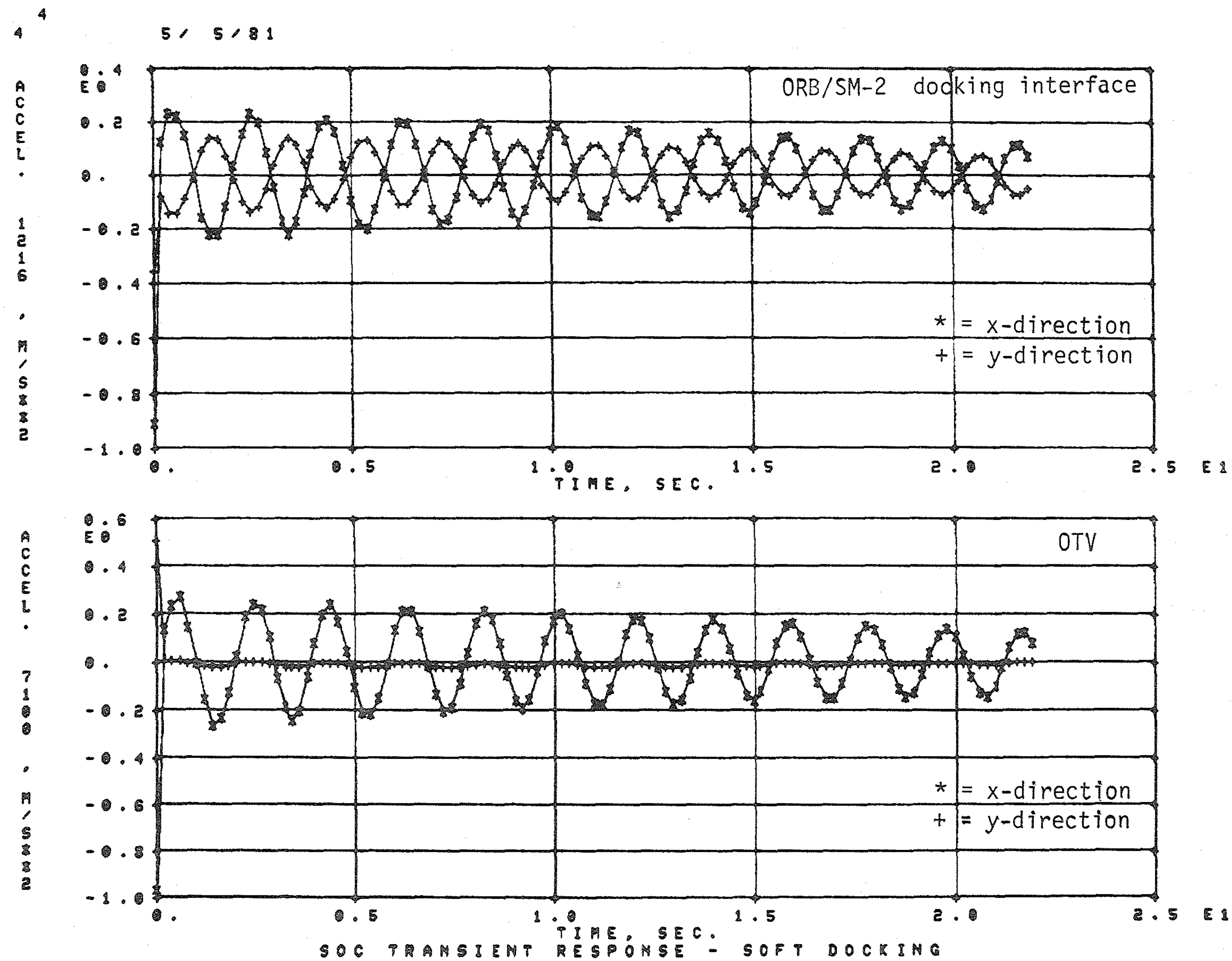


Figure 12-14: SOC Module Accelerations - Soft Docking (cont'd)

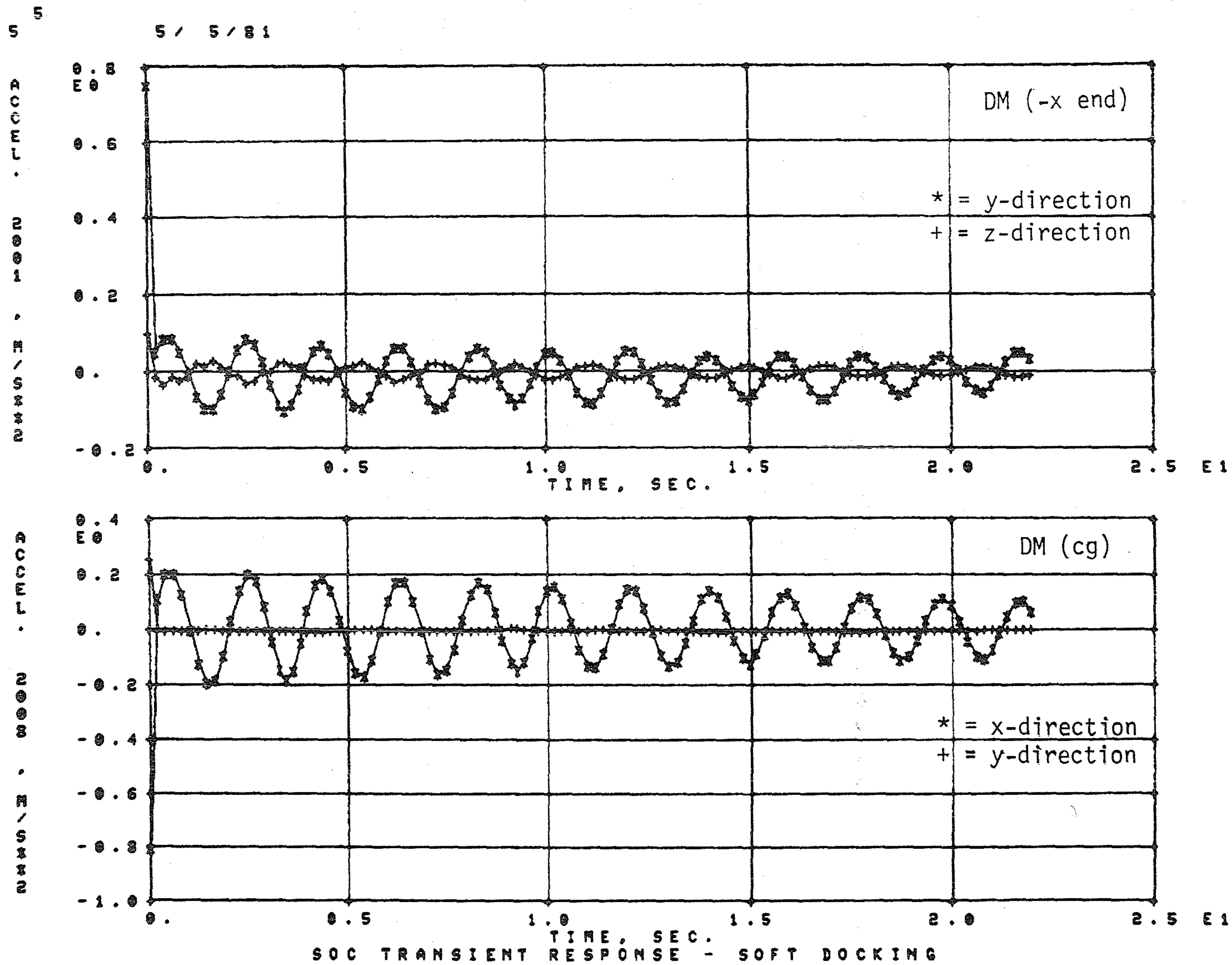


Figure 12-14: SOC Module Accelerations - Soft Docking (cont'd)

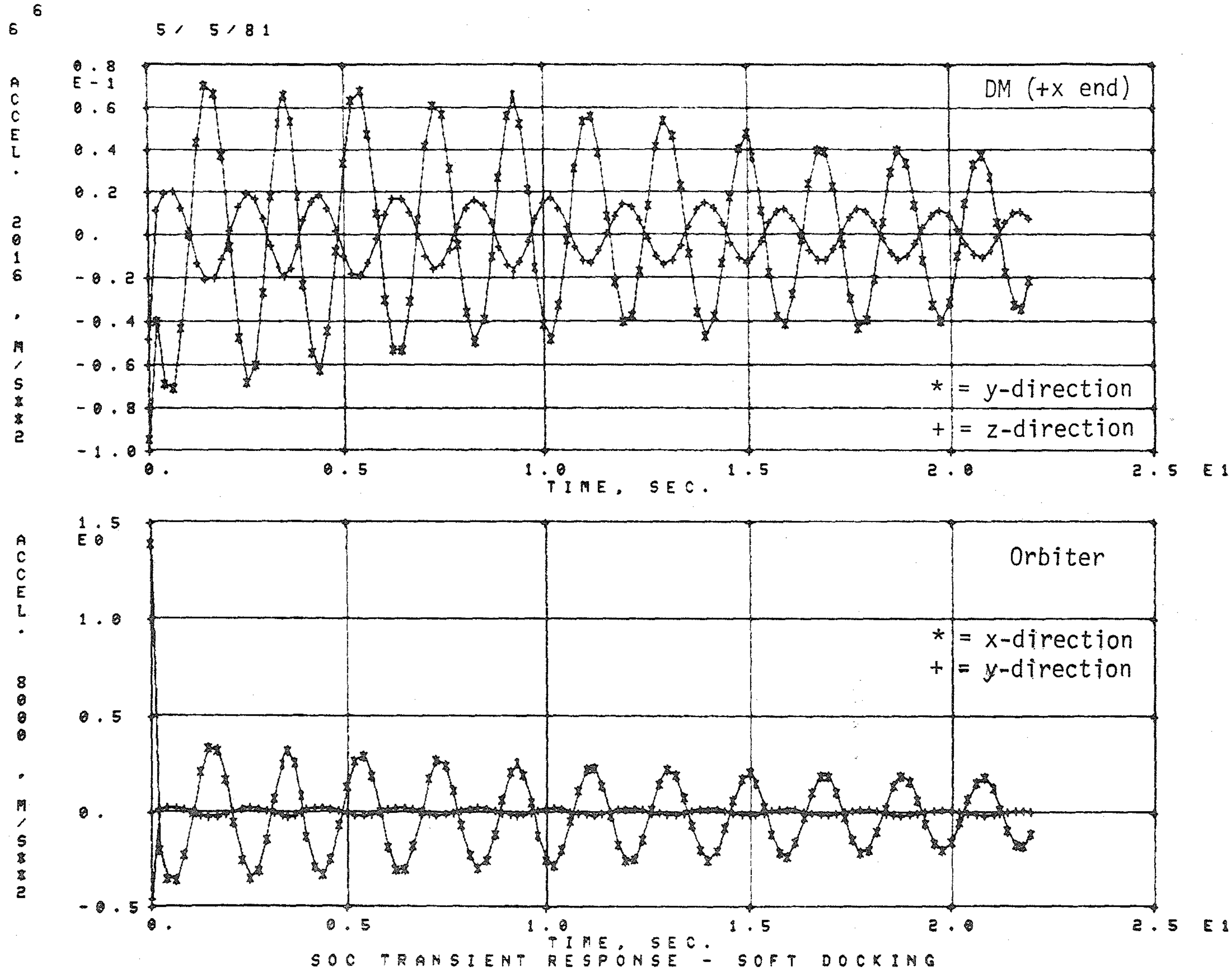


Figure 12-14: SOC Module Accelerations - Soft Docking (cont'd)

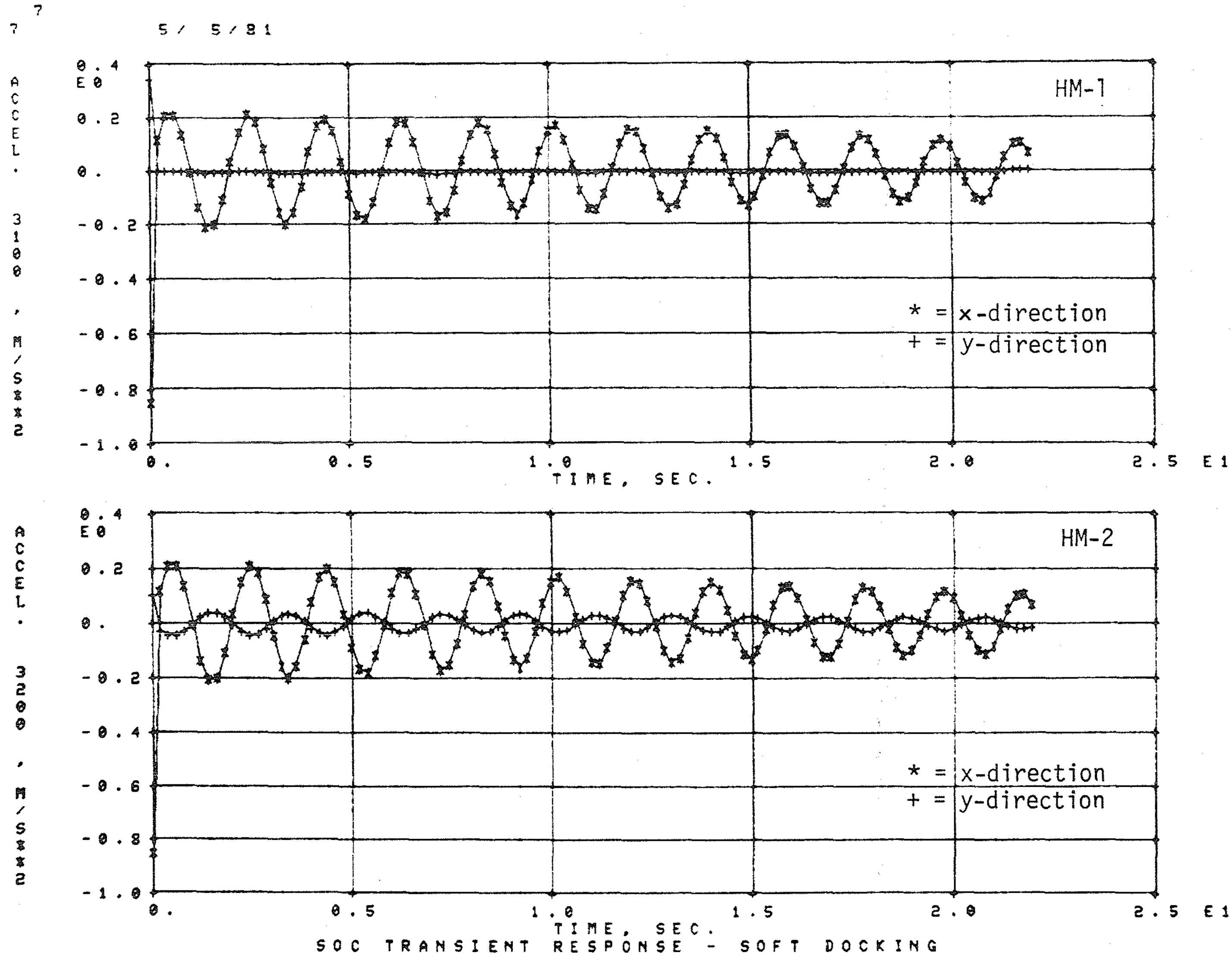


Figure 12-14: SOC Module Accelerations - Soft Docking (cont'd)

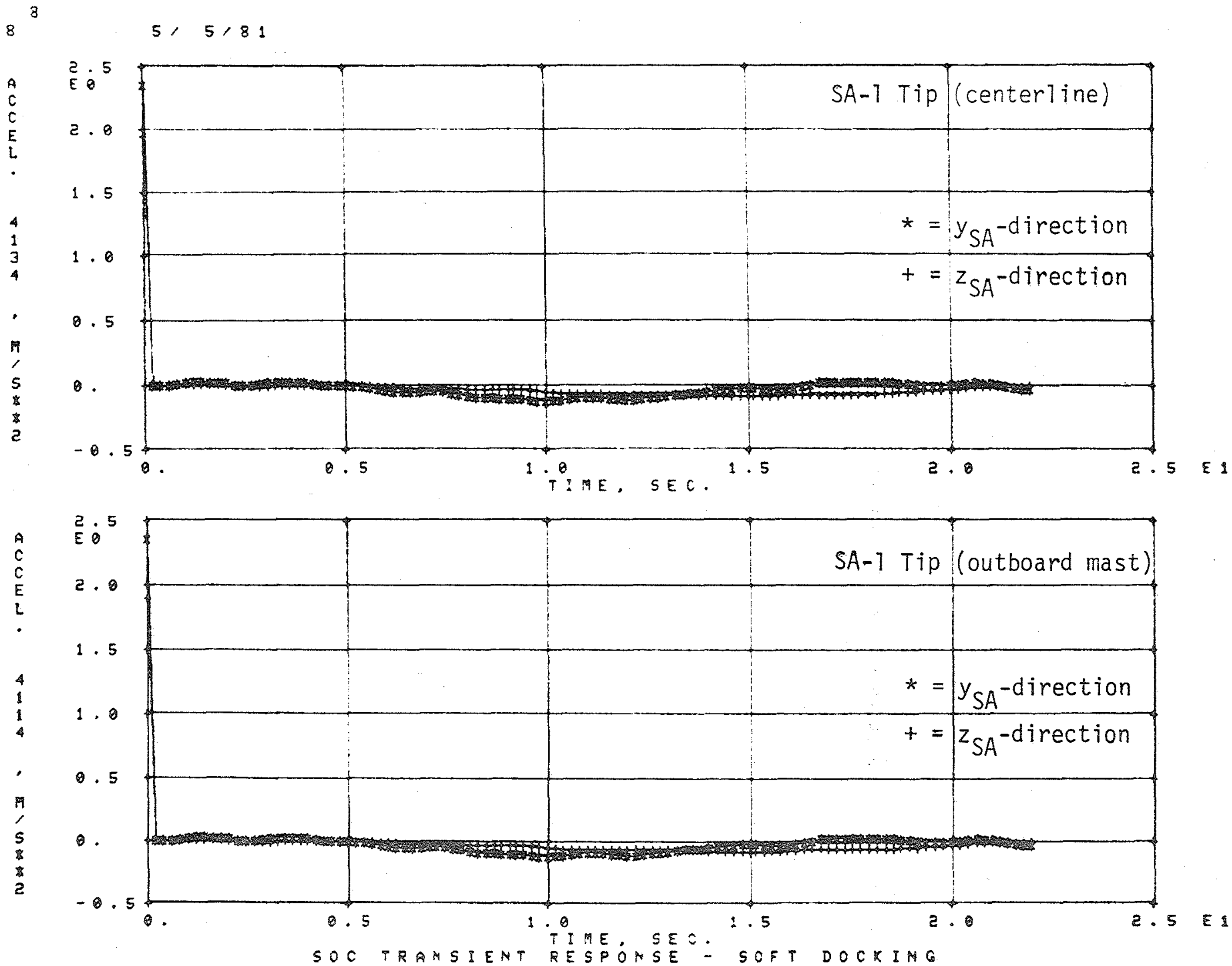


Figure 12-14: SOC Module Accelerations - Soft Docking (cont'd)

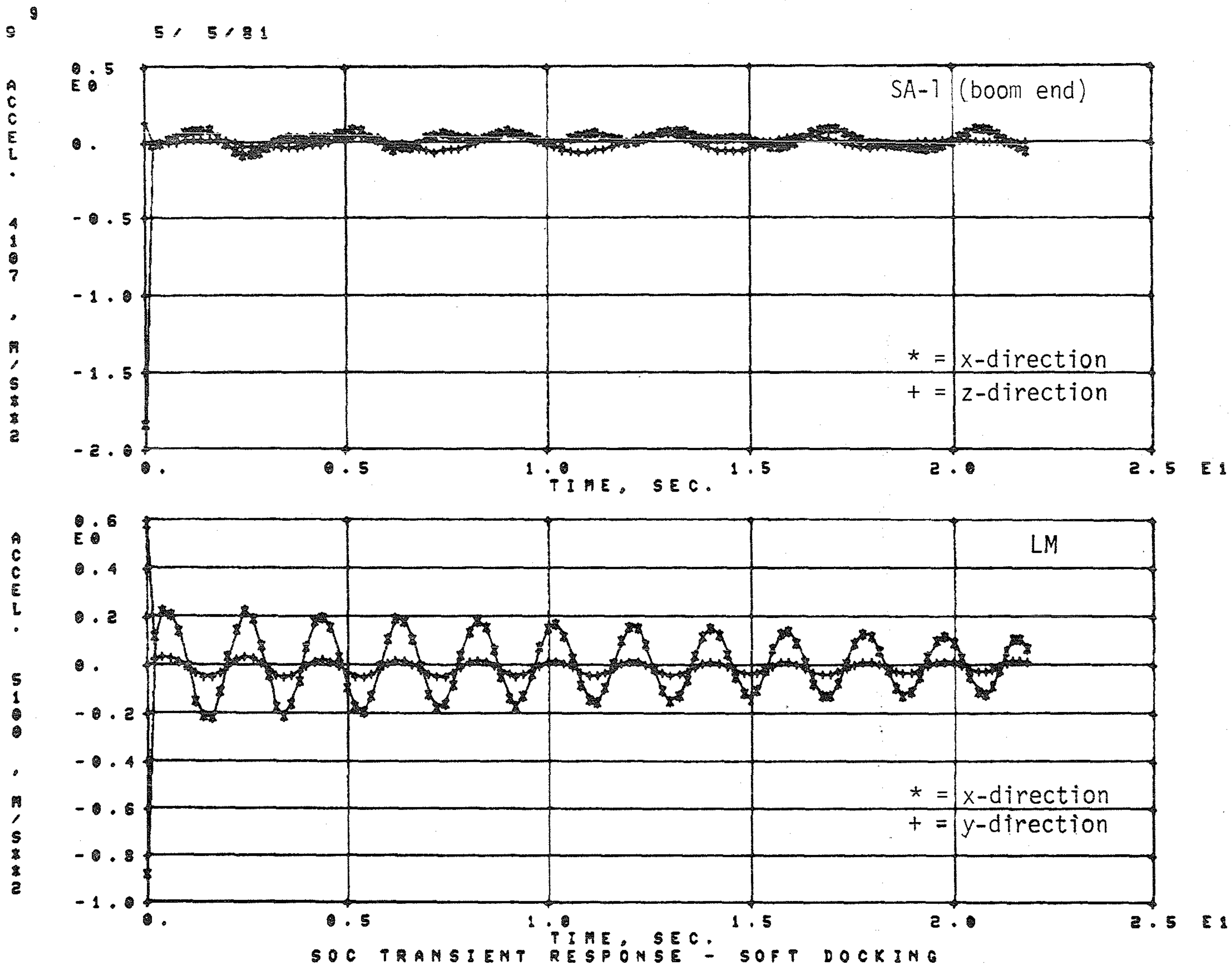


Figure 12-14: SOC Module Accelerations - Soft Docking (cont'd)

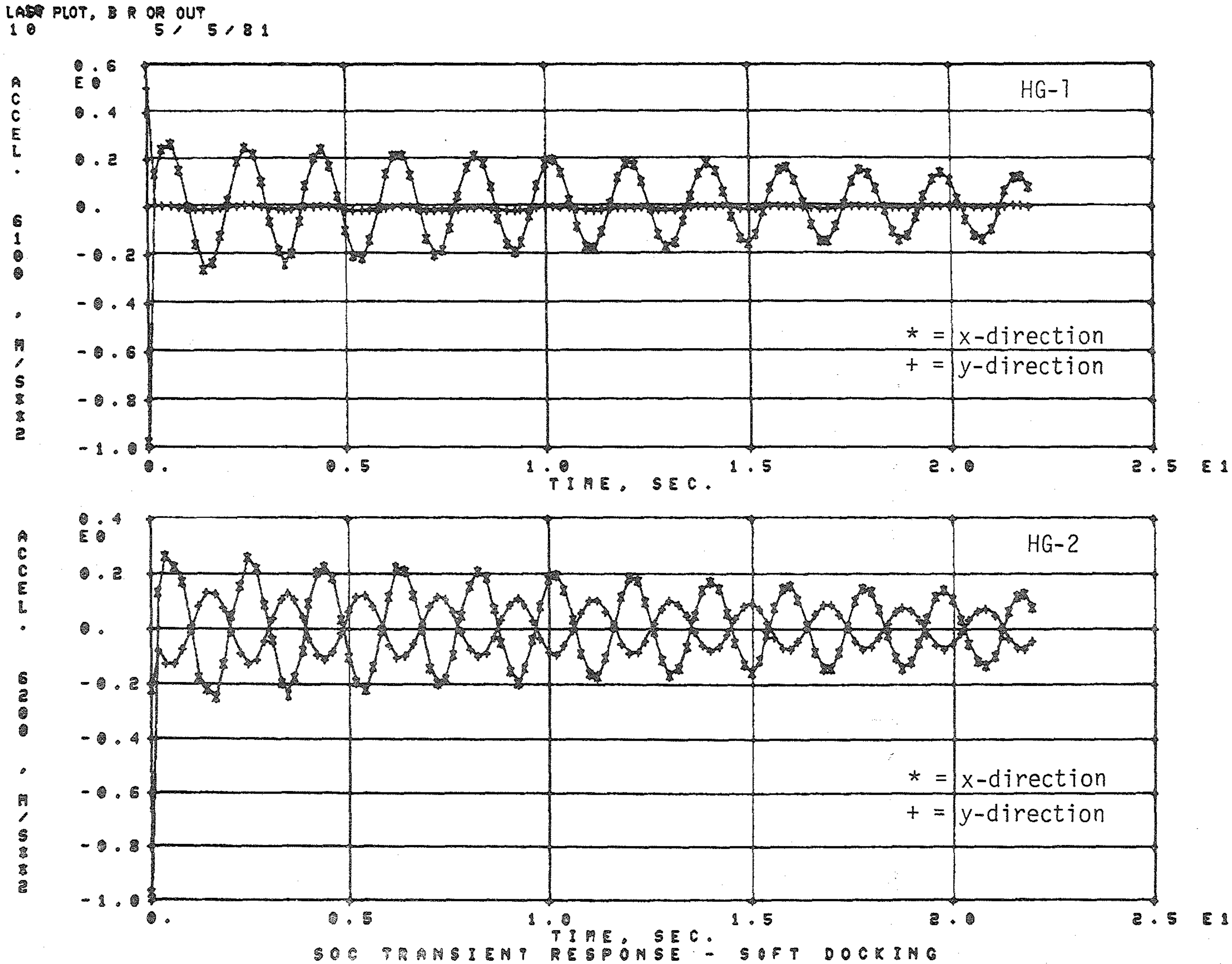


Figure 12-14: SOC Module Accelerations - Soft Docking (cont'd)

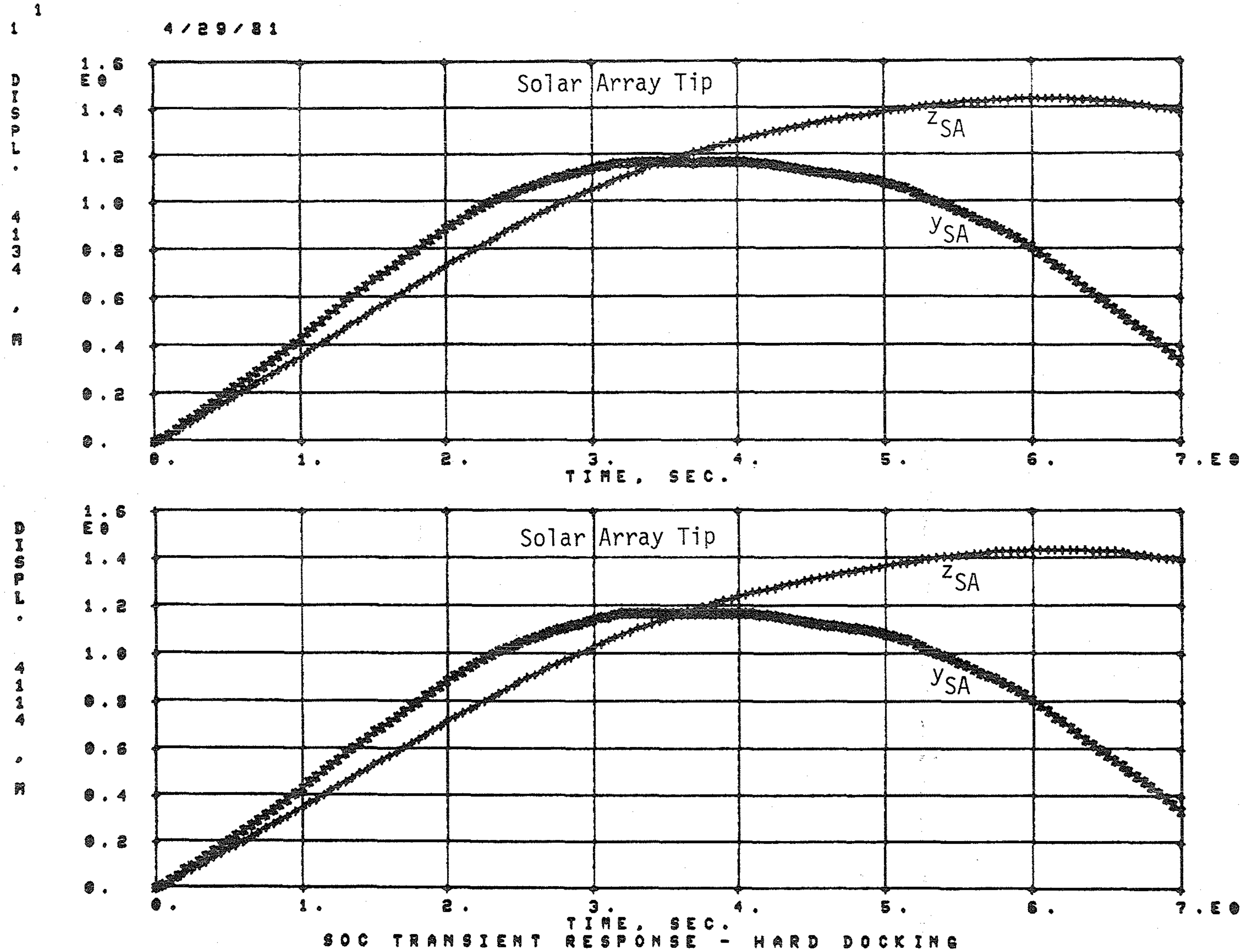


Figure 12-15: Solar Array Tip Displacement - Hard Docking

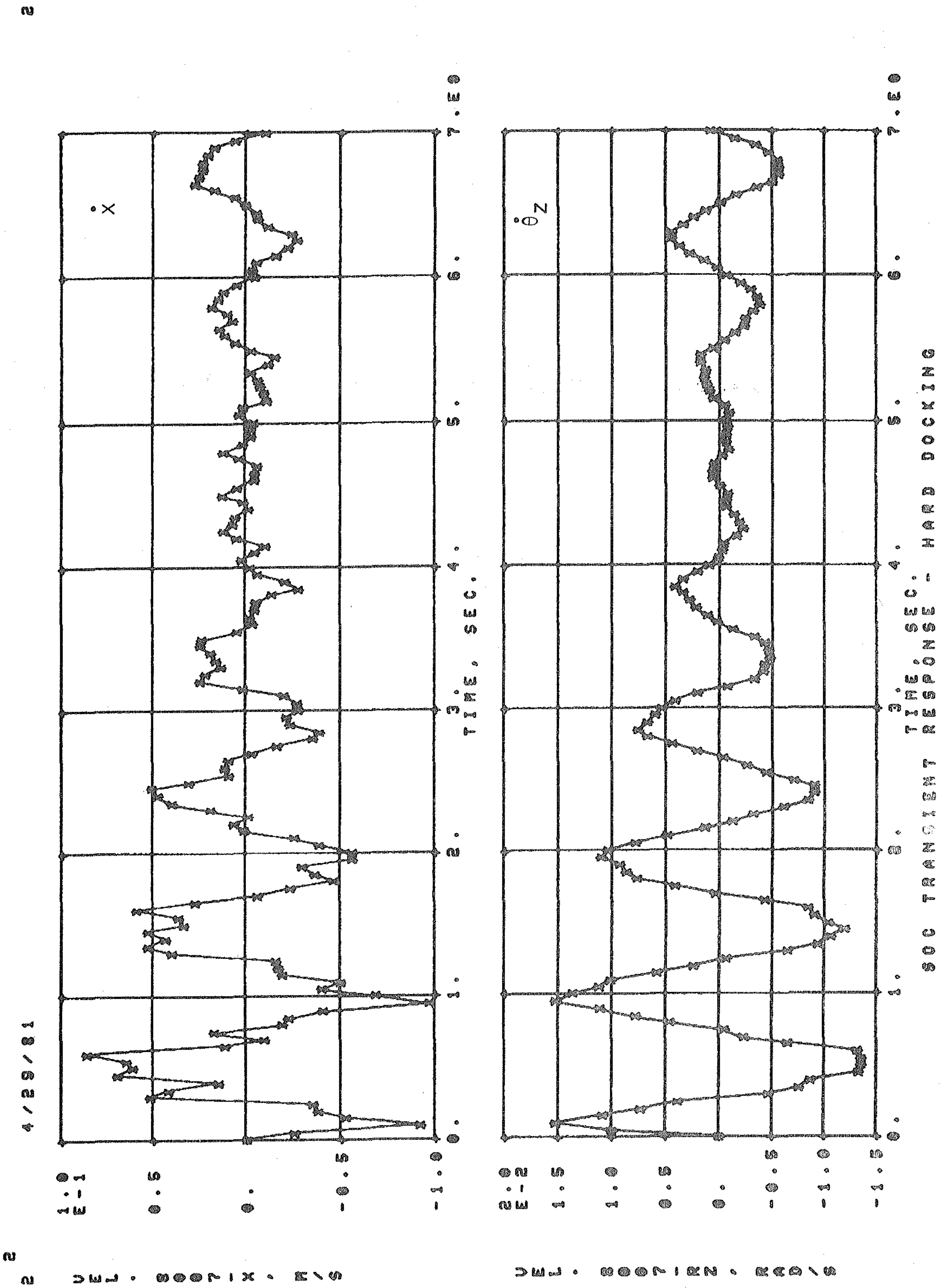


Figure 12-16: Orbiter Docking Interface Velocity - Hard Docking

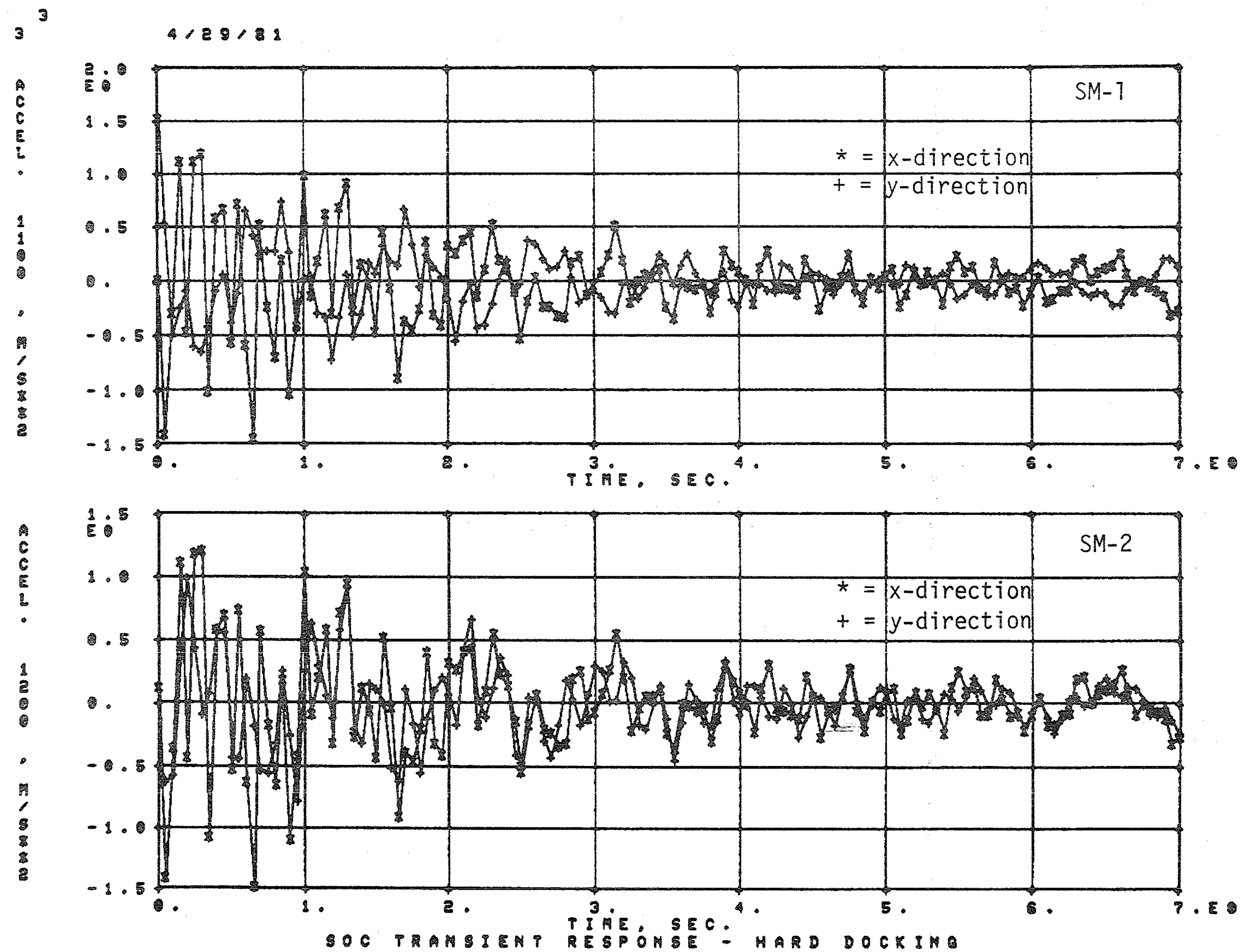


Figure 12-17: SOC Module Accelerations - Hard Docking

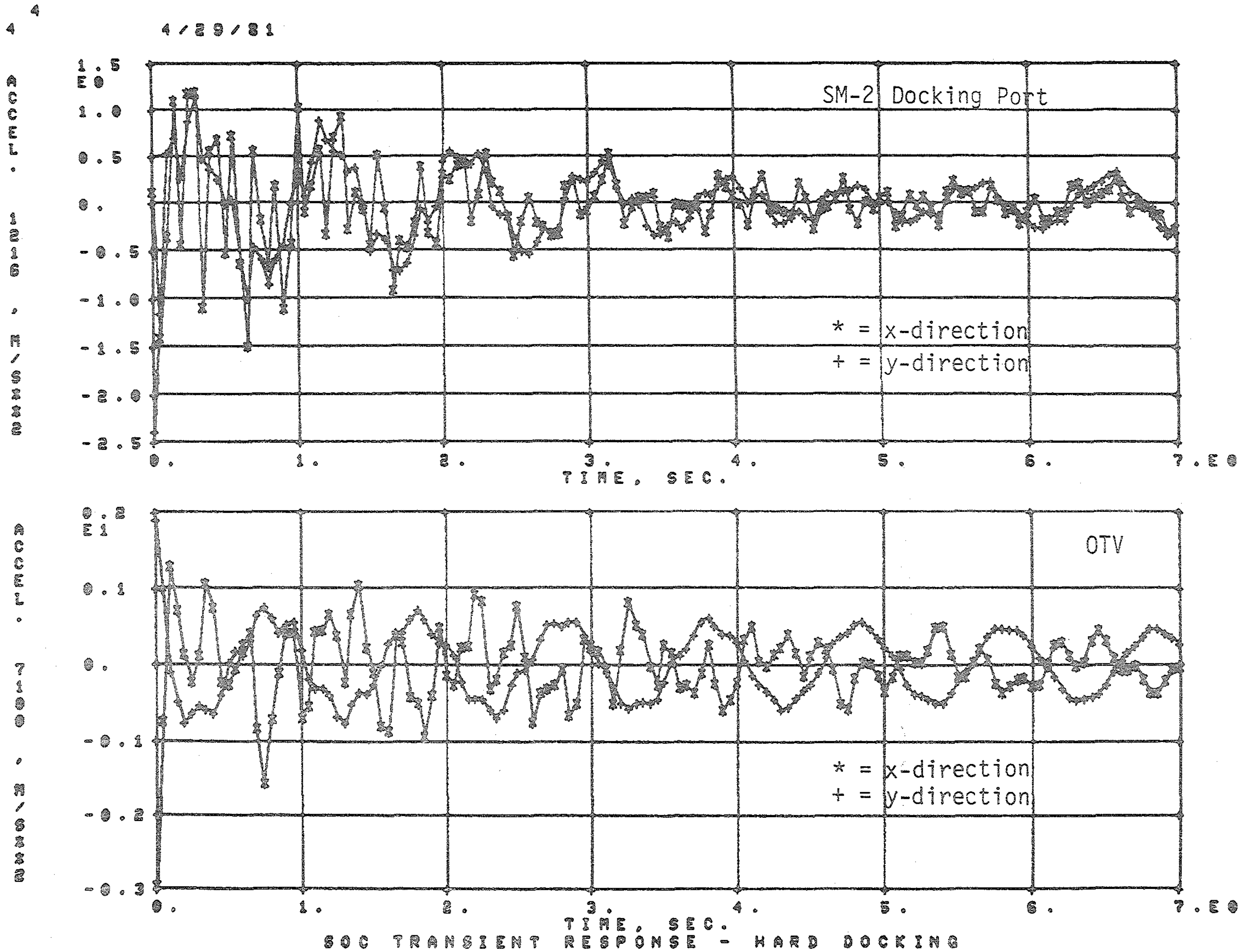


Figure 12-17: SOC Module Accelerations - Hard Docking (cont'd)

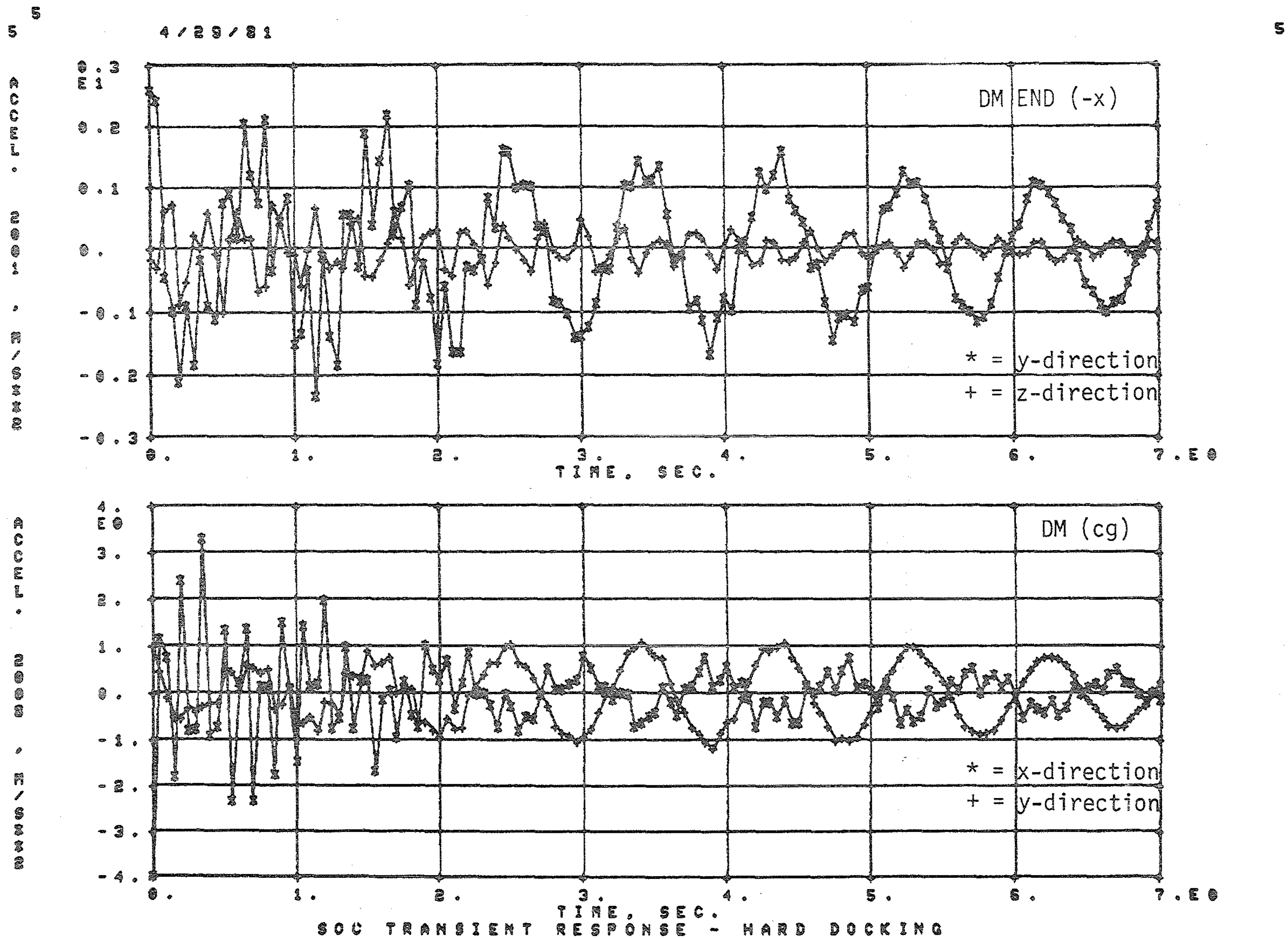


Figure 12-17: SOC Module Accelerations - Hard Docking (cont'd)

