NASA CR-172090

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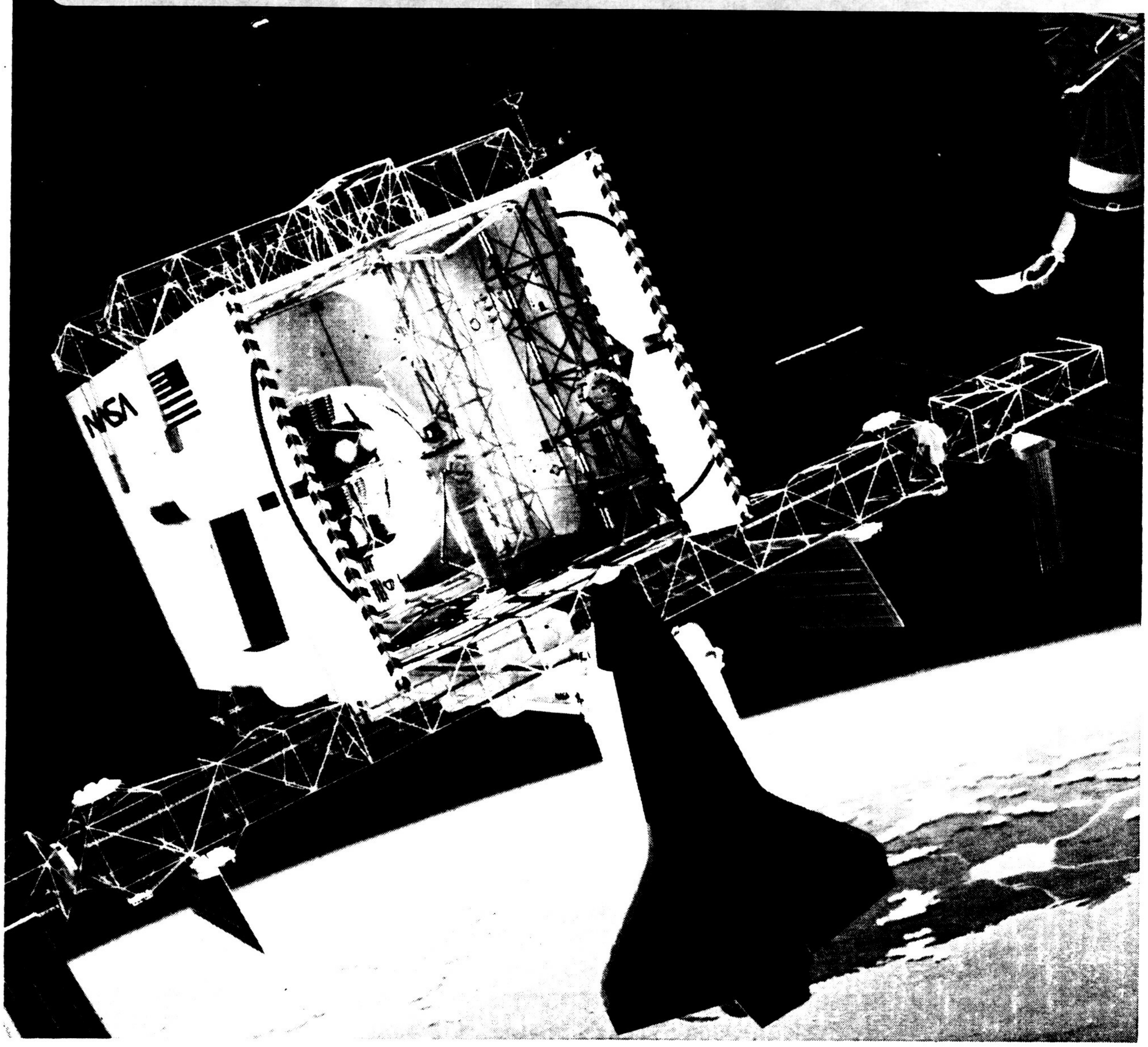
Transportation Node Space Station Conceptual Design



NASA Contract No. NAS 9-17878
Eagle Engineering, Inc.
Report No. 88-207
September 30, 1988

EAGLE

Transportation Node Space Station
Conceptual Design

September 30, 1988

National Aeronautics and Space Administration
Lyndon B. Johnson Space Center
Advanced Projects Office

Lunar Base Systems Study Tasks 2.3 and 2.4

Prepared by:
Eagle Engineering, Inc.
Houston, Texas

NASA Contract NAS9-17878
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Foreword

This report was prepared during March - September, 1988. Its purpose is to define a concept for a low Earth orbit space station to service a lunar surface base transportation system. In addition it examines the pros and cons of a lunar orbit transportation node.

Dr. John Alred was the NASA JSC technical monitor for the contract of which this study was a part. Ms. Jonette Stecklein was the NASA task manager for this study. Mr. Andy Petro provided valuable technical advice.

Mr. Bill Stump was the Eagle project manager for the contract of which this study was a part. Mr. Stump and Dr. Charles Simonds co-managed this particular study. Other Eagle participants included Mr. Mike Amoroso, Mr. Eric Christiansen, Mr. Bill Davidson, Mr. Mark Dowman, Mr. Jack Funk, Mr. John Euker, Mr. John Hirasaki, Mr. Tony Hodgson, Mr. Jeff Kline, Mr. Owen Morris, Mr. John Nagel, Mr. Sam Nassiff, Mr. John Philips, Mr. Paul Phillips, Mr. Pat Rawlings, and Mr. R.G. Ruiz.