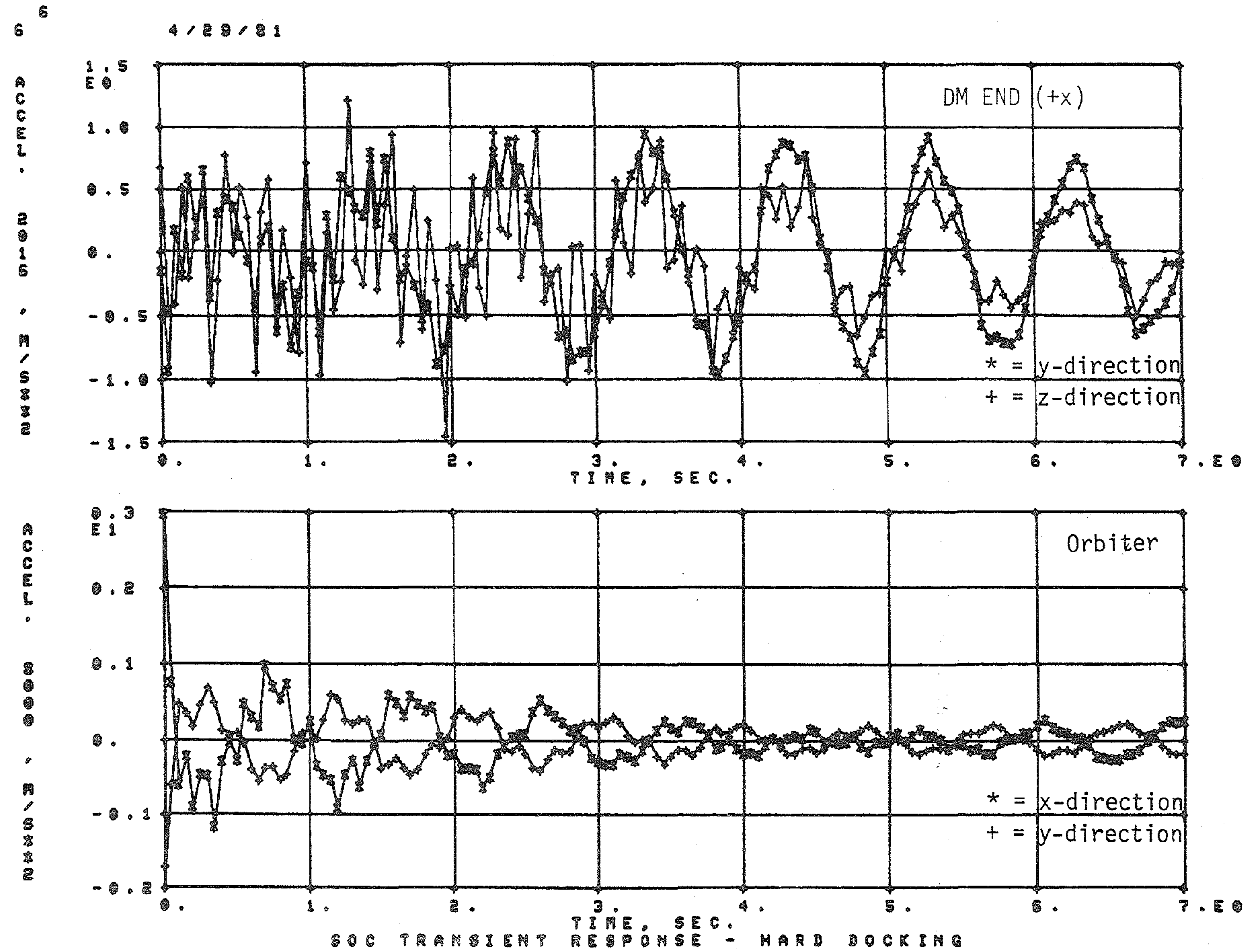


Figure 12-17: SOC Module Accelerations - Hard Docking (cont'd)

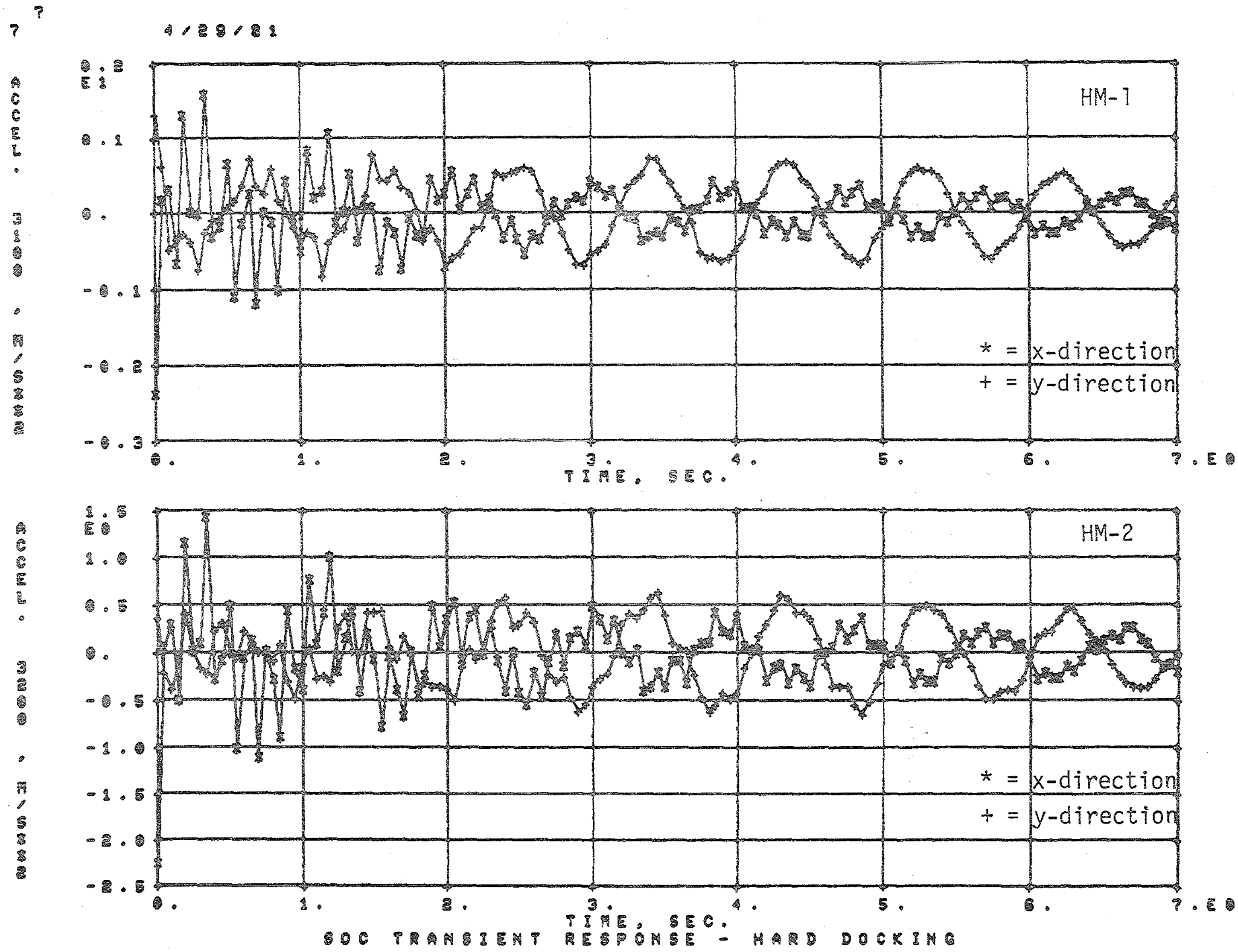


Figure 12-17: SOC Module Accelerations - Hard Docking (cont'd)

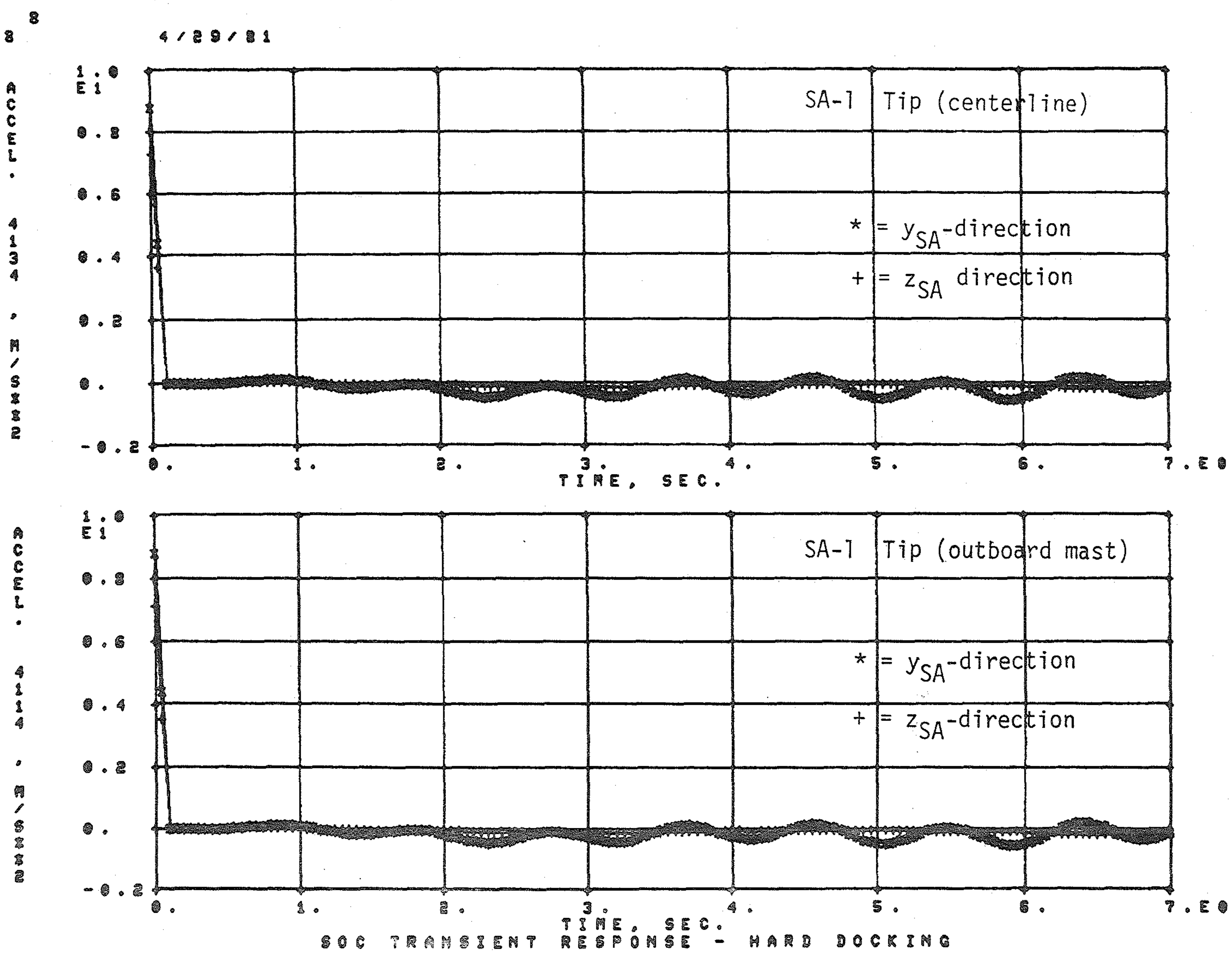


Figure 12-17: SOC Module Accelerations - Hard Docking (cont'd)

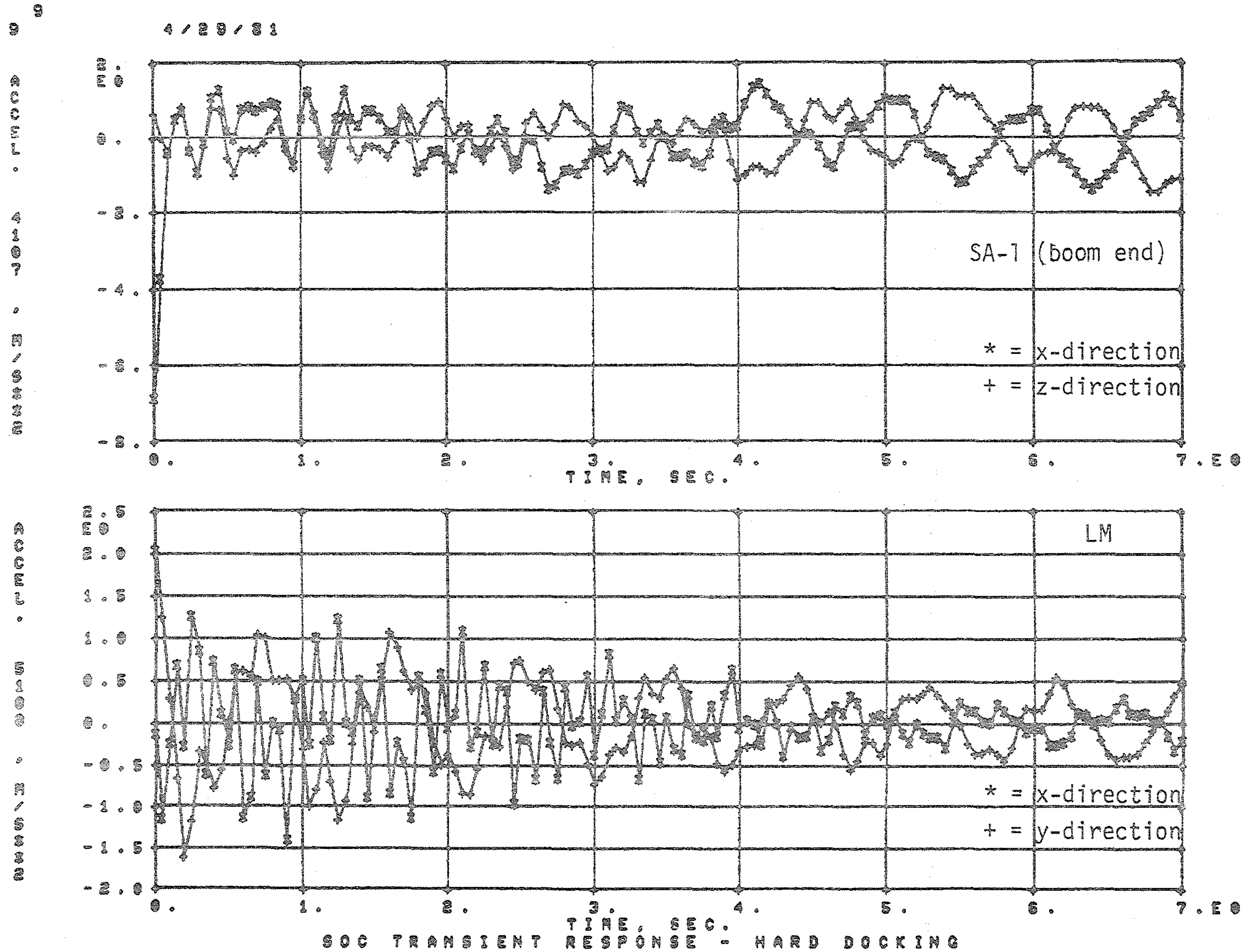


Figure 12-17: SOC Module Accelerations - Hard Docking (cont'd)

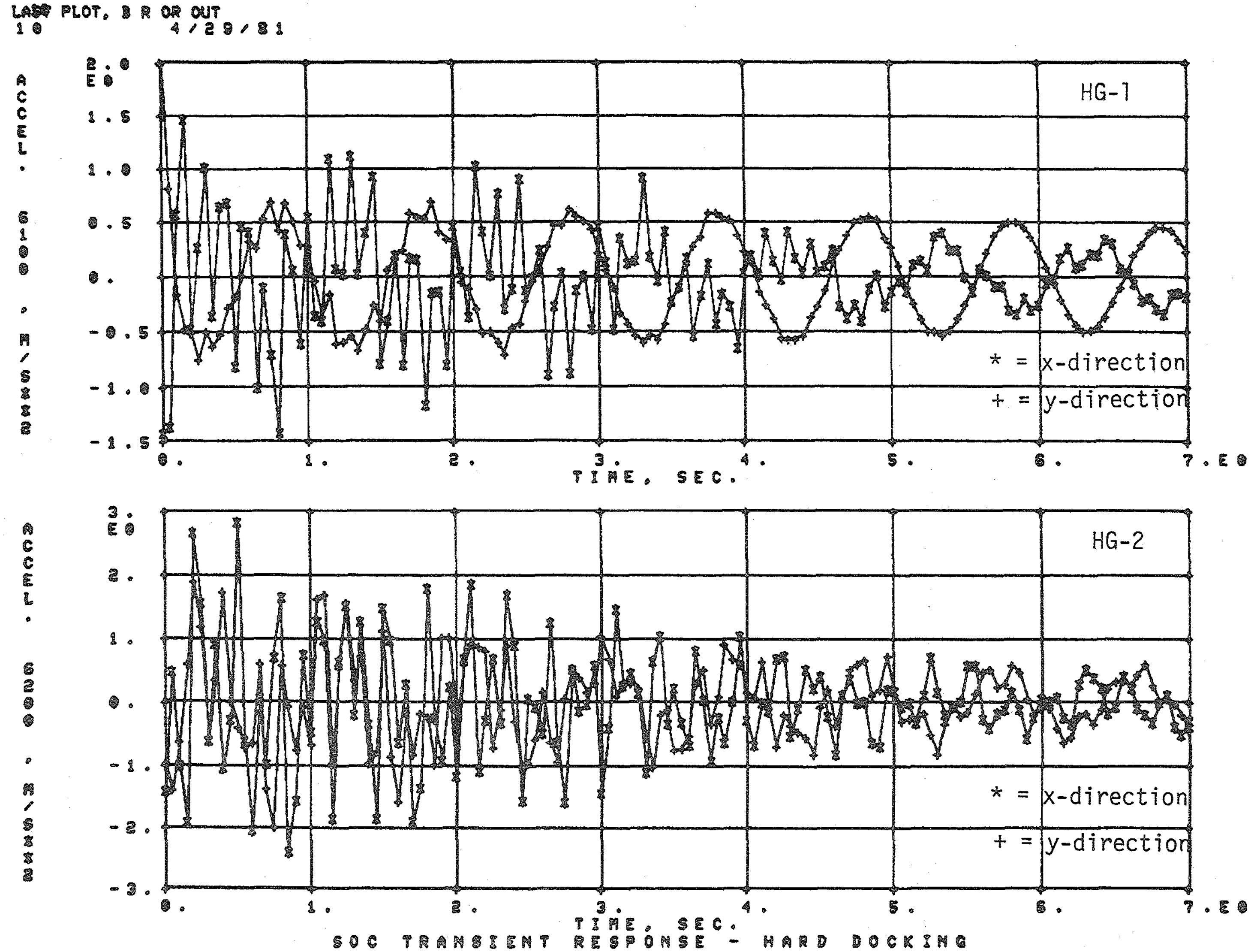


Figure 12-17: SOC Module Accelerations (cont'd)

Table 12-3: Residual SOC Velocities Due to Docking

Direction	Residual Velocity (m/sec, rad/sec)
x	-0.20
y	0.
z	0.
Rx	9.46E-4
Ry	1.43E-3
Rz	-6.63E-3

12.5 CONCLUSIONS

The dynamic analysis of the SOC shows that the lowest frequency resonances are associated with the solar array assemblies, and that there will be many solar array resonances below the first module frequencies. The distribution of modal frequencies is a function of the SOC configuration and will also depend on the orientation of the solar arrays.

The low modal frequencies (0.04 Hz range) and the high density of modal frequencies must be considered in the design of the attitude control system to assure stability for all configurations and to prevent unwanted structure/control interaction.

Docking transients may be the largest single disturbance that the SOC will experience. Significant flexible excitation occurs for both soft and hard docking. The residual rigid body rates are also significant and must be controlled by the attitude control system.

13.0 FLIGHT CONTROL ANALYSES

13.0 FLIGHT CONTROL ANALYSES

13.1 INTRODUCTION

The Space Operations Center (SOC) will be constructed in stages, a module at a time. Relative to conventional spacecraft, the modules are massive, and the inertia properties will vary significantly at each stage of buildup. The attitude control system (ACS) design must therefore account for the variation of requirements, as well as the unconventional bulky structure. It is also proposed that the ACS provide the control for station-keeping and orbit-makeup. Therefore, determination of the requirements for the SOC ACS is needed. Once the requirements are known, ACS designs need to be developed, simulated and tested.

The objective of this study is to derive mathematical models of several specific configurations, and evaluate the external disturbances acting on them. Given a specific design of an ACS, the capability to control the modeled configurations is determined. Additionally, further studies needed to design the most effective ACS are discussed.

13.2 DESCRIPTION

This section describes how the various SOC configurations are modeled and what coordinate systems are used in the analysis. The disturbances and flight parameters are described, and a specific ACS is introduced. The use of control moment gyro's (CMG's) is discussed.

13.2.1 Individual Components

For the estimation of mass properties, the SOC modules are modeled as simple geometric figures. The Service Module (SM), Habitat Module (HM), Docking Module (DM), Logistics Module (LM), Orbiting Transfer Vehicle (OTV), and Propellant Tank (PT) are modeled as right cylinders; the Hanger (HG) is modeled as a right hexagonal prism. The Orbiter (OR) and Rockwell Platform (RP) mass properties are estimated from other information provided (References 1 and 2)*. The Storage Facility (SF) mass is estimated as 25% of the DM. The Solar Array (SA) mass properties are provided by the SEPS study group. A summary of the individual component mass properties is listed in Table 13-1. The center of mass of all modules are assumed to be in the center of the

*References found in Section 13.8.

Table 13-1. *Individual Component Mass Properties*

Component Name	Mass (kg)	Diameter (m)	Length (m)	Center of Mass Location w.r.t. Reference–Axes (m) x	y	z	Moments of Inertia w.r.t. Body–Axes (1000–kg–m^2) I_x	I_y	I_z
Service module (SM)	18600	3.15	15.19	±6.076	0	0	26.449	426.8	426.8
Habitat Module (HM)	18692	4.27	14.97	±3.5	0	9.045 ±1.496	412.999	412.999	44.099
Docking Module (DM)	9100	3.15	16.36	0	0	18.09	12.501	233.5	233.5
Logistics Module (LM)	8800	4.572	8.0	-11.689	0	5.575	58.401	58.401	23.0
OTV Hangar (HG)	6800	8-10	16.	±11.689	0	-9.576	207.4	207.4	124.6
Orbiting Transfer Vehicle (OTV)	38650	4.48	14.17	-11.69*	0*	-8.66*	695.2	695.2	97.0
Propellant Tank (PT)	52000	4.48	15.0	±3.5	0	-9.075	1040.2	1040.2	130.5
Orbiter (OR)	90909	–	–	18.26	-12.11	0	7786.	1072.	7494.
Storage Facility (SF)	2275	–	–	0	0	26.09	180.9	193.5	13.1
Rockwell Platform (RP)	38806	–	–	-8.18	-4.13	108.3	57063.	58492.	1458.
Solar Array (SA)	582.8	–	28.8	±.96	∓43.	.5	10.564	95.889	106.443
Air Lock (AL)	230.	–	–	-4	0	-2.4	.143	.161	.161
Pallet (PL)	1000.	–	–	-11.689	1.146	3.075	1.077	2.39	1.967
Logistics (LG)	1801.	–	–	-11.689	0	2.31	1.526	1.526	2.163

*OTV x,y,z values listed are for configurations 2 and 2_{NO}
For configurations 3 and 3_{NO}, the values are -8.18, -4.13, 24.42 ±7.085

Table 13-1. *Individual Component Mass Properties (Continued)*

Component	Cross-Sectional Area (m^2) xz	yz	Centers of Pressure w.r.t. Body-Axes (m) xz-plane x	z	yz-plane y	z
SM	46	10	±7.595	0	0	0
HM	74	74	±3.5	9.045	0	9.045
DM	50	10	0	18.09	0	18.09
LM	36	36	-11.689	5.575	0	5.575
HG	128	160	±11.689	-9.576	0	-9.576
OTV	60	60	-11.69*	-8.66*	0*	-8.66*
PT	67	67	±3.5	-9.075	0	-9.075
OR	65	370	18.26	0	-12.11	0
SF	10	0	0	26.09	0	26.09
RP	874	612	-7.989	139.126	-4.13	152.359
SA	300	373	±.96	.5±11.7	∓46.	.5
AL	—	—	—	—	—	—
PL	12	—	-11.689	3.075	—	—
LG	—	—	—	—	—	—

*OTV listings for configurations 2 and 2_{NO}
For configurations 3 and 3_{NO}, centers of pressure are -8.18, 24.42 ±7.085, -4.415 ±.285, 24.42 ±7.085

module except for the HM and OTV. For these modules, the center of mass is displaced from the center by one-tenth of the length. The center of mass for the OR, RP, SF and SA components is given by the x,y,z coordinates as measured with respect to the reference-axes system. Axes systems are discussed in the next section.

13.2.2 SOC Configurations and Axes Systems

As shown in Figures 13-1 through 13-4, four specific configurations are analyzed. Configurations 2 and 3 are studied both with and without the attachment of the Orbiter (OR). The configuration mass properties are summarized in Table 13-2. The configuration center of mass is located with respect to the reference-axes system; the inertia tensor and the centers of pressure are measured with respect to the body-axes system. The reference-axes system has its origin in the center of the berthing port between the two SM's. The x-axis runs down the centerline of SM2, and the z-axis runs parallel to the HM centerlines as shown in Figures 13-1 through 13-4. The y-axis completes the right-hand orthogonal system. The body-axes system is parallel to the reference-axes system and has its origin at the configuration center of mass. A principal-axes system is also used, its origin also at the center of mass. The axes are located with a 3-2-1 Euler angle transformation, initially coinciding with the body-axes system. The principal moments of inertia are measured with respect to the principal axes system. The orbit-axes system also has its origin at the center of mass. The x-axis points in the direction of the flight path, the z-axis points toward the center of the earth, and the y-axis completes the right-hand orthogonal system. This axes system rotates in pitch (about its y-axis) with respect to an inertial system by the negative of the orbit rate.

The values of the principal moments of inertia are determined analytically and ordered in magnitude such that $I_y > I_x > I_z$. Then the Euler angles are calculated. This sequence of the principal moments of inertia provides stability when the SOC is flown with principal-axes aligned to orbit-axes.

Figure 13-5 through 13-10 give the relative relations of the axes systems.

13.2.3 Disturbances and Assumptions

There are three external disturbances most prevalent in the considerations of this SOC study. They are:

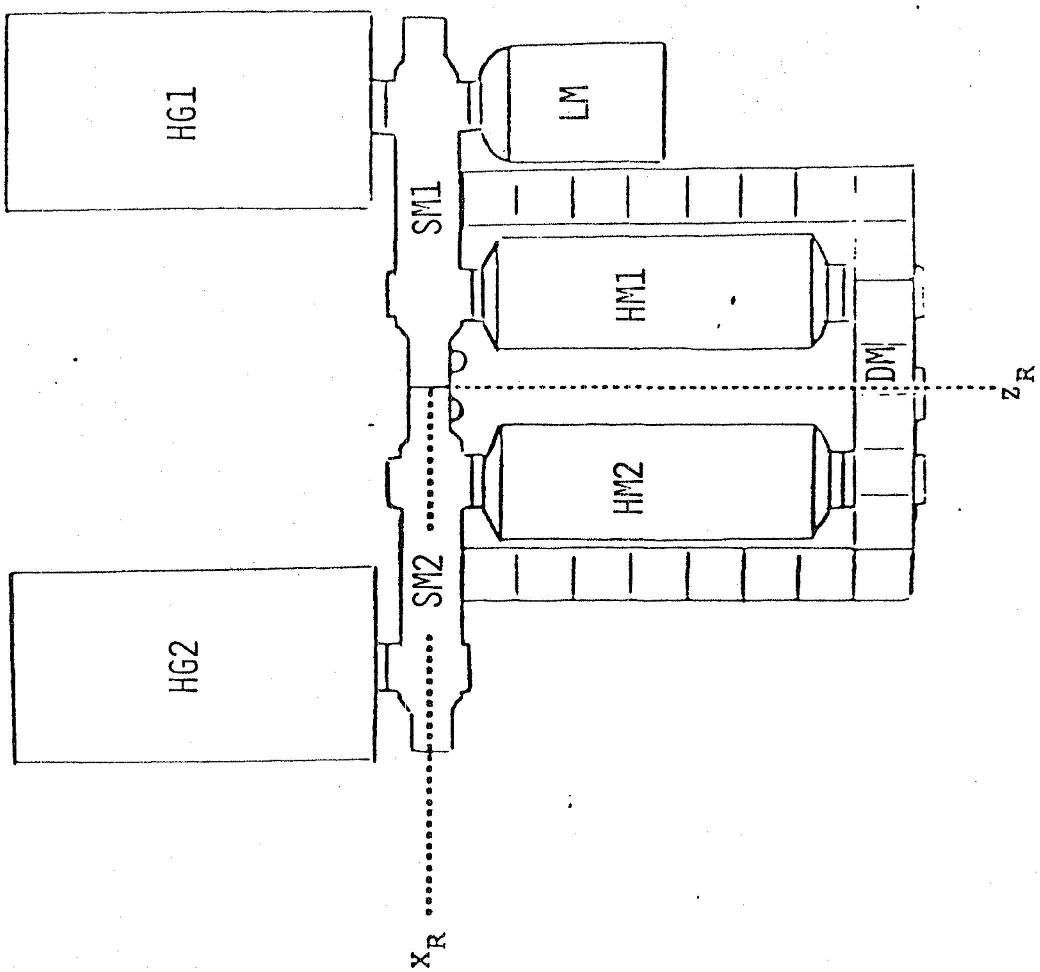


FIGURE 13-1. CONFIGURATION 1_{NO}

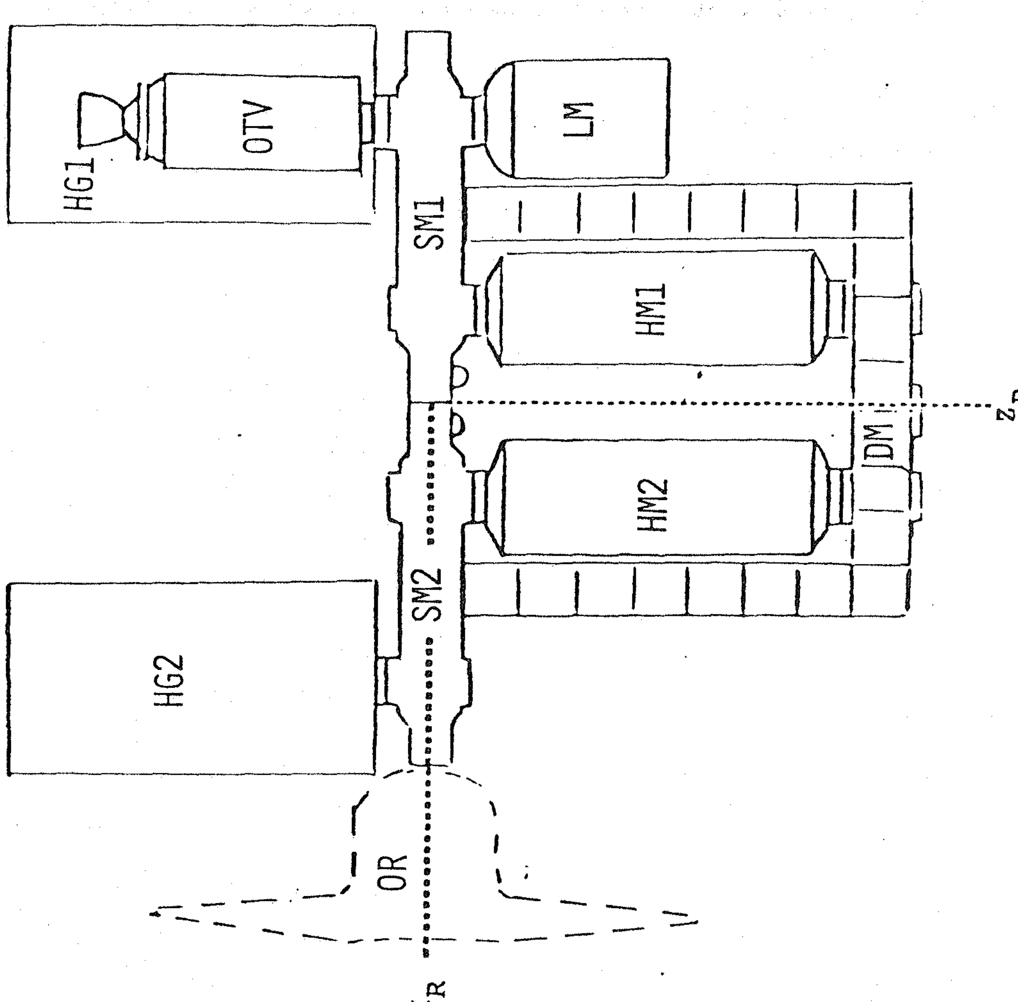


FIGURE 13-2. CONFIGURATIONS 2 AND 2_{NO}

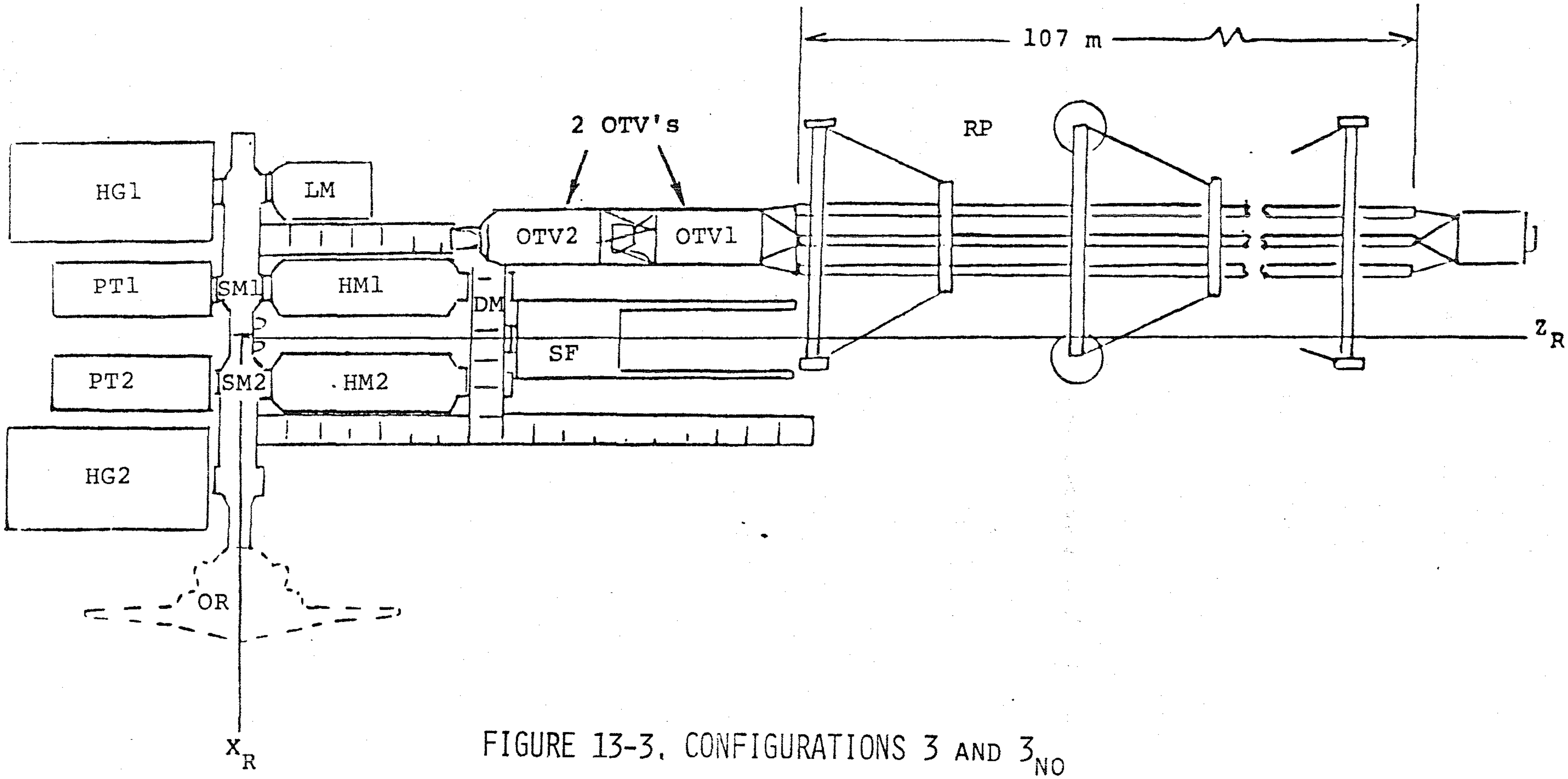


FIGURE 13-3. CONFIGURATIONS 3 AND 3_{NO}

FIGURE 13-4.
INITIAL SOC CONFIGURATION

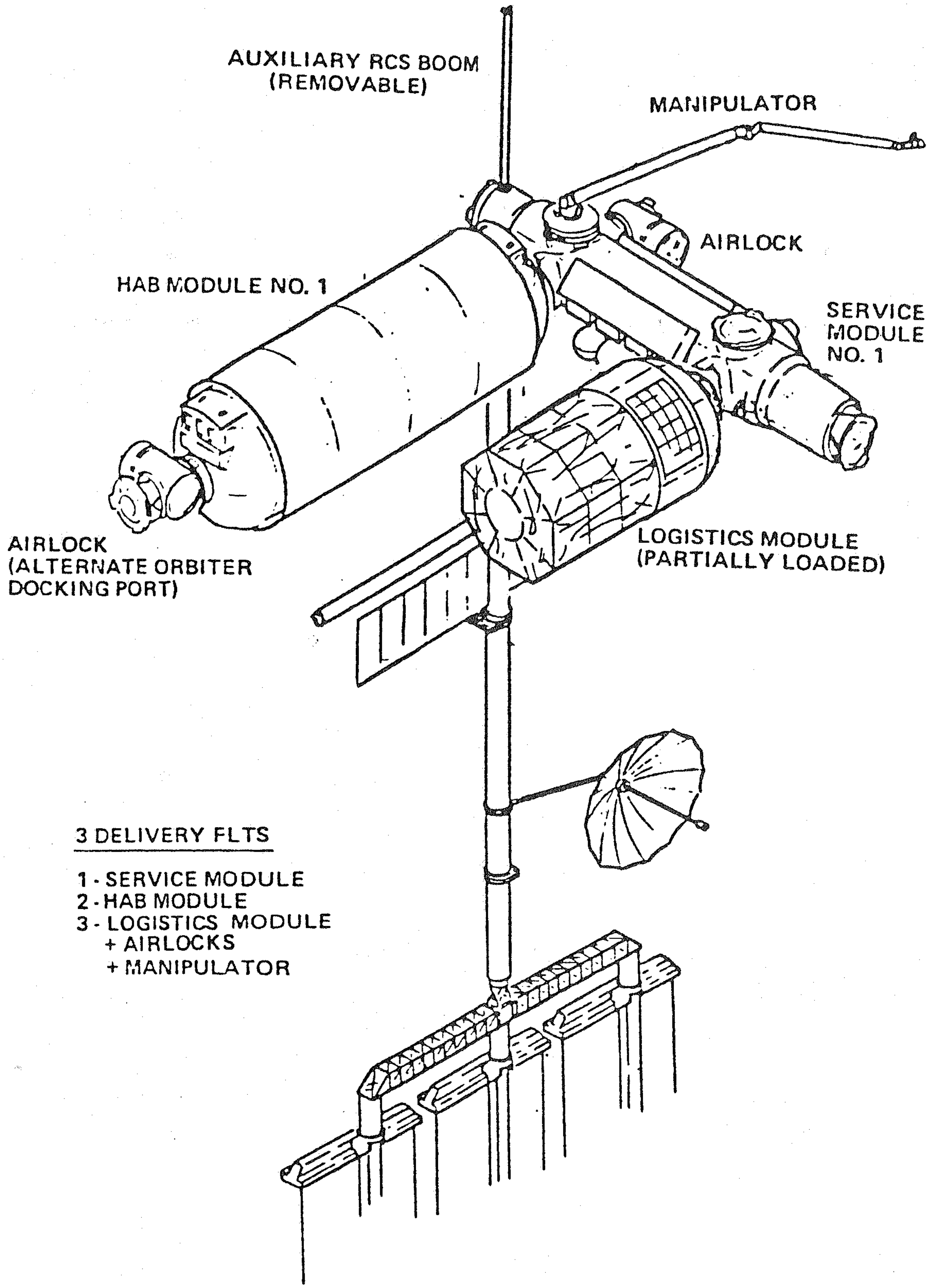


Table 13-2. *Configuration Mass Properties*

Parameter		Configuration ^{1}NO	^{2}NO	2	^{3}NO	3	^{4}NO
Total Mass (kg)		107250	145900	236809	329631	420540	41254
Center of Mass w.r.t. Reference—Axes (m)	x	-.959	-3.802	4.668	-3.193	1.444	-5.904
	y	0	0	-4.649	-1.455	-3.758	.635
	z	3.937	.6	.369	17.074	13.383	3.59
Inertia Tensor w.r.t. Body-Axes ($1000-kg-m^2$)	I_{xx}	9520	14724	30744	492327	528977	2069
	I_{yy}	13352	21827	50180	502098	556738	1716
	I_{zz}	8610	11978	54948	17599	65979	1866
	$-I_{xy}$	48	48	15012	-2343	13946	-117
	$-I_{xz}$	364	-3476	-2735	32616	58717	-298
	$-I_{yz}$	0	0	-407	16966	4002	78
Principal Moments of Inertia ($100-kg-m^2$)	I_x	9648	17088	54399	494390	528923	1684
	I_y	13352	21827	59044	502864	564132	2315
	I_z	8482	9613	22428	14770	58638	1652
3-2-1 Euler Angle Rotation such that Principal-Axes is aligned with Orbit-Axes (Degrees)	ψ	.8	.6	-74.3	-8.2	27.2	-82.9
	θ	19.3	-34.2	-61.4	3.9	7.1	-37.0
	ϕ	.2	-.2	-81.8	-2.0	-.3	10.3
3-2-1 Euler Angle Rotation, Orbit to Principal (Degrees)*	ψ	-.7	-.6	70.8	8.1	-27.4	84.1
	θ	-19.3	34.2	-66.9	-4.2	-6.2	14.5
	ϕ	.06	-.2	80.0	1.4	3.5	35.7
Total Areas (m^2)	xz-Plane	882	882	947	2020	2085	432
	yz-Plane	1008	1008	1348	1792	2132	457
Center of Pressure w.r.t. Body Axes (m)	xz-Plane X	.809	3.651	-3.554	-.815	-4.758	3.504
	Z	-7.755	-4.418	-3.926	42.433	44.269	-10.428
	yz-Plane Y	0	0	1.594	-.212	.426	36.91
	Z	-4.199	-.862	-.566	35.963	31.196	-1.718

*For gravity gradient torque calculation

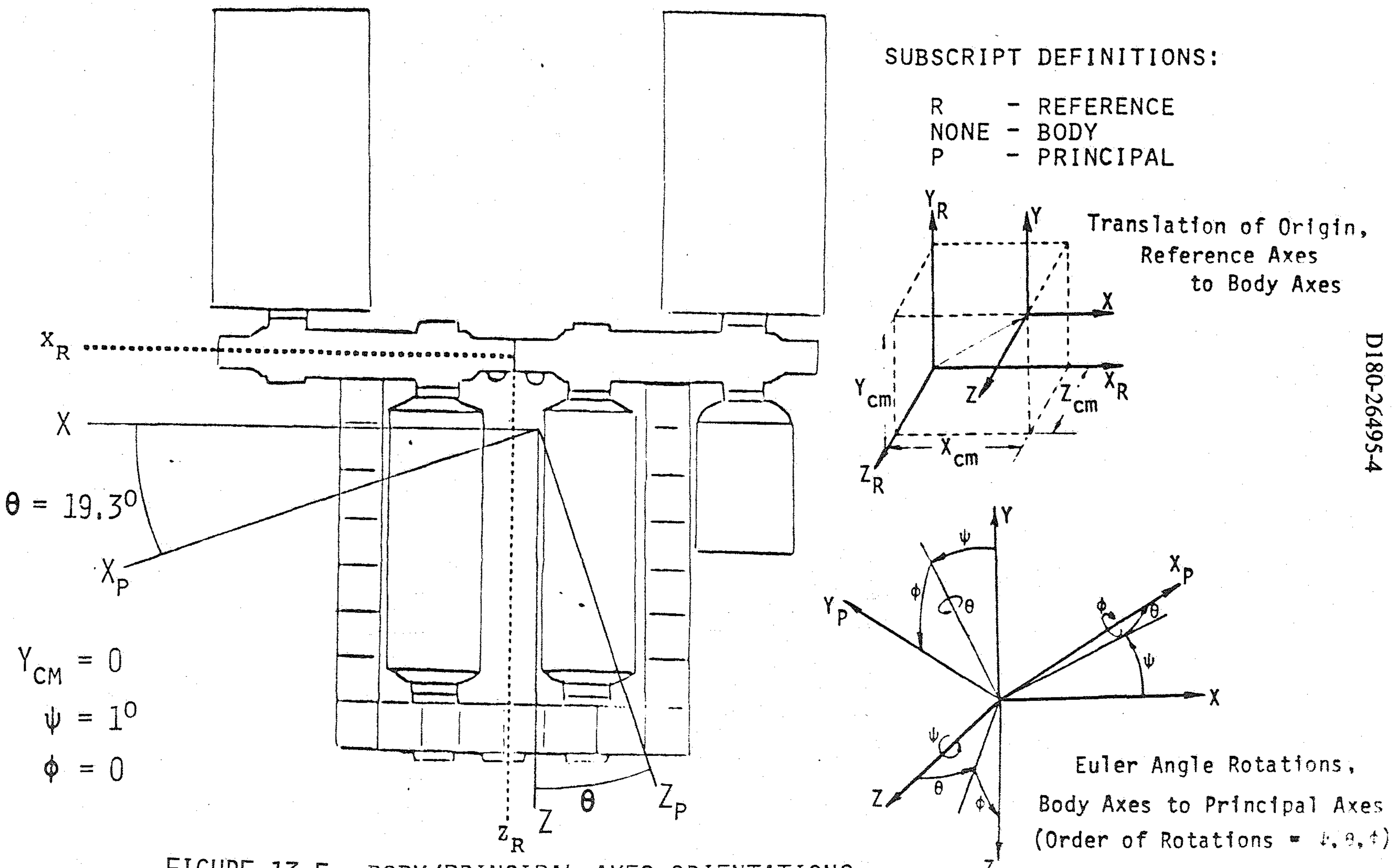


FIGURE 13-5. BODY/PRINCIPAL AXES ORIENTATIONS CONFIGURATION 1_{NO}

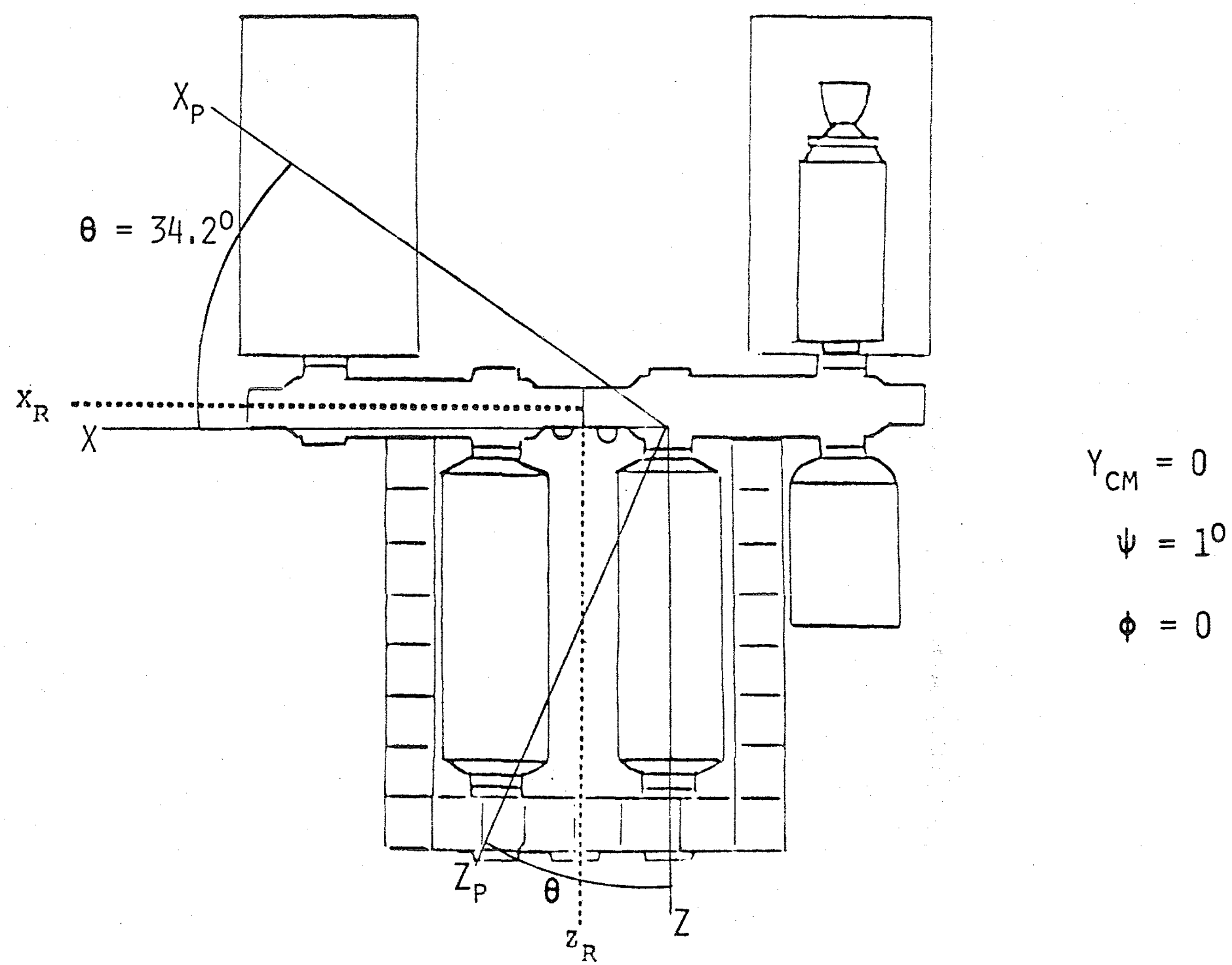


FIGURE 13-6. BODY/PRINCIPAL AXES ORIENTATIONS
CONFIGURATION 2_{NO}

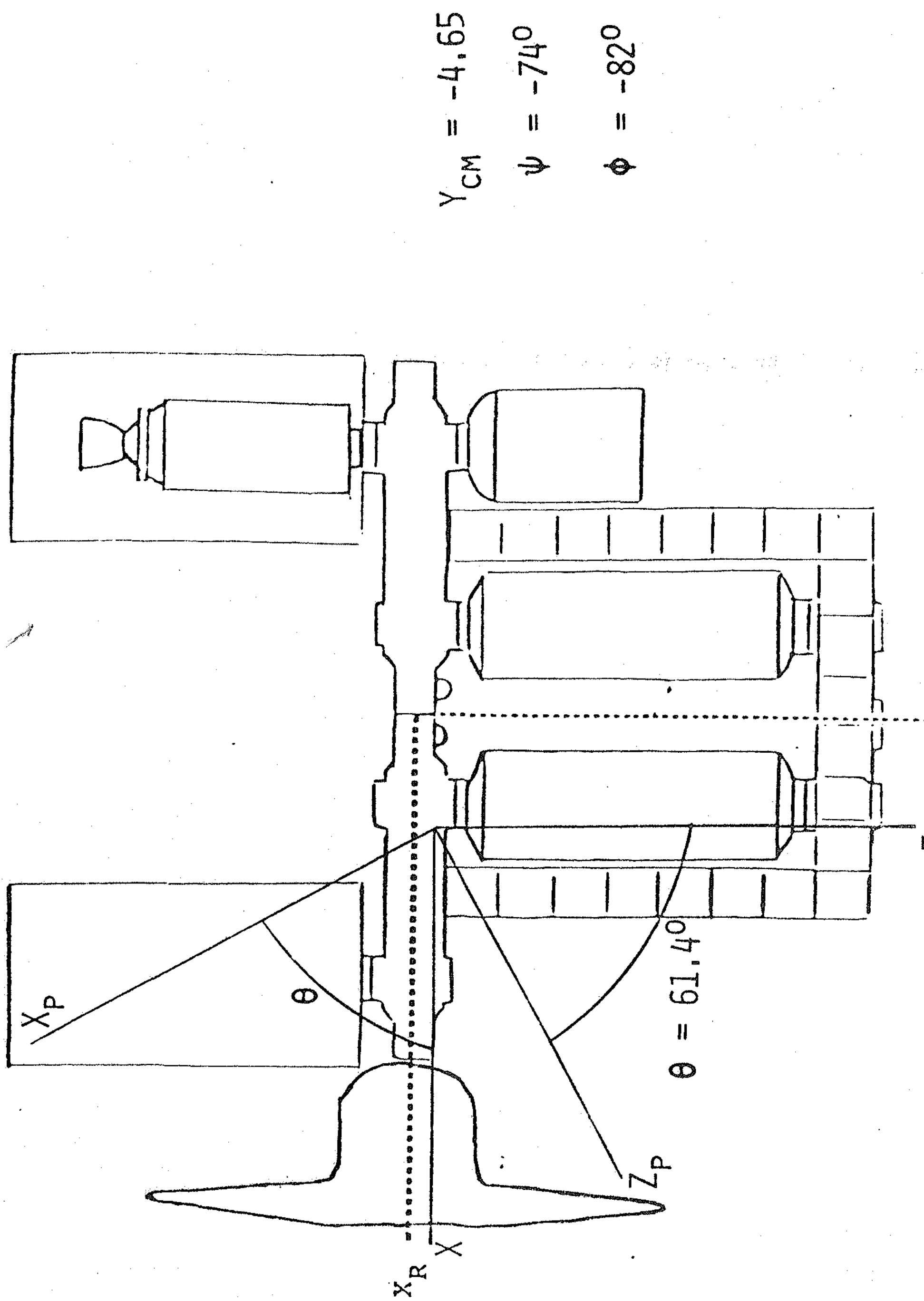


FIGURE 13-7. BODY/PRINCIPAL AXES ORIENTATIONS
CONFIGURATION 2

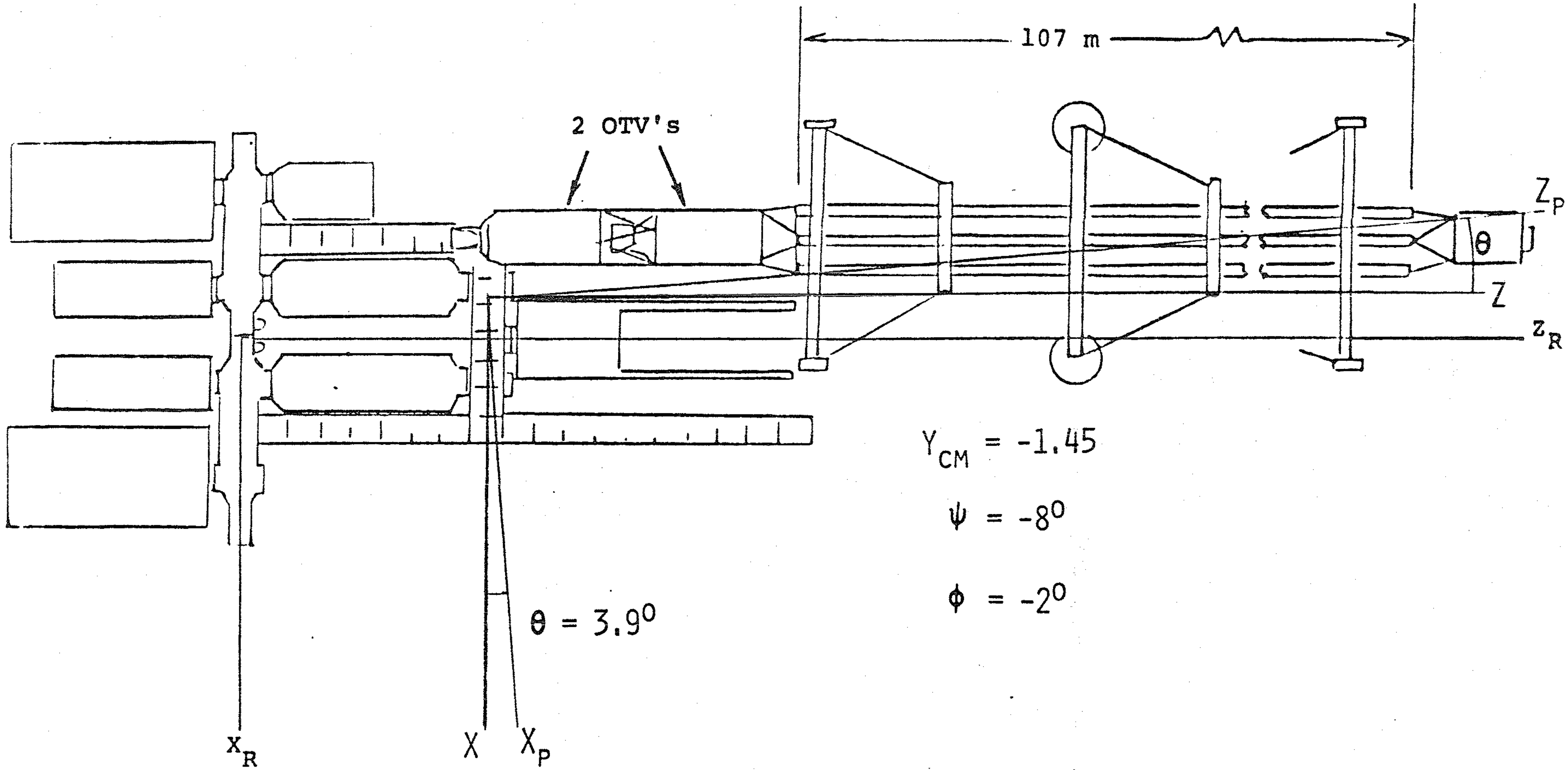


FIGURE 13-8. BODY/PRINCIPAL AXES ORIENTATIONS
CONFIGURATION 3_{NO}

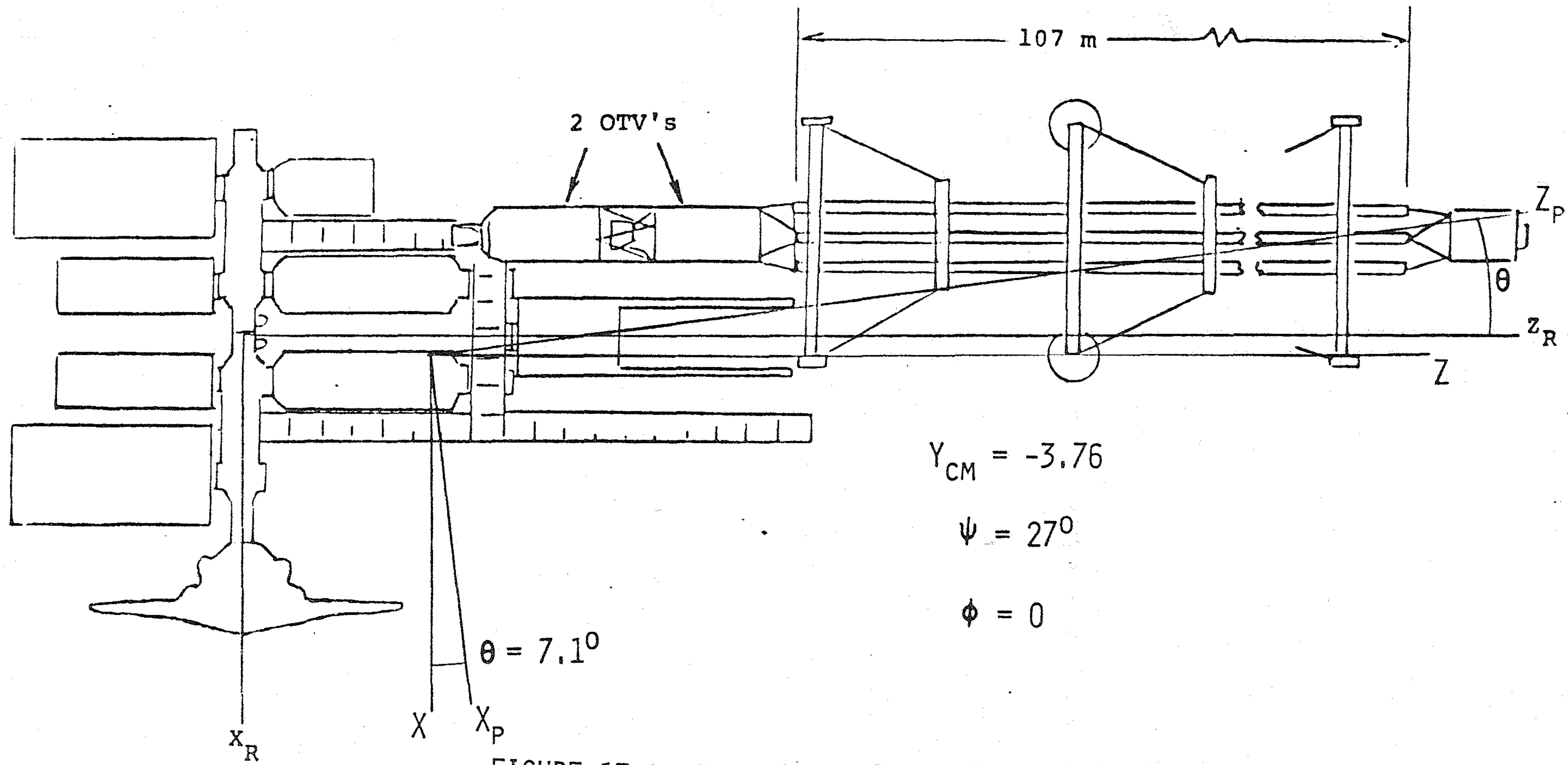
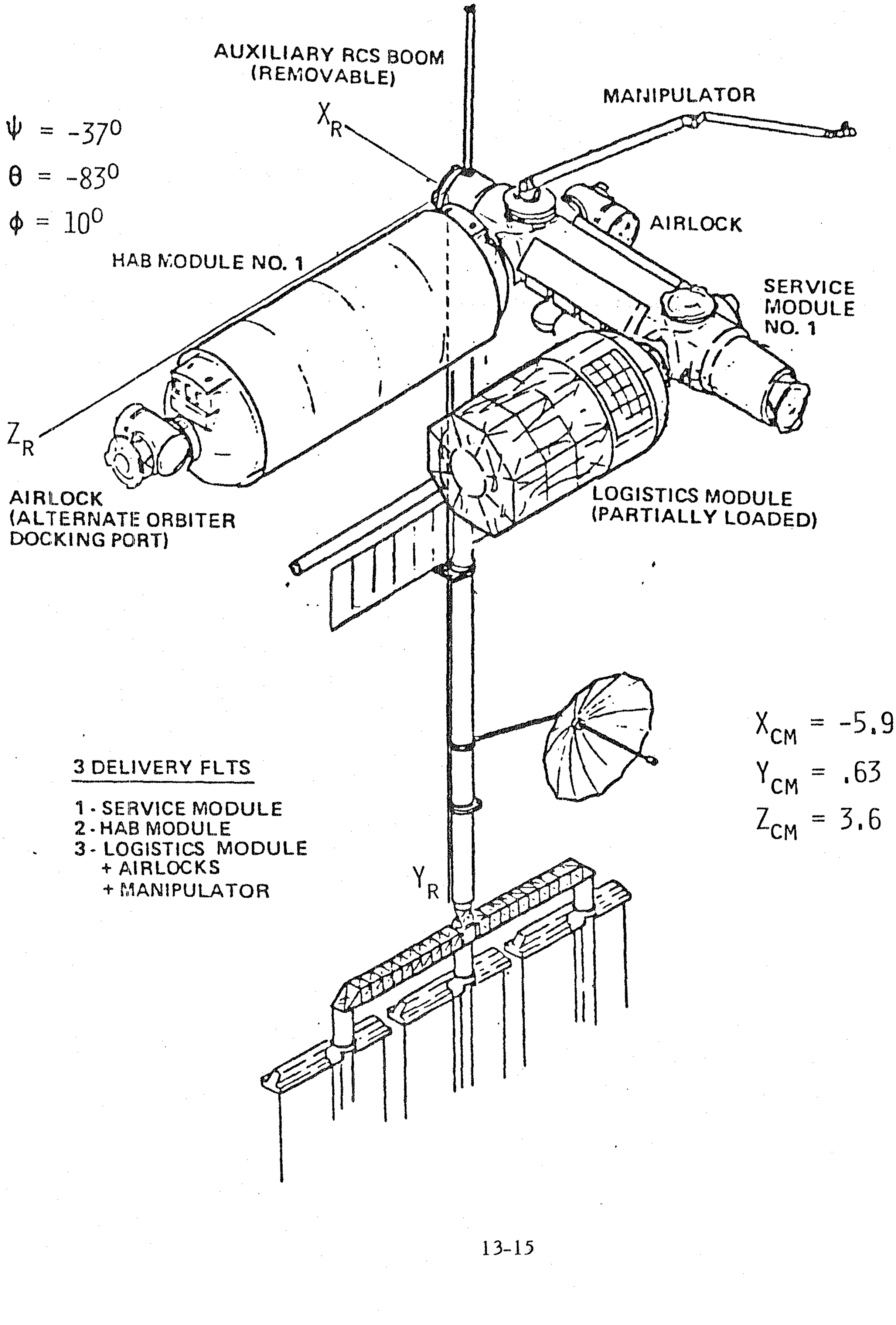


FIGURE 13-9. BODY/PRINCIPAL AXES ORIENTATIONS CONFIGURATIONS 3

FIGURE 13-10. BODY/PRINCIPAL AXES ORIENTATIONS
INITIAL SOC CONFIGURATION



Aerodynamic Drag
Gravity Gradient
Solar Pressure

The SOC is assumed to fly oriented to the earth in a circular orbit at an altitude between 370 to 405 km. The SOC will be free to seek its own orientation most of the time; that is, in the attitude in which the net torque is zero. It is desired to dock the OR to the SOC while flying the SOC with body-axes aligned to orbit-axes. This will introduce torques, and the effects of the above disturbances while flying in the docking orientation are evaluated. Only one solar aspect is considered: the sunline perpendicular to the orbit plane. The disturbance due to the solar pressure is greatest in this orientation, although relatively smaller than the other disturbances.

13.2.4 RCS Jet Control System

A specific design of an ACS using RCS jets is shown in Figures 13-11 and 13-12. The system of Figure 13-12 is used on configuration 4, the half-up SOC. The RCS clusters are located at the end of 8 meter booms which are attached to the solar booms about 9 meters from their base. The odd boom on the half-up SOC configuration is 10 meters long. The torque capability of this system is evaluated using a thrust level for the jets of about 133 Newtons (30 lbs).

13.2.5 Control Moment Gyro's

CMG's can be used to absorb the momentum of all cyclic torques. The gravity gradient torques in the roll-axis and the yaw-axis (T_x and T_z), and the solar pressure torques in roll and yaw are cyclic. The use of CMG's for secular torques has little advantage since the CMG's quickly saturate, require desaturation with the use of an external control source such as RCS jets, and do not reduce propellant requirements.

13.3 ANALYSIS

The torques due to the external disturbances and the control jets are analyzed in this section. The relative motion of two configurations are also estimated. The required momentum absorption capability of CMG's is calculated.

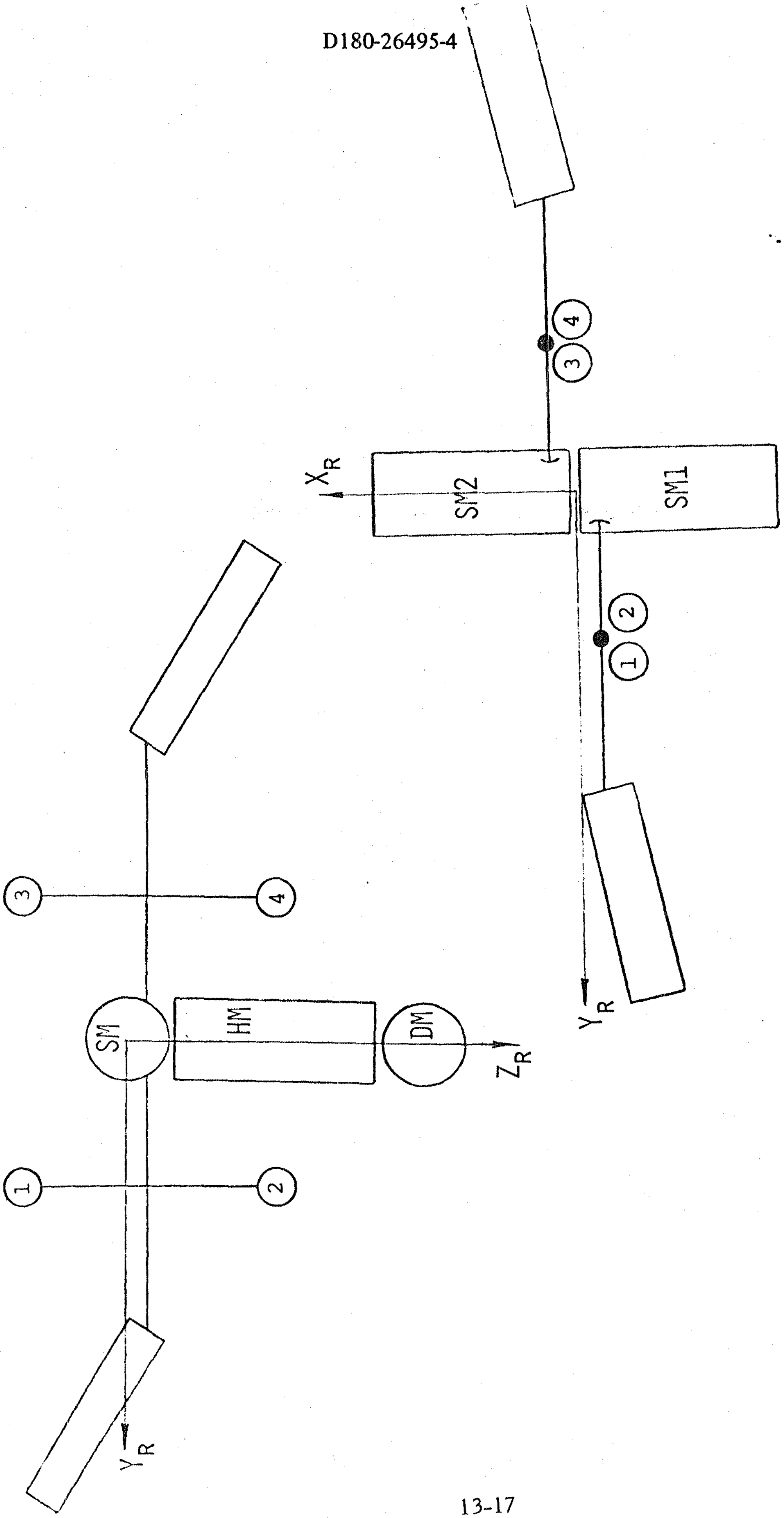


FIGURE 13-11. RCS CONTROL SYSTEM, CLUSTER LOCATIONS

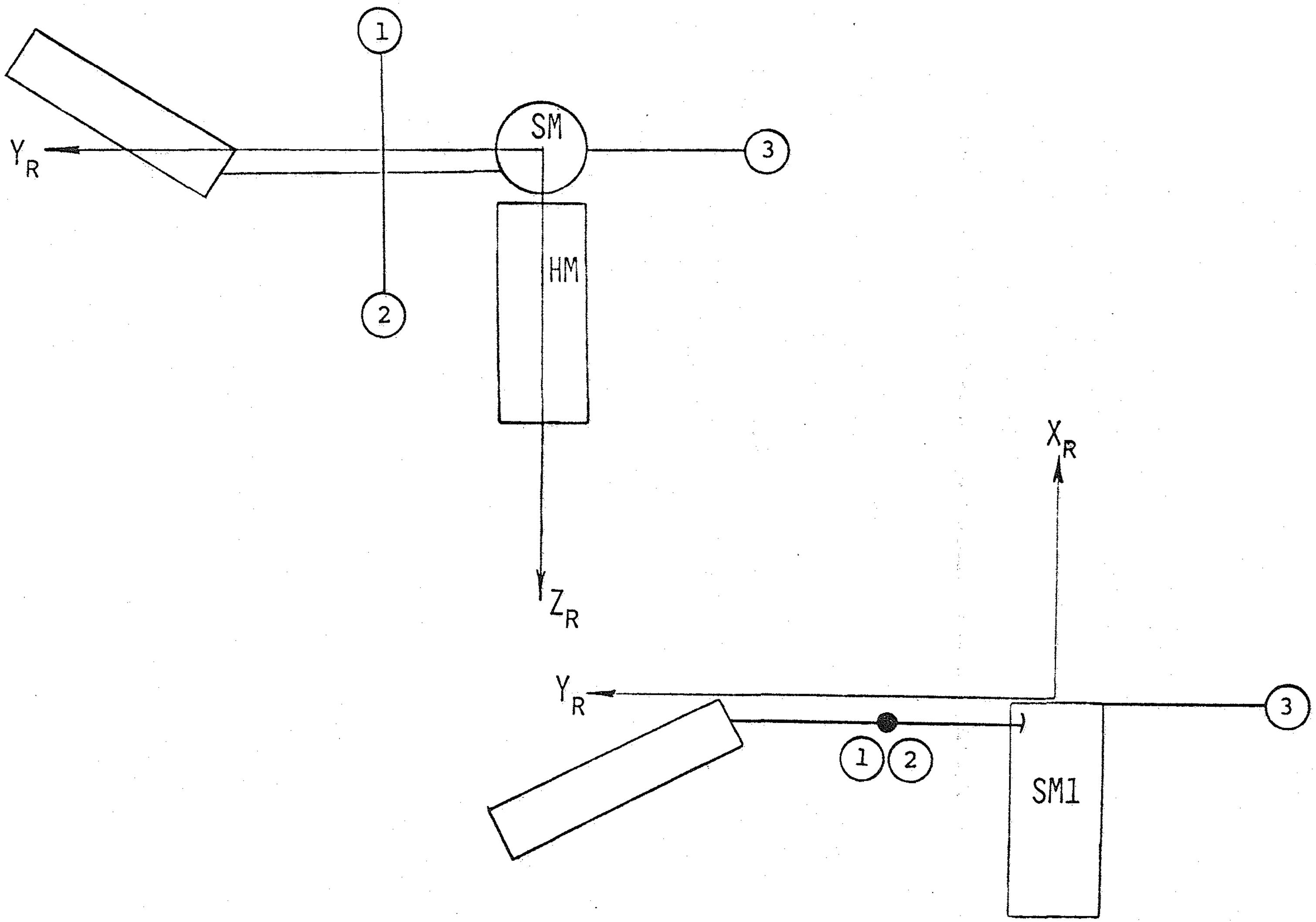


FIGURE 13-12. RCS CONTROL SYSTEM, CLUSTER LOCATIONS FOR CONFIGURATION 4_{NO}

13.3.1 Disturbance Torques

The disturbance torques are summarized in Table 13-3.

The aerodynamic torque governing equation is:

$$T_{AD} = q\, C_D\, S\, R$$

where q = dynamic pressure at the altitude of concern

C_D = drag coefficient (assumed 3)

S = projected surface area

R = moment arm from center of pressure as measured in body-axes

Several atmospheric models are used in the study of the SOC. For the purpose of this calculation, q is evaluated using the short-time maximum atmosphere at 370 km altitude; thus, q is approximately 0.28E-10 N/m^2.

The aerodynamic torque is separated into its y-component (body-axes), where the negative of the z-coordinate of the center of pressure (C_p) is used for R in the governing equation, and its z-component, where the y-coordinate of C_p is used for R. The surface area used is the surface area in the YZ-plane.

The solar pressure torque governing equation is:

$$T_{sp} = (1 + \gamma)\, U\, S\, R$$

where γ = reflectivity (assumed 1 for worst case)

U = solar pressure constant (4.673E-6 N/m^2)

S = projected surface area

R = moment arm from center of pressure

The solar pressure torque is separated into its x-component, where the negative of the z-coordinate of C_p is substituted for R in the governing equation, and its z-component, where the x-coordinate of C_p is used. The XZ-plane projected surface area is used for this calculation.

Table 13-3. *Disturbance Torques (N-m)*

Configuration	Gravity Gradient T_x	Gravity Gradient T_y	Aerodynamic Drag [1] T_y	Aerodynamic Drag [1] T_z	Solar Pressure T_x	Solar Pressure T_z
1_{NO}	-.017	1.42	10.5	0	-.064	-.007
2_{NO}	.114	-13.5	2.16	0	-.036	-.030
2	-3.75	7.77	1.89	5.33	-.035	.031
3_{NO}	-47.4	135	-160	-.94	.801	.015
3	-118	196	-165	2.25	.863	.093
4_{NO}	-1.15	-.024	1.95	41.8	-.042	-.014

[1] Short-time maximum atmosphere
$C_D = 3$
attitude = 370 km
$\Omega = .00113892$ rad/sec
$q = .28E\text{-}10\ N/m^2$

For the gravity gradient torques, both the x and y components are calculated. The governing equations for these components are:

$$T_x = \frac{3}{2}\Omega^2(I_x - I_y)\cos^2\theta \sin 2$$

$$T_y = \frac{3}{2}\Omega^2(I_z - I_x)\cos\phi \sin 2$$

where Ω = orbital rate
I_x, I_y, I_z = principal moments of inertia
ϕ = roll Euler angle
θ = pitch Euler angle

The orbital rate at 370 km is approximately 0.00113892 rad/sec. The calculation of the principal moments of inertia and the Euler angles are described in Section 13.7.

13.3.2 Control Torques

The control torque capability using the previously described design is summarized in Table 13-4. The calculations are based on jets firing in the negative x,y,z directions (body-axes) at each cluster location.

13.3.3 Relative motion of Uncontrolled SOC

Since configurations 1 and 2 are symmetrical with respect to the body-axes YZ plane, the relative motion in pitch can be estimated by the governing equations:

$$\ddot{\theta} = \frac{3}{2}\Omega^2\left(\frac{I_z - I_x}{I_y}\right)\sin 2\theta$$

and $$\theta_R = \frac{1}{2}\theta t^2$$

where Ω = orbital rate
I_x, I_y, I_z = principal moments of inertia
θ = pitch Euler angle
θ_R = relative pitch angle in time t
t = time (10 minutes)

Table 13-4. *Total RCS Torque Capability (N-m)*

	RCS Cluster	Direction of Firing	With Orbiter Attached T_x	With Orbiter Attached T_y	With Orbiter Attached T_z	Without Orbiter Attached T_x	Without Orbiter Attached T_y	Without Orbiter Attached T_z
Configuration 1	1	-x				—	-1613	-1237
		-y				1613	—	-.1
		-z				1237	.1	—
	2	-x				—	698	-1237
		-y				-698	—	-.1
		-z				1237	.1	—
	3	-x				—	-1613	1237
		-y				1613	—	256
		-z				-1237	-256	—
	4	-x				—	698	1237
		-y				-698	—	256
		-z				-1237	-256	—
Absolute Total						9572	5136	5461
Configuration 2	1	-x	—	-1137	-1858	—	-1168	-1237
		-y	1137	—	-751	1168	—	379
		-z	1858	751	—	1237	-379	—
	2	-x	—	1174	-1858	—	1144	-1237
		-y	-1174	—	-751	-1144	—	379
		-z	1858	751	—	1237	-379	—
	3	-x	—	-1137	617	—	-1168	1237
		-y	1137	—	-495	1168	—	635
		-z	-617	495	—	-1237	-635	—
	4	-x	—	1174	617	—	1144	1237
		-y	-1174	—	-495	-1144	—	635
		-z	-617	495	—	-1237	-635	—
Absolute Total			9572	7115	7440	9572	6653	6978

Table 13-4. *Total RCS Torque Capability (N-m) (Continued)*

	RCS Cluster	Direction of Firing	With Orbiter Attached T_x	With Orbiter Attached T_y	With Orbiter Attached T_z	Without Orbiter Attached T_x	Without Orbiter Attached T_y	Without Orbiter Attached T_z
Configuration 3	1	-x	—	-2874	-1739	—	-3366	-1431
		-y	2874	—	-321	3366	—	298
		-z	1739	321	—	1431	-298	—
	2	-x	—	-562	-1739	—	-1055	-1431
		-y	562	—	-321	1055	—	298
		-z	1739	321	—	1431	-298	—
	3	-x	—	-2874	736	—	-3366	1043
		-y	2874	—	-65	3366	—	554
		-z	-736	65	—	-1043	-554	—
	4	-x	—	-562	736	—	-1055	1043
		-y	562	—	-65	1055	—	554
		-z	-736	65	—	-1043	-554	—
Absolute Total			11821	7643	5720	13791	10547	6653
Configuration 4	1	-x				—	-1567	-1152
		-y				1567	—	-660
		-z				1152	660	—
	2	-x				—	744	-1152
		-y				-744	—	660
		-z				1152	-660	—
	3	-x				—	-479	1419
		-y				479	—	660
		-z				-1419	-660	—
Absolute Total						6515	4770	5703

For configuration 1_{NO}, $\theta_R = 1.1^{o}/10$ minutes
and for configuration 2_{NO}, $\theta_R = -6.4^{o}/10$ minutes.

13.3.4 CMG Sizing

The required momentum absorption capability is the impulse of the cyclic torque over one-half of the orbit. For a circular orbit, the governing equation is:

$$I = \frac{2T}{\Omega}$$

where T = cyclic torque
Ω = orbital rate

Table 13-5 summarizes the half-orbit impulses for the cyclic torques given in Table 13-3.

Table 13-5. *Cyclic Impulses (N-m-sec)*

Configuration	Gravity Gradient I_x	Solar Pressure I_x	Solar Pressure I_z
1_{NO}	29.9	112	12.3
2_{NO}	200	63.2	52.7
2	6585	61.5	54.4
3_{NO}	83237	1407	26.3
3	207200	1515	163
4_{NO}	2019	73.8	24.6

13.4 CONCLUSIONS

It is apparent from Table 13-3 that the effect of the solar pressure disturbance is negligible in comparison with the others. More importantly, it is apparent that the gravity gradient torques and the aerodynamic drag torques are of the same order for the configurations studied. The gravity gradient torques can be eliminated by flying the SOC with principal-axes aligned to orbit-axes, an attitude change which can be accurately predicted if the moments of inertia in body-axes are known. Torques due to aerodynamic drag (and solar pressure), however, would still be present. This is of major concern since the aerodynamic drag disturbance is of the same order as gravity gradient. Thus, if changes in attitude relative to earth are desired to be eliminated with minimum use of the ACS, a zero-torque attitude must be sought such that the effects due to aerodynamic drag and solar pressure exactly cancel the effect due to gravity gradient. This orientation is more difficult to determine since the projected surface area and C_p location are a function of attitude, but it is a problem that can be resolved with additional study. The sunline to solar arrays must be considered, however, as orientations without the body pitch-axis nearly normal to the orbit plane will not allow a full-faced solar array to be maintained. The zero-torque attitude would also vary, in general, through the cycle of the orbit since the solar arrays rotate continuously. This rotation, however, may be eliminated with the use of CMG's which will be effective in controlling the resulting cyclic torques.

When not flying in the zero-torque attitude, large torques are experienced as shown by Table 13-3. The torques listed in the table are those that would be experienced when the body-axes are aligned with the orbit-axes, the orientation presently chosen for docking operations. Thus, if the normal flying orientation is the zero-torque attitude, then the ACS is needed for reorientation to the docking attitude. Analysis of two configurations in the docking position indicates a possible controllability with RCS jets only, since estimation of the attitude rates and rate changes are relatively small.

Aerodynamic drag is seen to be most prevalent for configuration 4, the half-up SOC. The center of pressure is nearer the solar array and the center of mass is closer to the modules, thus producing a large moment arm. Reducing the projected surface area of the solar array would help to alleviate this situation.

The capability of the proposed ACS with RCS jets is seen from Table 13-4 to be quite adequate.

Although CMG's would be useful for the control of all cyclic torques, the results of Table 5 show that very large CMG's would be needed.

13.5 ADDITIONAL STUDY AREAS

The problem of designing an adequate ACS for the variable configuration SOC is quite complex and will require extensive study. The complexity arises primarily from the need for the ACS to be adaptable to extreme changes in the SOC inertia during buildup, and to mass movement during the construction phase.

Using the control system design as described herein as a baseline, control laws involving the cross-coupling effects of the RCS jets need to be evaluated. Selection laws, i.e. software governing the choice of RCS jet used for a particular maneuver, need to be developed in order to obtain pure torques most effectively. Criteria to measure effectiveness, such as propellant expenditure, rotation rate, etc., need to be defined. Studies to better determine the ACS requirements, and studies to evaluate the integration of control elements at every stage of buildup are necessary. Further evaluation of the use of CMG's is also needed.

13.6 CALCULATION OF PRINCIPAL MOMENTS OF INERTIA AND EULER ANGLES

The principal moments of inertia and the corresponding principal axes are determined by solving an eigenvalue problem. The procedure is outlined in Reference 3. Given the inertia tensor in body-axes as

$$\tilde{I} = \begin{bmatrix} I_{xx} & -I_{xy} & -I_{xz} \\ -I_{xy} & I_{yy} & -I_{yz} \\ -I_{xz} & -I_{yz} & I_{zz} \end{bmatrix} \tag{1}$$

the eigenvalue problem is written as

$$\tilde{I}[\omega] - I[\omega] = 0 \tag{2}$$

or equivanently as

$$\begin{bmatrix} (I_{xx} - I) & -I_{xy} & -I_{xz} \\ -I_{xy} & (I_{yy} - I) & -I_{yz} \\ -I_{xz} & -I_{yz} & (I_{zz} - I) \end{bmatrix} \begin{bmatrix} \omega_x \\ \omega_y \\ \omega_z \end{bmatrix} = 0 \tag{3}$$

It is assumed that $\omega_x, \omega_y, \omega_z \neq 0$; therefore, the determinant of the coefficient matrix must be zero.

The determinant yields a cubic equation in I. The three roots of this equation are the three principal moments of inertia.

The principal directions corresponding to the three principal moments of inertia are found by substituting the three roots, one at a time, into Equation 3. Only the ratios of $\omega_x, \omega_y, \omega_z$ can be solved, not their absolute magnitudes. Only two amplitude ratios are required; hence, only two of the three simulations equations given in Equation 3 need to be used. Arbitrarily omitting the first equation and dividing the other two by ω_x, gives in matrix form:

$$\begin{Bmatrix} \frac{\omega_y}{\omega_x} \\ \frac{\omega_z}{\omega} \end{Bmatrix} = \begin{bmatrix} (I_{yy} - I_{yz}) & -Iyz \\ -Iyz & (I_{zz} - I) \end{bmatrix} \begin{Bmatrix} I_{xy} \\ I_{xz} \end{Bmatrix} \tag{4}$$

These ratios indicate the direction of the axis corresponding to the given prinicpal moment of inertia. For example, arbitrarily setting $\omega_x = 1$, the values of ω_y and ω_z, together with ω_x, determine the direction of the corresponding principal axis.

These values can be converted to directional cosines by defining as

$$\bar{\omega} = (\omega_x^2 + \omega_y^2 + \omega_z^2)^{1/2} \tag{5}$$

Then, the directional cosines are given by

$$\cos \theta_i = {}^{\omega_i}/_{\bar{\omega}} \qquad \text{where} \quad i = x, y, \text{ or } z \tag{6}$$

A directional cosine matrix is evaluated, the columns of which are the directional cosines to each of the three principal axes. The directional cosine matrix is equated to the 3-2-1 Euler angle transformation matrix given as

$$\begin{bmatrix} C\Theta C\psi & C\Theta S\psi & -S\Theta \\ -C\phi S\psi + S\phi S\Theta C\psi & C\phi C\psi + S\phi S\Theta S\psi & S\phi C\Theta \\ S\phi S\psi + C\phi S\Theta C\psi & -S\phi C\psi + C\phi S\Theta S\psi & C\phi C\Theta \end{bmatrix}$$

where C = cosine

S = sine

ϕ = roll Euler angle

θ = pitch Euler angle

ψ = yaw Euler angle

The Euler angles can then be determined by the following:

$$\phi = \tan^{-1}\left(\frac{d_{23}}{d_{33}}\right)$$

$$\theta = \sin^{-1}(-d_{13}) \tag{7}$$

$$\psi = \tan^{-1}\left(\frac{d_{12}}{d_{11}}\right)$$

where d_{ij} = the element of the directional cosine matrix located on the i^{th} row and j^{th} column

13.7 DYNAMICS SIMULATION

A simple dynamics simulation was performed to assess controllability of the SOC. The simulation was conducted for configuration I_{NO} and included pitch-axis control only. The gain value was low enough that the control system would not excite the SOC first modes at approximately 0.04 Hz.