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List of Acronyms

ATCS	Active Thermal Control System
ALS	Advanced Launched System
AR	Air Revitalization
A/L	Airlock
ACS	Atmosphere Control and Supply
ARS	Atmosphere Revitalization System
ACA	Attitude Control Assemblies
CG	Center of Gravity
CCZ	Command and Control Zone
C&T	Communication and Tracking
CAD	Computer Aided Design
CMG	Control Moment Gyro
CETA	Crew and Equipment Translation Aid
CHeCS	Crew Health Care System
CETF	Critical Evaluation Task Force
DMS	Data Management System
DGCMG	Double Gimbal Control Moment Gyro
EPDS	Electrical Power Distribution System
ECLSS	Environmental Control and Life Support System
EVAS	Extra-Vehicular Activity System
EMU	Extra-vehicular Mobility Unit
EVA	Extra-Vehicular Activity
FDS	File Detection and Suppression
FTS	Flight Teleoperated Servicer
GN&C	Guidance, Navigation & Control
HAB	Habitation
HLV	Heavy Lift Vehicle
ISA	Inertial Sensor Assemblies
IOC	Initial Operational Capability
ITA	Integrated Truss Assembly
IVA	Intra-Vehicular Activity
LAB	Laboratory
LaRC	Langley Research Center
LRU	Line Replaceable Unit
LAD	Liquid Acquisition Device
LH_2	Liquid Hydrogen
LO_2	Liquid Oxygen
LOG	Logistics
LEO	Low Earth Orbit
LLO	Low Lunar Orbit
LOI	Lunar Orbit Insertion
MMU	Manned Maneuvering Unit
MSC	Manned Spacecraft Center
MH	Man-hours
MLI	Multi-Layer Insulation

NASA	National Aeronautics and Space Administration
OMV	Orbital Maneuvering Vehicle
ORU	Orbital Replaceable Unit
OTV	Orbital Transfer Vehicle
PEPS	Payload Experiment Processing System
RCM	Reaction Control Module
RCS	Reaction Control System
RMS	Remote Manipulator System
RFP	Request for Proposal
SSCE	Space Station Core Equipment
STN	Space Transportation Node
SOI	Sphere of Influence
SDP	Standard Data Processors
THC	Temperature and Humidity Control
TCS	Thermal Control System
TEI	Trans-Earth Injection
TLI	Trans-Lunar Insertion
WM	Waste Management
WRM	Water Recovery Management
WP-1	Work Package 1
WP-2	Work Package 2

1.0 Executive Summary

A low Earth orbit space station was conceptually designed to support a reusable transportation system for lunar flights. Figure 1.0-1 illustrates the overall concept showing a departing stack and arriving heavy lift tanker. Figure 1.0-2 shows a computer-generated version of the Station alone. Figure 1.0-3 shows a three-view with 75 kw of solar power. This Space Transportation Node (STN) station is oriented exclusively toward the assembly, refurbishment, maintenance, propellant loading, checkout, and repeated reuse and launch of cargo and piloted vehicles going to the lunar surface.

Up to eight flights per year to the lunar surface are to be supported. The transportation system consists of a large single-stage reusable OTV that delivers a single-stage reusable lander/launcher to low lunar orbit (LLO). Figure 1.0-4 shows the OTV and landers stacked and separate. The OTV waits in orbit for the lander to return. Both then aerobrake back to the low Earth orbit (LEO) station using separate aerobrakes. Both vehicles are reloaded with propellant and refurbished at the LEO station. Though a specific transportation system is used, a range of different transportation system options can be accommodated. The emphasis however, is on reusability for space-maintainable vehicles.

The STN supports two stacks, each consisting of an OTV, lunar lander/launcher, and a payload. The single stage reusable lander/launcher delivers 25 m tons one way to the lunar surface or a 6 m ton crew capsule round trip from low lunar orbit (LLO). A stack departing LEO weighs on the order of 200 m tons, including 158 m tons of cryogenic propellant.

The dry weight of the STN, without propellants or OTVs and landers is approximately 400 metric tons. 182 m tons of cryogenic hydrogen and oxygen propellant is stored in four tanks. The storage uses liquid acquisition devices to acquire the propellant for transfer to the OTVs and landers. Passive thermal control is used and boil-off is used for orbital make-up propellant. With two stacks fully loaded with propellant, and the storage tanks full, the station has a maximum weight of approximately 1,000 m tons. A typical weight might be more in the range of 800 m tons. The STN was originally planned to keep the majority of the mass centrally located in the plane of the front hangar door.

75 kilowatts of continuous power is provided by a Phase 1 Space Station photo-voltaic system. Figure 1.0-3 shows this 75 kw system. 75 kw of heat is rejected via a Space Station thermal control system.

Many of the STN subsystems are derived from the Freedom Space Station design. In addition to the power system, two habitation modules, two airlocks, numerous nodes, truss structure, and other subsystems are Phase 1 Freedom Station designs.

The two stacks can be assembled or serviced in parallel. Each stack is docked to a rotating fixture that turns to allow 360° access to the entire stack from a manipulator running up and down the truss. The rotating fixture also allows pressurized access to the lunar crew module or cargo from the STN interior.

Habitation facilities for a permanent crew of 6 and a transient crew of 7 (13 total) are provided. Pressurized workshops and a workstation in the hangar are also supplied. Figure 1.0-5 shows the overall pressurized volume configuration.

The two major assumptions of the design are; 1) the fully reusable, space-maintainable OTV and lander, and 2) the high maximum flight rate (8/year). These assumptions require careful examination in future work.