The control system block diagram is shown in Figure 13-13. A summary of the analysis and the equations is as follows:

EASY5 Simulation - Reference Configuration
(Configuration 1_{NO})

1. State Vector $= \{X\} = [p\ q\ r\ \phi\ \theta\ \psi]^T$

where p = roll rate
q = pitch rate
r = yaw rate
ϕ = roll angle
θ = Pitch angle
ψ = yaw angle

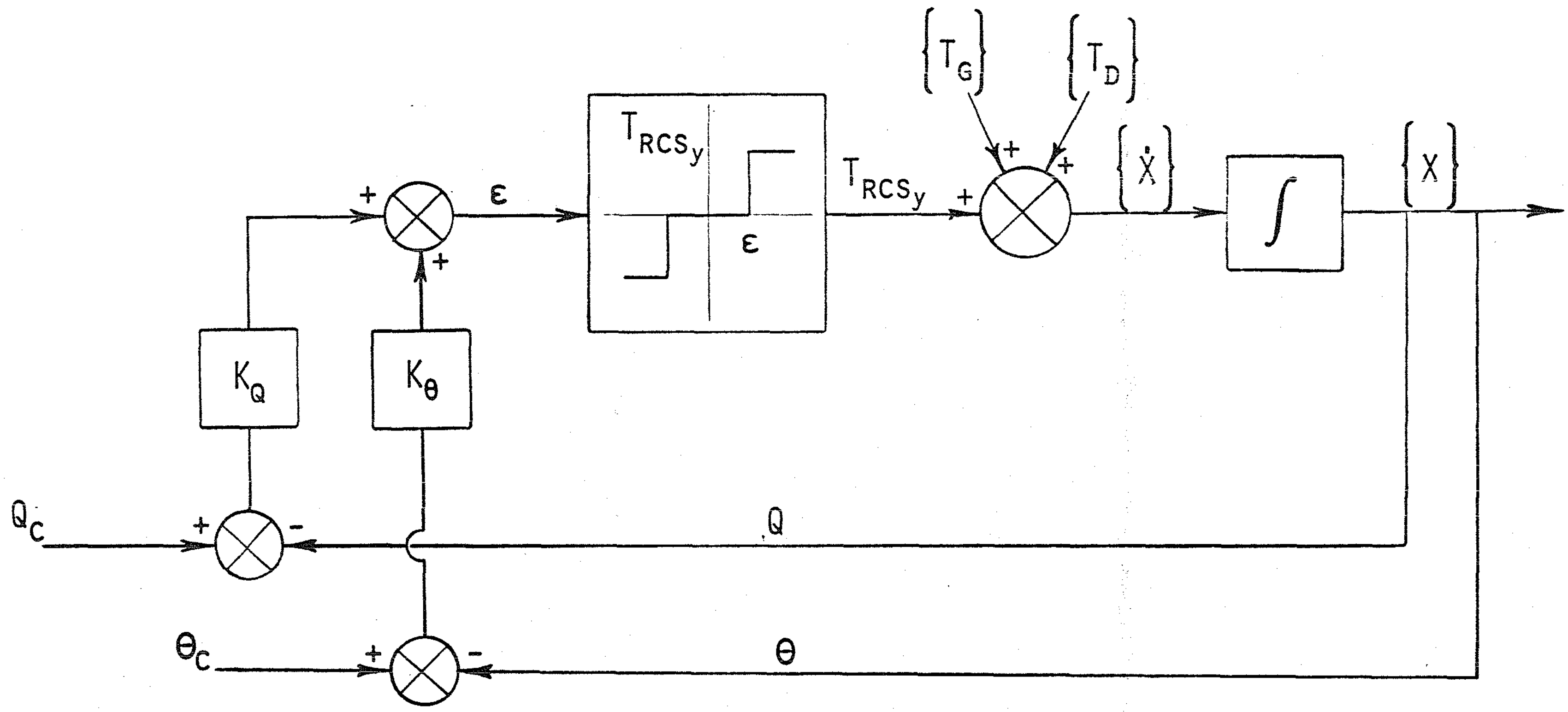
2. Gravity gradient torque $= \{T_g\} = [T_{gx}\ T_{gy}\ T_{gz}]^T$

$$T_{gx} = \frac{3}{2}\Omega^2 (I_z - I_y)\cos^2\theta \sin 2\phi$$
$$T_{gy} = \frac{3}{2}\Omega^2 (I_z - I_x)\cos\phi \sin 2\theta$$
$$T_{gz} = \frac{3}{2}\Omega^2 (I_x - I_y)\sin\phi \sin 2\theta$$

where

Ω = orbital rate
I_x, I_y, I_z = principal moments of inertia $(I_y > I_x > I_z)$
ϕ = roll angle
θ = pitch angle

3. Aerodynamic drag torque $= \{T_d\} = [T_{dx}\ T_{dy}\ T_{dz}]^T$
(assumed constant)



Q = Pitch Rate

Θ = Pitch Euler Angle

Q_C, Θ_C = Pitch Commands

K_Q, K_θ = Gains

ε = Error

$\{X\} = [P\ Q\ R\ \phi\ \Theta\ \psi]^T$ = State Vector

$\{T_G\} = [T_{G_x}\ T_{G_y}\ T_{G_z}]^T$ = Gravity Gradient Torque Vector

$\{T_D\} = [T_{D_x}\ T_{D_y}\ T_{D_z}]^T$ = Aerodynamic Drag Torque Vector

T_{RCS_y} = Pitch Control Torque Using RCS Jets

FIGURE 13-13. EASY-5 SIMULATION BLOCK DIAGRAM

4. State Equations

$$\dot{p} = [T_{g_x} - \int p + T_{d_x} + (I_y - I_z)qr]/I_x$$
$$\dot{q} = [T_{g_y} - \int (q + \Omega) + T_{d_y} + T_{RCS_y} + (I_z - I_x)pr]/I_y$$
$$\dot{r} = [T_{g_z} - \int r + T_{d_z} + (I_x - I_y)pq]/I_z$$
$$\dot{\phi} = p + (q \sin\phi + r\cos\phi)\tan\theta + \Omega \sin\psi \sec\theta$$
$$\dot{\theta} = q\cos\phi - r\sin\phi + \Omega\cos\psi$$
$$\dot{\psi} = (q\sin\phi + r\cos\phi)\sec\theta + \Omega\tan\theta\sin\psi$$

where the factors involving $\int$ are introduced for artificial damping

5. RCS Jet Control Torque (pitch axis) $= T_{RCS_y}$
(assumed constant)

6. Quantities Used

$I_x = 9.651 \times 10^6$ kg-m²
$I_y = 13.357 \times 10^6$ "
$I_z = 8.483 \times 10^6$ "
$\Omega = .00113892$ rad/sec
$T_{d_x} = -.03465$ N-m
$T_{d_y} = 1.3916$ "
$T_{d_z} = -.005657$ "

$\int = .01 \ \frac{kg-m^2}{sec}$
$K_Q = 100$ sec.
$K_\theta = 1$
$T_{RCS_y} = -10$ N-m
$\varepsilon = .001$

INITIAL CONDITIONS

$p = .945101 \text{ E-3}$ deg/sec

$q = -.065248$ "

$r = .104976 \text{ E-3}$ "

$\phi = .382°$

$\theta = -19.25°$

$\psi = -.879°$

State rates as calculated by equations in Section 4.
in rad/sec
State positions in radians
In EASY5 plots, rates are in degrees/sec
, positions in degrees.

The steady-state pitch angle θ is seen to be offset from the pitch command θ_c. It is offset by the amount of the error tolerance ε, i.e.,

$$\varepsilon = .001 \text{ rad} = .0573 \text{ degrees}$$

$$|\theta_{ss} - \theta_c| = |-19.18° + 19.25°| = .07°$$

The slight difference between error and offset is due to a partial contribution to the offset from the rate error.

The initial impulse seen on the pitch rate plot is due to transience of the pitch angle, and the pitch rate outside the boundaries of the duty cycle.

Figures 13-14 through 13-16 show results of the simulation.

Since the simulation was pitch only, the motions are nutations resulting from the once-per-orbit rotation in pitch and a slight asymmetry in distributions of mass. These cyclic motions could be controlled by a CMG set.

The commanded rate and position were given as the initial rate and positions; i.e., $q_c = q_o$ and $\theta_c = \theta_o$.

13.8 REFERENCES

1. J.A. Roebuck, Jr., Shuttle Considerations for the Design of Large Space Structures, NASA Contractor Report 160861, Contract NAS9-15718, Amend/Mod 4S, Rockwell International Corporation, November 1980.

2. Space Construction System Analysis, Part 2, Final Report, Platform Definition, SSD 80-0037, Contract NAS9-15718, Rockwell International, April 1980.

3. D. T. Greenwood, Principles of Dynamics, Prentice-Hall, Inc., Englewood Cliffs, New Jersey, 1965.

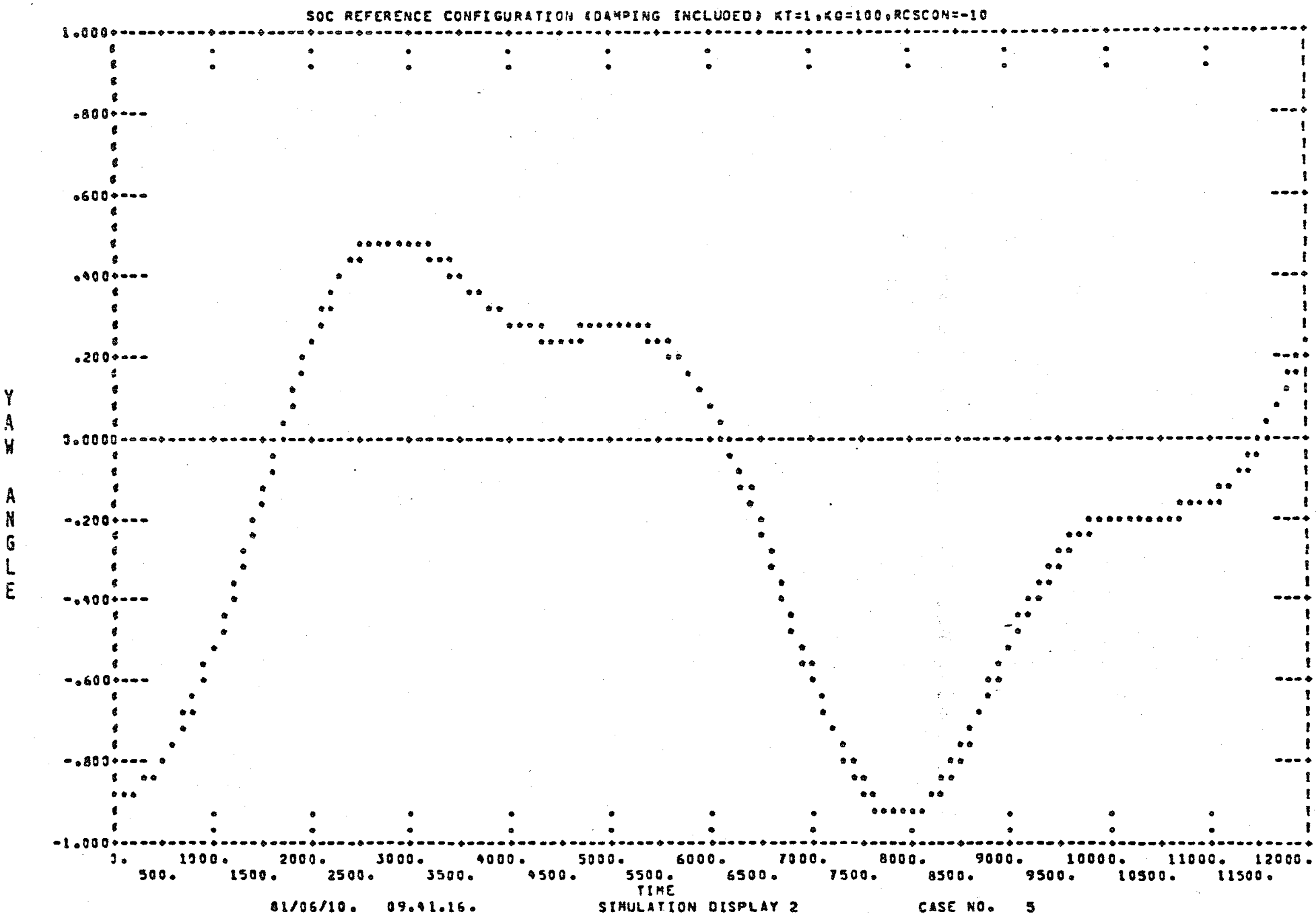


Figure 13-14. Yaw Angle vs. Time

SOC-1128

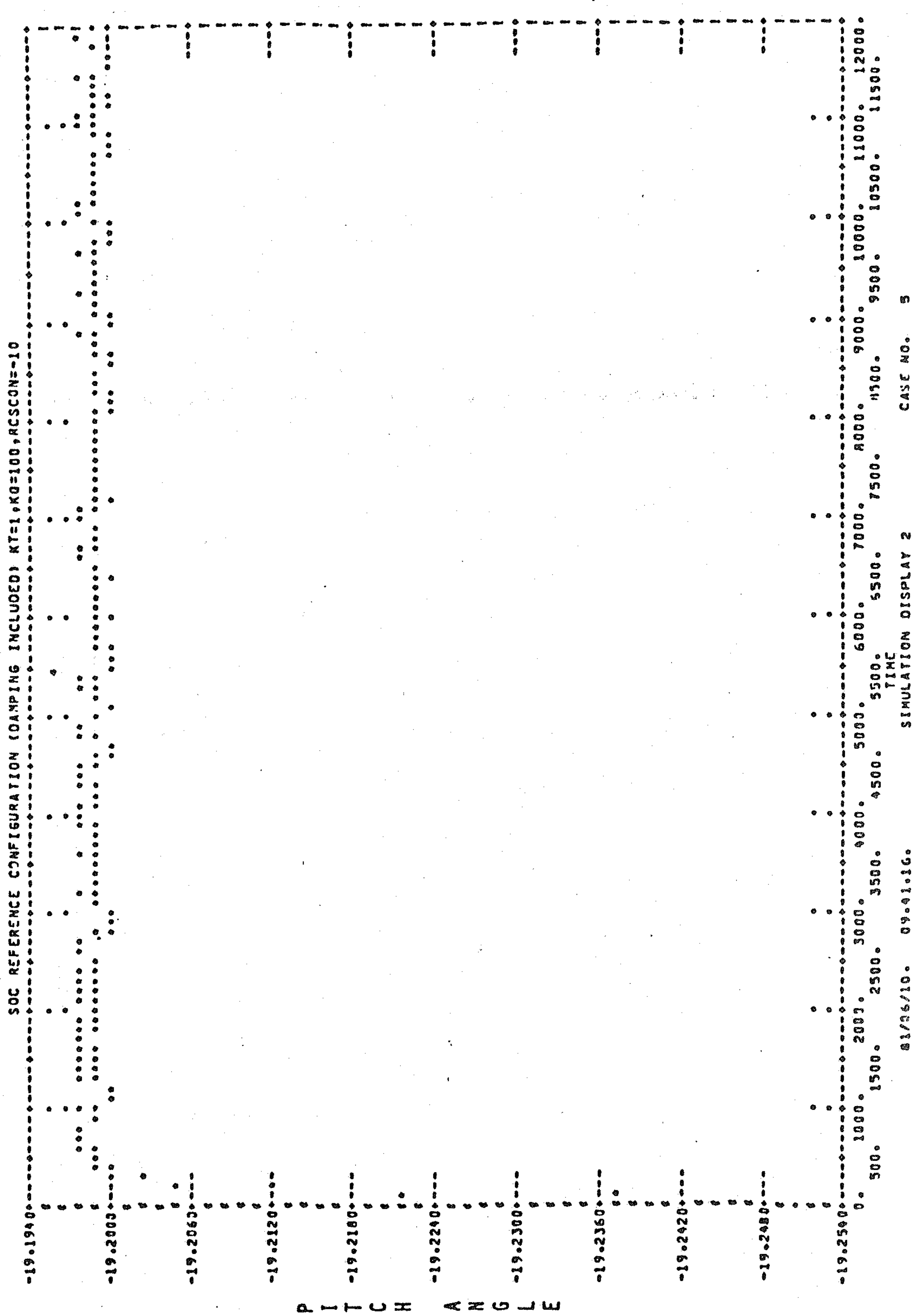


Figure 13-15. Pitch Angle vs. Time

SOC-1129

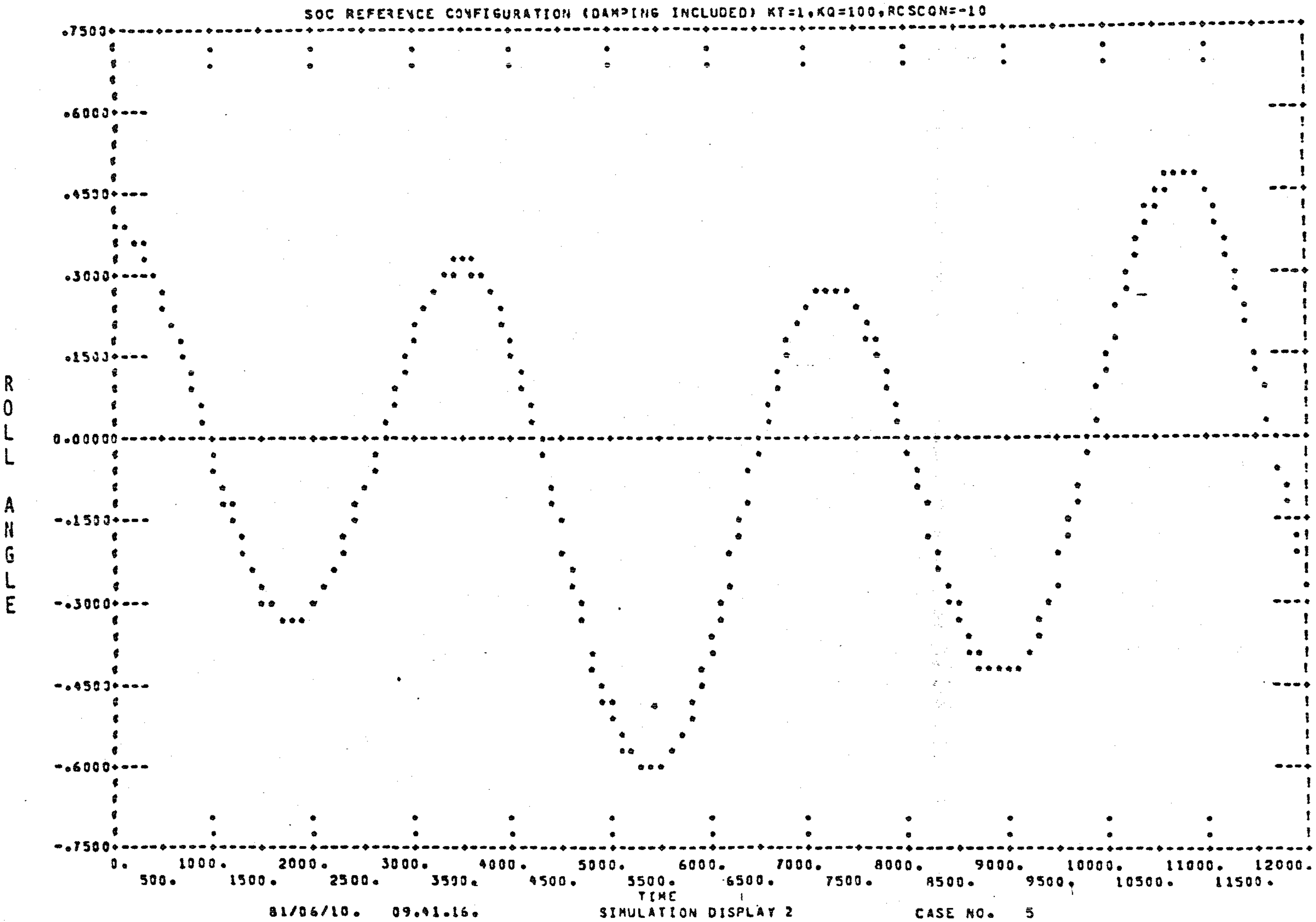


Figure 13-16. Roll Angle vs. Time

14.0 SOFTWARE AND DATA MANAGEMENT ANALYSIS

14.0 SOFTWARE AND DATA MANAGEMENT ANALYSES

14.1 INTRODUCTION

The objectives of the software and data management analyses were to (1) establish an approach for controlling software and processing cost, reliability and standardization, and (2) to develop a plan for controlling software sizing, flexibility, and redundance. The results of the analyses of these issues are given in the System Description Document (Boeing - 19) under WBS 1.2.1.1.10. Additional information on concepts and costs is presented in Section 14.2 below. The only data management analysis not summarized in the referenced document was an analysis of the use of ground-based software for test and checkout of satellites constructed at the SOC. This analysis is presented in Section 14.3 below.

14.2 SOFTWARE AND DATA MANAGEMENT CONCEPTS AND COSTS

Several data management considerations and issues were taken into account in assessing the SOC data processing functional requirements. These are summarized in Table 14-1.

The SOC Data Management System has both unique requirements and requirements which are functionally similar to existing command and control systems. The SOC DMS has the requirement to be incrementally built-up during early stages of the mission. The first segments of the SOC put into space will eventually become part of the SOC DMS but initially it must function autonomously. These parts of the DMS must first perform independently without the resources of the full SOC DMS and later become integrated into the full DMS. These system considerations must be taken into account early in the design of the DMS.

More common requirements of the DMS include the relatively long lifetime for SOC with its changing operational environment and the presence of a crew. The latter provides great flexibility in the design of the DMS with the added burden of defining an appropriate man-machine interface. These design considerations are common to several existing command, control and surveillance systems such as AWACS.

SOC-715

- SERVICE MODULE(S) MUST FLY AUTONOMOUSLY DURING BUILDUP PHASE
  - OFF-NOMINAL FLIGHT ATTITUDES
  - SPECIAL FLIGHT CONTROL LAWS
  - GROUND AND SHUTTLE COMMAND AND CONTROL
- SYSTEM WILL BE USED FOR 10 YEARS OR MORE IN CHANGING OPERATIONAL ENVIRONMENT
- PRESENCE OF CREW REQUIRES INTERACTIVE OPERATION BUT PERMITS:
  - MAINTENANCE AND REPAIR
  - HOT/COLD RESTARTS
  - OVERRIDES
  - EXCHANGE OF MASS STORAGE MEDIA
- CREW-SOFTWARE INTERFACE MUST EMPLOY FLEXIBLE AND EASY-TO-USE COMMAND LANGUAGE

SIMILAR TO AWACS

Table 14-1. SOC Data Management Design Considerations

Redundancy of the data management system is very important for the SOC given the 10 year design life goal and the complexity of the data management system. In order to provide sufficient redundancy, it has been concluded that we should employ a redundant or bypass bus architecture similar to one of those described in the systems description document. Further, the principal data management functions provided by Habitat Module 1 are backed up by the same functions provided by Habitat Module 2. In order to avoid system malfunction as a result of data bus breakdown, all processors will be operable standalone for critical functions. Although the operation of the processors in this mode may be somewhat of a nuisance to the crew and may not provide optimal results, this mode of operation will ensure crew survival and critical system operation until normal function can be restored.

The critical processors will be provided with redundancy and self-check. Utilizing advanced microprocessor technology, this can be readily accomplished. Finally, software will be backed up by non-erasable mass memory such that system crashes or breakdowns can be accommodated by cold restarts with back-up software.

A preliminary sizing estimate was made for the Space Operations Center software. The result, reported in the System Description document, predicted slightly more than 1.6 million lines of machine instruction code distributed among a number of processors.

Cost benefits of distributed processing are one of the more significant motivations for selecting this approach. The estimation of software cost is less well developed than the estimation of hardware cost. Techniques vary widely in their results. One of the more popular techniques is to assign a certain cost based on the number of lines of code that must be written. This technique is summarized on the left hand side of Figure 14-1. It makes no distinction between centralized and distributed systems and also does not allow for complexity or schedule impacts.

On the right side is illustrated a concept developed by Joe Gauger of Boeing. This concept correlates the software cost with the number of input and output data items that must be manipulated. It assumes that in a centralized system all

LINES-OF-CODE CONCEPT	
10% IN ASSEMBLER	–(162K)
20% IS UNIQUE	– 32K
COMMON	– 16K
REMAINING IN HLL at 5:1	– 292K
TOTAL	340K
PRODUCTIVITY 3 LINES/DAY	–433MY
GROUND SUPPORT SOFTWARE (X3)	1300MY
TOTAL	1733MY
COST	≈ $150M

RCA PRICE-S

CENTRALIZED

$1.1 BILLION

SEVERE SCHUDULE

CONSTRAINT–

ESTMATED

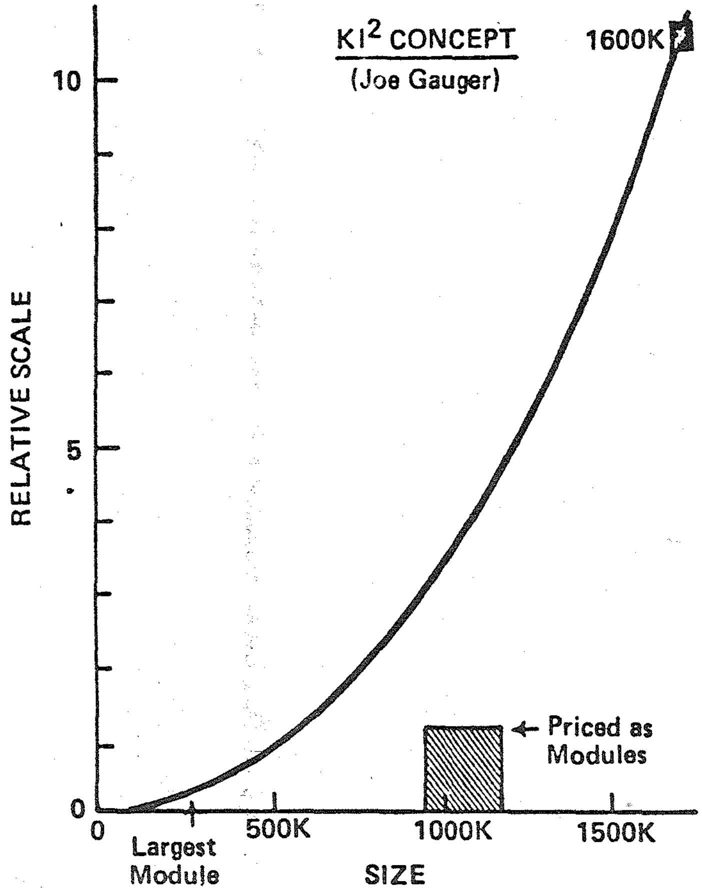
NEED 192 MONTHS

DISTRIBUTED

$370 MILLION (55 MO)
$320 MILLION (66 MO)

MODERATE
SCHEDULE
IMPACT

IDEAL SCHEDULE
97 MONTHS



- 66 MONTH SCHEDULE ASSUMES SOFTWARE STARTED IN PHASE B
- SAVINGS POTENTIAL FOR ADVANCED MICROPROCESSORS ON THE ORDER OF $100 MILLION

Figure 14-1. Software Costing Concepts elements interact but that in a properly designed distributed system the number of interactions is confined to the module level. Excellent historical correlations were developed by Gauger showing that the software cost scales as the square of the number of input/output items that must be manipulated. (The historical data base did not include distributed systems.) This costing concept indicates that, if the distributed system can be designed to isolate the modules with a minimum of interaction, great savings should be achieved by distributed processing as indicated.

A third method is use of the RCA Price-S software cost-estimating model. Price-S is an extension of the early RCA Price cost model developed for hardware cost. Like other models, Price-S has not been designed for distributed processing estimation; however, indications were obtained by pricing the SOC software system as a single unit of 1600 K lines of code and pricing the software by individual module sizes and adding an integration cost for integrating the modules together. Price-S includes schedule impact and complexity factors. It prices manned spacecraft software considerably higher than missile or aircraft software. The estimates are summarized in the middle of the figure. The Price-S estimates for centralized processing include a severe schedule impact. Much less cost was estimated for the distributed system with a less severe schedule impact. Although the available estimating models do not give an adequate distinction between distributed and centralized processing, they indicate a clear cost advantage for distributed processing.

If one removes the schedule impact penalties, the Price-S distributed cost estimate is not too different from the lines-of-code estimate. The importance of schedule emphasizes the necessity of getting an early start on software design during a Phase B systems definition study or an SOC technology program.

Illustrated in Figure 14-2 is a cost estimating relationship based on the Price-S results for the particular software characteristics estimated to be applicable for the Space Operation Center. As may be seen, the cost trends are non-linear, suggesting that modularization of software will have substantial cost benefits.

SOC-1108

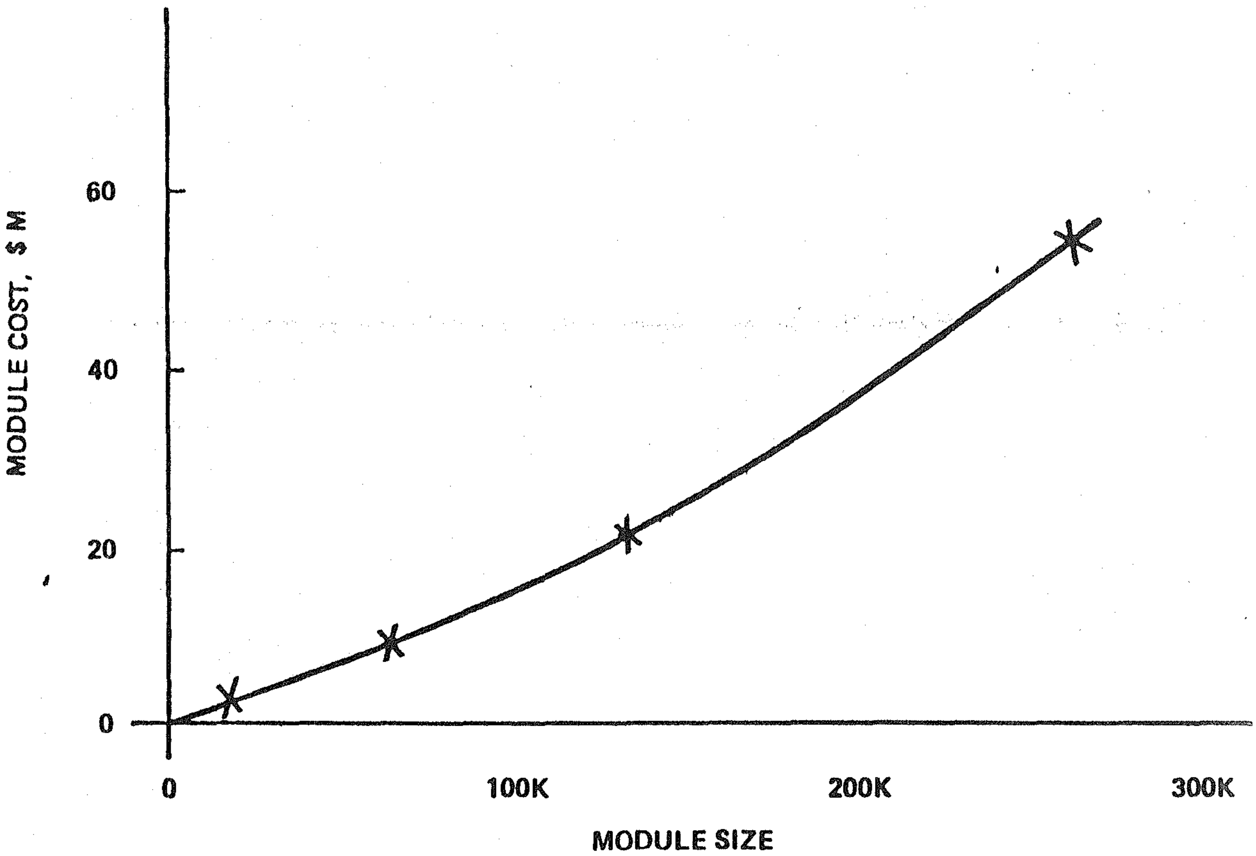


Figure 14-2. Software Cost Trend

The software elements estimated for use in the Space Operation Center cover the range of requirements from initial SOC through growth SOC. Accordingly, it was important to subdivide and allocate the software element costs and requirements to the various SOC configurations. The results are summarized in Table 14-2.

The evaluation of distributed processing techniques to satisfy SOC functional requirements has uncovered no major technology obstacles but the Data Management System design schedule is challenging. Evaluation results are summarized in Table 14-3. Major benefits can be achieved using the distributed approach but tight management control is essential. Early development of subsystem functional allocations and Interface Control Data is mandatory. The subsystem allocations of hardware vs. software must be made using system engineering techniques to provide an integrated and efficient design approach. Since several technologies can be traded off in this process, flexibility of design still remains.

Several specific needs for Phase B have been identified.

- Technology tradeoffs and selections need to be accomplished, especially in the areas of:
  - hardware
  - data bus design
  - programming language selection
- The preliminary functional design of the software needs to be completed to identify any problem areas early.
- The development/selection of a common operating system for the subsystems must be accomplished to allow proper definition of the system before full scale development begins.
- The data bus architecture and accompanying protocol must be defined and validated, probably requiring some amount of system simulation.

A review of advanced microprocessor technology was conducted as a part of the software task. The Intel 432 was taken as representative of this technology. The Intel devices were not compared with alternative 32-bit machines; such a comparison must be made before specific hardware is selected.

SOC-1080

FUNCTION	SIZE	INITIAL SOC	OPERATIONAL SOC	GROWTH SOC
MAIN SYSTEM	256K	40	14.8	-0-
DISPLAYS	256K	54.8	-0-	-0-
COMMUNIC.	128K	20.8	-0-	-0-
DATA BASE	128K	10	10.8	-0-
INSTRUMENTATION	16K	2.2	-0-	-0-
HEALTH MAINT.	32K	-0-	4.6	-4-
EC/LSS	64K	7	2.6	-0-
SHUTTLE INTERFACE	16K	2.2	-0-	-0-
FLT CONTROL	256K	40	14.8	10
POWER	64K	8	1.6	2
PROPULSION	16K	2.2	-0-	-0-
UPPER STG C/O	64K	4	5.6	4
FACIL. EQUIP.	128K	10	21	10
SPACECRAFT C/O	64K	4.6	5	5
EXPERIMENT	64K	6.6	3	3
CRYO SFER	64K	-0-	-0-	9.6
TEST & INTEG		16.9	10	10
TOTALS		229.3	93.8	67.6

Table 14-2. Software Elements Cost (Millions of Dollars)

SOC-696

- NO MAJOR OBSTACLES IDENTIFIED BUT SCHEDULE IS CHALLENGING
- SEMI-AUTONOMOUS SUBSYSTEMS WITH DISTRIBUTED PROCESSING OFFER SUBCONTRACT MANAGEMENT ADVANTAGES
- EARLY DEVELOPMENT OF SPECS AND ICD'S IS ESSENTIAL TO REALIZE BENEFITS OF DISTRIBUTED PROCESSING
- TO PREDICT AND CONTROL LIFE CYCLE COSTS, SOFTWARE MUST BE SYSTEM ENGINEERED CONCURRENTLY WITH HARDWARE
- ADVANCED MICROPROCESSORS VERY PROMISING

- SPECIFIC PHASE B NEEDS:
  - COMPLETE TECHNOLOGY TRADEOFFS AND SELECT:
    - HARDWARE (PROCESSORS, BUS IMPLEMENTATION, MASS STORAGE)
    - BUS ARCHITECTURE & PROTOCOL
    - CODING LANGUAGE(S)
  - PRELIMINARY DESIGN SOFTWARE IN LOCKSTEP WITH HARDWARE
  - GET HEAD START ON OPERATING SYSTEM DEVELOPMENT
  - VALIDATE ARCHITECTURE & PROTOCOLS BY LABORATORY SIMULATION

Table 14-3. Evaluation Results

The Intel 432 possesses several potential advantages compared with earlier systems. These features are intended to reduce software costs and are summarized in Table 14-4. If these features prove out, software cost savings of 25 to 50% are not inconceivable.

14.3 FEASIBILITY STUDY OF USING GROUND BASED SOFTWARE FOR TEST/CHECKOUT OF SATELLITES CONSTRUCTED AT THE SOC

The Space Operations Center will be used for the space construction of large structures and systems. These mission payloads will be relatively ill-defined when the SOC is defined. Great flexibility in the design of the SOC will be required.

A problem in the definition of the mission payload data processing concerns a requirement for computer checkout of the system under construction (i.e., a computer and its software which is not part of the mission payload but which will be required during the construction and test phase to ensure that the payload is operating properly).

Three general approaches could be utilized to meet this requirement. First, the computer that is required could be space-qualified, launched with the payload, utilized and returned. Second, a test computer could be provided in the SOC for which all checkout software would be written. Third, the information required for the checkout of the mission payload could be telemetered between the SOC and the ground. These three alternatives are depicted in Figures 14-3, 14-4 and 14-5. The advantages and disadvantages of each of these approaches will be discussed.

The use of a <u>mission payload unique computer</u> will give the payload designer the maximum flexibility in designing both the payload and the test portion of the system. In fact, it is not unlikely that certain construction tasks will require this approach due to special requirements that are unforeseeable at the time of design of the SOC. The disadvantages are that the test computer will have to be space-qualified and that it will have to physically be transported to the SOC.

SOC-1077

ITEM	ADVANCED MICROPROCESSOR	CURRENT S.O.A.
INSTRUCTION LENGTH	6 to 300 BITS	8 to 32 BITS
INSTRUCTION TYPE	ADA (NO ASSEMBLY LANGUAGE WILL EXIST)	MACHINE LANGUAGE–HOL COMPILERS AVAILABLE
DATA TYPES	CHARACTER, INTEGER, SINGLE & DOUBLE PRECISION FLOATING POINT	BINARY: CONVERSION TO OTHER TYPES DONE WITH SOFTWARE. ARITHMETIC CHIPS AVAILABLE
OPERATIONS SYSTEM	SOME INHERENT: SOME DONE WITH ADA	ASSEMBLY–LANGUAGE SOFTWARE
DATA PROTECTION	INHERENT IN CHIP & LANGUAGE DESIGN	MUST BE DONE WITH SOFTWARE: OFTEN NOT DONE
PARALLEL PROCESSING, MULTITASKING, INTER–PROCESSOR COMMUNICATIONS	INHERENT IN CHIP & LANGUAGE DESIGN	MUST BE DONE WITH SOFTWARE
FUNCTIONAL REDUNDANCY CHECKING	INHERENT IN CHIP DESIGN	MUST BE DONE WITH SOFTWARE

Table 14-4. Features of Advanced Microprocessor (INTEL 432)

SOC-400

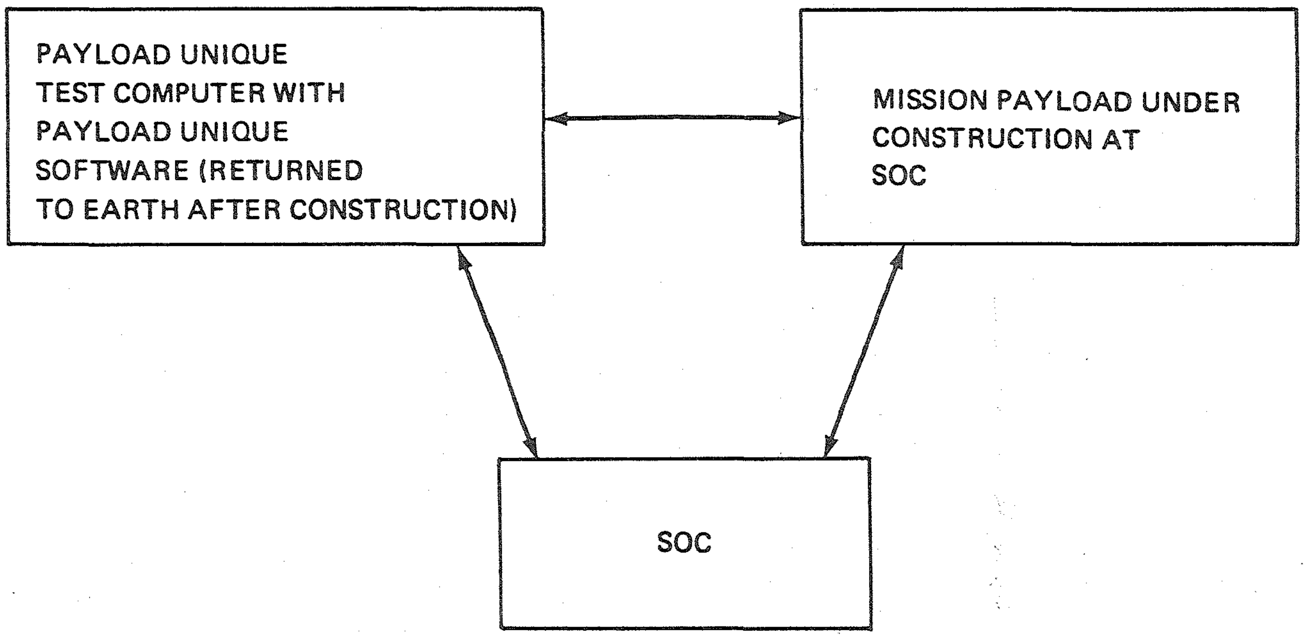


Figure 14-3. Payload Unique Test Computer

SOC-401

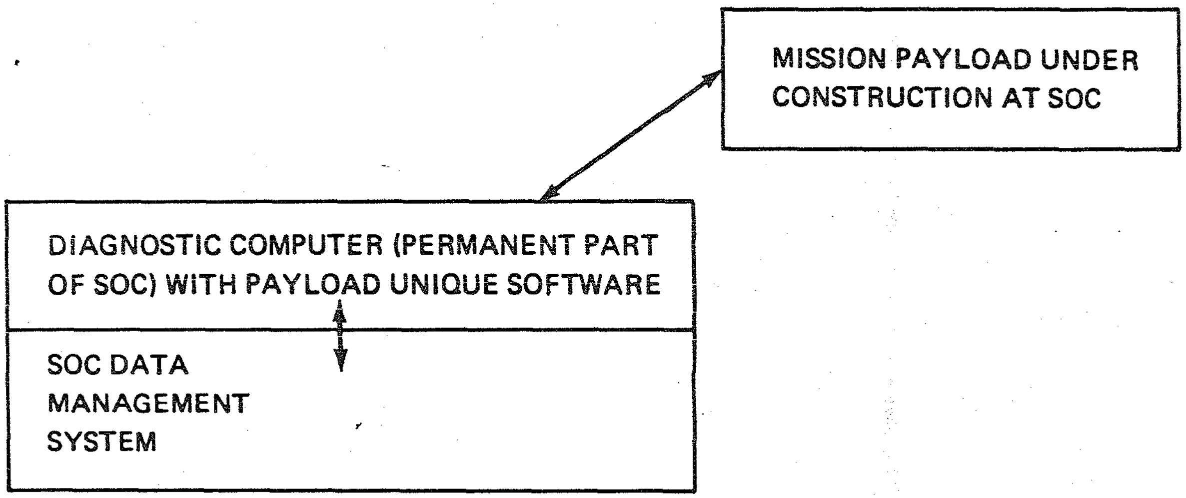


Figure 14-4. SOC Standard Test Computer

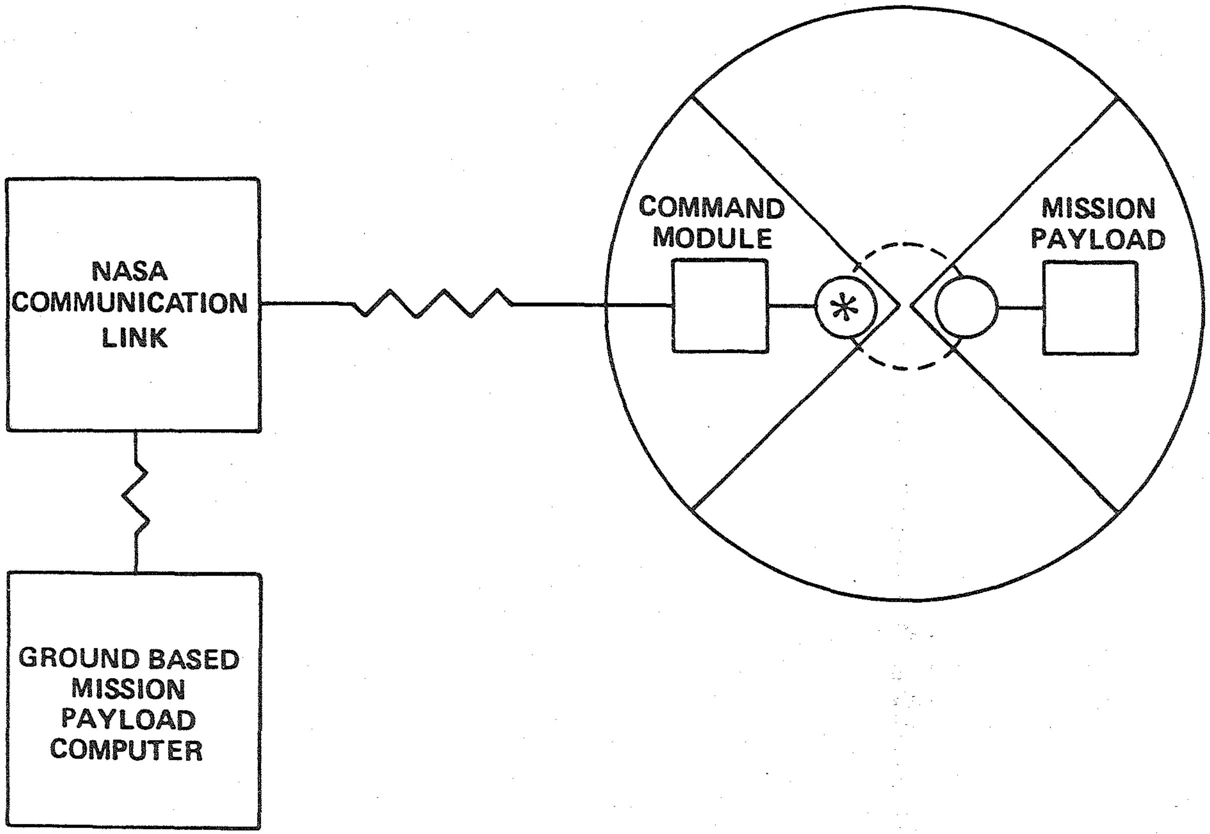


Figure 14-5. Use of Ground-Based Software

The design of a permanent test computer in the SOC would be an approach to eliminate the necessity for lifting the test computer to the SOC for each different payload.

The disadvantages are that the designer of the payload would be constrained in that the test software he/she wishes to write would have to execute in this computer which was made for a general purpose test system and not for his/her unique application. Furthermore, the general test system would have to be designed at a time when the mission payloads it is designed to test are still undefined. It would probably be overspecified for every task it was ever used for and be missing critical capabilities for the more stringent tasks. The general purpose would also have to be lifted into space but it would only have to be done once.

The third approach utilizes a ground based computer system to provide the testing with telemetry of data from the mission payload through the SOC to the ground with test instructions going the other way. This approach minimizes the amount of test hardware that must be flown while leaving the choice of test computer to the payload designer. Also the system designers (who probably are not able to go into space themselves) can be intimately involved in the checkout of the hardware. The main disadvantage is that the data bandwidth between the test computer and the mission payload is limited to the SOC/ground telemetry bandwidth less the bandwidth required for SOC critical functions. Further, the delays introduced by multiple computer communications as well as by the actual transmission of data may preclude real-time testing of the payload.

The actual testing of mission payloads should probably require a combination of the three general approaches such as shown in Figure 14-6. A telemetry link to the ground would permit the payload designers themselves to check out the construction as well as minimize the amount of hardware flown. Since most of the checkout and system checkout can be done in a static mode, the delay and bandwidth limitations need not pose an insurmountable problem during construction. The final testing may, however, require tester response times less than that permitted through a space to ground link. Therefore, a certain amount of general purpose test equipment including interfaces and mission payloads and a small test

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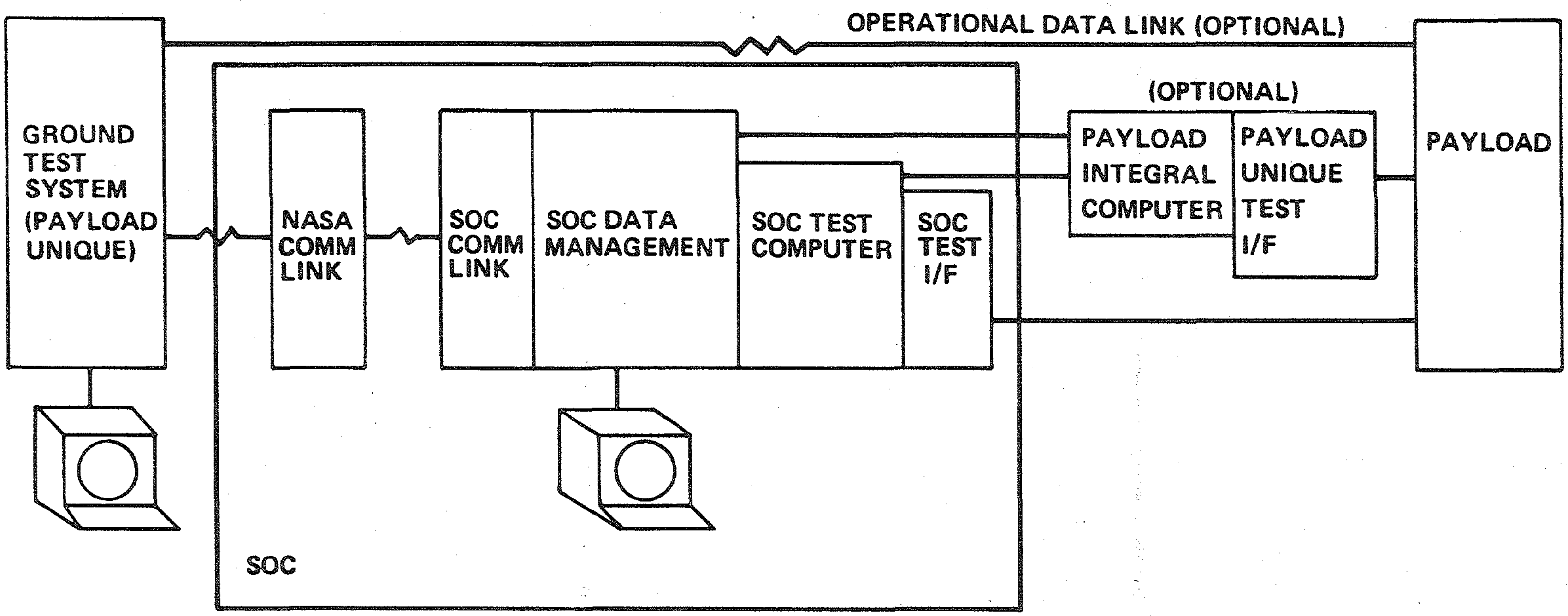

Figure 14-6. SOC Mission Payload Testing

computer should be made a part of the construction module. While this test approach might handle all of the test requirements of small construction projects and of the requirements of large construction projects, provision will have to be made for some designers to launch special test interfaces which may include computers and their software to orbit for the checkout of their payloads. Planning for a ground link should be made to minimize the need for special test equipment rather than to try to eliminate it.

The planning for use of a ground-based computer to aid in payload checkout will require planning in the design of the SOC for a link (with a well defined interface) from the ground-based communications system to the payload designers' computer. This interface will require not only a hardware specification but also a message format for data that the payload designer may wish to have delivered to his unique test equipment on the SOC. The interface must also specify messages to be displayed to NASA ground personnel, the command module on the SOC, as well as in the construction module in the SOC.

15.0 PROPULSION AND PROPELLANTS ANALYSIS

15.0 PROPULSION AND PROPELLANTS ANALYSIS

15.1 INTRODUCTION

The Space Operations Center requires a propulsion system to maintain, and occasionally to change, its orbital altitude. The selection of orbital altitude and the resulting propellant resupply requirements are discussed in section 8.0 of this report.

Propellant quantity and resupply requirements are set by drag considerations. Thrust levels are set by control authority and controlled deorbit considerations.

15.2 THRUST LEVEL SELECTION

The thrust level is established by the control authority criteria listed in Figure 15-1. These criteria are based on emergency conditions. In order to establish a worst case torque, it was assumed that the SOC was on the verge of reentry under worst atmosphere density conditions with one solar array drive out (maximum asymmetry of the configuration) and that the propulsion system is near the end of its blowdown. This is a highly unlikely situation. Even so, the thrust requirement is adequately met by the use of 30-lb hydrazine thrusters. 30-lb thrusters are available as developed hardware.

Also indicated in the figure is the condition under which propellant consumption is dominated by drag makeup rather than control authority. The thruster locations established by the JSC reference design are adequate for all normal operational conditions. The drag symmetry will place the center of pressure within the envelope noted and the booms are long enough to maintain the center of gravity of the system also within the envelope.

It is necessary to fly the vehicle in a principal axis mode when the shuttle is attached or other gravitational asymmetries exist, to avoid very frequent propulsion operations to maintain altitude control.

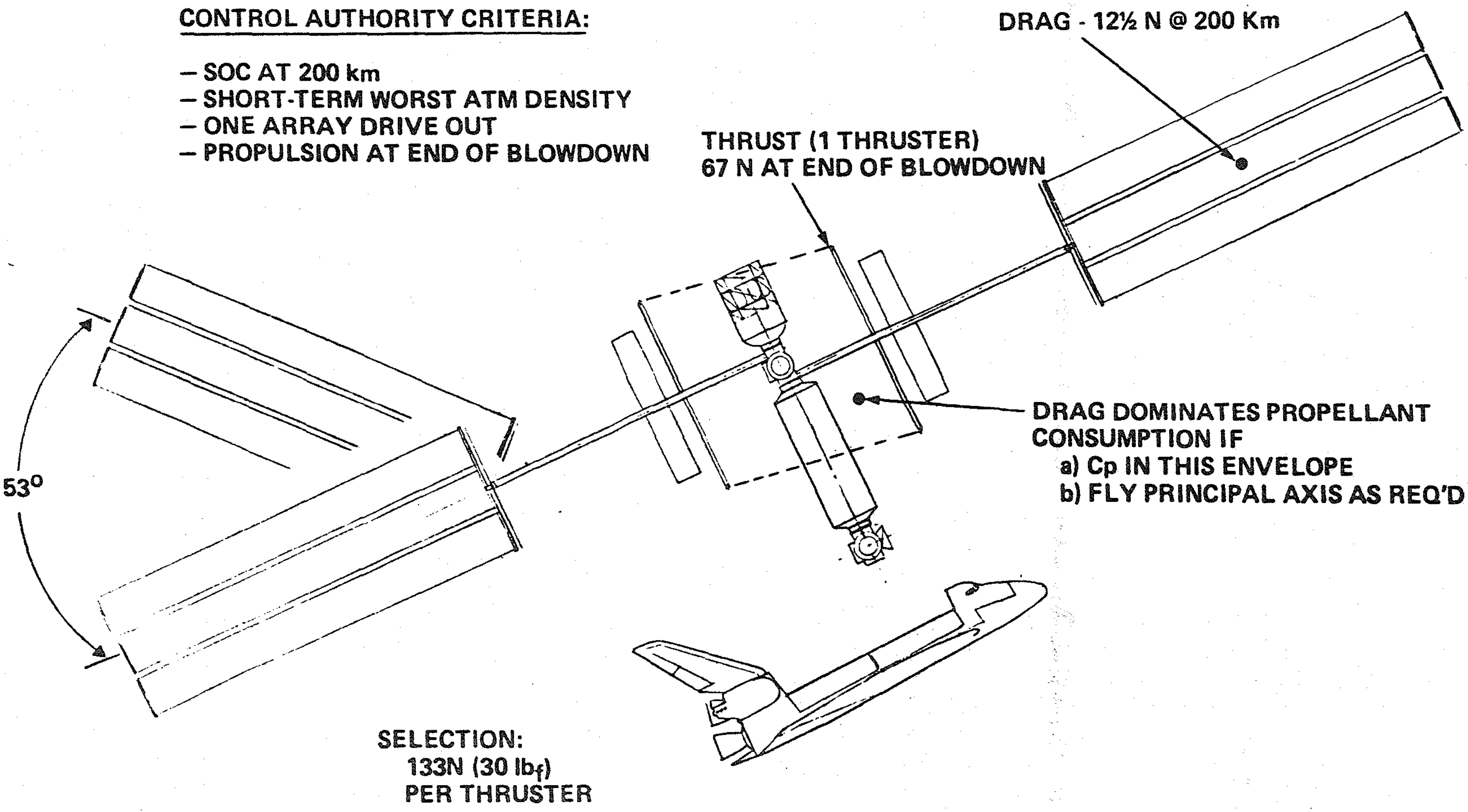

Figure 15-1. SOC Thrust Level

The present SOC requirements include a requirement to accomplish a controlled deorbit from 200 km altitude as a measure of last resort in order to avoid uncontrolled deorbit. The delta v required is 42 m/sec, to be delivered in 20 minutes or less. The required acceleration is 0.035 m/sec^2, leading to a thrust level of about 4400 newtons for the SOC reference configuration. Allowing for cg offsets and redundancy, this requirement indicates a need for 2-1100N thrusters at each thruster location (1100N = 250 lb). These thrusters must be forward-firing and would be used only for controlled deorbit.

15.3 PROPELLANT SELECTION

Several options were considered for the Space Operations Center propulsion system. There is considerable temptation to select a high specific impulse system to minimize propellant consumption. However, when operational practicality is considered, the case for monopropellant hydrazine is very strong, as summarized in Table 15-1.

15.4 SOC PROPULSION SYSTEM

The propulsion system is designed as a resuppliable and maintainable hydrazine blowdown system. A schematic is shown in Figure 15-2. The service module system is provided with paired, normally opened and normally closed squib valves at all terminals in order to meet Shuttle launch safety requirements. In addition, solenoid valves are provided at the service-module-to-service-module interface and at the service-module-to-logistics-module interface to allow control of propellant transfer and propellant sharing. Redundant manifold lines allow leaks to be isolated and one manifold line to be replaced while the system is still operable. The principal normal thrusting requirement is aft firing. Aft firing thrusters are redundant whereas the others are not. Thruster modules are replaceable by an EVA astronaut. The tank pressures are selected such that the service module system can be resupplied by the logistics module in a blowdown mode.

Table 15-1. Propellant Selection

	HYDRAZINE	BI-PROP	LO_2-LH_2	ELECTRIC RESIST.	ELECTRIC ION
HARDWARE AVAILABILITY	✓	✓			
SIMPLICITY	✓				WORST
RELIABILITY	✓				
MAINTAINABILITY	✓				
THRUSTER LIFE	✓	PROBLEM	?	?	PROBLEM
BLADDER LIFE	✓		N/A	✓	✓
POSITIVE EXPULSION	✓	✓		✓	✓
USED BY ECLS	✓			?	
POWER REQ'T	HEATERS REQ'D	HEATERS REQ'D		HIGH	PROHIBITIVE IF CONTROL AUTHORITY CRITERIA APPLIED
CONTAMINATION	✓	PROBLEM	✓	?	WORST (IF MERCURY) BEST (IF ARGON)
SAFETY	✓				WORST
RESUPPLY PER 90 DAYS (TONNES)	~2	~1	~1	~1	NIL

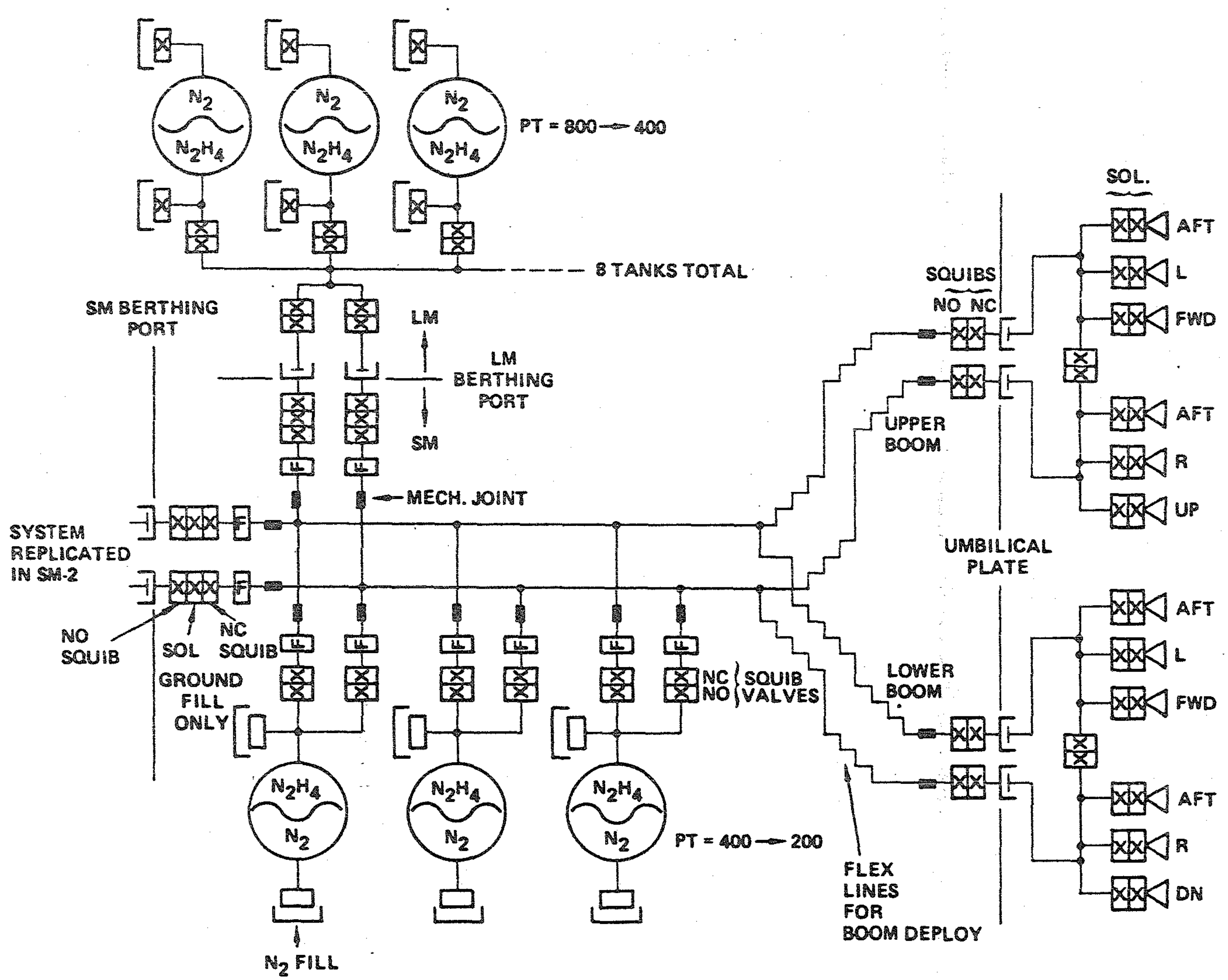

Figure 15-2. SOC Propulsion System Schematic

(This schematic does not include the deorbit thrusters; they are to be added. Mass and cost allowances for deorbit thrusters are included in the SOC service module mass and cost statements in the system description.)

The module shown in figure 15-3 is based on the Hamilton Standard REA 22-16 IUS hydrazine thruster. The thruster is designed for 30 lbs maximum thrust and can operate in a blowdown mode. The module is essentially identical to one selected for the space-based OTV by the Future OTV study.

15.5 PROPULSION OPERABILITY AND MAINTAINABILITY

The main propulsion features for operability and maintainability are as follows:

- Thrusters are redundant.
- Tanks and lines are redundant.
- Leaks can be isolated by squib valves.
- Engine modules are quickly replaceable by an EVA crew member.
- Tanks and lines are also replaceable (more difficult).
- The system is operable while being maintained, except that thruster firings must be inhibited if an EVA crew member is near thruster installation.
- The service modules and logistics modules are sized to accommodate worst-case drag requirements for 90 days.
- The normal operation is to replenish SMs from LM. LM load equals SM available capacity plus the least expected 90-day consumption.

15.6 ISP ENHANCEMENT POTENTIAL

The specific impulse of the SOC hydrazine propulsion system could be enhanced by electrical resistance heating of the decomposition products emitted by the thruster catalyst bed. ISPs exceeding 300 seconds are potentially achievable.

The power requirements for this could be drawn from the SOC solar array on the sunlit side of the orbit. The electrical power needed is quite high as shown in

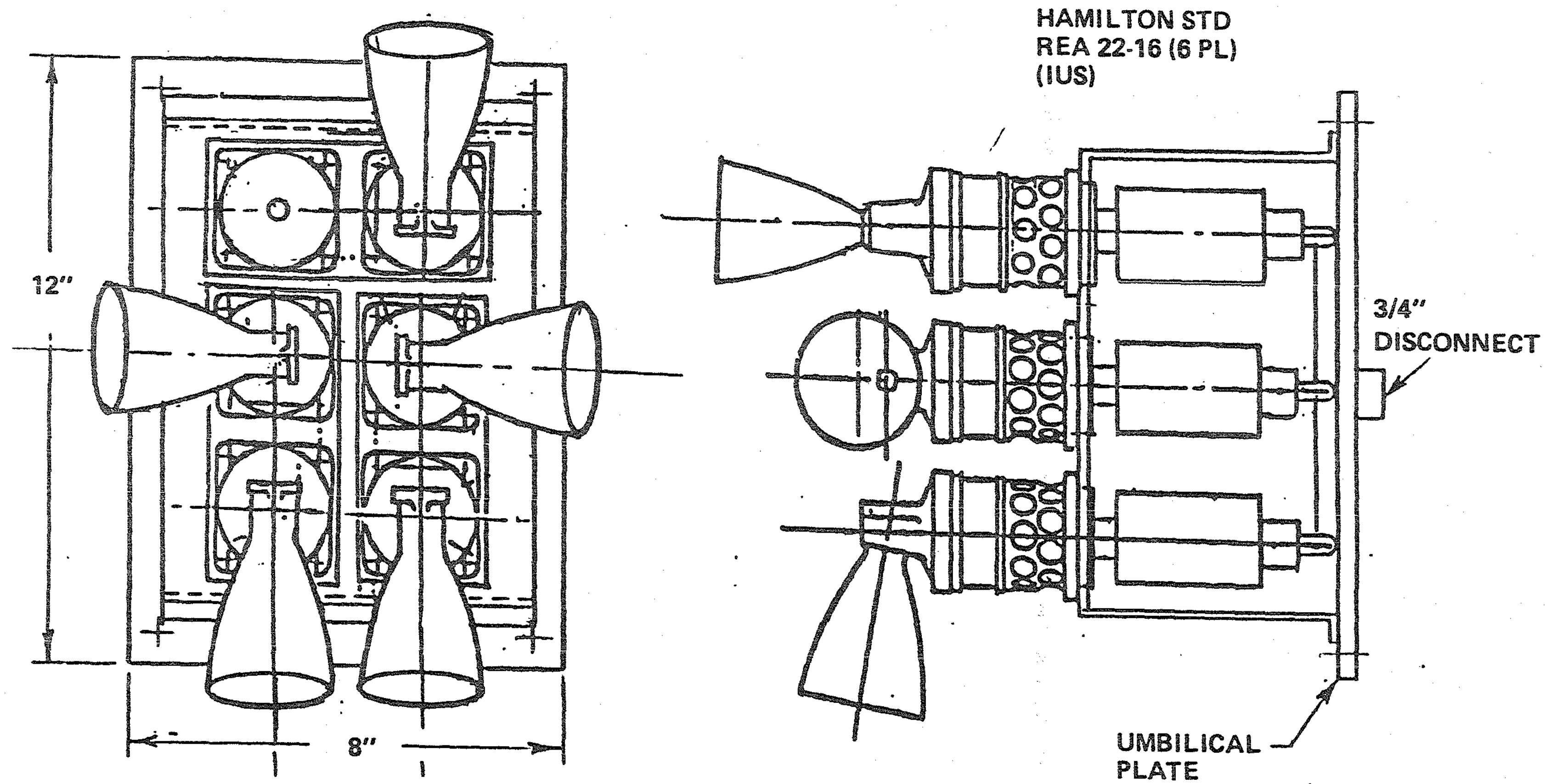

Figure 15-3. SOC Propulsion—Removable REM Module

Figure 15-4. A set of four 1-lb thrusters delivering 325 seconds ISP would draw nearly 10 kW.

Since the maximum drag to which the SOC will normally be subjected is less than one newton (1/4 lb), 1-lb thrusters would provide plenty of margin over drag forces. The 30 lb thrusters described above would be retained to provide control authority in off-normal situations.

An estimate of heated gas temperatures versus ISP is presented in Figure 15-5.

The small orbit makeup thrusters, if used, would accumulate many hundreds of hours' firing time per year. Lifetime of their catalyst beds is an issue needing further evaluation.

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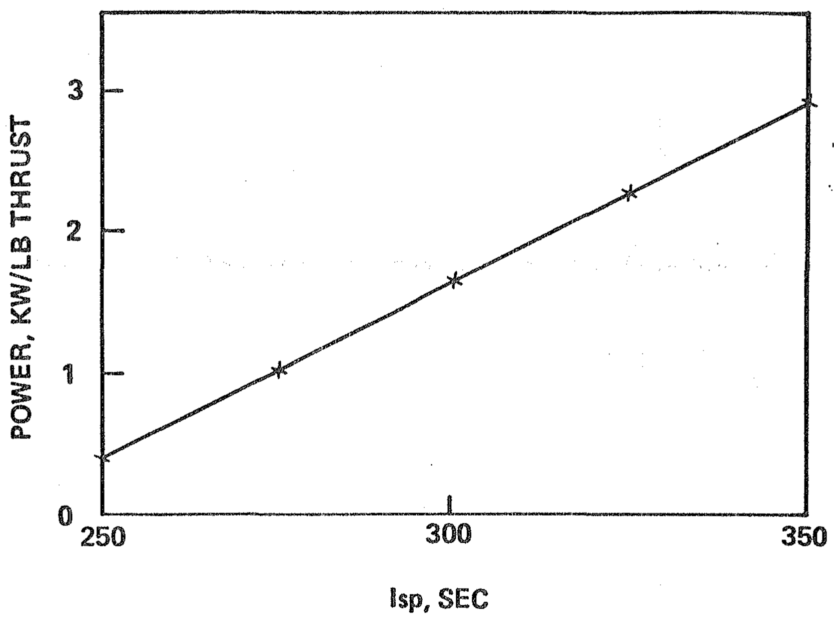

Figure 15-4. Power Requirements for Isp Enhancement

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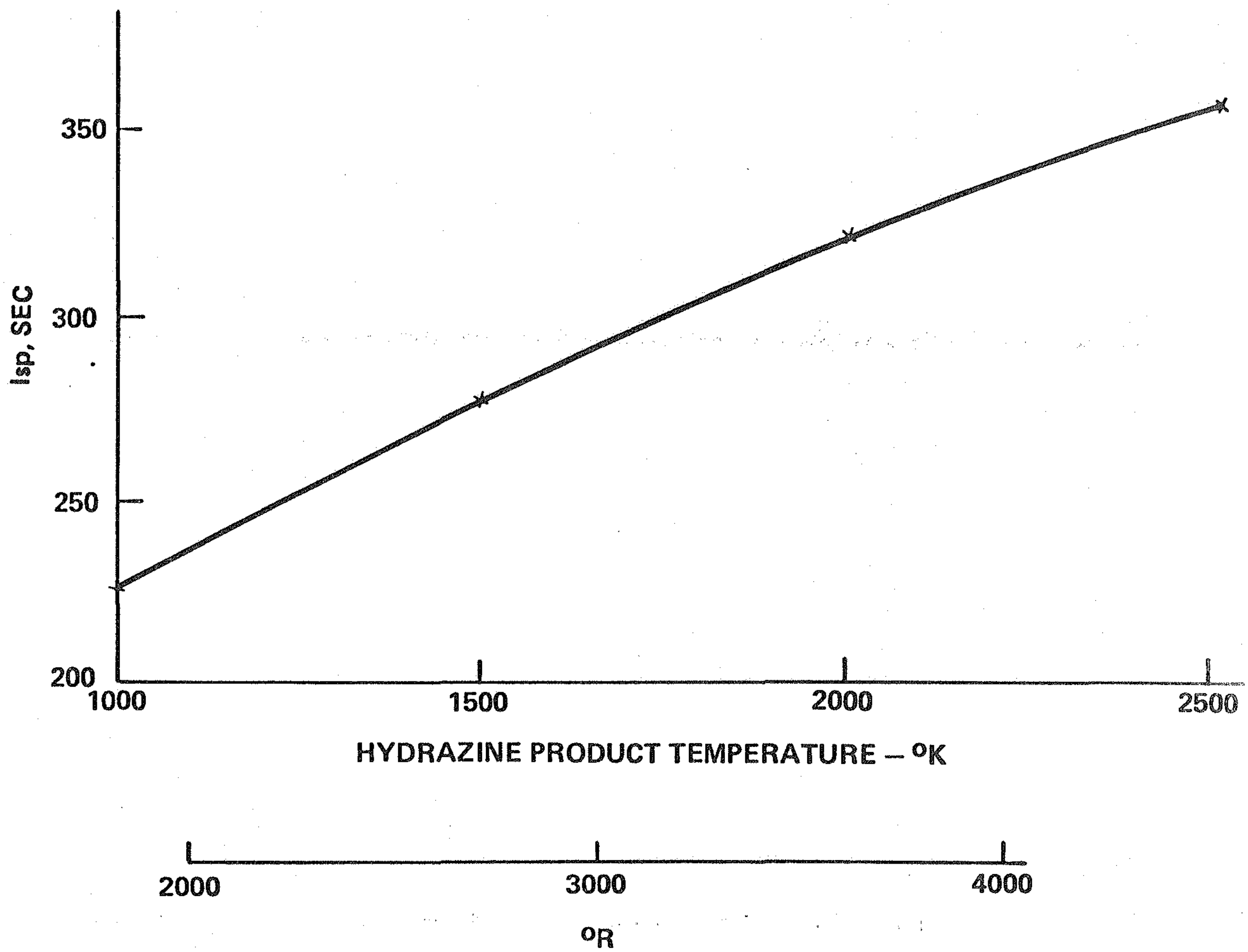

Figure 15-5. Hydrazine Isp vs. Temperature

16.0 SUBSYSTEM INTERRELATIONSHIPS ANALYSIS

16.0 SUBSYSTEMS INTERRELATIONSHIPS ANALYSIS

16.1 INTRODUCTION

A subsystems interrelationships analysis was conducted to characterize the technical and operational interfaces among the various subsystems and to define the dependency SOC resupply requirements on operational factors. Both quantitative and qualitative models were constructed.

The quantitative model was concerned mainly with the interrelationships between the environmental control and life support system, crew operations, electrical power, and resupply requirements. The qualitative model considered all subsystem as well as operational factors.

16.2 INTERRELATIONSHIPS MODEL

The quantitative model is based on a model of the control and life support systems that consist of a set of independent variables and a set of dependent variables defined as functions of other variables (dependent or independent). Output were derived as a function of crew size and metabolic rate. It is also possible to see how the output varies when different choices are made for the interrelationships among the variables. The model is constantly being refined and updated as new information becomes available.

The model is a set of simultaneous equations which is solved by the ISAIAH software. ISAIAH can handle up to 50 independent variables and 135 dependent variables, and automatically varies the crew size and metabolic rate. Output is obtained as both a regular computer printout and as plot files which are made into graphs by the VAXPLOT facility.

The model shows the variations induced by off-optimal conditions. This is especially important as the systems must be designed to handle the maximum predicted loads not just the optimal loads.

The other major contribution is that the model shows the relative sensitivities of the variables to the parameters. Among other things, it showed that the ECLS hydrazine (used to replenish nitrogen) resupply requirements depend on crew size only to the extent that crew size affects the number of airlock cycles per day, and depends not at all on metabolic rate. The water resupply requirements were shown to depend almost exclusively on the amount of EVA time, since the only major loss is from the EVA suit cooling system.

Figures 16-1 through 16-3 show the principal interrelationships. Table 16-1 presents model details and Table 16-2 shows nominal outputs.

The qualitative model employed a high-level qualitative matrix. The matrix, and the main results, are shown in Table 16-3.

The main conclusions of the analysis were:

- o SOC subsystems will be capable of autonomous or manually-controlled operation as backup (not dependent on data bus).
- o The normal mode will be automatic.
- o The critical subsystems are power and EC/LSS—all subsystems require power and thermal control.
- o The interrelationships are simple—subsystems can be developed independently of one another with interface simulators.

16.3 RESUPPLY LOGISTICS

A 90 day resupply cycle has been baselined for the SOC operations. Resupply will be accomplished by using a dedicated Logistics Module (LM) delivered in the payload bay of a Shuttle Orbiter. Two modules will be used in the resupply cycle, one berthed at the SOC and one on the ground being prepared for flight. When a freshly loaded LM arrives at the SOC it is berthed to a Service Module and remains there for 90 days. The depleted LM, now containing spent LiOH canisters, compacted waste products, replaced parts, etc., is placed in the Shuttle bay and returned to Earth. A more complete description of the LM may be found in the SOC System Definition Final Report, Reference Boeing 19.

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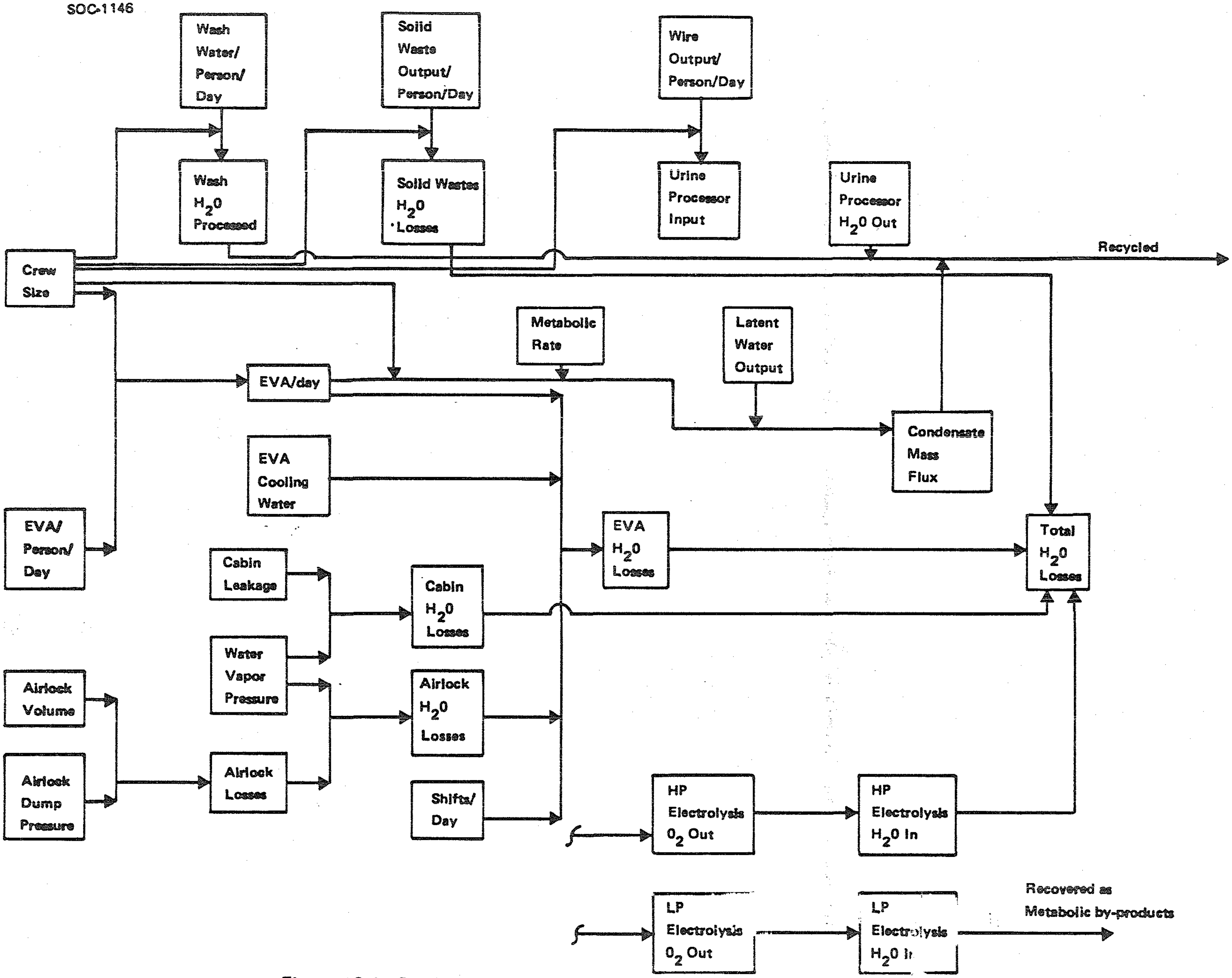

Figure 16-1. Partial Water Processing Subsystem Interrelationships Model

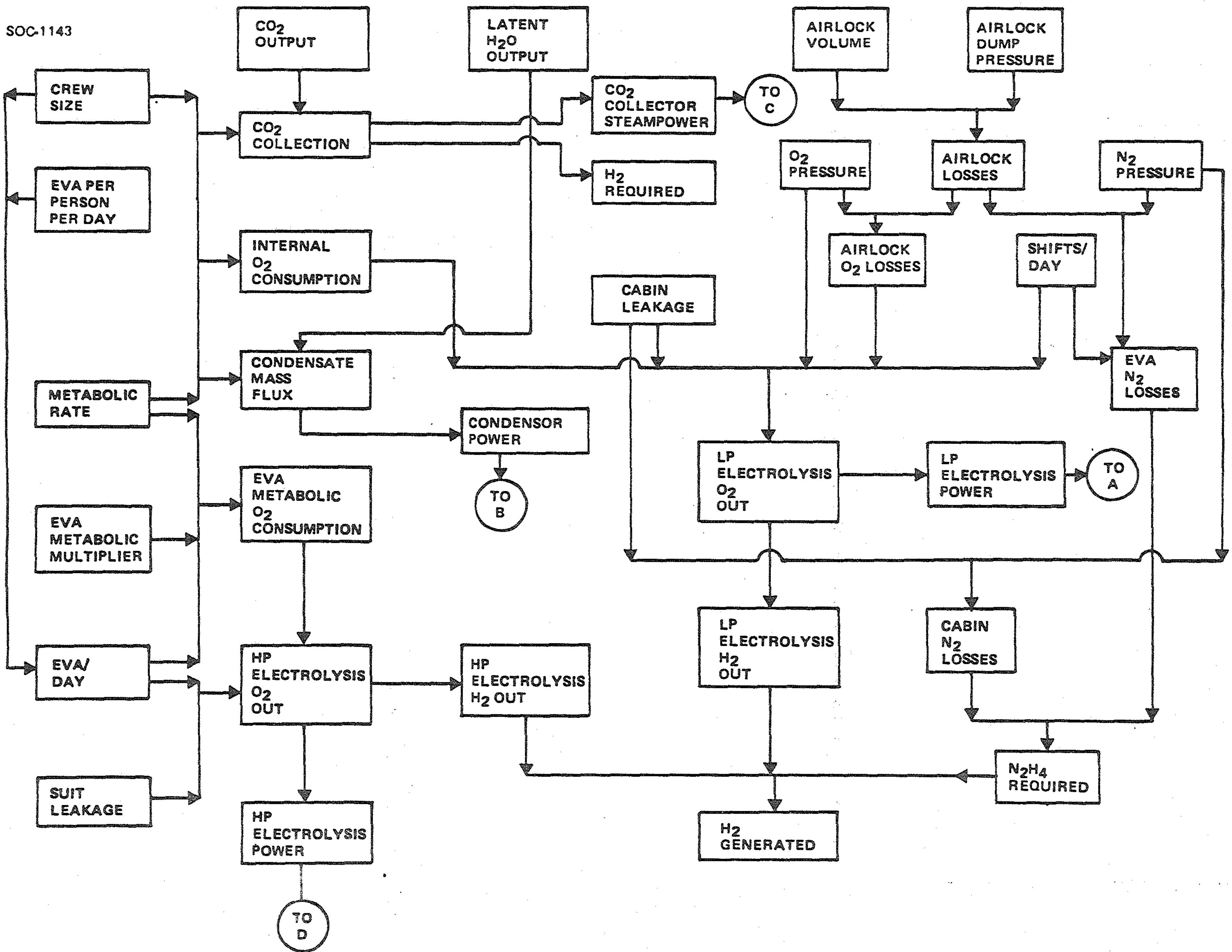

Figure 16-2. Partial ECLS Atmosphere Supply Subsystem Interrelationships

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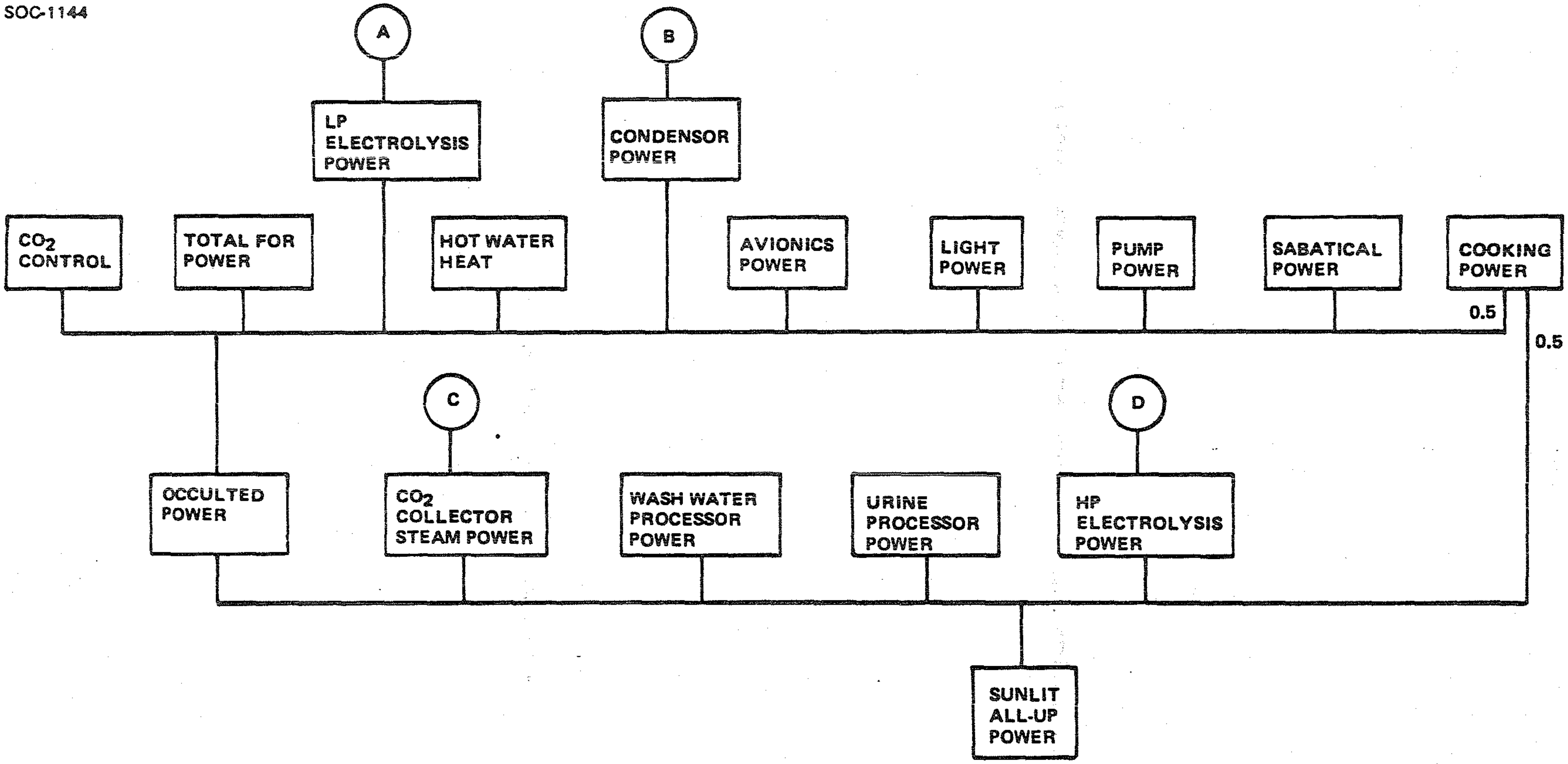

Figure 16-3. Total Power Summation Subsystem Interrelationship Model

TABLE 16-1

INTERRELATIONSHIPS MODEL DETAILS

Var.#	Name	Value	Comments
I1	Crew Size	8	Software varies value from two to eight.
I2	EVA/person/day	.5	Allows the number of EVAs to vary according to crew size.
I3	Metabolic Rate	133 w	From Hamilton Standard data. Software varies value from 66.5 w to 266 w.
I4	Food Consumption	1.63 kg/person/day	Dry weight of shelf-stable food plus total weight of frozen food. From Hamilton Standard data.
I5	Potable Water	2.49 kg/person/day	Drinking water plus rehydration water and water content of shelf-stable food. From Hamilton Standard data.
I6	Wash Water	18.94 kg/person/day	From Hamilton Standard data.
I7	Hot Water	5 kg/person/day	From Hamilton Standard data.
I8	Wash Evaporation Fraction	.05	
I9	Flush Factor	.3	
I10	Urine Output	1.5 kg/person/day	From Hamilton Standard data.
I11	Latent Water Output	1.32 kg/person/day	
I12	Solid Waste Output	.15 kg/person/day	From Hamilton Standard data.
I13	Laundry Load	10 kg	
I14	CO_2 Output	1 kg/person/day	From Hamilton Standard data.
I15	Cabin Wall Thickness	.01 m	
I16	O_2 Pressure	.22 atm	
I17	N_2 Pressure	.765 atm	
I18	CO_2 Pressure	.005 atm	
I19	Water Vapor Pressure	.01 atm	
I20	Heat Rejection Temperature	$280^{o}K$	
I21	Maximum Temperature	$300^{o}K$	
I22	Real Dummy	1	
I23	Cabin Leakage	2.2 kg/day	From Hamilton Standard data.
I24	PPU Efficiency	.8	
I25	Avionics Power	15,720 w	Includes communications and tracking, control functions, caution and warning, entertainment, etc.
I26	Shifts/Day	2	
I27	Light Power	5,000 w	

TABLE 16-1 (Continued)

Var.#	Name	Value	Comments
I28	Pump Power	1,200 w	
I29	CO_2 Control	1,080 w	
I30	Cooking Power	10,000 w	
I31	Sabatier Power	10 w	
I32	EVA H_2O	5.03 kg/EVA	Cooling water for EVA backpacks. Hamilton Standard figure.
I33	Airlock Volume	8 m^3	
I34	Airlock Dump Pressure	.136 atm	
I35	EVA Matabolic Multiplier	1.5	To account for greater physical exertion by EVA crewpersons.
I36	Airlock Scavenge Time	.01 day	
I37	Suit Leakage	1 kg/EVA	Hamilton Standard figure.
I38	Resupply Interval	90 days	
I39	Emergency Staytime	90 days	
I40	Emergency Crew Size	8 persons	
I41	Number of Airlocks	2	One airlock per habitat module.
I42	Habitat Decks	4	
I43	Habitat Window Area	1 m^2	
I44	Number of OTV Docks	1	
I45	Number of Habitats	2	
I46	Number of Service Modules	2	
I47	Number of Tunnels	1	
I48	Number of Logistics Modules	1	
I49	Number of Habitat Pressure Hatches	3	

TABLE 16-1 (Continued)

Var.#	Name	Input Estimate	Interrelationship Description	Comments
D1	Sunlit All-up Power	30,000 w	D2+D13+D21+D26+D36 +.5 (I30)	Includes everything under occulted power, plus those power draws which can safely and conveniently be shut down during the occulted part of the orbit. The airlock pump power is not included in either the sunlit or occulted summation. It is planned to handle this draw by power switching.
D2	Occulted Power	20,000 w	D12+D31+D48+D54+ I25+I27+I28+I29+ 0.5 (I30)+I31	Includes power to vital internal ECLSS equipment and, under avionics, items such as communications and entertainment. Cooking power includes refrigerator and freezer power, so half of it is included under occulted power.
D3	Peak Power	30,000 w	1.5 (D58)	
D4	Average Supplied Power	25,000 w	$\frac{32(D2)+60(D1)}{92}$ I24	The average of the sunlit all-up power and the occulted power over the orbit period times the PPU efficiency.
D5	Maximum Heat Transfer	200 w/	$\frac{D3}{I21-I20}$	by definition, $K = \frac{Q}{T}$
D6	Metabolic Power	800 w	(I1)(I3)	Total heat load due to crew metabolisms.

TABLE 16-1 (Continued)

Var.#	Name	Input Estimate	Interrelationship Description	Comments
D7	Internal O_2 Consumption	4 kg/day	(I1-0.4(D74))(I3)(0.00623)	EVA crewpersons have to be subtracted from the crew size since their oxygen comes from a different source. They are supposed to be outside for one-third of a day, but the factor of 0.4 is used since the EVA time is assumed to be the period of heaviest activity. The factor 0.00623 is an empirical conversion factor to get the crew size time metabolic rate down to Hamilton Standard's figure.
D8	H_2 Generated	2 kg/day	0.125(D19)+D30+D35	The factor of .125 is the percentage weight of H_2 produced from the hydrazine (N_2H_4) dissociated.
D9	H_2 Requirement	1 kg/day	0.181818(D11)	Two unit weights of H_2 are needed to reduce eleven unit weights of CO_2, hence the factor of .181818.
D10	Cabin N_2 Losses	2 kg/day	(I23)(I17)	
D11	CO_2 Collection	10 kg/day	(I1-0.4(D74))(I3)(I14)(0.00752)	The crew size reasoning is as in the internal O_2 consumption calculation (D7). The factor of 0.00752 is a conversion factor to get the output down to Hamilton Standard's figure.
D12	Total Fan Power	1800 w	D50+D52	
D13	CO_2 Collector Steam Power	30 w	4(D11)	The factor of four is to get the output near the estimated power requirement.