The advantages and disadvantages of a low lunar orbit station were also examined as part of this effort. A LLO STN in lunar equatorial orbit would allow the OTV and landers to always deliver maximum payload which may be required in some lunar oxygen schemes to achieve reasonable mass efficiency. These scenarios generally assume a lunar based and maintained reusable lander/launcher however and are probably not practical until well after a permanent lunar base is established. As the inclination of the lunar orbit goes up, the number of opportunities to arrive and depart the Moon without excessive delta-V penalties goes from three per month to one per month. For these higher inclination lunar orbits, an LLO STN adds another constraint that further complicates the window problem. Delta V plots were generated that indicated inclinations of 10° and less can be made essentially equal to equatorial in their accessibility (three arrival/dep. opportunities per month) for a 15% penalty in LEO stack mass.

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Figure 1.0-1, LEO Transportation Node Space Station for Lunar Base Support

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Figure 1.0-2, CAD Generated LEO Transportation Node Space Station

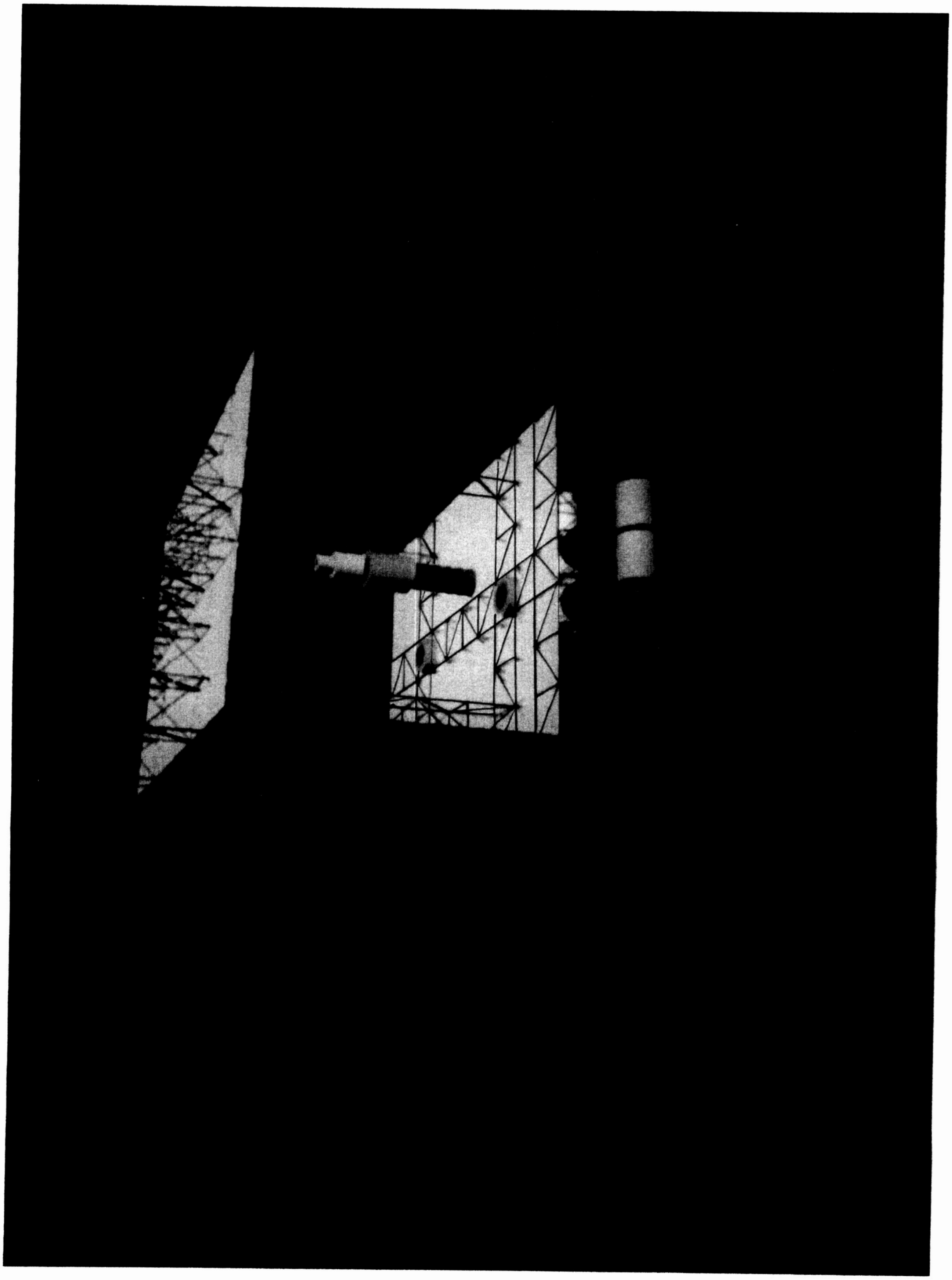


Figure 1.0-3, Three-View of Transp. Node Station

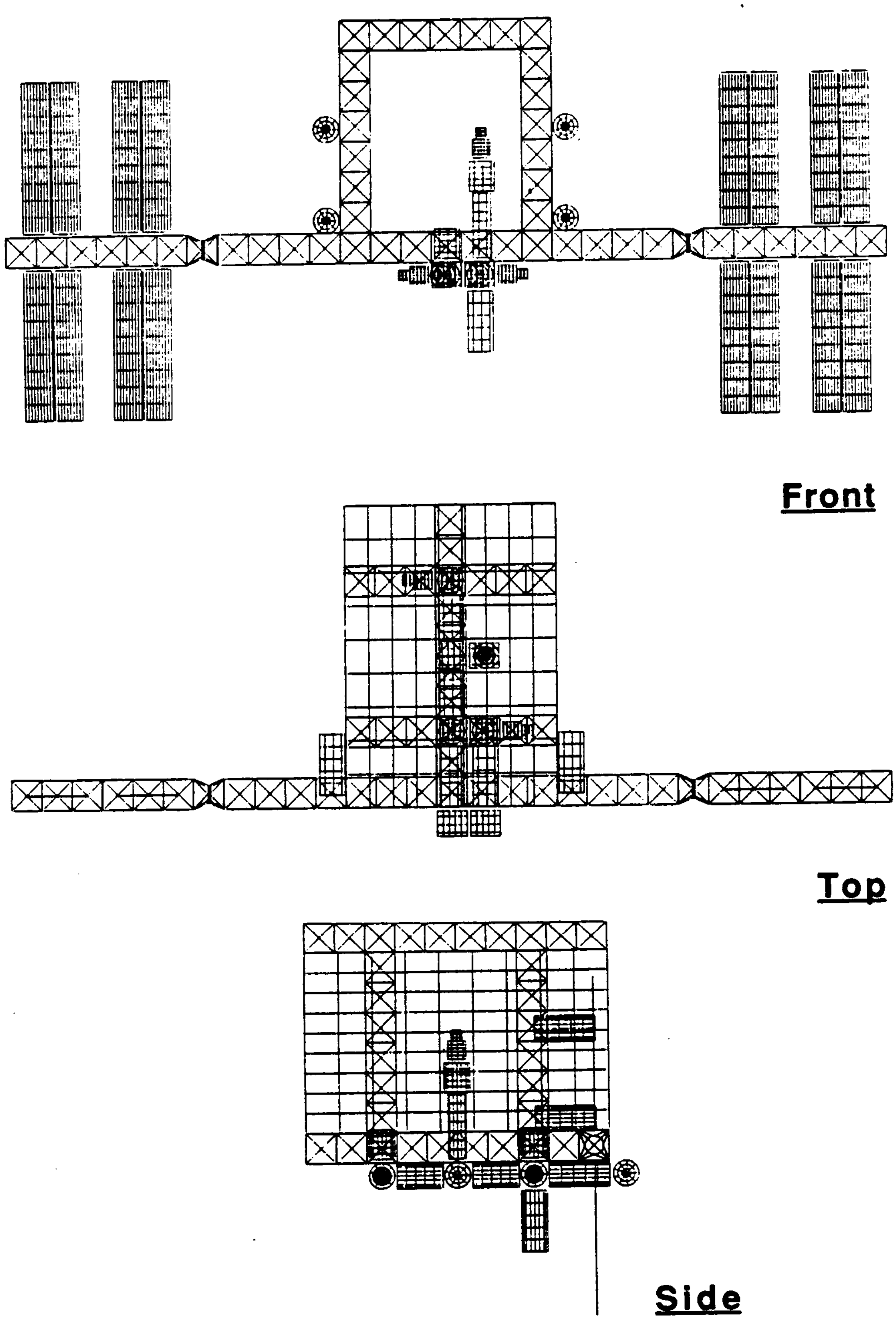


Figure 1.0-4, OTV and Lander

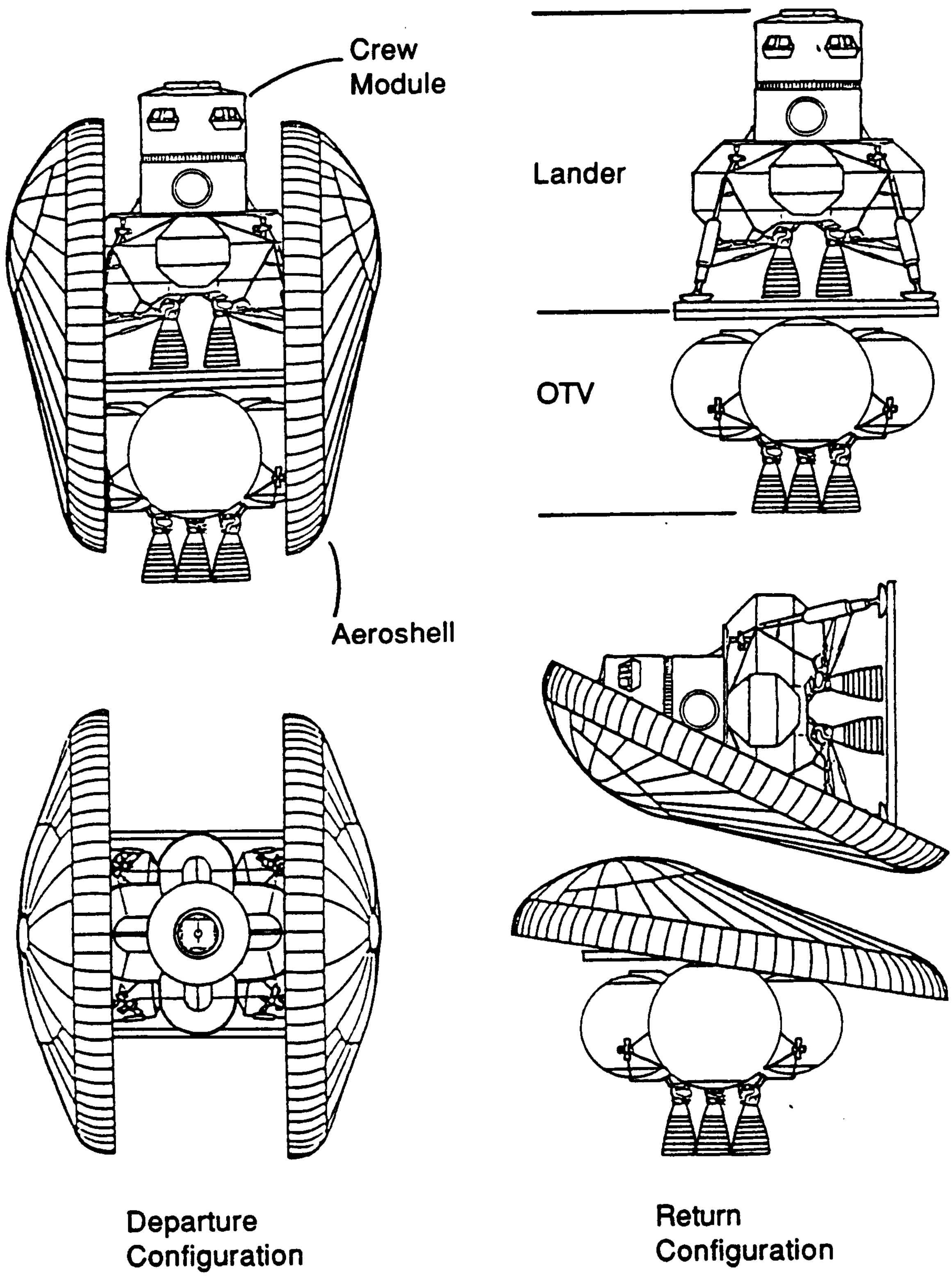
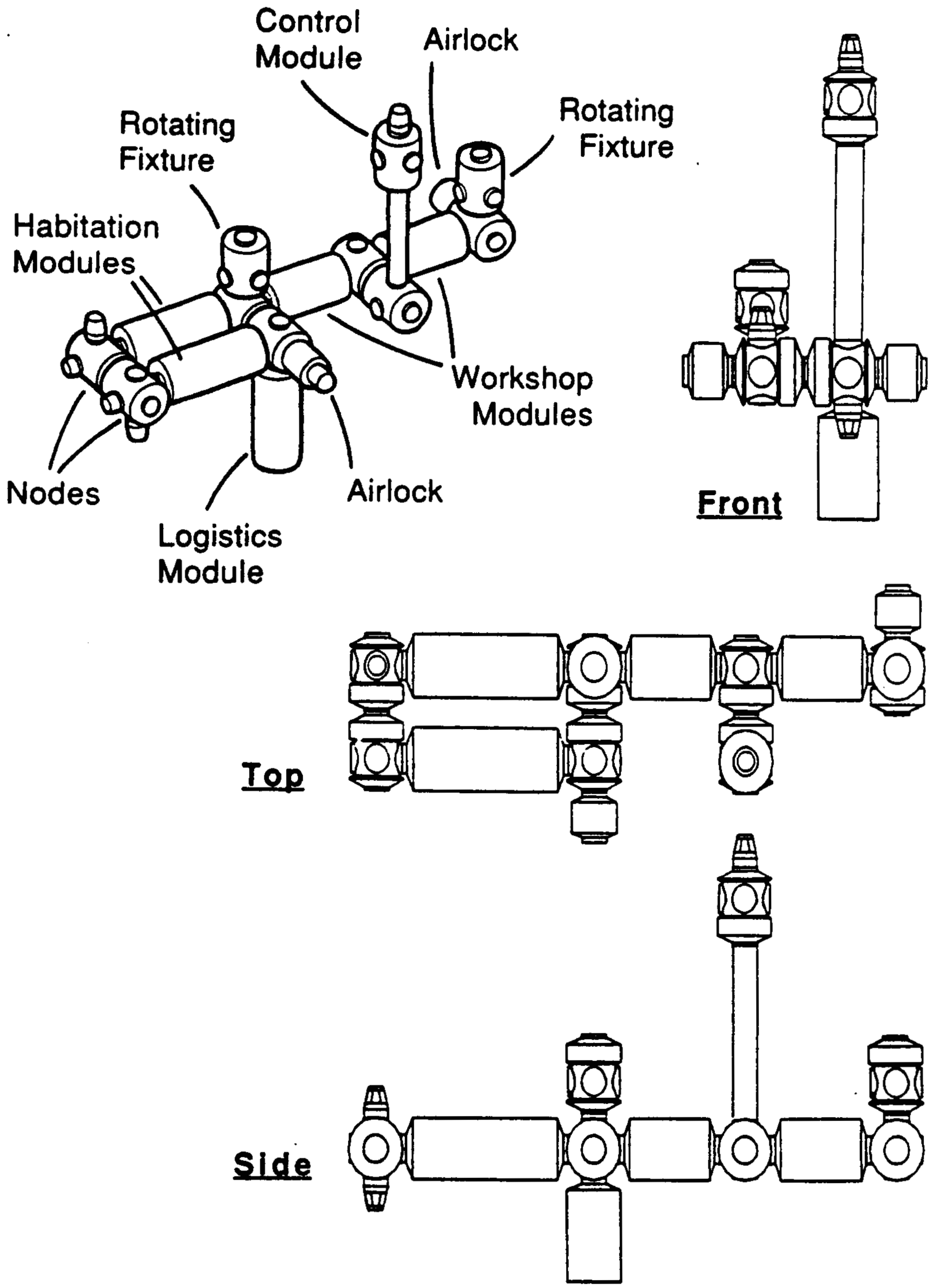


Figure 1.0-5, Pressurized Volume Configuration



2.0 Introduction/Assumptions/Groundrules

A number of recent studies (Weidman, et al, 1988, Cordell, et al, 1988) have addressed the problem of a transportation node space station. The current space station program is focused more on the problems of micro-gravity research and space science. How things would change or what additional facilities would be needed to support a major lunar or Mars initiative is an often asked question. Other studies are addressing the problem of assembling a Mars vehicle, which may weigh over 1,000 metric tons in LEO. This study addresses only the support of a lunar base, requiring stacks on the order of 200 metric tons each to land 25 m tons on the lunar surface with reusable vehicles. A lunar base, first under construction, and then permanently manned is assumed. Steady-state support for such a base is expected to require four to eight flights per year to the lunar surface.

This study concentrates on the problem of maintaining and reusing large single stage OTVs and single stage lander/launchers in space. The required people and equipment needed, to maintain these vehicles are only vaguely known at present. The people and equipment needed depend on how well the OTV and lander/launcher can be designed for easy reuse. Since the OTV and lander/launcher are only conceptually defined at present, the real maintenance and refurbishment requirements are unobtainable. An estimate of what is needed, based on previous studies and obvious requirements was therefore made. An attempt was made to err on the conservative side.

The OTV and lander/launcher used in this study are at the heavy end of the spectrum of proposed vehicles for this purpose, again an attempt to err on the conservative side.

2.1 Commonality with the Freedom Space Station

The concept developed in this report was not constrained to use Freedom Space Station systems or configuration, but these were generally used if better ways were not obviously available. Freedom Station Truss, Habitation Modules, Nodes, Airlocks and distributed systems were used. Commonality provides a method of reducing program cost if the Space Transportation Node (STN) is either made at the same time as enhancements of the Freedom Station or if tooling is maintained in a useful form after the initial production runs have been completed. Figures 6.2-4 and 6.2-5 show recent versions of the Freedom Space Station.

2.2 Use of Freedom Space Station

The design ground rule for the STN was that it would be a separate entity from the Freedom Space Station and would not be dependent on the Space Station in any way. Thus it is not co-orbiting with the Space Station.

Previous studies have indicated that activities currently planned for the Freedom Station would be adversely affected by transportation node activities. The microgravity limits would be exceeded, the vacuum would be contaminated, viewing experiments disrupted, and the crew and resources of the Station in general diverted from research activities to the preparation of the lunar vehicles. Previous studies have therefore favored completely separate or at best co-orbiting facilities. Other studies have claimed that co-orbiting is not practical because of large amounts of propellant required to station-keep. This study did not address these questions, but simply assumes the STN will be a completely separate facility.

3.0 Scenario

A transportation node station is defined by the transportation system it must support. The transportation system for a lunar base is defined by the size and nature of the base, and the functions it performs. The lunar base is assumed to be an evolutionary facility, starting out man-tended, but becoming permanently manned with a small crew within a short period of time. The permanently manned aspect is a key assumption, driving the transportation system toward reuse and refurbishment in space.

3.1 Definition of Heavy Lift Vehicle

The largest U.S. launch vehicles that are currently under detailed study are the NASA Shuttle-C and the Air Force Advanced Launch System (ALS). The performance characteristics of these vehicles are:

	Shuttle-C	ALS	ALS (expanded)
Nominal Apogee km (nm)	407 (220)	278 (150)	278 (150)
Nominal Perigee km (nm)	407 (220)	148 (80)	148 (80)
Inclination°	28.5	28.5	90
Payload capacity kg	57,168	49,900	72,595
lbs	126,000	110,000	160,000
Payload length m (ft)	23.5 (77)	24.4 (80)	38.1(125)
Payload diameter m (ft)	4.6 (15)	4.6 (15)	12.2 (40)
Shroud Clear Dia. m (ft)		10 (33)	12.2 (40)
Reference		(USAF, 1988)	

For purposes of this study the largest proposed vehicle is baselined, the expanded ALS with a 72.6 m ton (160,000 pound) launch capacity to polar low earth orbit. This same vehicle will launch on the order of 85 to 90 m tons to a 28.5°, 150 x 80 nm orbit. The STN will be in something on the order of a 250 nm circular, 28.5° inclination, so the actual payload to this orbit will be lower due to an upper stage boost needed to get to that altitude. The upper stage propellant and stage mass needed to raise the orbit to 250 nm circular, is around 5,000 kg for a storable propulsion system, thus the actual payload of a 90 m ton launcher would be 85 m tons. Another 10% of so of this would be structure and tankage, therefore the actual propellant delivered would be around 75 to 77 m tons.

In reality, a wide range of heavy lift vehicles with different payloads could service the STN. For lunar operations, a heavy lift vehicle (HLV) that brings up all the propellant for one mission at once would be optimum. This minimizes storage requirements and boil-off losses and reduces the number of operations (and potential failures) to a minimum. For the vehicles baselined in this study, 158 m tons of propellant are required for the heaviest maximum diameter of the OTV is 10 meters (3 ft). The OTV can be delivered assembled, without aerobrake within the expanded ALS shroud. The aerobrake may have to go up in two halves and be joined together in space. The general concept for the OTV and lander aerobrake system comes from Petro, 1988, Lunar Base Transportation Concepts.