TABLE 16-1 (Continued)

Var.#	Name	Input Estimate	Interrelationship Description	Comments
D14	CO_2 Collector Mass	100 kg	10(D11)	
D15	Sabatier CH_4 Out	4 kg/day	.363636(D11)	Four unit weights of CH_4 are produced for every eleven unit weights of CO_2 collected, hence the factor of .363636.
D16	Sabatier H_2O Out	5 kg/day	.8181818(D11)	Nine unit weights of H_2O are produced for every eleven unit weights of CO_2 collected, hence the factor of .8181818.
D17	EVA H_2O Losses	13 kg/day	(I32)(D74)+(D45)(I26)	Assumes that the airlock will be cycled just once per shift.
D18	Solid Wastes H_2O Losses	.96 kg/day	.8(I1)(I12)	Fecal water is lost to the system because feces will not be reclaimed. The assumptions are fecal water content of 80% and insignificant variation with metabolic rate.
D19	N_2H_4 (Hydrazine) Required	3 kg/day	1.143(D10+D69)	The factor of 1.143 is the ratio of the unit weights of hydrazine and nitrogen.
D20	Wash Water Processed	150 kg/day	(I1)(I6)	Includes shower, handwash, and laundry.

TABLE 16-1 (Continued)

Var.#	Name	Input Estimate	Interrelationship Description	Comments
D21	Wash Water Processor Power	760 w	5(D20)	The factor of five is to get the output close to the estimated power requirement.
D22	Wash Water Processor Mass	20 kg	10(D20)	
D23	Urine Processor Input	12 kg/day	(I1)(I10)	Assumes insignificant variation due to metabolic rate.
D24	Urine Processor Solids Out	1 kg/day	0.05(I1)(I10)	
D25	Urine Processor Water Out	11 kg/day	0.95(I1)(I10)	Assumes urine composition of 95% water and 5% solids. The factor of 20 is to get the output near the expected power requirement.
D27	Urine Processor Mass	20 kg	10(I1)(I10)	
D28	LP ELectrolysis H_2 Out	7 kg/day	1.125(D29)	Nine unit weights of H_2O are needed to produce eight unit weights of O_2, hence the factor of 1.125.
D29	LP Electrolysis O_2 Out	6.3 kg/day	(I26)(D42)+(I23)(I16)+D7	Summation of all O_2 usages/losses inside the SOC.

TABLE 16-1 (Continued)

Var.#	Name	Input Estimate	Interrelationship Description	Comments
D30	LP Electrolysis H_2 Out	3 kg/day	.125(D29)	One unit weight of H_2 is produced for every eight unit weights of O_2 produced, hence the factor of .125.
D31	LP Electrolysis Power	1800 w	215(D29)	The factor of 215 is to get the output near the predicted power requirement.
D32	LP Electrolysis Mass	30 kg	2(D29)	
D33	HP Electrolysis H_2O In	4 kg/day	1.125(D34)	The reason for the factor of 1.125 is the same as in the LP Electrolysis H_2O In calculation (D25).
D34	HP Electrolysis O_2 Out	3.9 kg/day	D46+(I37)(D74)	Total amount of oxygen used or lost from EVA backpacks and suits.
D35	HP Electrolysis H_2 Out	.5 kg/day	.125(D34)	One unit weight of H_2 is produced for every eight unit weights of O_2 produced, hence the factor of .125.
D36	HP Electrolysis Power	950 w	310(D34)	The factor of 310 is to get the output near the predicted power requirement.
D37	HP Electrolysis Mass	30 kg	2(D34)	
D38	CO_2 Collector Airflow Volume	50,000 m^3/day	(D11/I18)(63.6)	
D39	Humidity Controller Airflow Volume	15,000 m^3/day	(D53/I19)(26.1)	

TABLE 16-1 (Continued)

Var.#	Name	Input Estimate	Interrelationship Description	Comments
D40	Airlock Pump Power	500 w	$\frac{I33(1-I34}{1.16(I36)}$	
D41	Airlock Losses	1 kg/EVA	(I33)(I34)	Mass of air dumped each time the airlock is cycled.
D42	Airlock O_2 Losses	.2 kg/EVA	(I16)(D41)	Oxygen fraction of dumped air.
D43	Airlock N_2 Losses	.7 kg/EVA	(I17)(D41)	Nitrogen fraction of dumped air.
D44	Airlock CO_2 Losses	.05 kg/EVA	(I18)(D41)	Carbon dioxide fraction of dumped air.
D45	Airlock H_2O Losses	.1 kg/EVA	(I19)(D41)	Water vapor fraction of dumped air.
D46	EVA Metabolic O_2 Consumption	2 kg/day	.00363(D74)(I3)(I35)	The factor of 0.00363 serves the same purpose as it did in the calculation of D7, Internal O_2 Consumption. The EVA metabolic multiplier is based on the assumption that the EVA crew will have higher metabolic rates due to the extra stresses of working in a weightless environment, in addition to the work they will be doing.
D47	HP/LP H_2O-O_2-H_2 Ratio	1	D33/D28	HP/LP ratios are the same for H_2O, O_2, and H_2.
D48	Hot Water Heat	50 w	2(I1)(I7)	The factor of 2 is to get the output near the predicted power requirement.

TABLE 16-1 (Continued)

Var.#	Name	Input Estimate	Interrelationship Description	Comments
D49	Number of Vent Packs	10	4(I45)+2(I46)	There are four vent packs in each habitat module and two in each service module.
D50	Vent Fan Power	1000 w	100(D49)	The factor of 100 is to get the output near the predicted power requirement.
D51	Number of Revitalized Packs	4	2(I45)	There are two revit. packs in each habitat module.
D52	Revit. Fan Power	300 w	100(D51)	The factor of 100 is to get the output near the predicted power requirement.
D53	Condensate Mass FLux	5 kg/day	(I1-0.4(D74)(I3) (I11)(.00752)	The factor of .0752 is the same factor as appears in D11. The reasoning on "equivalent crew size" is the same as for D7 and D11.
D54	Condensor Power	400 w	54(D53)	The factor of 54 is to get the output near the predicted power requirement.
D55	Gross Volume	600 m^3	201(I45)+57(I46) +36(I47)+64(I48)	The factors are the volumes of each element. They are multiplied by the number of each type of module in the SOC.
D56	Atmospheric Volume	500 m^3	.95(D55)	The factor of .95 is from the assumption that 5% of the gross volume will be occupied by solids.

TABLE 16-1 (Continued)

Var.#	Name	Input Estimate	Interrelationship Description	Comments
D57	Habitable Volume	300 m^3	.5(D55)	The assumption is that 50% of the gross volume will be in the ceilings and floors, occupied by storage lockers, or otherwise used.
D58	Sunlit Internal Heat Load	25,000 w	D1-.75(I25)+D6	75% of the avionics power is subtracted from the internal heat load as a lot of this power is already being radiated away from the station in the form of radio waves, etc.
D59	Occulted Interior Heat Load	20,000 w	D2-.75(I25)+D6	Same reasoning as for D58, Sunlit Interior Heat Load.
D60	Habitat Dry Mass	100,000 kg	100,000(I45)	
D61	Service Module Dry Mass	10,000 kg	10,000(I46)	
D62	Tunnel Mass	1,000 kg	1,000(I47)	
D63	Logistic Module Dry Mass	2,000 kg	2,000(I48)	
D64	OTV Dock Dry Mass	3,000 kg	1,000(I44)	
D65	Construction Facility Mass	6,000 kg	3,000(I22)	
D66	OTV Dry Mass	3,000 kg	3,000(I22)	

TABLE 16-1 (Continued)

Var.#	Name	Input Estimate	Interrelationship Description	Comments
D67	Cabin H_2O Losses	.02 kg/day	(I19)(I23)	Water vapor fraction of total losses due to leakage.
D68	Total H_2O Losses	15 kg/day	D17+D18+D67+ 0.9(D33)	Summation of all water losses. The EVA O_2 is considered lost because CO_2 is not reclaimed.
D69	EVA N_2 Losses	1.5 kg/day	(I26)(D43)	Total daily nitrogen losses due to airlock dumping.
D70	Total Dry Mass	200,000 kg	D60+D61+D62+D63 +D64+D65+D66	
D71	Atmospheric Mass	500 kg	(D56)(I22)	Assumes that 1 m^3 of air at atmosphere pressure masses about 1 kg.
D72	H_2O Mass	2000 kg	1000(I22)	
D73	Organics Mass	1000 kg	1000(I22)	
D74	EVA/day	3	(I1)(I2)	Allows the number of EVAs to be dependent upon crew size.
D75	Crew Mass	500 kg	70(I1)	Assumes an average crew mass of 70 kg.
D76	SOC Hydrazine Mass	2,000 kg	2,000(I22)	
D77	OTV H_2 Mass	50,000 kg	50,000(I22)	
[illegible]	OTV O_2 Mas	50,000 kg	20,000(I22)	

TABLE 16-1 (Continued)

Var.#	Name	Input Estimate	Interrelationship Description	Comments
D79	Excess H_2	2 kg/day	D8-D9	Difference of the amount of H_2 produced and the amount required.
D80	OTV Depot H_2 Mass	50,000 kg	50,000(I22)	
D81	OTV Depot O_2 Mass	50,000 kg	20,000(I22)	
D82	OTV Hydrazine Mass	1,000 kg	2,000(I22)	
D83	Food Consumption	13 kg/day	((I1-0.4(D74))+ (0.4)(D74)(I35) (1.25(I3)-33.25) x(I4)(.00752)	The first, long factor is the "equivalent crew size." The second factor allows small variations in food consumption due to variations in metabolic rate. The factor of .00752 is to get the output back down near the estimated consumption.
D84	Dummy	1	(I22)	
D85	Total SOC Expendables Mass	5,000 kg	D71+D72+D73+D76	
D86	Total SOC Only Mass	200,000 kg	D70-D66+D75+D85	
D87	Total Non-SOC Expendables Mass	100,000 kg	D77+D78+D80+D81 +D82	
D88	Total System Mass	300,000 kg	D86+D66+D87	

1	SUNLIT ALL UP POWER	=	3.965E+04 W		
2	OCCULTED POWER	=	3.149E+04 W		
3	PEAK POWER	=	4.339E+04 W		
4	AVG. SUPPLIED POWER	=	2.945E+04 W		
5	MAX HEAT XFR COEFF	=	2.170E+03 W/K		
6	METABOLIC POWER	=	1.064E+03 W		
7	INTERNAL 02 CONSUMPTION	=	5.303E+00 KG/DAY	(	1.169E+01 LBM/DAY)
8	H2 GENERATED	=	2.124E+00 KG/DAY	(	4.682E+00 LBM/DAY)
9	H2 REQUIREMENT	=	1.164E+00 KG/DAY	(	2.566E+00 LBM/DAY)
10	CABIN N2 LOSSES	=	1.683E+00 KG/DAY	(	3.710E+00 LBM/DAY)
11	CO2 COLLECTION	=	6.401E+00 KG/DAY	(	1.411E+01 LBM/DAY)
12	TOTAL FAN POWER	=	1.600E+03 W		
13	CO2 COLLECTOR STEAM PUMP	=	2.560E+01 W		
14	CO2 COLLECTOR MASS	=	6.401E+01 KG	(	1.411E+02 LBM)
15	SABATIER CH4 OUT	=	2.328E+00 KG/DAY	(	5.132E+00 LBM/DAY)
16	SABATIER H20 OUT	=	5.237E+00 KG/DAY	(	1.155E+01 LBM/DAY)
17	EVA H20 LOSSES	=	2.014E+01 KG/DAY	(	4.440E+01 LBM/DAY)
18	SOLID WASTES H20 LOSSES	=	9.600E-01 KG/DAY	(	2.116E+00 LBM/DAY)
19	N2H4 REQUIRED	=	3.826E+00 KG/DAY	(	8.436E+00 LBM/DAY)
20	WASH H20 PROCESSED	=	1.515E+02 KG/DAY	(	3.340E+02 LBM/DAY)
21	WASH H20 PROC. POWER	=	7.576E+02 W		
22	WASH H20 PROC. MASS	=	1.515E+03 KG	(	3.340E+03 LBM)
23	URINE PROC. INPUT	=	1.200E+01 KG/DAY	(	2.646E+01 LBM/DAY)
24	URINE PROC. SOLIDS OUT	=	6.000E-01 KG/DAY	(	1.323E+00 LBM/DAY)
25	URINE PROC. H20 OUT	=	1.140E+01 KG/DAY	(	2.513E+01 LBM/DAY)
26	URINE PROC. POWER	=	2.400E+02 W		
27	URINE PROC. MASS	=	1.200E+02 KG	(	2.646E+02 LBM)
28	LP ELECTROLYSIS H20 IN	=	7.049E+00 KG/DAY	(	1.554E+01 LBM/DAY)
29	LP ELECTROLYSIS 02 OUT	=	6.266E+00 KG/DAY	(	1.381E+01 LBM/DAY)
30	LP ELECTROLYSIS H2 OUT	=	7.832E-01 KG/DAY	(	1.727E+00 LBM/DAY)
31	LP ELECTROLYSIS POWER	=	1.347E+03 W		
32	LP ELECTROLYSIS MASS	=	1.253E+01 KG	(	2.763E+01 LBM)
33	HP ELECTROLYSIS H20 IN	=	7.759E+00 KG/DAY	(	1.710E+01 LBM/DAY)
34	HP ELECTROLYSIS 02 OUT	=	6.897E+00 KG/DAY	(	1.520E+01 LBM/DAY)
35	HP ELECTROLYSIS H2 OUT	=	8.621E-01 KG/DAY	(	1.901E+00 LBM/DAY)
36	HP ELECTROLYSIS POWER	=	2.138E+03 W		
37	HP ELECTROLYSIS MASS	=	1.379E+01 KG	(	3.041E+01 LBM)
38	CO2 COLLECTOR AIRFLOW VO	=	8.142E+04 M3/DAY		
39	HUMIDITY CONTROLLER AIRF	=	2.205E+04 M3/DAY		
40	AIRLOCK PUMP POWER	=	8.016E+02 W		
41	AIRLOCK LOSSES	=	1.088E+00 KG/EVA	(	2.399E+00 LBM/EVA)
42	AIRLOCK 02 LOSSES	=	2.394E-01 KG/EVA	(	5.277E-01 LBM/EVA)
43	AIRLOCK N2 LOSSES	=	8.323E-01 KG/EVA	(	1.835E+00 LBM/EVA)
44	AIRLOCK CO2 LOSSES	=	5.440E-03 KG/EVA	(	1.199E-02 LBM/EVA)
45	AIRLOCK H20 LOSSES	=	1.088E-02 KG/EVA	(	2.399E-02 LBM/EVA)
46	EVA METABOLIC 02 CONSUMP	=	2.897E+00 KG/DAY	(	6.386E+00 LBM/DAY)
47	HP/LP H20-02-N2 RATIO	=	1.101E+00		
48	HOT WATER HEAT	=	6.000E+01 W		
49	NO. OF VENT PACKS	=	1.200E+01		
50	VENT FAN POWER	=	1.200E+03 W		
51	NO. OF REVIT PACKS	=	4.000E+00		
52	REVIT FAN POWER	=	4.000E+02 W		
53	CONDENSATE MASS FLUX	=	8.449E+00 KG/DAY	(	1.863E+01 LBM/DAY)
54	CONDENSOR POWER	=	4.563E+02 W		
55	GROSS VOLUME	=	6.160E+02 M3		
56	ATMOSPHERIC VOLUME	=	5.852E+02 M3		
57	HABITABLE VOLUME	=	3.080E+02 M3		
58	SUNLIT INT HEAT LOAD	=	2.893E+04 W		
59	OCCULTED INT HEAT LOAD	=	2.077E+04 W		
60	HABITAT DRY MASS	=	4.700E+04 KG	(	1.036E+05 LBM)
61	SERVICE MODULE DRY MASS	=	4.080E+04 KG	(	8.995E+04 LBM)
62	TUNNEL MASS	=	7.500E+03 KG	(	1.653E+04 LBM)
63	LOGISTIC MODULE DRY MASS	=	1.780E+04 KG	(	3.924E+04 LBM)
64	OTV DOCK DRY MASS	=	1.000E+03 KG	(	2.205E+03 LBM)
65	CONST FACILITY MASS	=	3.000E+03 KG	(	6.614E+03 LBM)
66	OTV DRY MASS	=	3.000E+03 KG	(	6.614E+03 LBM)
67	CABIN H20 LOSSES	=	2.200E-02 KG/DAY	(	4.850E-02 LBM/DAY)
68	TOTAL H20 LOSSES	=	2.112E+01 KG/DAY	(	4.657E+01 LBM/DAY)
69	EVA N2 LOSSES	=	1.665E+00 KG/DAY	(	3.670E+00 LBM/DAY)
70	TOTAL DRY MASS	=	1.201E+05 KG	(	2.648E+05 LBM)
71	ATM MASS	=	5.852E+02 KG	(	1.290E+03 LBM)
72	H20 MASS	=	1.000E+03 KG	(	2.205E+03 LBM)
73	ORGANICS MASS	=	1.000E+03 KG	(	2.205E+03 LBM)
74	EVA/DAY	=	4.000E+00		
75	CREW MASS	=	5.600E+02 KG	(	1.235E+03 LBM)
76	SOC HYDRAZINE MASS	=	2.000E+03 KG	(	4.409E+03 LBM)
77	OTV H2 MASS	=	5.000E+04 KG	(	1.102E+05 LBM)
78	OTV 02 MASS	=	2.000E+04 KG	(	4.409E+04 LBM)
79	EXCESS H2	=	9.597E-01 KG/DAY	(	2.116E+00 LBM/DAY)
80	OTV DEPOT H2 MASS	=	5.000E+04 KG	(	1.102E+05 LBM)
81	OTV DEPOT 02 MASS	=	2.000E+04 KG	(	4.409E+04 LBM)
82	OTV HYDRAZINE MASS	=	2.000E+03 KG	(	4.409E+03 LBM)
83	FOOD CONSUMPTION	=	1.435E+01 KG/DAY	(	3.163E+01 LBM/DAY)
84	DUMMY	=	1.000E+00		
85	TOTAL SOC EXPENDABLES MA	=	4.585E+03 KG	(	1.011E+04 LBM)
86	TOTAL SOC ONLY MASS	=	1.201E+05 KG	(	2.648E+05 LBM)
87	TOTAL NON-SOC EXPENDABLE	=	1.420E+05 KG	(	3.131E+05 LBM)
88	TOTAL SYSTEM MASS	=	2.651E+05 KG	(	5.844E+05 LBM)

Table 16-2. Nominal Output

EFFECT OF

	ORBIT ALT.	POWER	MASS	CMG	CMG DES	PROP EQPT	O/M PROP	SHUTTLE	CONSTR	FSF	CREW ROT.	LM	CREW OPs
ORBIT ALTITUDE		SOLAR ARRAY IS MAIN DRAG PRODUCER	NO EFFECT	MAY INFLUENCE ALTITUDE SELECTION	INFLUENCES FEASIBILITY OF VARIABLE ALTITUDE WITH INFREQUENT MAKEUP	NO EFFECT	NO	NO	PROJECTS MAY INFLUENCE DRAG. ALSO DESIRE NO OMS KIT	LOWER ALTITUDE IMPROVES PROPELLANT DELIVERY CAPABILITY	NO	NO	NO
POWER	LOWER ALT. INCREASES % OF ORBIT SHADOWED		NO EFFECT	UNINTERR-UPTIBLE LOAD	NO EFFECT	NO EFFECT	NO	SHADOWING DURING APPROACH & DEPARTURE	LIGHTING & EQUIPMENT LOADS	PUMPING, OTV MAINT. LIGHTING REFRIGERATION?	MORE EC/LSS POWER DURING OVERLAP PERIODS	NO	LIGHTING, EC/LSS POWER CREW [illegible].
MASS DISTRIBUTION & SYMMETRY	SLIGHT VARIATION IN GRAVITY GRADIENT	SOLAR ARRAY IS NOT SYMMETRICAL		MINOR EFFECT	CONSTRAINING FACTOR	MAKEUP PROPELLANT IS SIGNIFICANT MASS	NO	ROUGHLY DOUBLES MASS TO BE CONTROLLED	CONFIGURATION VARIABLE	CONFIGURATION VARIABLE & PROPELLANT MASS	NO	SIGNIFICANT MASS	NO
CMG SIZING	SLIGHT VARIATION IN GRAVITY GRADIENT	NO EFFECT	DIRECT INFLUENCE ON SIZING		DESIRE FOR INFREQUENT DESAT. TRENDS TO LARGER CMG'S	NO EFFECT	NO	DIRECT INFLUENCE ON SIZING	THRU MASS DISTR.	THRU MASS DISTR.	FREQUENCY OF SHUTTLE PRESENCE	THRU MASS DISTR.	NO
CMG DESATURATION FREQUENCY	LOWER ALTITUDE INCREASES	NO EFFECT	ASYMMETRY INCREASES DESIRED FREQUENCY	LARGER CMG'S REQUIRE LESS FREQ. REQUIRED		NO EFFECT	NO	ASYMMETRY MAY REQUIRE FREQUENT DESAT.	THRU MASS DISTR.	THRU MASS DISTR.	FREQUENCY OF SHUTTLE PRESENCE	THRU MASS DISTR.	NO
PROPULSION EQUIPT LOCATION	MAY AFFECT THRUSTER SIZE	THRUSTER PLUMES SHOULDN'T IMPINGE ON ARRAY	THRUSTERS MUST BE ABLE TO PROVIDE CONTROL MOMENTS	THRUSTERS MUST BE ABLE TO DESATURATE CMG'S	NO EFFECT		MAY INFLUENCE INSTALLATION DESIGN	NO	AVOID PLUME IMPINGEMENT	NO	NO	NO	MAINTENANCE & SAFETY CON-SIDERATIONS
ORBIT MAKEUP PROPELLANT SELECTION	LOWER ALTITUDE INCREASES VALUE OF ISP	NO EFFECT (UNLESS FUEL CELLS USED)	NO DIRECT EFFECT	NO DIRECT EFFECT	NO EFFECT	LENGTH OF PLUMBING RUNS COULD INFLUENCE		NO	CONTAMINATION WOULD PREFER O_2 - H_2	POTENTIAL ECONOMY OF USING OTV PROPELLANT	NO	NO	SAFETY CONSIDERATIONS
SHUTTLE DOCKING LOCATION	NO EFFECT	NO EFFECT	CONSIDER SHUTTLE-DOCKED EFFECTS	NO EFFECT	NO EFFECT	NO EFFECT	NO		MUST BE ABLE TO OFF LOAD PAYLOADS	MUST BE ABLE TO OFF LOAD PAYLOADS & PROPELLANT	NO	COULD BE A HANDLING PROBLEM	FLIGHT CONTROL & MONITORING
CONSTRUCTION FACILITY	DRAG EFFECT ON DEBRIS CONTROL	SELECT LOCATION TO AVOID SHADOWING	SELECT TO MINIMIZE DISTRIBUTION PROBLEMS	MAY CONSTRAIN LOCATION	CONSTRUCTION OPS MAY BE INTERRUPTED DURING MANEUVERS	POSSIBILITY OF PLUME IMPINGEMENT	NO	SELECT TO ENABLE PAYLOAD TRANSFER		NON INTERFERENCE	NO	NON-INTERFERENCE	SCHEDULING
FLIGHT SUPPORT FACILITY	INFLUENCES ORBIT PHASING	SELECT LOCATION TO MINIMIZE SHADOWING	SELECT LOC. TO MINIMIZE PROBLEMS & AID PROP. TRANSFER	MAY CONSTRAIN LOCATION	FS OPS MAY BE INTERRUPTED BY MANEUVERS	NO EFFECT	POTENTIAL COMMONALITY WITH OTV PROPELLANT	SELECT TO ENABLE PAYLOAD & PROPELLANT TRANSFER	NON-INTERFERENCE		NO	NON-INTERFERENCE	SCHEDULING
CREW ROTATION & RESUPPLY FREQUENCY	NO EFFECT	NO EFFECT	NO EFFECT	NO EFFECT	NO EFFECT	NO EFFECT	NO	NO	WOULD LIKE CREW OVERLAP	WOULD LIKE CREW OVERLAP		NO	WOULD LIKE CREW OVERLAP
LOGISTICS MODULE SIZE	LOWER ALTITUDE INCREASES PROPELLANT SUPPLY	NO EFFECT	SELECT DOCK-ING LOCATION TO MINIMIZE	NO EFFECT	NO EFFECT	NO EFFECT	PROPELLANT RESUPPLY VOLUME	NO	NO	OTV SPARES	MORE OFTEN REDUCES SIZE		RESUPPLY ROTS.
CREW OPERATIONS	NO EFFECT	MAINTENANCE WORKLOAD	NO EFFECT	CMG DESAT MAY BE MANUALLY CONTROLLED	AFFECTS CREW WORKLOAD	THRUSTER PLUMES ARE POTENTIAL EVA HAZARD. ALSO EFFECTS MAINT OPS	NO	SELECT FOR CONVENIENT CREW TRANSFER	MAJOR WORKLOAD	MAJOR WORKLOAD	WOULD LIKE CREW OVERLAP	ADDITIONAL HABITABLE VOLUME, PRIVATE?	

Table 16-3. System Interrelationships

16.4 RESUPPLY REQUIREMENTS

The resupply requirements to sustain a crew of eight for 90 days are shown in Table 16-4. Food will be supplied to the SOC with about 80% as rehydrable food and the other 20% as frozen food. The ECLS hydrazine is used to replace N_2 lost due to cabin leakage including airlock activity. Even though water will be recycled for SOC, a substantial amount will be lost through backpack cooling during EVA. This water must be resupplied and is substantial as shown in the table. The other supplies shown include such items as filters, commode liners, and LiOH canisters (required for EVA).

In addition to the normal supplies, emergency supplies must be provided for the SOC. These supplies must be sufficient to sustain the crew for 90 days starting anyplace in the resupply cycle. These items are listed in Table 16-5 along with the amounts required. The 1990 lbs of food will provide approximately 57% of the normal rations for 90 days. 534 lbs of atmosphere will repressurize one habitable module. The water and EVA supplies will support one half the number of EVA's. Hydrazine and ECLS supplies are for a full 90 day requirement. All other emergency considerations for SOC are based on redundant systems to provide fail-operational capability.

16.5 RESULTS

With the exception of power and thermal control, which all subsystems need, the SOC subsystems are essentially independent in terms of basic function. For this reason, under emergency situations, the subsystems can be either autonomous or manually-operated, not requiring control from the SOC main computer. Highlights of the results are shown in Table 16-6.

Table 16-4. Resupply Requirements (Crew of 8 for 90 Days)

SOC-1147

ITEM	LBS.
• TOTAL CONSUMABLES	16,140
– SHELF STABLE FOOD	2,592
– FROZEN FOOD	720
– HYDRAZINE (ECLS)	2,182
– HYDRAZINE (ATTITUDE CONTROL)	4,850
– WATER	4,960
– ATMOSPHERE	534
– SPARES	300
• TOTAL SUPPLIES	6,272
– PERSONAL SUPPLIES	1,327
– SHIP STORES	104
– HOUSEKEEPING/HYGIENE	708
– ECLS RESUPPLY	1,018
– EVA RESUPPLY	3,011
– MAINTENANCE	104

Table 16-5. Stored Emergency Supplies (Crew of 8 for 90 Days)

SOC-1148

ITEM	LBS.
• FOOD	1,990
• ATMOSPHERE	534
• WATER	2,500
• HYDRAZINE	
– ATTITUDE CONTROL	4,850
– ECLS	2,182
• ECLS SUPPLIES	1,018
• EVA SUPPLIES	1,500

Table 16-6. Subsystems Interrelationships Highlights

SOC-772

INPUT TO ↓ / INPUT FROM →	POWER	TRACKING & COMMUNI-CATIONS	EC/LSS & THERMAL CONTROL	FLIGHT CONTROL	PROPULSION	CONTROLS/ DIS & MAIN PROCESSOR	CREW
POWER	—	Ground Commands*	Thermal Control	None	None	• Power Switching Requests	• Manual Backup Operation
TRACKING & COMMU-NICATIONS	Power	Ground Commands*	Thermal Control	Status	None	• SOC Data	• Audio & Video Comm • Backup Op
EC/LSS & THERMAL CONTROL	Power	Ground Commands*	—	None	None	• Autonomous Operation Overrides	• Manual Backup Operation
FLIGHT CONTROL	Power	• State Vector Updates • Attitude Commands	Thermal Control	—	Execution Verification	• Flight Control Overrides	Manual Backup Operation
PROPULSION	Power	None	Thermal Control	Thrust Commands	—	• Manual Thrust Control	None
CONTROLS, DISPLAYS, & MAIN PROCESSOR	• Power • Status Data	• Data Updates • Software Updates	• Thermal Control • Status Data	• Status Data	• Status Data	—	• Normal Control & Data Inputs
CREW	• Lighting	• Audio & Video Communi-cations	Thermal Control & Life Support	None	None	• Status Displays • C&C Data • Scheduling	—

*During Unmanned Periods, e.g. Buildup

17.0 SYSTEM DESIGN/OPERATION ANALYSIS

17.0 SYSTEM DESIGN/OPERATION ANALYSIS

17.1 INTRODUCTION

The main results of the SOC system design and operations analysis activity were presented in section 2 of this report, and in the SOC System Description. Two subjects are presented in additional detail in this section.

17.2 INITIAL SOC CONFIGURATION ANALYSIS

The objective of this analysis is to characterize a concept for an early SOC configuration sized for 4 people and which could be declared to be permanently habitable after the minimal number of SOC module delivery flights. This early configuration, to be called the "Initial SOC Configuration", should be designed using the SOC elements previously defined. This configuration should be designed so that it can grow into the previously defined Reference SOC Configuration, now known as the "Operational SOC Configuration". The Initial SOC should represent a relatively low cost initial plateau in the SOC evolution. The requirements for a Space Operations Center (D180-26135-1) should be observed to the extent that they would be applicable to this early stage of SOC evolution.

Preliminary Initial SOC Concept

Figure 17-1 illustrates a concept for the Initial SOC Configuration that could be put in place with 3 delivery flights. Table 17-1 lists the "scars" required to make the Initial SOC a reality.

The areas that were studied to resolve pertinent design and operational issues are listed below:

o Habitability - what habitability provisions are required to accomodate the 4-man crew in the emergency mode where either pressurized module required

SOC-803

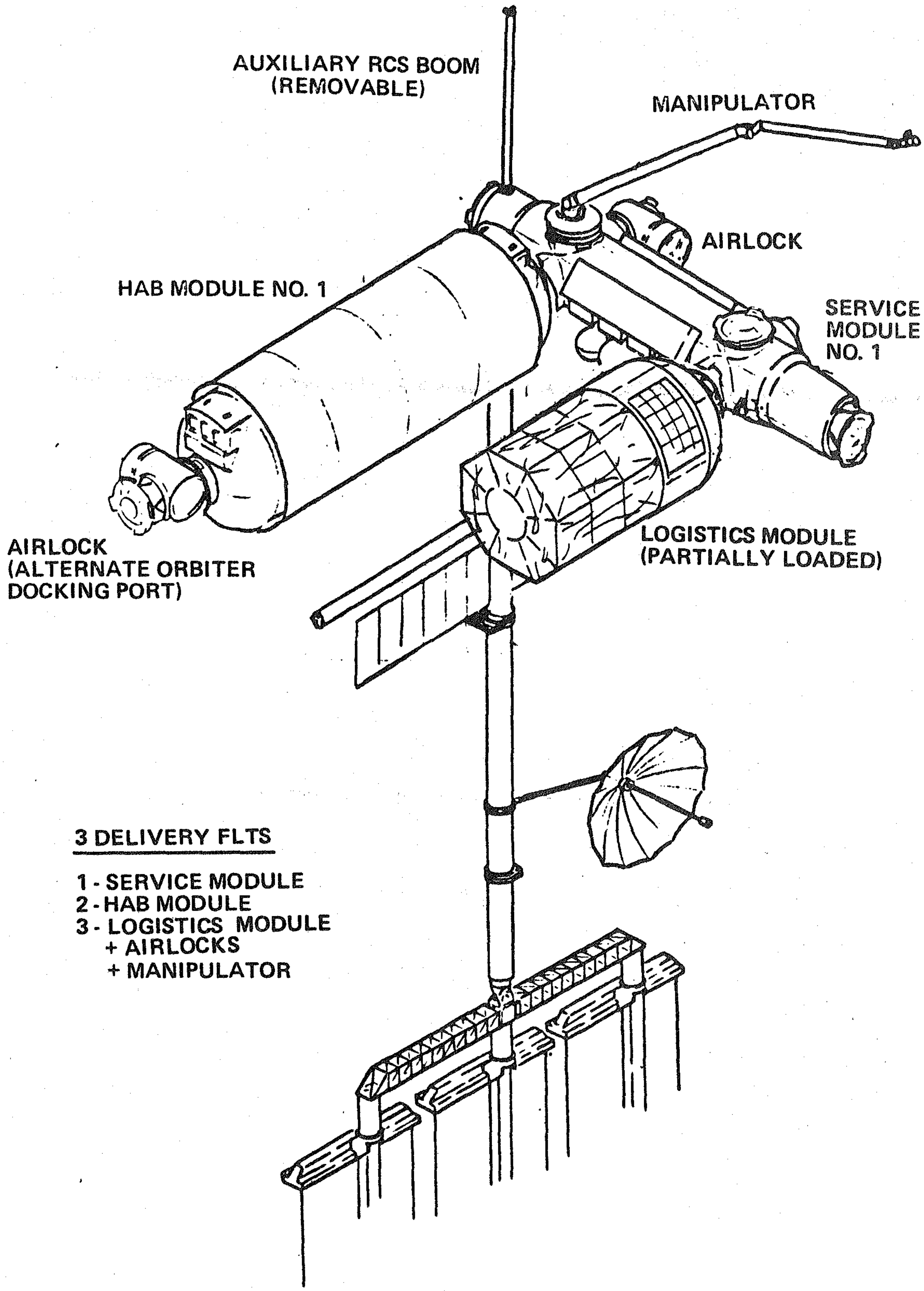

Figure 17-1. Initial SOC Configuration

TABLE 17-1
INITIAL SOC SCARS*

AUXILIARY RCS BOOM

MANIPULATOR

SERVICE MODULE NO.1 EC/LSS ADDITIONS

- o Dehumidifier
- o 2 Waste Water Storage Tanks
- o Atmosphere Monitor
- o Cold Water Supply
- o Hot Water Supply
- o Water Pump
- o Dry John
- o Backpack Reconditioning Unit
- o Emergency CO_2 Removal
- o Control/Display Console
- o 8 Ft_3 Food Stowage

HABITAT MODULE NO. 1 EC/LSS ADDITIONS

- o Clothes Washer/Dryer
- o Power Jumper Cables
- o Voice Comm Antenna

* Things added to make the Initial SOC feasible but which will have little or no utility in the Operational and Growth SOC's.

evacuation?

o Resupply - what provisions must be made to resupply the 4-man crew?

o Flight Control - what provisions are necessary to make this asymmetric configuration controllable?

o Mission Utility - what missions could be accommodated with this configuration?

o SOC Build-up Sequence - how would the Initial SOC be built-up? How to grow into the Operational SOC?

Habitability

In order that the Initial SOC can be permanently declared to be habitable, it is necessary to meet the requirements given in the Requirements document (D180-26495-2). The only requirement that needs to be amended is Requirement 2.102 which declared that the crew should be able to evacuate any module and carry on business as usual for up to 90 days. For the initial SOC, it was deemed reasonable to amend this requirement to require only 21 days of survival in an emergency mode in the event that one of the pressurized modules were vacated. This 21 day period is the time required for a Shuttle rescue visit.

In discussions between Boeing, Hamilton Standard, and NASA-JSC SOC study team members, the environmental control/life support provisions for the Initial SOC listed in table 17-2, were mutually agreed upon. This table cites the delta provisions that have to be made to meet the requirements. Figure 17-2 illustrates the Service Module No. 1 interior configuration that results from this analysis. (The second Service Module will not require the provisions shown in shading on this figure.)

Note that an external airlock has been added to the SM to provide EVA capability under emergency conditions when only the SM is habitable.

If an external airlock is to be provided for the SM, program cost savings are

FUNCTION	HABITAT MODULE PROVISIONS	SERVICE MODULE PROVISIONS
CABIN VENTILATION & THERMAL CONTROL	• 4 VENTILATION FANS • 4 AIR COOLING HEAT EXCHANGERS • 10 COLD PLATES	• 2 VENTILATION FANS • 2 AIR COOLING HEAT EXCHANGERS • 26 COLD PLATES
AIR REVITALIZATION	• 2 DEHUMIDIFIERS • 2 CO_2 REMOVAL UNITS • 2 CATALYTIC OXIDIZERS UNITS • 2 ODOR CONTROL UNITS • 1 ATMOSPHERE MONITORING UNITS	• 1 EMERGENCY CO_2 REMOVAL SYS** • 1 CO_2 REDUCTION UNIT • 1 DEHUMIDIFIER** • 1 ODOR CONTROL UNIT ** • 1 TRACE CONTAMINENT REMOVAL UNIT **
HEAT TRANSPORT & REJECTION	• 2 RADIATORS • 2 FREON PUMP PACKAGES • 2 FREON–TO–WATER HEAT EX. • 2 WATER PUMP PACKAGES	• 2 RADIATORS • 1 FREON PUMP PACKAGE • 1 FREON–TO–WATER HEAT EX. ** • 1 WATER PUMP PACKAGE **
ATMOSPHERIC SUPPLY		• 1 O_2 GENERATION UNIT • 1 HYDRAZINE DECOMP/N_2 SUPPLY • EMERGENCY O_2 STORAGE **** • EMERGENCY N_2 STORAGE ****

* INITIAL SOC = 1 SERVICE MODULE + 1 HABITAT MODULE + LOGISTICS MODULE

** PROVISION INCLUDED FOR USE DURING INITIAL SOC CONFIGURATION THAT WILL NOT BE REQ'D FOR OPERATIONAL SOC

**** O_2 AND N_2 STORAGE VOLUME INCREASED TO COVER EMERGENCY MODE FOR INITIAL SOC

*Table 17-2. Environmental Control/Life Support Provisions for the Initial SOC**

SOC-826

FUNCTION	HABITAT MODULE PROVISIONS	SERVICE MODULE PROVISIONS
WATER PROCESSING & MANAGEMENT	• 2 EVAPORATION PURIFICATION UNITS • 1 WATER QUALITY MONITOR • 3 WASTE WATER STORAGE TANKS • 3 POTABLE WATER STORAGE TANKS • 4 EVA/EMERGENCY WATER ST. TANKS	• 2 WASTE WATER STORAGE TANK **
HEALTH & HYGIENE	• 1 WASTE COLLECTION/STORAGE • 1 HOT WATER SUPPLY • 1 COLD WATER SUPPLY • 1 SHOWER *** • 1 HAND WASHER *** • 1 CLOTHES WASHER/DRYER ** • 1 TRASH COMPACTOR • 1 FOOD FREEZER *** • 1 OVEN *** • 1 DISHWASHER	• 1 WASTE COLLECTION/STORAGE ** EMERGENCY WASTE COLLECTION BAGS ** • 1 HOT WATER SUPPLY ** • 1 COLD WATER SUPPLY ** • WET WIPES ** • TRASH DISPOSAL BAGS **

* INITIAL SOC-1 SERVICE MODULE + 1 HABITAT MODULE + LOGISTICS MODULE
**PROVISION INCLUDED FOR USE DURING INITIAL SOC CONFIGURATION THAT WILL NOT BE REQ'D FOR THE OPERATIONAL SOC
*** COULD BE A RETRO FIT ITEM

Table 17-2. Environmental Control/Life Support Provisions for the Initial SOC (Cont'd)*

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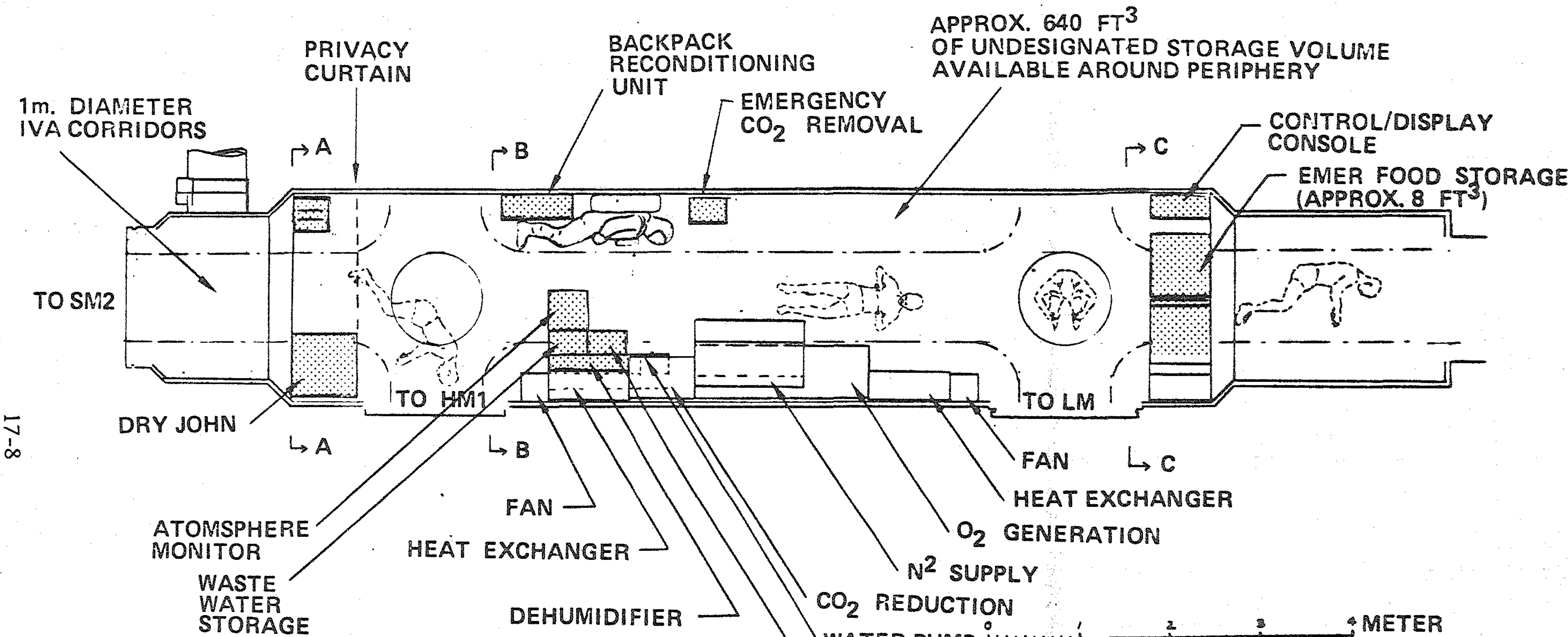
FUNCTION	HABITAT MODULE PROVISIONS	SERVICE MODULE PROVISIONS
EVA SUPPORT NOTE—EXTERNAL AIRLOCKS LOCATED ON SM AND ON HM	• 2 SUITS & BACKPACKS • 1 RECHARGE STATION	• 1 SUIT & BACKPACK • 1 RECHARGE STATION **
SYSTEM CONTROL	• 1 CENTRAL ECLS CONT/DISP	• 1 PORTABLE MAINT. CONT/DISP
FOOD	• SHELF STABLE FOOD STORAGE • FROZEN FOOD	• SHELF STABLE FOOD STORAGE **

* INITIAL SOC 1 SERVICE MODULE + 1 HABITAT MODULE + LOGISTICS MODULE

** PROVISION INCLUDED FOR USE DURING INITIAL SOC CONFIGURATION THAT WILL NOT BE REQ'D FOR THE OPERATIONAL SOC

Table 17-2. Environmental Control/Life Support Provisions for the Initial SOC (Cont'd)*



NOTE: SHADED ITEMS ARE THOSE ELEMENTS ADDED TO MAKE THE SERVICE MODULE HABITABLE FOR A CREW OF 4 FOR 14 DAYS IN AN EMERGENCY MODE

Figure 17-2. Service Module Interior Arrangement for the Initial SOC Configuration

D180-26495-4

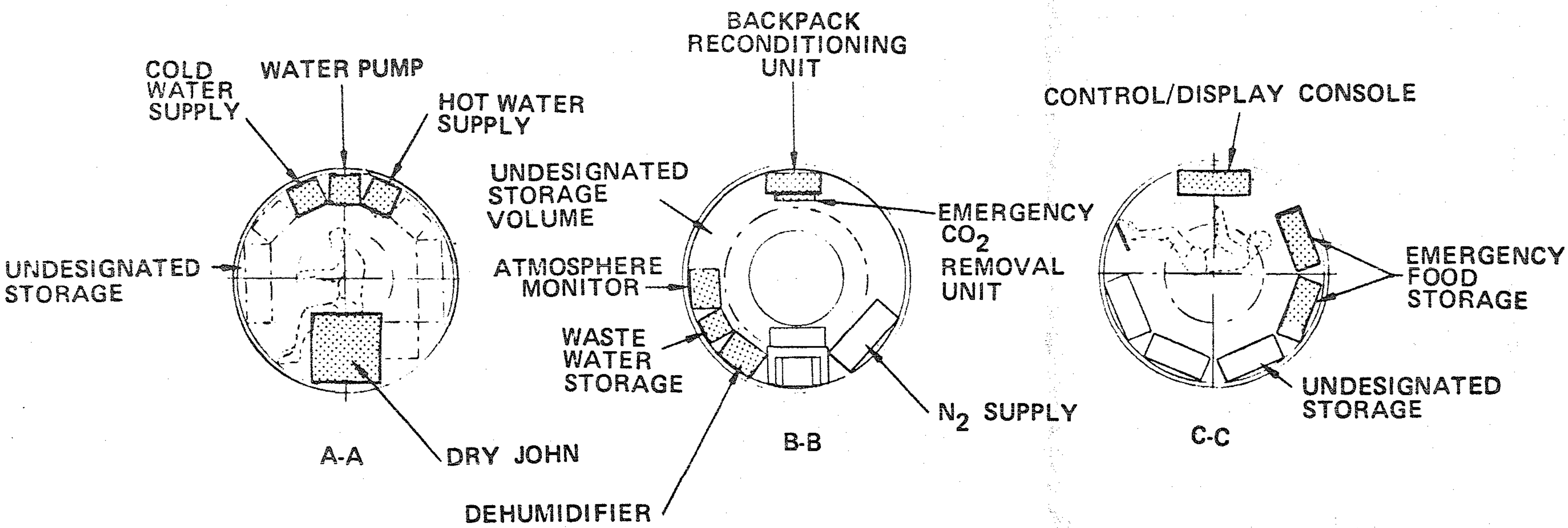


Figure 17-2. Service Module Interior Arrangement for the Initial SOC Configuration (Cont'd)

possible if the same design is used for the HM, eliminating the internal airlock. (This avoids producing two airlock designs. The external airlock can be a modification of the design proposed for the Shuttle airlock and docking adapter.) The second airlock, located on the HM in the initial configuration, can be relocated to the Docking Module in the operational configuration.