Figure 3.2-1, OTV/Lander Stack and Aerobrakes

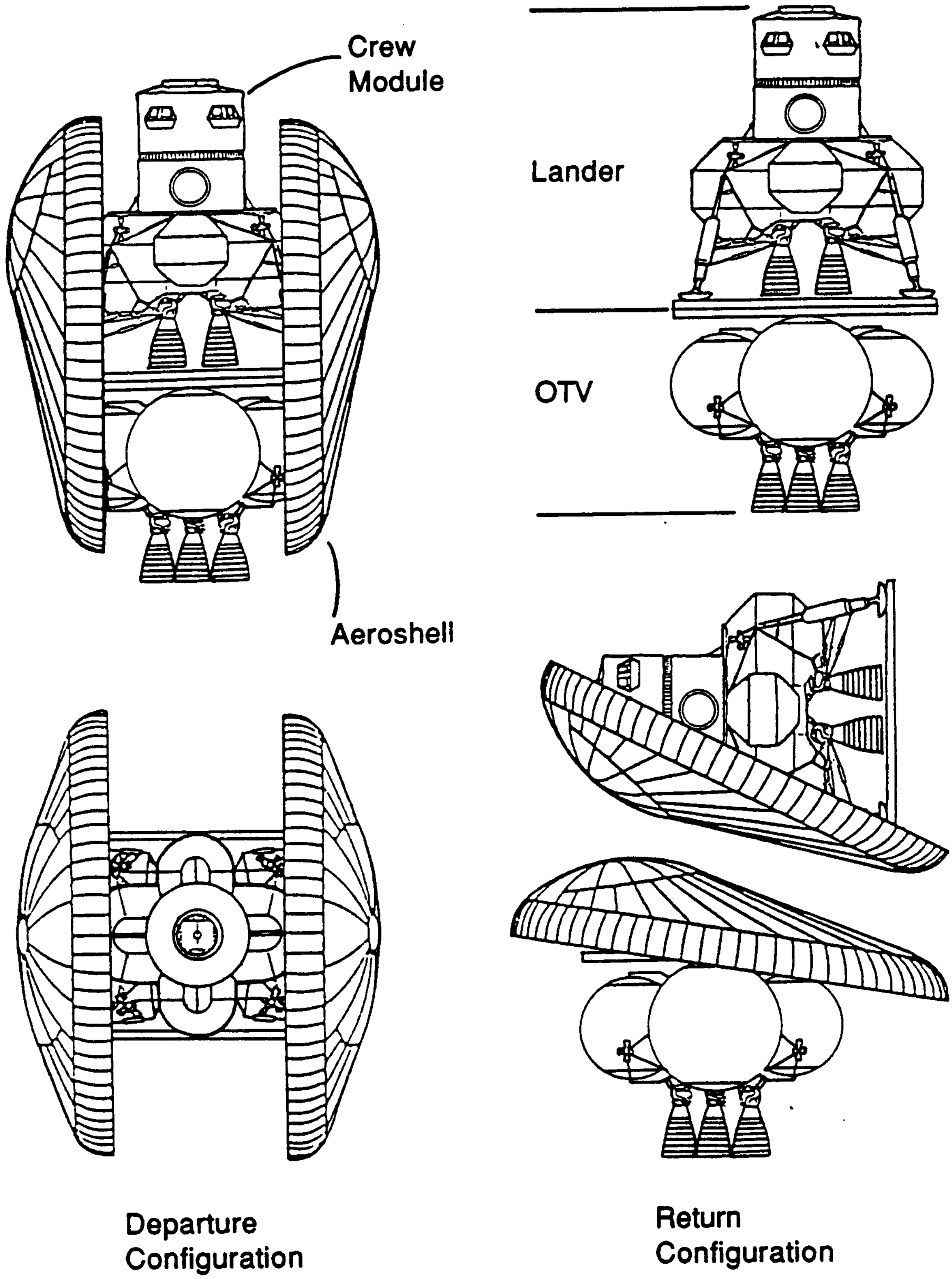


Table 3.3-1, Single Stage Orbit Transfer Vehicle Weight Statement*
kg (pounds)

	6 m ton crew module round trip		25 m ton cargo to lunar surface one way (lander expended)	
	kg	lbs	kg	lbs
Payload from LEO to LLO (lander + cargo)	48,218	(106,300)	60,074	(132,400)
Payload from LLO to LEO (returned lander inert and crew module)	16,000	(35,200)	0	
Lander Aerobrake	2,500	(5,500)	0	
OTV Aerobrake	3,000	(6,600)	3,000	(6,600)
OTV Inert	11,000	(24,200)	11,000	(24,200)
OTV LO_2**	107,400	(236,280)	107,400	(236,280)
OTV LH_2**	18,000	(39,600)	18,000	(39,600)
OTV Total	139,400	(337,480)	139,400	(337,480)
Stack Mass at LEO Dep.	190,118	(418,260)	199,474	(438,842)
Total OTV Prop.**	125,400	(275,880)	125,400	(275,880)
Total Lander Prop.**	32,395	(71,269)	25,251	(55,552)
Total Stack Prop.	157,795	(347,149)	150,651	(331,432)

*See Eagle (March 30, 1988) for more details.

**Includes flight performance reserve (FPR) and unusables.

3.4 Definition of Reusable Lunar Lander

The baseline Lunar Lander used for this point design is a 60 metric ton gross, multi-purpose liquid oxygen/hydrogen vehicle. Figure 3.2-1 shows the geometry of the vehicle. When the vehicle is collapsed it is 10.5 meters (34.4 feet) high, by 8.8 meters (29 feet) by 8.5 meters (27.9 feet). With the landing gear extended, the vehicle fits in an envelope of 11.0 (36 feet) by 9.5 (31.1 feet) by 12 (39.4 feet) meters high. The vehicle is designed to be launched complete in a 30 foot diameter payload shroud. The vehicle can be launched in components to fit within a 15 foot diameter payload envelope of the currently proposed launch vehicle fleet. The weight statement for three different lander missions are given in Table 3.4-1. The weight statements are derived from Eagle (March 30, 1988).

Table 3.4-1, LO_2/LH_2 Multi-purpose Lander Weight Statement weights in kg(lbs)

	Expendable Lander kg lbs	w/Crew Module kg lbs	Unmanned Reusable kg lbs
Payload to Moon	25,000 (55,125)	6,000 (13,230)	14,000 (30,856)
Payload from Moon	0	6,000 (13,224)	0, (inert mass returned to LLO)
Inert Mass	9,823 (21,650)	9,823 (21,650)	9,823 (21,650)
Total LO_2	21,644 (47,703)	27,767 (61,198)	26,261 (57,879)
Total LH2	3,607 (7950)	4,628 (10,200)	4,377 (9,647)
Gross Weight	60,074 (132,403)	48,218 (106,272)	54,461 (120,032)
Total Lander Prop. (includes FPR, unusables, and RCS)	25,251 (55,552)	32,395 (71,269)	30,368 (67,404)

3.5 Definition of use of Shuttle

The Shuttle is used principally to ferry crew between Earth and the Transportation Node. Some payload bay compatible cargos such as the crew capsules, replacement engines, etc., may also be flown on the Shuttle. The ALS is used to place the majority of the Transportation Node in Orbit and to provide propellants for the transportation system.