Removal of the internal airlock from the HM frees up a significant internal volume. A review of mass and center-of-gravity for the reference HM revealed that (a) while the total launched mass is less than the Shuttle target capability of 65,000 lbs., it slightly exceeds a probably actual capability of about 55,000 lbs., (b) the cg, incuding the Shuttle airlock-docking module and the manipulator is inside but close to the Shuttle landing cg limit. Accordingly, a reconfiguration of the HM has removed the internal airlock and about 1 meter (3 ft.) of length. The resulting mass reduction is 3400 lb. this reduced the estimated launched mass from 56,800 lbs. to 53,400 lbs. The revised cg has not been estimated but should be comfortably within the limits as the reconfiguration has also moved the HM cg aft in the launched configuration. It should be noted that the above mass estimates include about 8000 lbs. of growth margin and about 5000 lbs. of mission equipment that could be moved to the first logistics launch if necessary.

It is, of course, desirable that as much internal volume as possible be retained in the HM. The length removed was taken out of the recreation area. It can be replaced later by a simple extension of the HM cylinder if continuing mass and cg reviews indicate more margin is available.

An unresolved issue is where to locate the control moment gyros (CMG's). There is some question as to whether or not CMG's are required. If they are required, we need to define their size and find suitable installation locations.

Resupply

Table 17-3 lists the resupply requirements and resupply options that are being considered. Table 17-4 gives the results of the trades of these options. The preferred resupply concept is the one where the previously defined Logistics Module would be used but it would only be half loaded on each trip.

Flight Control

The initial configuration is asymmetric in drag characteristics because it has only one solar array wing. Flight control authority can be gained by attachment of a simple thruster boom on the opposite side of the SM from the array. This boom can be removed later when the second SM is added. The thruster module can be retained as a spare.

The propellant consumption for this configuration will be dictated by drag torque compensation rather than orbiter makeup. This is a tolerable situation for the initial configuration provided that this configuration is an interim step, as presently conceived. If solar activity is low during the existence of this configuration (as is presently forecast), the nuisance will be minimal. If solar activity is high, propellant consumption at the nominal (high-activity) SOC altitude of 405 Km would force revisits more frequent than 90 days. In such an event, a practical workaround is to use the SOC onboard propulsion to raise the orbit to about 450 Km. Shuttle OMS kits would be required for logistics visits. When it is planned to continue the buildup, the orbit could be lowered to the 370 Km altitude necessary for delivery of the second SM and HM.

Build-up Sequence

Figure 17-3 illustrates the build-up sequence for Initial SOC and the build-up sequence to grow from the Initial SOC into the Operational SOC.

Mission Utility

An analysis was conducted to ascertain what equipment would be required for

TABLE 17-3
INITIAL SOC RESUPPLY OPTIONS

REQM'TS - 90 day resupply for 4 people

Pressurized Storage 260 ft^3 (if no frozen food); add 20 ft^3 if frozen food included.

Unpressurized Storage for H_2O and Hydrazine.

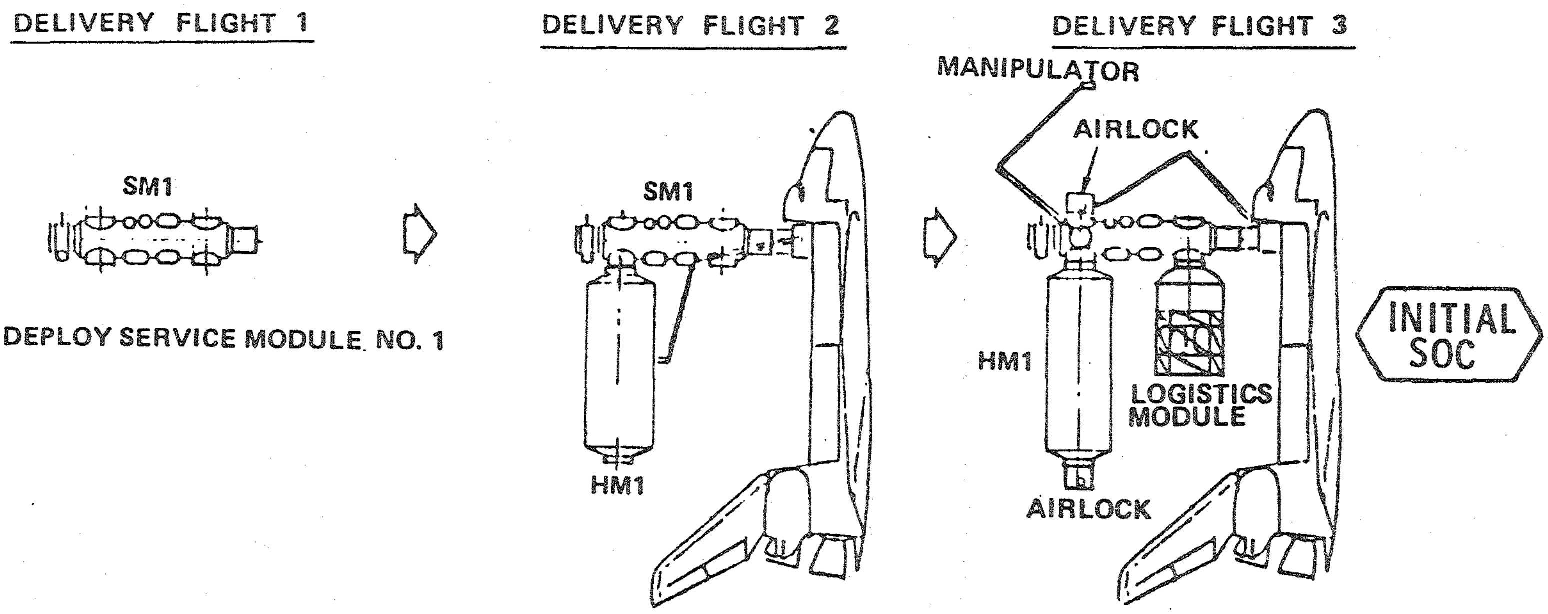
RESUPPLY OPTIONS -

1. Use previously defined Logistics Module (LM)
   - 1A - Transport 90 days supply (LM only partially loaded)
   - 1B - Transport 180 days supply (LM fully loaded)
2. Use a Logistics Pallet (based on Spacelab pallet)
   - 2A - Pallet contains fluids tanks only. Resupply items requiring pressurized environment transported in Orbiter mid-deck (approx. 160 ft^3). This would require a resupply frequency of 28 days as the total of 260 ft^3 could not be transported.
   - 2B - Pallet contains fluid tanks and 100 ft^3* of pressurized storage containers. Containers would be small enough to hand-carry via EVA into the SOC.
3. Use Spacelab Pressurized Module and a Spacelab 3 meter pallet. A pressurized module composed of 2 Spacelab Experiment Segments (without overhead viewing window) and using standard storage racks yields 226 ft^3 of storage volume. The balance (260 - 226 = 34) of 34 ft^3* could be transported in the Orbiter mid-deck. The fluids would be transported on a spacelab pallet as was indicated in Option 2A.

* Add 20 ft^3 if frozen food to be included.

TRADE FACTORS	OPTIONS				
	1A LM Half Loaded	1B LM Fully Loaded	2A LP, 55 Day Resupply Freq.	2B LP + Press. Containers	3 Spacelab Elements
● COST (SCALE OF 1 TO 10)					
• INITIAL COST	10	10	5 *	7	8-9
• OPERATIONS COST	7	5 *	10	7	7
● OPERATIONS					
• EVA REQ'D?	NO *	NO *	NO *	YES	NO *
• VOL. OF FOOD TO REDISTRIBUTE					
• Redistribute Residual food	$8.5 ft^3$	$8.5 ft^3$	0 *	0 *	$8.5 ft^3$
• distribute new food	$58 ft^3$ *	$116 ft^3$	$160 ft^3$	$260 ft^3$	$58 ft^3$ *
● FOOD STORAGE VOLUME REQM'T					
• Service Module	$32 ft^3$ *	$64 ft^3$	$67.5 ft^3$	$56.5 ft^3$	$32 ft^3$ *
• Habitat Module	$55 ft^3$ *	$97 ft^3$	$87.5 ft^3$	$76.5 ft^3$	$55 ft^3$ *
● USE BEYOND INITIAL SOC?	YES *	YES *	NO	NO	NO
	PREFERRED CONCEPT				

Table 17-4. Initial SOC Resupply Options Trade



- INSTALL HABITAT MODULE NO. 1 ONTO SM1 USING ORBITERS RMS
- HABITABLE IF SHUTTLE-TENDED

- INSTALL
  - LOGISTICS MODULE
  - AIRLOCKS
  - MANIPULATOR
- INSTALLATION OF HM AIRLOCK REQUIRES AN EXTENSION TOOL FOR THE SOC'S MANIPULATOR
- 4 MAN SOC CREW MOVES ON-BOARD FOR 90 DAY STAYTIME
- INITIAL SOC NOW PERMANENTLY HABITABLE

Figure 17-3. SOC Build-Up Sequence

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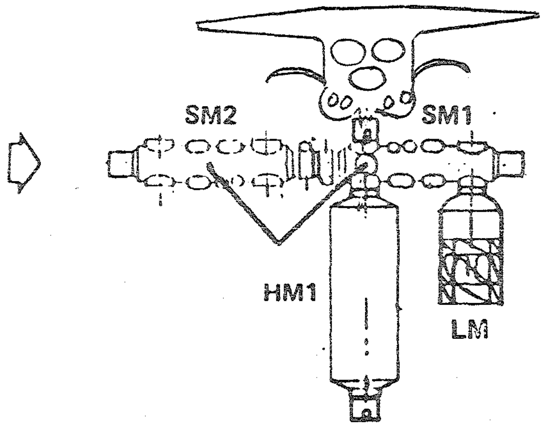
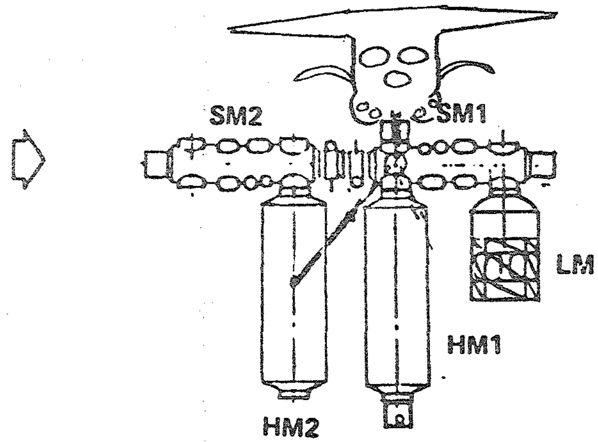
DELIVERY FLIGHT NO. 4



- DOCK ORBITER TO SM AIRLOCK
- INSTALL SERVICE MODULE NO. 2

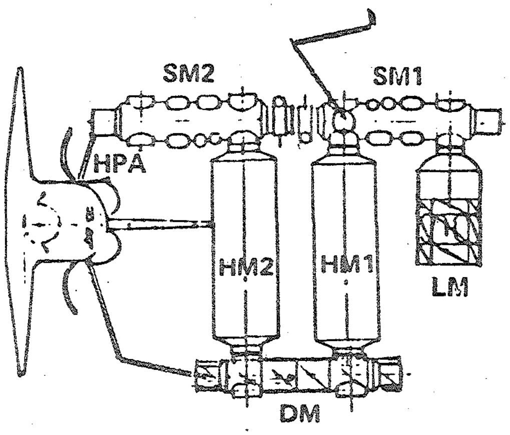
DELIVERY FLIGHT NO. 5



- DOCK ORBITER TO SM AIRLOCK
- INSTALL HABITAT MODULE NO. 1
- COULD HOUSE MORE THAN 4 PEOPLE FOR SHORT PERIODS (LESS THAN 45 DAYS)

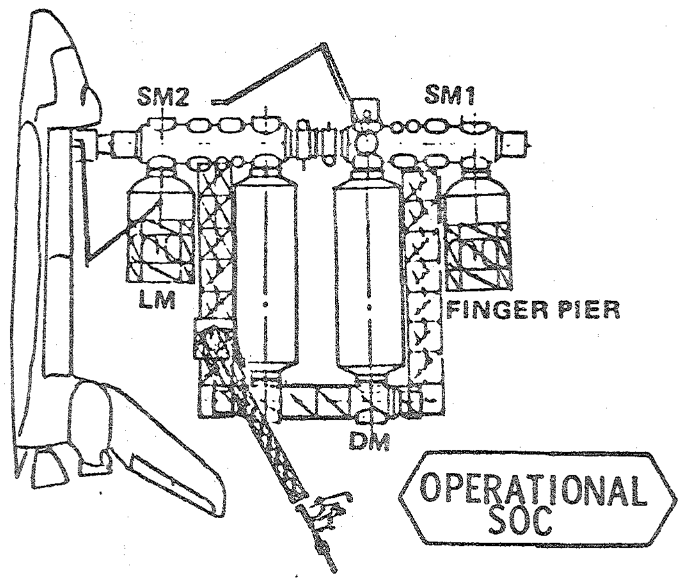
Figure 17-3. SOC Build-Up Sequence (Cont'd)

DELIVERY FLIGHT NO. 6



- ORBITER EQUIPPED WITH A HPA
- HM AIRLOCK REMOVED AND TEMPORARILY STOWED ON SM1 (REQUIRES EXTENSION TOOL FOR SOC'S MANIPULATOR)
- INSTALL DOCKING MODULE ONTO HABITAT MODULES
- MOBILE CHERRYPICKER DELIVERED WITH THE DM
- AFTER DM INSTALLED, AIRLOCK MOVED TO DM

DELIVERY FLIGHT NO. 7



- DELIVER LOGISTICS MODULE, FINGER PIER COMPONENTS, AND OTHER MISC. ELEMENTS
- CAN NOW BE PERMANENTLY INHABITED BY CREW OF 8

Figure 17-3. SOC Build-Up Sequence (Cont'd) the Initial SOC to support the various SOC missions. Table 17-5 summarizes the results.

For the Initial SOC configuration shown in figure 17-1, the payload size constraints were ascertained (see table 17-6.)

17.3 SOC HABITAT MODULE RADIATOR PERFORMANCE

A thermal analysis was performed to determine the heat rejection capability of radiators mounted on the SOC habitat modules. The full SOC configuration 1 was used for the analysis. The thermal model included two (2) habitat modules, two (2) service modules and one (1) docking module. Figure 17-4 shows a computer-generated plot of the thermal model. (Half cylinders were used for the docking and service modules since the other halves do not affect the radiation balance.)
The radiation simulation (RADSIM) and orbital payload environmental radiation analyzer (OPERA) computer programs were used to calculate radiation view factors and orbital heat loads. These programs account for blockage and multiple reflections. The results from these programs were used to calculate the heat rejection capability of the radiators.

The following assumptions were used in the analyses:

(1) 200 n.m. circular orbit
(2) Solar flux = 444 BTU/ft.^{2} hr.
(3) Earth emission = 83 BTU/ft.^{2} hr.
(4) Albedo = 0.40
(5) White paint on all surfaces ($\alpha = 0.9, \epsilon_s = 0.2$)
(6) All surfaces, except radiators, adiabatic (i.e., floating at the local sink temperature)
(7) Radiator area = 1890 ft.^{2}/module

Two Beta (sun to orbit plane) angles were considered. Figure 17-5 shows the

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EQUIPMENT	MISSIONS		
	FLIGHT SUPPORT MISSIONS	CONSTR MISSIONS	SATELLITE SERVICING MISSIONS
SOC STANDARD EQUIPMENT			
MANIPULATOR	●	●	●
MANIPULATOR END EFFECTOR	●	●	●
OPEN MRWS		●	●
UMBILICAL SYSTEM	●	●	●
TURNTABLE/TILTTABLE	●	●	●
EVA SUITS, TOOLS, ETC	●	●	●
MISSION DEDICATED EQUIPMENT			
STORAGE RACK		●	●
FIXTURES	●	●	●

Table 17-5. Summary of Equipment Required to Support Flight Support, Construction, and Satellite Servicing Missions on the Initial SOC

- IF MANIPULATOR NOT REQ'D TO SERVICE ATTACHED MISSION MODULE, MAX SIZE MODULE IS MAX SIZE THAT CAN BE TRANSPORTED BY THE SHUTTLE (15 ft Ø x 60 ft long)
- IF MANIPULATOR REQ'D TO SERVICE ATTACHED MISSION MODULE
  - ATTACHED TO BP–6 MISSION MODULE MAX LENGTH APPROX 9m
  - ATTACHED TO BP–5 MISSION MODULE MAX LENGTH APPROX 11m
- MAXIMUM SIZE OF PAYLOAD
  - ATTACHED TO BP–6
    - MAX DIAM = 10.6mϕ (limited by interference with airlock)
    - MAX LENGTH = 7m *
  - ATTACHED TO BP–5
    - MAX DIAM = 12mϕ (limited by interference w/ manipulator)
    - MAX LENGTH = 9m *
  - IF ORBITER IS STILL DOCKED
    - MAX DIAM = 8mϕ (limited by interference w/Orbiter cabin)
    - MAX LENGTH = 7 to 9m *

*THIS DIMENSION ASSUMES THAT PAYLOAD IS ATTACHED TO THE TURNTABLE/TILTTABLE

Table 17-6. Payload Size Constraints for Initial SOC

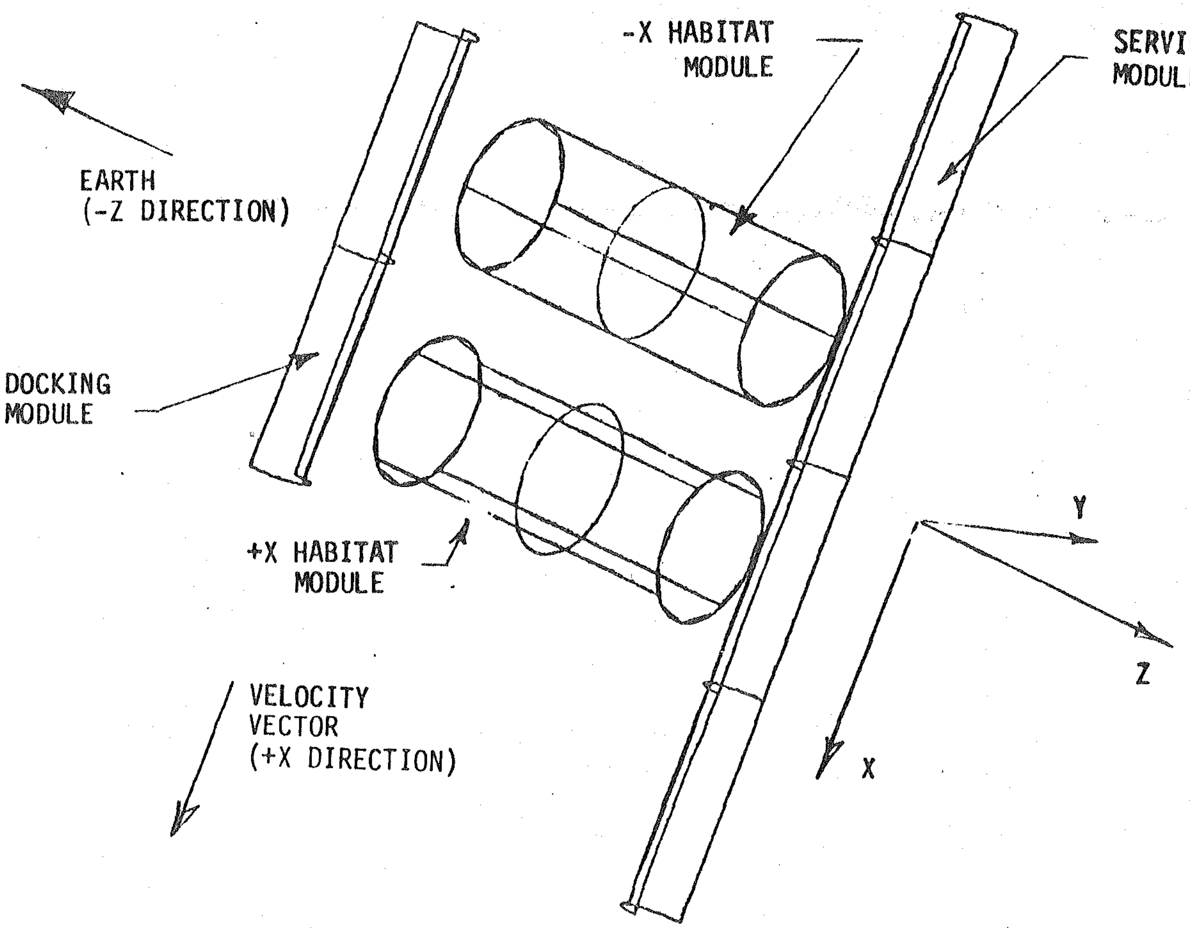

Figure 17-4. SOC Configuration 1 Mid 1990's

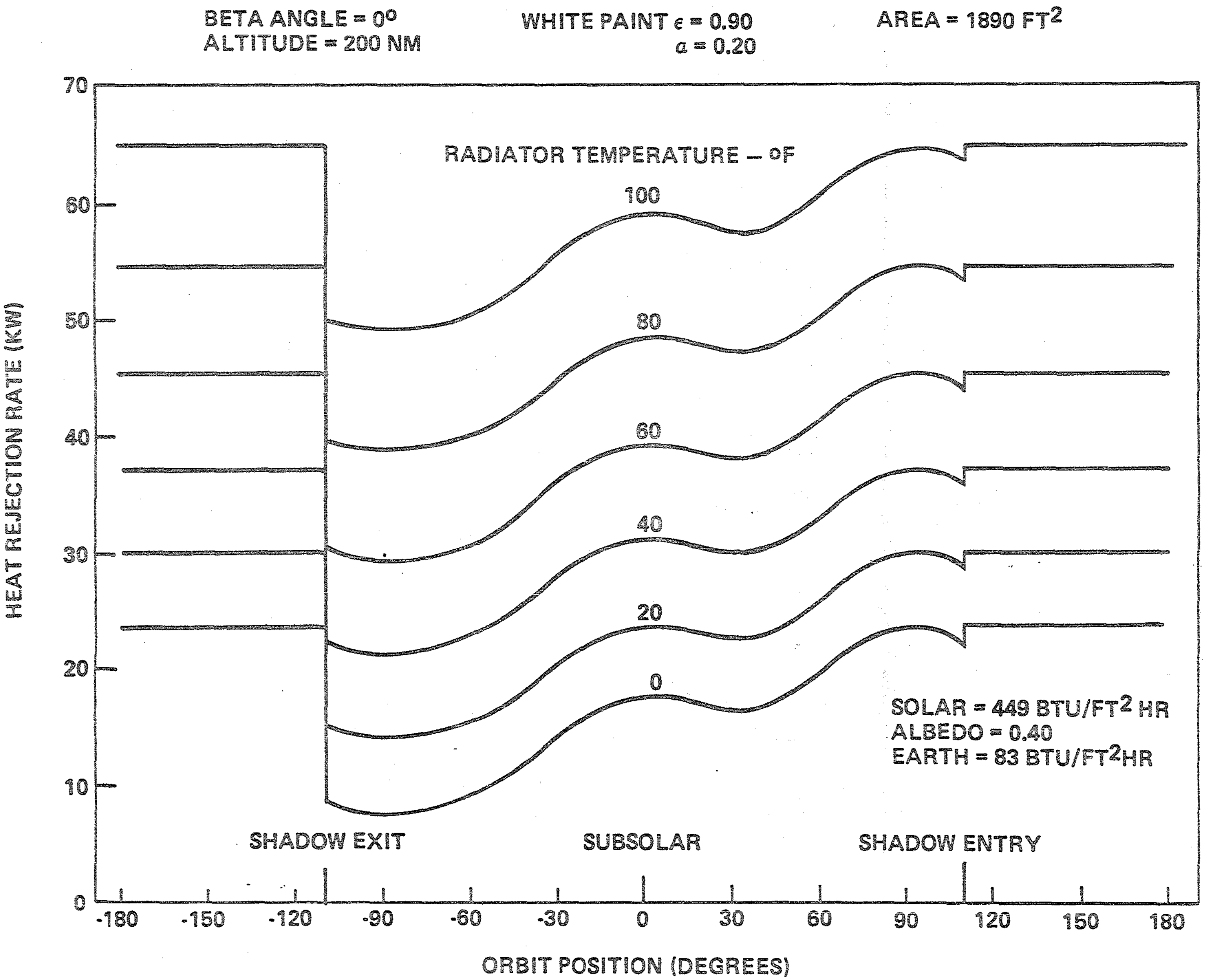

Figure 17-5. SOC Habitat Module Radiator Capability (+X Module)

heat rejection capability for a 0 degree Beta angle and figure 17-6 shows it for a 53 degree Beta angle. These figures give the instantaneous heat rejection rate as a function of orbit position for the +X habitat module radiator. The results for the -X habitat module radiator are identical if the sign of the orbit position angle is reversed.

Figure 17-7 shows the radiator heat rejection rate as a function of temperature for both the shadowed and sunlight portions of the orbit. The sunlight portion curves are based on a time average. Also shown is the required radiator area, to reject a kilowatt of heat, as a function of temperature. The recommended preliminary design point is a radiator temperature of $50^{o}F$ and radiator area of 80 $ft.^{2}/KW$.

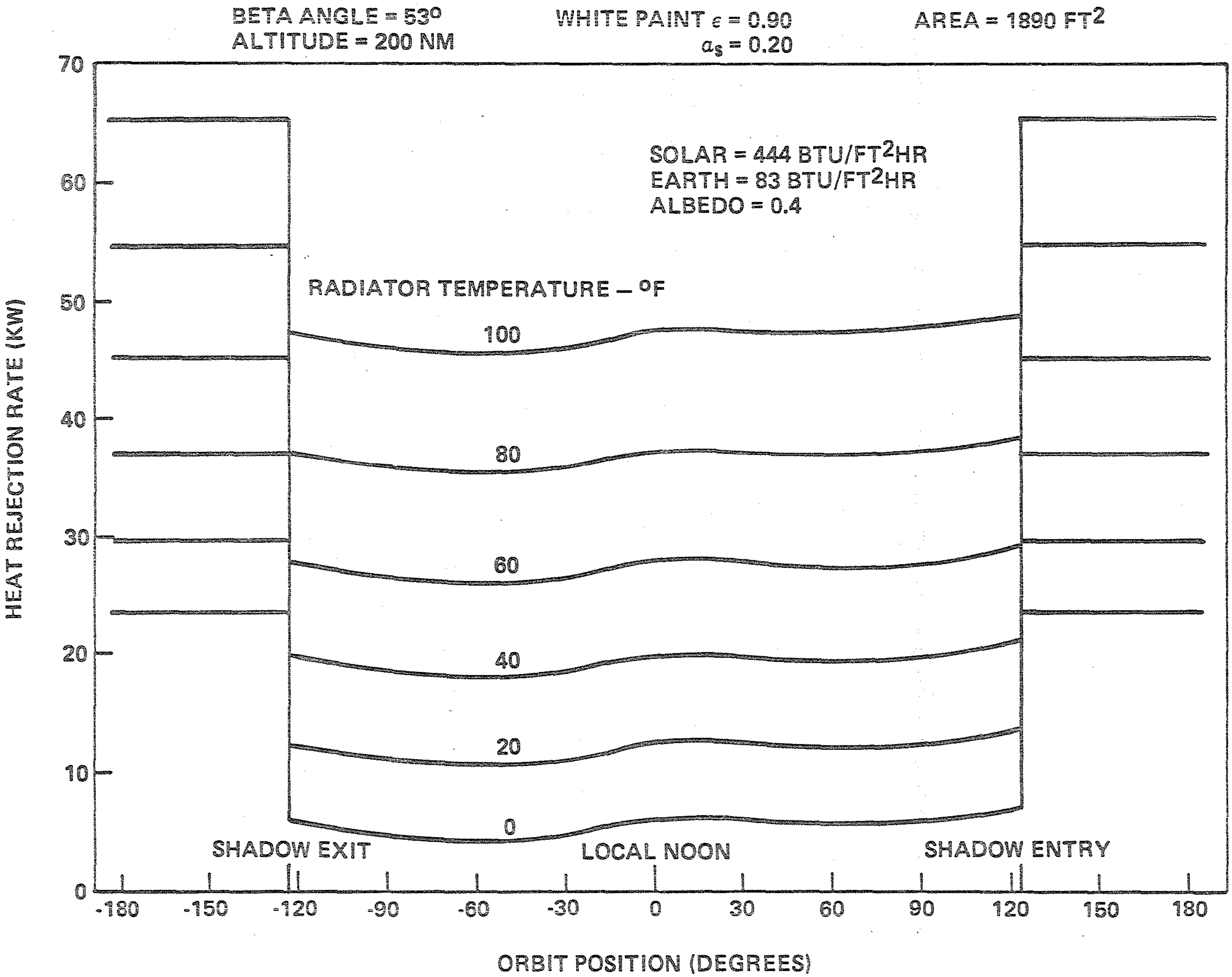

Figure 17-6. SOC Habitat Module Radiator Capability (+X Module)

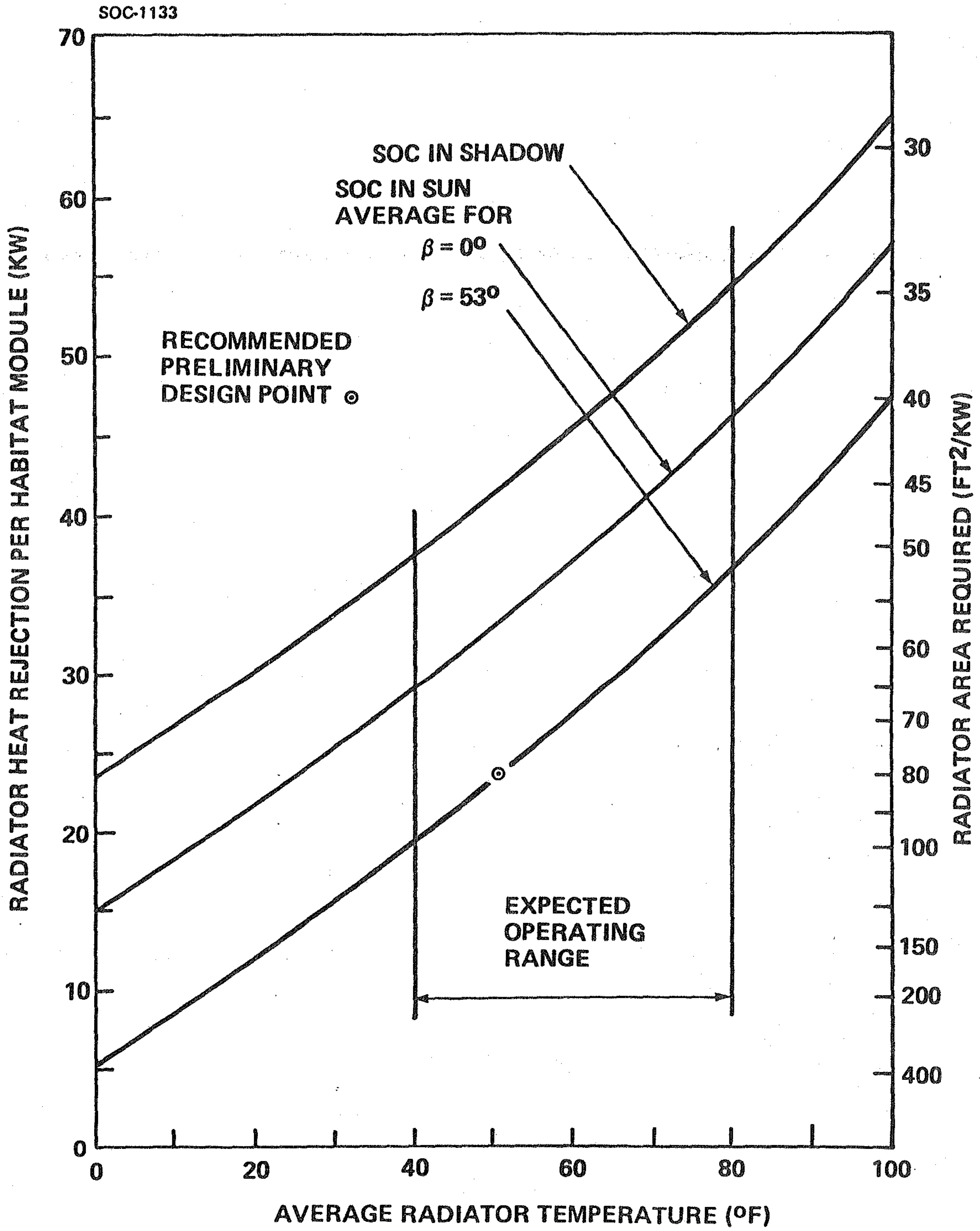

Figure 17-7. SOC Habitat Module Radiator Performance

18.0 PROGRAMMATICS

18.0 PROGRAMMATICS

18.1 INTRODUCTION

The study of Space Operations Center programmatics included considerations of program structure, cost, hardware commonality, schedules, and program phasing. The results of the cost analysis are reported in the System Description document (Boeing-19). A summary of program phasing options is included in section 2 of this report. This section presents more detailed discussions of (18.2) program structure and work breakdown structure, (18.3) schedules and schedule analysis, (18.4) system buildup options and funding profiles, and (18.5) recommended technology levels and status of the recommended technologies.

18.2 PROGRAM STRUCTURE AND WORK BREAKDOWN STRUCTURE

Selection of a work breakdown structure for a new program is an important factor in establishing the overall program structure and management approach. Decisions built into the arrangement of the WBS tend to be perpetuated in later decisions regarding contracting and management arrangements. As an example, one way of defining a work breakdown structure is to separate hardware, software, and supporting activities at the highest level. This approach presupposes that software will be developed independently of hardware, e.g., by a software house. Once such a decision is made in setting up the work breakdown structure, it tends to place a roadblock in the way of associating software closely with hardware, as may be desirable for a distributed software system. Similarly, those major hardware items identified at the top level of then WBS tend to become the contract end items in a hardware development phase.

Recognizing the potential of setting precedents in establishing a work breakdown structure, we set forth the criteria in table 18-1 as a precursor to preparing the WBS itself. These criteria are aimed at minimizing the program structural problems that could be introduced by an illogical WBS.

The SOC WBS that was used in the present study is shown in figure 18-1. This WBS formed the outline for the System Description document and was the basis for mass and cost analyses.

We elected to employ a three-part structure at the top level. The project management and integration element was created in recognition of the multi-element nature of the SOC hardware system. The SOC flight article will very likely consist of three to five separately-procured contract end items. A system-level integration activity will be essential. This activity may be carried out by NASA or by an integration contractor. Further, the integration contractor may be one of the end-item contractors or a separate integration contractor. The WBS allows these decisions to be deferred. Facilities are identified as a separate item in view of the NASA practice of separate C of F budgets.

The main part of the SOC WBS is the flight equipment. This is divided into six generic types of hardware and a services element. The six generic hardware types are subdivided further as appropriate. The groupings represent potentials for hardware commonality, e.g., HM-1 will be identical (or nearly identical) to HM-2 and the docking tunnel (DT) is very similar to the service module (SM) structure.

A separate services element was provided for those items closely associated with the flight hardware that may be procured independently from the SOC end items. Suits and EVA gear, for example, are expected to be common with the shuttle equipment and may be commonly procured. The division of SE&I responsibility between this element, the

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1) THE WBS SHOULD BE INDEPENDENT OF PROGRAM PHASE. EACH ELEMENT INCLUDES ACTIVITY AND COST BY PHASE.

2) RESPONSIBILITY FOR EACH ELEMENT SHOULD BE CLEARLY ASSIGNABLE.

3. THE WBS SHOULD PRESENT LOGICAL WORK PACKAGES AND INTERFACES.

4) THE WBS SHOULD FACILITATE DIRECT MANAGEMENT CONTROL.

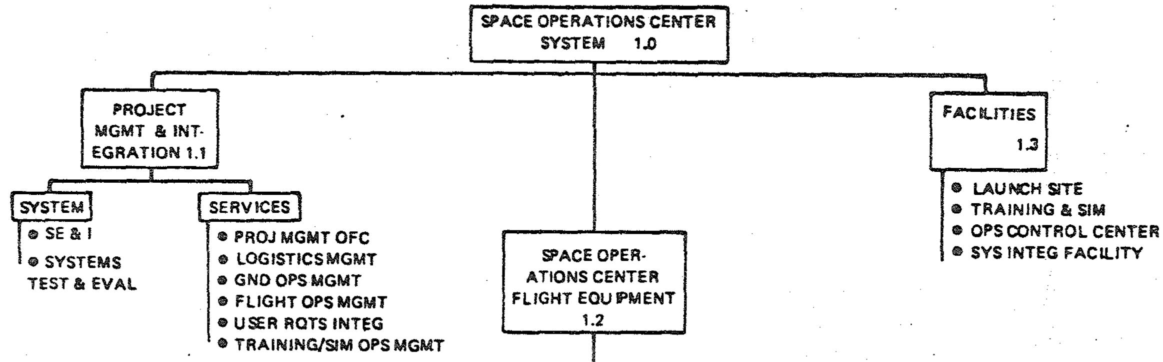
5) THE WBS SHOULD NOT INHIBIT FREEDOM OF CONTRACTING OPTIONS.

6) THE WBS SHOULD ENABLE STRAIGHTFORWARD COST MODELING.

7) THE WBS SHOULD ALLOW DIRECT DERIVATION OF SOC USER CHARGES.

8) THE WBS SHOULD BE A SUITABLE OUTLINE FOR REQUIREMENTS SPECIFICATION, SYSTEM DESCRIPTIONS, AND MASS AND COST ESTIMATES.

Table 18-1. SOC Work Breakdown Structure Criteria



GENERIC SUBSYS. HARDWARE & SOFTWARE	HABITAT MODULE 1.2.1		SERVICE MODULE			GENERAL PURPOSE SUPPORT EQUIPMENT			CONSTRUCTION SUPPORT EQUIPMENT			TRANSPORTATION SUPPORT EQUIPMENT				RESUPPLY & LOGISTICS SUPPORT SYSTEMS		
	HM-1,	HM-2	SM-1,	SM-2,	DT	MOB/ ACCESS	CP	MRWS	TT&F			HANG-AR	DOLLY	PROP SYS	SPARES	LM	TKR	
1.2.X.X.1 STRUCTURES	x	x	x	x	x	x	x	x	x			x	x	x		x	x	
2 MECHANISMS	x	x	x	x	x	x	x	x	x			x	x	x		x	x	
3 THERMAL CONTROL	x	x	x	x		x	x	x	x			x	x	x		x	x	
4 PRIMARY PROPULSION																		
5 AC PROPULSION			x	x														
6 ØRDNANCE	x	x	x	x	x	x	x	x									x	
7 ELEC POWER	x	x	x	x	x	x	x	x	x			x	x	x		x	x	
8 GN&C	x	x	x	x														
9 TRACKING & COMMUNIC.	x	x	x	x				x					x					
10 DATA MGMT & FLTS/W	x	x	x	x	x	x	x	x	x			x	x	x		x	x	
11 INSTRUMENTATION	x	x	x	x	x	x	x	x	x			x	x	x		x	x	
12 CREW ACCOMM.	x	x	x	x	x			x								x		
13 EC/LSS	x	x	x	x				x										
14 ASE, GSE, & STE	x	x	x	x	x	x	x	x	x			x	x	x		x	x	
15 ASSY & C/O	x	x	x	x	x	x	x	x	x			x	x	x		x	x	
16 SYSTEMS E&I	x	x	x	x	x	x	x	x	x			x	x	x		x	x	
17 MISSION EQUIPMENT	x	x	x	x			x	x							x	x		

SERVICES

- SE & I
- LOGISTICS
- OPS SUPPORT
- CREW
- EQUIPMENT
  - SUITS
  - EVA GEAR
  - PERS. EFFECTS
  - UTENSILS & TOOLS
  - ETC.

Figure 18-1. SOC Work Breakdown Structure (WBS) program-level element, and end-item SE&I can be deferred.

A common subsystems listing is applied to all hardware end items. The presence of a particular subsystem in a particular end item is indicated by a small x in the figure.

The other main dimension of the program structure is its time phasing. This is discussed below in sections 18.3 and 18.4.

18.3 SCHEDULES AND SCHEDULE ANALYSES

Schedules for SOC development were laid out using analogous experience with programs of similar size and complexity. Certain assumptions are implicit in the schedules:

(1) Significant technology advancements will be carried at least to the proof-of-concept stage by technology advancement activities prior to initiation of Phase C/D for SOC. If the technology advancement is critical, a full technology demonstration may be required.

(2) Accordingly, program delays to solve technology immaturity problems will not be encountered.

(3) Shuttle launch service will be available on a timely basis for SOC buildup; further, the SOC buildup will not be constrained by availability of facilities at KSC.

(4) End item fabrication and test activities are phased so that one set of tooling for each end item type, and one test crew, can accomplish the required fabrication and testing.

The schedule analyses keyed on the fabrication, test, and integration schedules incorporating assumptions (3) and (4). Two alternative schedules are shown in figures 18-2 and 18-3. The first schedule illustrates direct buildup to the reference SOC configuration; the second shows a gap in the buildup, with interim operation of an "initial SOC". The gap offers one option for reducing peak funding, but is not necessary from an assembly and test flow standpoint.

Because the flight SOC will be finally assembled in space by berthing modules together using the shuttle, it was seen as very important to validate, both mechanically and functionally, the berthing interfaces on the ground before launch. Subsystems such as electrical power, EC/LS, communications, and data management interface through these berthing ports. This need led to the concept of a ground test vehicle (GTV). The GTV is comprised of one service module, one habitat module, one logistics module, and a docking tunnel interface simulator. All subsystems in the GTV will be flight or flight prototype hardware.

The GTV will initially serve in an integration role to prove out the proper operation of the subsystems that interface through the berthing ports, and will later serve to validate flight hardware interfaces at KSC before each flight article is launched. Finally, after the flight system is fully built up in orbit, the GTV will be returned to JSC to serve as a "hangar queen" for simulation, training, and checkout of procedures, subsystem updates, and software changes before these are implemented in the flight system.

A high-level program schedule, based on the detailed schedules referenced above, is shown in figure 18-4. This high-level schedule includes the Phase B study activity and presumes a new start in FY85.

18.4 SOC BUILDUP OPTIONS AND FUNDING PROFILES

Several alternative approaches to buildup and evolution of the SOC were

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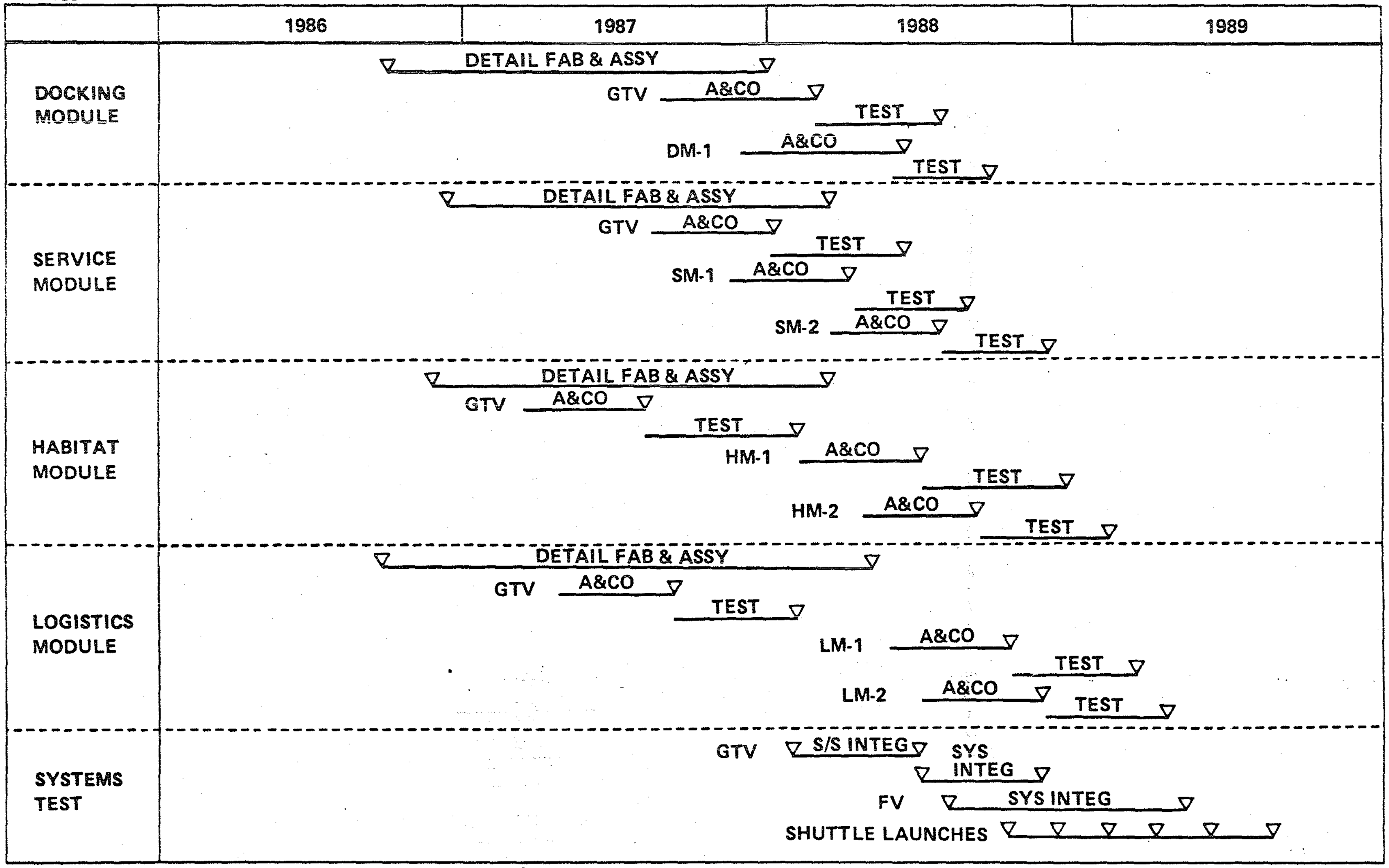


Figure 18-2. SOC Manufacturing and Test

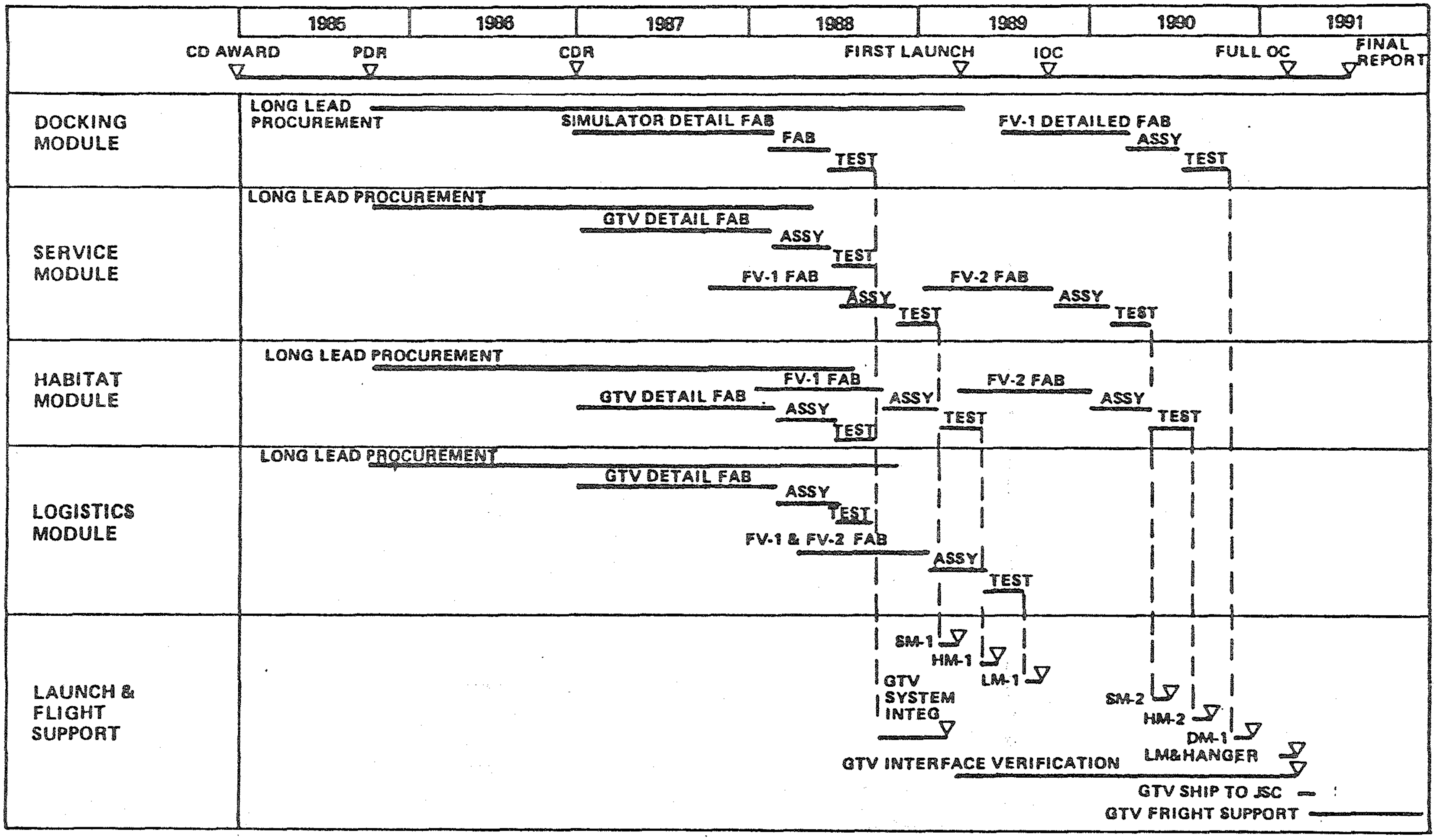
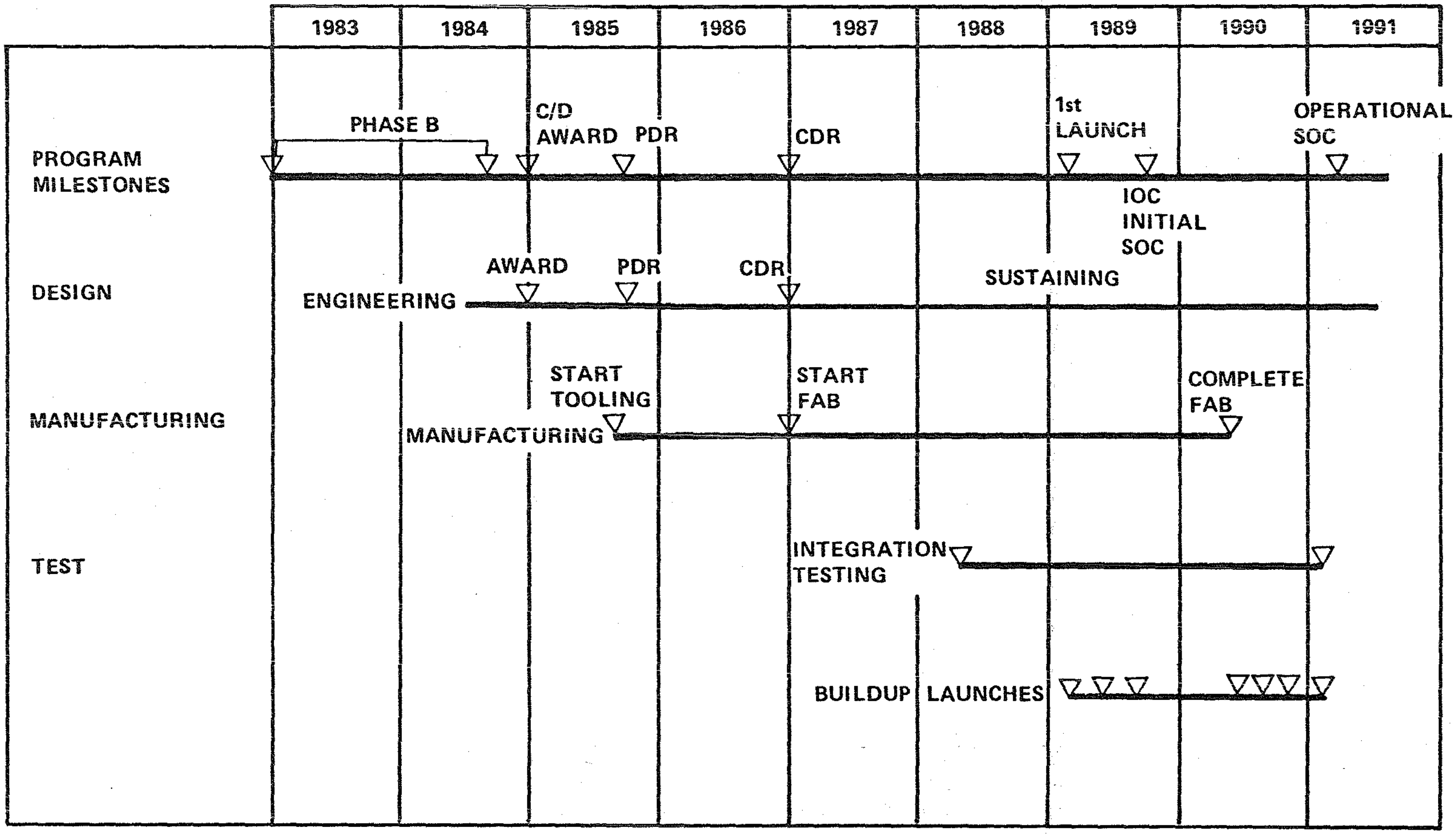


Figure 18-3. SOC Manufacturing & Test Evolutionary Buildup Schedule



NOTE: RESUPPLY LAUNCHES ON 3 – MONTH CENTERS AFTER IOC

Figure 18-4. SOC Program Summary Schedule explored. The evolution finally selected will depend on development of additional information on mission needs and applications. The options presented in figure 18-5 provide a reasonable menu of alternatives that can be adapted to future needs.

Although the figure shows a stepwise or evolutionary buildup of the SOC, the simplest approach is direct buildup of the core configuration, followed by addition of mission equipment and facilities. This approach, however, requires six to seven shuttle launches before the SOC can be permanently manned. Accordingly, the evolutionary alternatives were investigated.

The SOC can be occupied as soon as one habitat module, one service module, and a logistics module are in place; it is then capable of supporting a crew of four. This configuration is identified in figure 18-5 as the "initial SOC". Safety rules are violated by this configuration. There is no backup to the habitat module for crew life support, and only one path exists for escape from either module.

Modifications to the service module can qualify it as a backup habitable volume, reducing but not eliminating safety concerns for the initial SOC. These modifications consist mainly of added environmental control and life support equipment, with some communications equipment and crew supplies rearrangements, as summarized in figure 18-6 and table 18-2. These scars need not be added to the second service module.

The desire to occupy the SOC after only three launches led to a decision to delete the internal airlock from the habitat module. In the operational configuration (two habitat modules), the internal location is the most convenient from an operational viewpoint. However, in the initial configuration, if the crew is isolated in the service module, they have no access to an airlock for egress. Use of external airlocks, one each on the habitat and service modules, resolves this issue and

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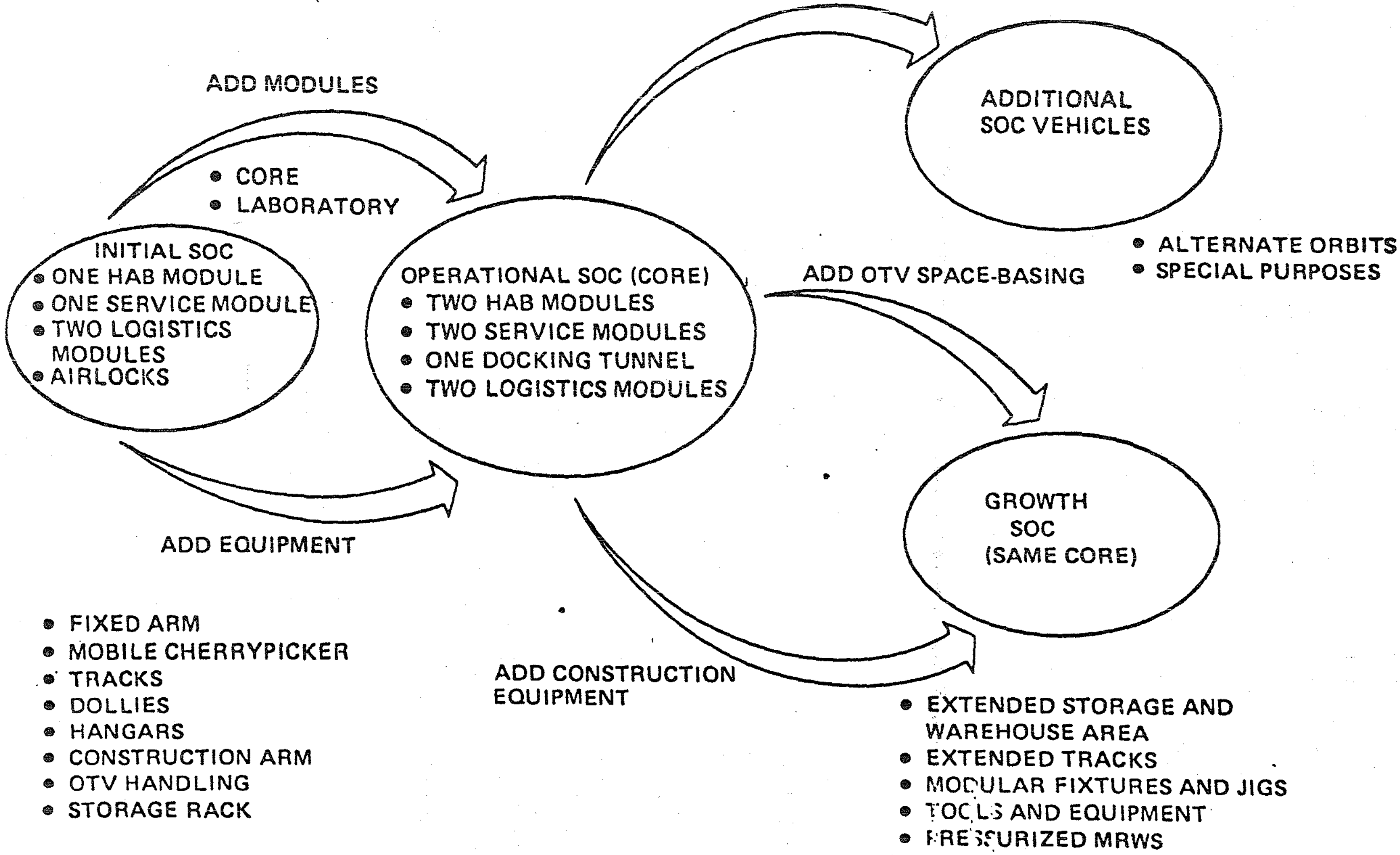
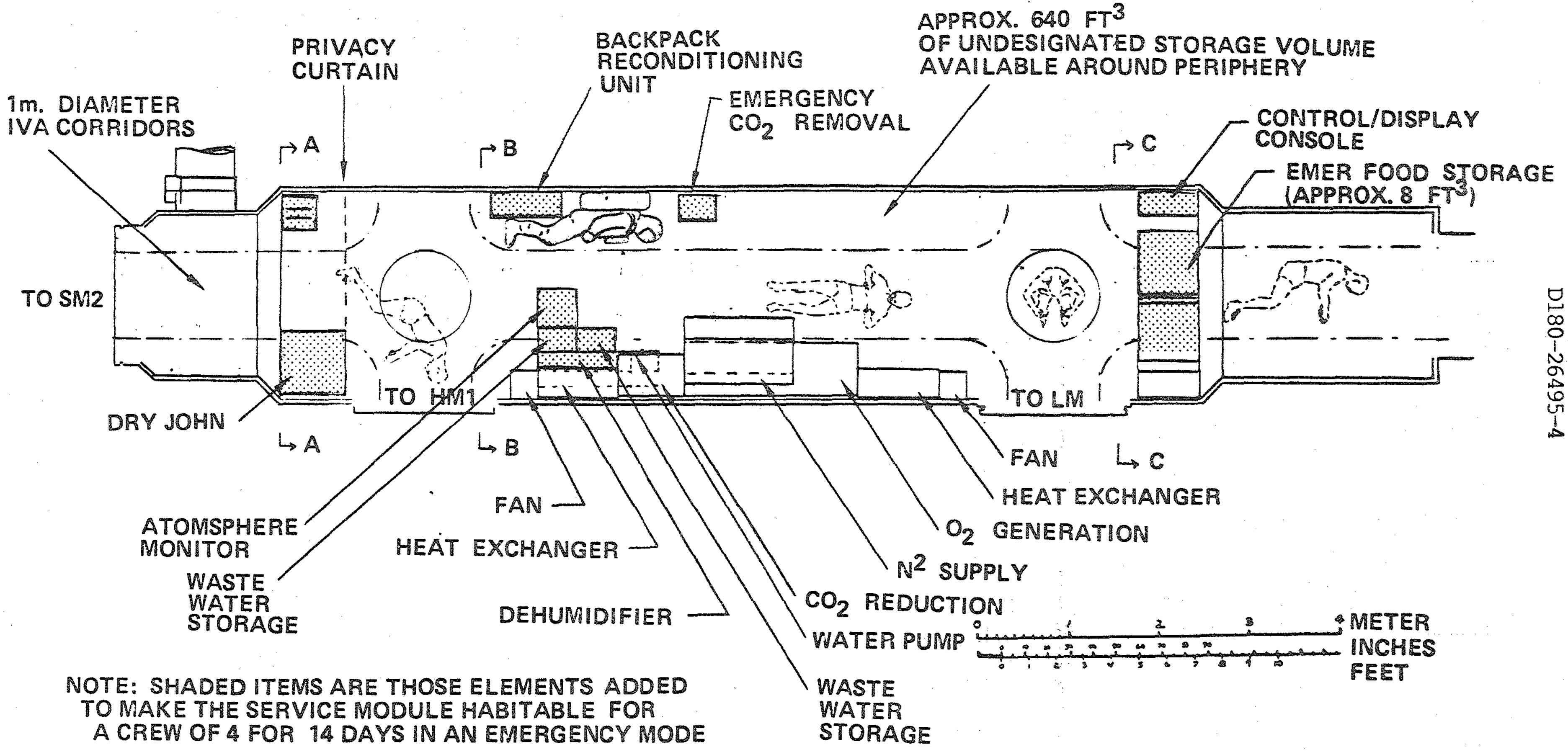


Figure 18-5. SOC Development Path Options



NOTE: SHADED ITEMS ARE THOSE ELEMENTS ADDED TO MAKE THE SERVICE MODULE HABITABLE FOR A CREW OF 4 FOR 14 DAYS IN AN EMERGENCY MODE

Figure 18-6. Service Module Interior Arrangement for the Initial SOC Configuration

AUXILIARY RCS BOOM

MANIPULATOR (?)

SERVICE MODULE NO. 1 EC/LSS ADDITIONS
- DEHUMIDIFIER
- 2 WASTE WATER STORAGE TANKS
- ATMOSPHERE MONITOR
- COLD WATER SUPPLY
- HOT WATER SUPPLY
- WATER PUMP
- DRY JOHN
- BACKPACK RECONDITIONING UNIT
- EMERGENCY CO_2 REMOVAL
- CONTROL/DISPLAY CONSOLE
- 8 FT^3 FOOD STOWAGE

HABITAT MODULE NO. 1 EC/LSS ADDITIONS
- CLOTHES WASHER/DRYER
- POWER JUMPER CABLES
- VOICE COMM ANTENNA

*ELEMENTS ADDED TO MAKE THE INITIAL SOC FEASIBLE BUT WHICH WILL HAVE LITTLE OR NO UTILITY IN THE OPERATIONAL AND GROWTH SOC's.

*Table 18-2. Initial SOC Scars**

eases concerns regarding habitat module mass and center of gravity.

The initial SOC configuration presents additional configuration issues relating to solar array configuration and drag asymmetry. These were discussed in section 2.3 of this report.

The buildup sequence initially conceived was one in which the two service modules were joined in orbit before any other elements were emplaced. We found, however, that the shuttle RMS reach is insufficient to assemble the SM's when the shuttle is docked to the end of one of them. A second buildup sequence was then visualized, still striving to achieve drag symmetry as soon as possible, and using the docking tunnel in a temporary berthed position on the first service module as an assembly aid. This concept permitted assembly, but resulted either in excess requirements for assembly tooling and test provisions in final assembly and test of the hardware, or in a stretched-out buildup schedule. Since the docking tunnel is a modified service module structure, three SM end items were to be fabricated and tested, followed by two HM end items. These programmatic considerations led to a final buildup sequence that proceeds through the "initial SOC" configuration.

Funding profiles were investigated for three program options. These were (1) direct buildup to the operational SOC configuration; (2) buildup to the initial SOC configuration with a two-year gap before resuming buildup to the operational SOC; (3) direct buildup to the operational SOC core with deferral of mission equipment such as the mobile crane and OTV hangars.

Funding profiles were created by using spread function methods to spread the lump-sum cost estimates from paramentric cost estimating. These spread functions provide representative funding profiles for program elements and include parameters to adjust the magnitude and timing of the peak funding for each element.

The funding profiles for the three options discussed are shown in figures 18-7 through 18-9. Each program element was subdivided into its parts of engineering and development, manufacturing, test, and buildup support. The last of these was included to represent the need for sustaining engineering and manufacturing activity during the period after delivery of hardware end items, when SOC buildup operations are taking place. Program-wide activities such as software and management and integration were then added.

The handling of software represents an artifact of the cost estimating method. In the actual SOC program, it is expected that software for operating each subsystem would be procured as a part of that subsystem. The cost estimating for the present study, however, treated software separately because of estimating methodology considerations.

Funding profile evaluations indicated that a stretchout of the nine-month buildup gap does not reduce peak funding very much because of stretchout cost penalties in the HM and SM programs. We assumed that a two-year gap between the initial SOC and the resumption of buildup to the operational SOC would incur a ten percent cost penalty for the major end items. This is approximately borne out by history.

Deferring the mission equipment (cherrypickers, hangars, etc.) is more effective, but delays full mission capability. Mission equipment, if deferred, can be started late and no penalty for stretchout was projected.

18.5 RECOMMENDED TECHNOLOGY

A key part of programmatic considerations is the selection of technology levels for implementation. This represents a tradeoff among cost, risk, schedule, and the desire to apply enough technolgy advancement that the

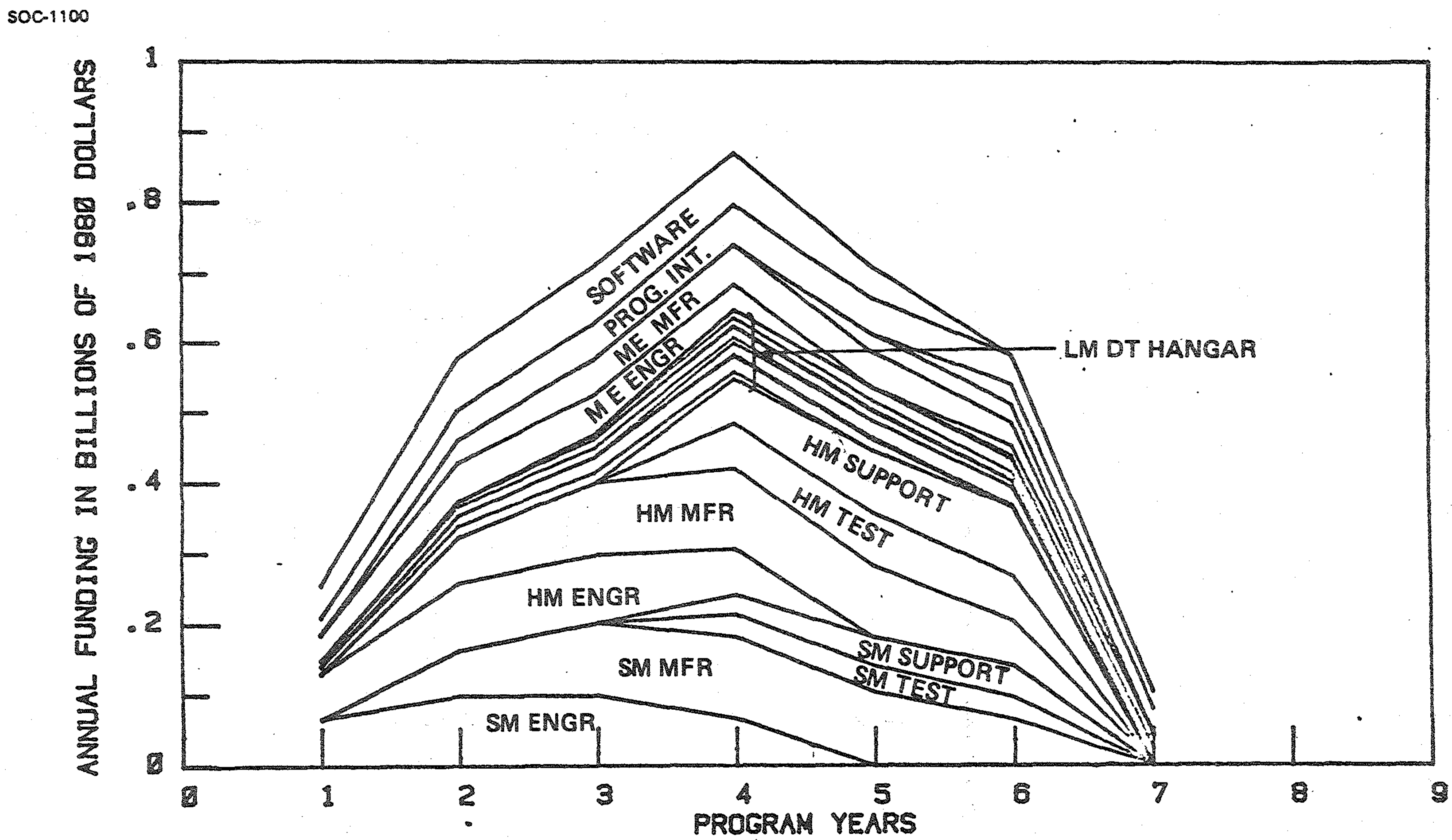


Figure 18-7. SOC Baseline Program Funding

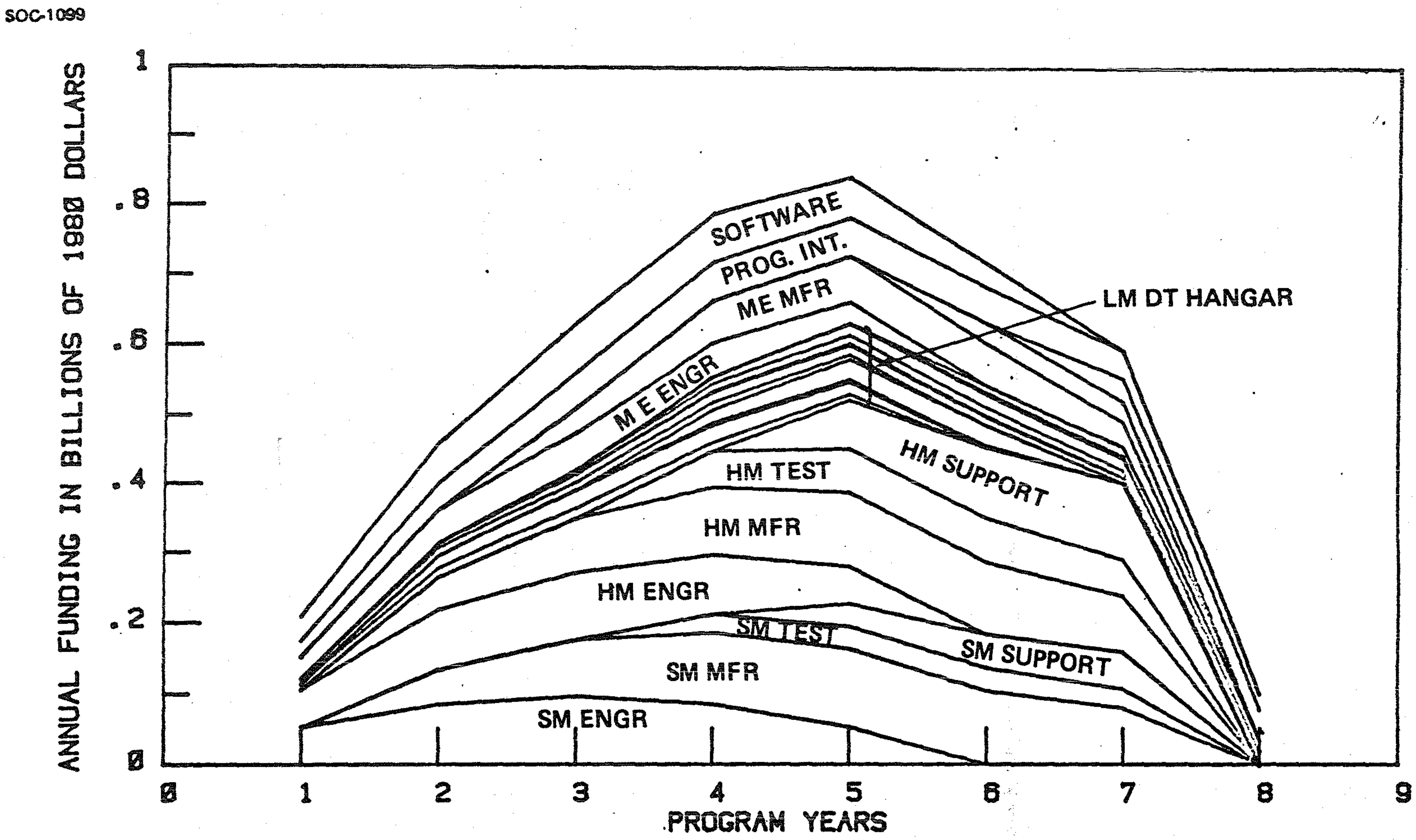


Figure 18-8. Modified SOC Program Funding (Deferred Buildup From Initial to Operational SOC)

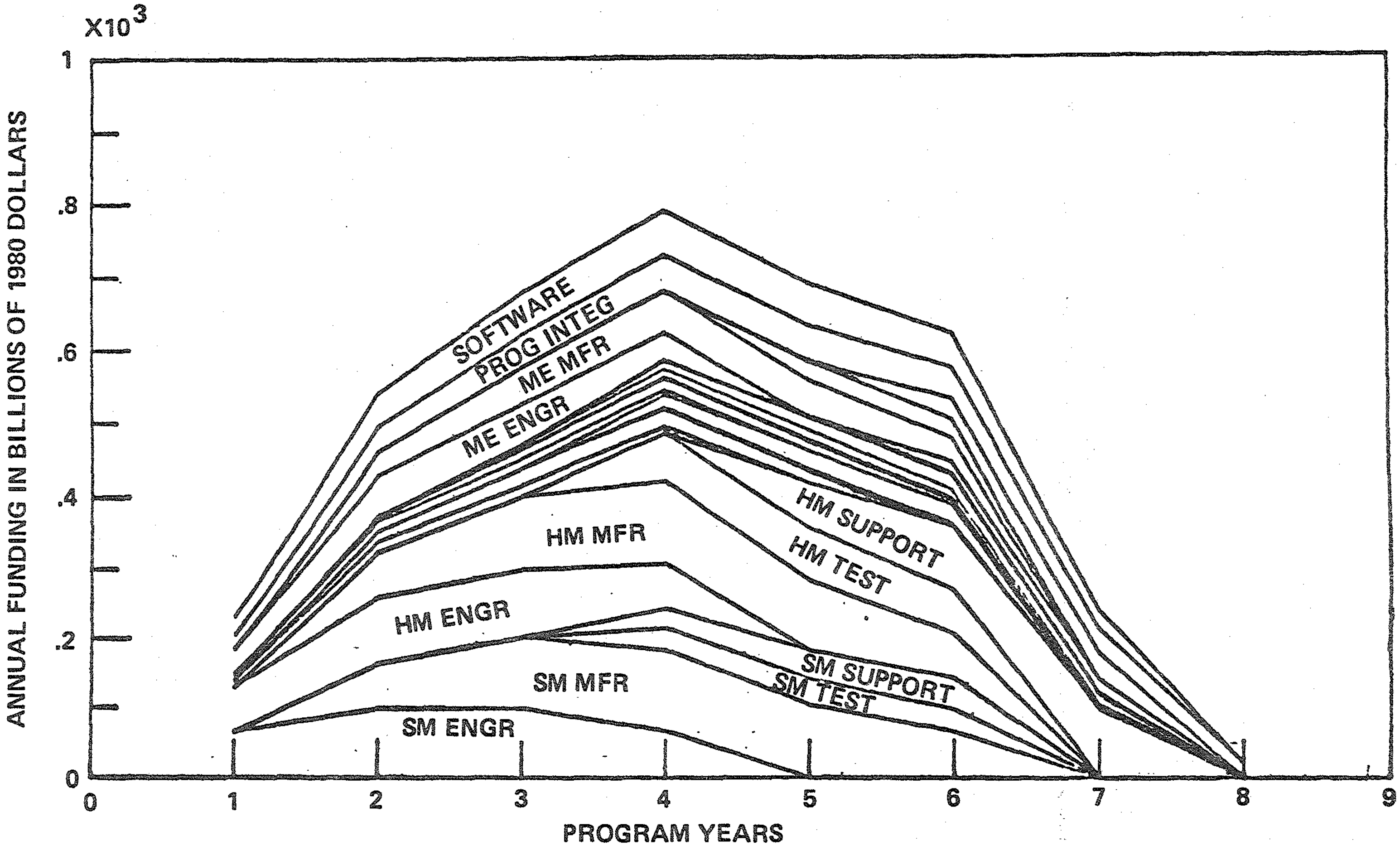


Figure 18-9. SOC Program Funding (Alternate 2)

planned system will not be obsolete when operational. Conscious technology selections were made for all of the SOC subsystems. Some of these were discussed in section 2 of this report. Table 18-3 presents a summary of the technology recommendations developed as a part of the present Phase A study. These recommendations also were used as a basis for technology advancement recommendations presented in the SOC Technology Assessment and Advancement Plan prepared as a part of this study.

Certain technolgy advancement needs carry with them significant schedule implications. Most important are the areas for which life testing of flight prototype hardware may be needed as a part of the development program. Two such areas for SOC are the EC/LS systems and the electrical power system. In both areas, technology advancements are proposed, the proper operation of the hardware is critical to crew safety, and the required hardware life is challenging. These areas merit special consideration in developing plans to proceed with technology advancement so as to accomplish the life tests in a timely manner.

Another area meriting special attention is software. Our estimates of the desired schedule for SOC software development showed that it will require longer than the hardware. The software schedule can be accelerated, but only at higher cost and greater risk. The problem can be alleviated by carrying out a data management architecture technology program and by initiating software design and development as a part of the SOC Phase B studies.

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SYSTEM OR SUBSYSTEM	RECOMMENDED LEVEL	STATUS	RATIONALE FOR SELECTION
PRIMARY STRUCTURE	WELDED ALUMINUM (NEW DESIGN)	DEVELOPED	- BECAUSE OF COLLISIONS & FIRE CRITERIA, NO SIGNIFICANT BENEFIT FROM ALTERNATIVES
BOOM & TRACK STRUCTURES	GR-EP	DEVELOPED; SOME CONCERN ABOUT LIFE IN SPACE	- NO SUITABLE EXISTING DESIGN STIFFNESS & MASS ADVANTAGES
SECONDARY STRUCTURES	ALUMINUM OR GR-AL	GR-AL IS IN DEVELOPMENT. NOT MUCH PROPERTIES DATA	- GR-AL OFFERS WEIGHT & STIFFNESS ADVANTAGES - FIRE CRITERIA PRECLUDE GR-EP - ALUMINUM IS ADEQUATE
THERMAL CONTROL - HAB MODULE COATING	REFRESHABLE SELECTIVE COATING	RESEARCH	- SELECTIVE COATINGS ESSENTIAL DUE TO SUN ANGLES - DEGRADATION IS A PROBLEM FOR 10-YEAR LIFE - ALTERNATIVE IS LONG-LIFE COATING

Table 18-3. Recommended Technology Levels

SYSTEM OR SUBSYSTEM	RECOMMENDED LEVEL	STATUS	RATIONALE FOR SELECTION
SERVICE MODULE RADIATORS	CONSTRUCTABLE HEATPIPE	IN TECHNOLOGY DEVELOPMENT	- NOT ENOUGH AREA ON SM EXTERIOR - PACKAGING ADVANTAGES - READILY REPAIRABLE
FLUID LOOPS	SHUTTLE	DEVELOPED; UPGRADE DESIGN TO ENHANCE LIFE & PROVIDE FOR ONBOARD MAINTEN-ANCE	- SHUTTLE TECHNOLOGY IS ADEQUATE
PROPULSION	HYDRAZINE MONO-PROPELLANT EVALUATE ELECTRICAL HEAT AUGMENTATION (RAISE ISP TO 300)	DEVELOPED; SMALL HEAT-AUGMENTED THRUSTERS IN DEVELOPMENT	- LOW CONTAMINATION; SIMPLE; RELIABLE - HEATED THRUSTERS APPROACH BI-PROP ISP - CRYO TOO MUCH RISK FOR THE PAYOFF
SOLAR ARRAY	PEP-TYPE WITH LARGE AREA CELLS	IN TECHNOLOGY DEVELOPMENT	- PACKAGING ADVANTAGES - LOW WEIGHT - LARGE AREA CELLS OFFER COST ADVAN-TAGES

Table 18-3. Recommended Technology Levels (Cont'd)

SYSTEM OR SUBSYSTEM	RECOMMENDED LEVEL	STATUS	RATIONALE FOR SELECTION
BATTERIES	NICKEL-HYDROGEN	IN TECHNOLOGY DEVELOPMENT; SMALL ONES FOR GEO SATELLITES NEARING FLIGHT READINESS	ABOUT HALF THE WEIGHT OF NICKEL-CADMIUM IN THIS APPLICATION. WEIGHT SAVINGS ENOUGH TO REDUCE NUMBER OF LAUNCHES
ENVIRONMENTAL CONTROL & LIFE SUPPORT	RECYCLED WATER AND OXYGEN	IN TECHNOLOGY DEVELOPMENT; INTEGRATED TESTING OF EXPERIMENTAL HARDWARE	GREAT SAVINGS IN RESUPPLY COST
CREW SYSTEMS (FOOD, WASTE, HYGIENE)	NEW DESIGN EXCEPT FOR SHUTTLE TOILET	SHUTTLE TOILET DEVELOPED OTHER ITEMS CONCEPTED	NO SUITABLE AVAILABLE EQUIPEMENT EXCEPT SHUTTLE TOILET
EVA EQUIPMENT	SHUTTLE WITH SLIGHTLY HIGHER PRESSURE SUIT AND ICE PACK THERMAL CONTROL	SHUTTLE EQUIPMENT DEVELOPED MODIFICATIONS CONCEPTUALLY DESIGNED	ELIMINATE EVA PREBREATHE MINIMIZE WATER RESUPPLY SHUTTLE TECHNOLOGY IS ADEQUATE WITH THESE IMPROVEMENTS

Table 18-3. Recommended Technology Levels (Cont'd)

SYSTEM OR SUBSYSTEM	RECOMMENDED LEVEL	STATUS	RATIONALE FOR SELECTION
FLIGHT CONTROL SENSORS & ACTUATION	CURRENT TECHNOLOGY WITH DESIGN MODS FOR IN-SPACE MAINTENANCE	TECHNOLOGY IS DEVELOPED	- CURRENT TECHNOLOGY IS ADEQUATE - IN-SPACE MAINTENANCE REQUIRED FOR 10-YEAR LIFE
FLIGHT CONTROL COMPUTATION	32-BIT MICROPROCESSORS AND ADAPTIVE CONTROL ALGORITHMS	- MICROS IN TECHNOLOGY DEVELOPMENT FOR COMMERCIAL & MILITARY APPLICATIONS - ALGORITHMS IN RESEARCH STAGE	- PROCESSING POWER NEEDED - SOFTWARE COST SAVINGS - ADAPTIVE CONTROL ESSENTIAL FOR VARIABLE CONFIGURATION
COMMUNICATIONS	- CONVENTIONAL S-BAND, K-BAND & UHF, PLUS	- DEVELOPED	- NECESSARY TO TALK TO EXISTING VEHICLES
	- MM-WAVE FOR RELAY & RADAR	- IN TECHNOLOGY DEVELOPMENT	- REDUCED RFI
	- POTENTIAL USE OF LF FOR EVA	- DEVELOPED	- ENHANCED COVERAGE

Table 18-3. Recommended Technology Levels (Cont'd)

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SYSTEM OR SUBSYSTEM	RECOMMENDED LEVEL	STATUS	RATIONALE FOR SELECTION
DATA MANAGEMENT			
- PROCESSORS	32-BIT MICROPROCESSORS	IN DEVELOPMENT FOR COMMERCIAL & MILITARY APPLICATIONS	- PROCESSING POWER - SOFTWARE COST SAVINGS
- SOFTWARE	ADA (NEW HIGH-ORDER LANGUAGE)	COMPILERS UNDER DEVELOPMENT BY DOD	- WILL BECOME A STANDARD - RICH & POWERFUL LANGUAGE
- ARCHITECTURE	DISTRIBUTED, HIERARCHICAL	- MANY DISTRIBUTED SYSTEMS EXIST - PROBABLY NEED NEW BUS PROTOCOL	- NECESSARY BECAUSE OF SOC MODULARITY - COST SAVINGS IN SOFTWARE AND SYSTEM INTEGRATION
- DATA BUS	FIBER OPTICS	PARTIALLY DEVELOPED	- HIGH SPEED - EMI IMMUNITY

Table 18-3. Recommended Technology Levels (Cont'd)

REFERENCES

Ref. No.	NASA DOCUMENTS
NASA-1	Request for Proposal 9-B0281-A12-0-33P Space Operations Center System Analysis.
NASA-2	Space Operations Center - A Concept Analysis, Vol. I -Executive Summary, JSC-16277, Nov. 29, 1979.
NASA-3	Space Operations Center - A Concept Analysis, Vol. II -Technical Report, JSC-16277, Nov. 29, 1979.
NASA-4	Space Operations Center - A Concept Analysis, Conference Presentation Slides, Nov. 29-30, 1979.
NASA-5	Minutes of the SOC System Analysis Study Orientation Briefing, EW4-80-80, April 10, 1980.
NASA-6	Space Operations Center System Analysis Study, Requirements for a Space Operations Center, JSC-16244.
NASA-7	Updated Information for SOC Requirements Document (JSC-16244), EW4-80-70, March 20, 1980.
NASA-8	Cost Specification for Space Operations Center Study, LR-1S-1561, Feb. 27, 1980.

Ref. No.	BOEING DOCUMENTS

Boeing-1 SOC System Analysis Technical and Management Proposal, Vol. 1, D180- 25834-1, Feb. 18, 1980.

Boeing-2 SOC System Analysis Business Proposal, Vol. 2, D180-25834-2, Feb. 15, 1980.

Boeing-3 SOC System Analysis Study Plan, D180-25941-1, May 1980.

Boeing-4 SOC Technology Assessment and Advancement Plan Study -Statement of Work, D180-25941-2, May 1980.

Boeing-5 SOC System Analysis Monthly Progress Report No. 1, June 19, 1980.

Boeing-6 SOC System Analysis Monthly Progress Report No. 2, July 19, 1980.

Boeing-7 SOC System Analysis Monthly Progress Report No. 4, Sept. 19, 1980.

Boeing-8 SOC System Analysis Monthly Progress Report No. 5, Oct. 17, 1980.

Boeing-9 SOC System Analysis Monthly Progress Report No. 7 & 8, Jan. 19, 1981.

Boeing-10 SOC System Analysis Monthly Progress Report No. 9, Feb. 19, 1981.

Boeing-11 SOC System Analysis Monthly Progress Report No. 10, April 19, 1981.

Boeing-12 SOC System Analysis Monthly Progress Report No. 11, May 19, 1981.

Boeing-13 SOC System Analysis, First Quarterly Briefing, D180-26127-1, Sept. 3, 1980.

Boeing-14 SOC System Analysis, Midterm Briefing, D180-26209-1, Dec. 2, 1981.

Ref. No. **BOEING DOCUMENTS**

Boeing-15 SOC System Analysis, Third Quarterly Briefing, D180-26433-1, March 31, 1981.

Boeing-16 Requirements for a Space Operations Center, D180-26135-1, Dec. 1980 (Draft).

Boeing-17 Space Operations Center System Analysis, Final Report, Volume I, Executive Summary, D180-26495-1, July 1981.

Boeing-18 Space Operations Center System Analysis, Final Report, Volume II, Requirements (NASA CR-160944), D180-26495-2, July 1, 1981.

Boeing-19 Space Operations Center System Analysis, Final Report, Volume III, SOC System Definition Report, D180-26495-3, July 1981.

Boeing-20 Space Operations Center System Analysis, Final Report, Volume IV, System Analysis Report, D180-26495-4, July 1981.

Boeing-21 Space Operations Center System Analysis, Final Report, Volume V, Data Book, D180-26495-5, July 1981.

Boeing-22 (Reserved)

Boeing-23 Space Operations Center Technology Identification Support Study, Final Report, D180-26495-7, July 1981

Boeing-24 Space Operations Center System Analysis, Final Report, Volume VI, Final Briefing, D180-26495-8, June 25, 1981

ROCKWELL DOCUMENTS

Rockwell-1 Space Operations Center - Shuttle Interaction Study Monthly Progress Report No. 1, August 1980.

Rockwell-2 Space Operations Center - Shuttle Interaction Study Monthly Progress Report No. 2, September 1980.

Rockwell-3 Space Operations Center - Shuttle Interaction Study Monthly Progress Report No. 3, October, 1980.

Rockwell-4 Space Operations Center - Shuttle Interaction Study Monthly Progress Report No. 4, November 1980.

Rockwell-5 Space Operations Center - Shuttle Interaction Study Monthly Progress Report No. 5, December 1980.

Rockwell-6 Space Operations Center - Shuttle Interaction Study Monthly Progress Report No. 6, January 1981.

Rockwell-7 Space Operations Center - Shuttle Interaction Study Monthly Progress Report No. 7, February 1981.

Rockwell-8 Space Operations Center - Shuttle Interaction Study Monthly Progress Report No. 8, March 1981.

Rockwell-9 Space Operations Center - Shuttle Interaction Study First Quarterly Review, Sept. 3, 1980.

Rockwell-10 Space Operations Center - Shuttle Interaction Study Midterm Review, Dec. 3, 1980.

Rockwell-11 Space Operations Center - Shuttle Interaction Study Final Review, March 31, 1980.

OTHER CONTRACTOR DOCUMENTS

MacDac-1 Space Platform Advanced Technology Study, Contract NAS9-16001, Midterm Review, MDCG8550, McDonnell Douglas, July 1, 1980.

MacDac-2 Space Platform Advanced Technology Study, Contract NAS9-16001, Final Review, MDCG9333, McDonnell Douglas, January 21, 1981.

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