3.6 Earth to Moon Flight Scenario

Numerous scenarios for a lunar base have been proposed. At the start of the Lunar Base Systems Study in the fall of 1987, the mission scenario shown in Table 3.6-1 (Eagle, 1987) was developed and used in an initial study and compilation of assumptions and requirements for a transportation node station supporting a lunar base (Eagle, 1988). Though some of the assumptions have changed, this baseline mission set is retained in this study because it is conservative and because it allows the use of conclusions drawn based upon it in the earlier study (Eagle, 1988). The chief number of interest is the maximum number of missions per year. This study assumes it could be as high as eight. More recent, less ambitious mission scenarios have also been proposed, as shown in Table 3.6-2 (Alred, 1988).

The gross features of the STN are for the most part independent of a small variation in the maximum number of missions (6 or 8). Two stacks are required for safety in this design. Based only on through put timelines however, the hangar size could be reduced by one half to a one stack configuration at some point as number of missions/year is reduced.

The minimum number of missions per year is in the range of one per year proposed in support of an unmanned observatory facility on the far side, a man-tended base. One mission per year could perhaps be supported by expendable stages. For a permanently occupied base the minimum number of missions is more in the range of 3 or 4 per year, two missions to change out crew and one or two for resupply of consumables, spares, etc. As the flight rate increases, reusable vehicles become more desirable.

Table 3.6-1, Flight Schedule for Ambitious Lunar Base (Eagle, 1987)

	1999	2000	2001	2002	2003	2004	2005
Piloted	1	2	3	4	4	4	4
Cargo	0	3	3	4	3	3	3
Total	1	5	6	8	7	7	7

Table 3.6-2, Flight Schedule for Less Ambitious Lunar Base (Alred, 1988)

	1999	2000	2001	2002	2003	2004	2005
Piloted	0	1	2	3	3	3	3
Cargo	0	2	2	2	2	3	3
Total	0	3	4	5	5	6	6

4.0 Assumed Design Criteria

A previous study (Eagle, 1988) developed the assumptions and requirements which are to guide the current point design. Certain aspects of the program definition and understanding have changed since the initial formulation of the requirements due to the dynamic nature of program planning. However, the basic assumptions which provide the concept design criteria are continued in this task because they remain generically valid and because maintaining continuity in this planning process where possible is important. These design criteria are presented in the remainder of section 4.0. The requirements derived from the design criteria are addressed in section 5.0.

4.1 Basis of Criteria

As explained in (Eagle, 1988) "a representative but generic scenario was desired to reduce the sensitivity of the results to fluctuations in detail definition as the program changes and evolves." In the intervening months, plans have been dynamic and the definition has tended toward a less active schedule of flights to the Moon. However, the definition of the vehicles and mission is still evolving, so the previous generic missions baseline still represents a reasonable model of an upper limit of lunar flight activity. The baseline of eight flights in calendar year 2002 is restated briefly below in Table 4.1-1. In this year, hardware is being delivered to the surface and a permanent base is under construction, but crews are still constrained to living in temporary facilities and can therefore only stay for short periods. Table 4.1-2 summarizes the elements the STN must support.

Table **4.1-1,** Missions in Baseline Year

Mission Number	Date	Purpose
15	1/2/2002	LO_2 pilot plant automated delivery
16	2/8/2002	30-day crew stay on lunar surface
17	4/3/2002	Airlock, node, and radiator automated delivery
18	5/9/2002	30-day crew stay on lunar surface
19	7/4/2002	Life sciences research facility auto. delivery
20	8/8/2002	30-day crew stay on lunar surface
21	10/4/2002	Rovers and garage, automated delivery
22	11/8/2002	30-day crew on lunar surface

Table 4.1-2, Summary of Vehicle Characteristics (all weights are in kilograms)

Vehicle	*Wet mass	Prop. Load	*Inert Mass	Max Dimension, meters
Aerobrake	-	-	3,000	18
OMV	5,900	3,600	2,300	4.6
OTV**	136,400	125,400	11,000	10
Lander**	42,200	32,400	10,000	11
Crew Capsule	6,000	-	-	4.6
Max. Cargo	25,000	-	-	-
Heavy Lift Veh. (upper stage payload)	90,000	80,000	~10,000	-

* Does not include aerobrake, an additional 3,000 kg or so.
** Payload (crew capsule or cargo) not included.

More detailed definitions of the vehicles are provided in sections 3.1, 3.2, 3.3, and 3.4. The specific LEO STN activities to support the more current versions of the vehicles would be different from those in Eagle, 1988. In a brief and general review of the probable changes in derived requirements caused by the vehicle changes, it was concluded that the impact was small and task resources should not be expended to perform a new timeline/schedule analysis.

4.2 Summary of Assumptions

The assumptions of Eagle, 1988 provide the design criteria for this conceptual design task. The assumptions most directly affecting the design effort are summarized in Table 4.2-1.

Table 4.2-1, Summary of Assumed LEO STN Design Criteria

Assumption ID Number	Abbreviated Description of Design Criteria Assumption
1.06	OTV maintained in readiness status at STN for emergency lunar crew return
1.07	Protection required in hangar from radiation, meteors, and orbital debris; hangar side facing Earth may acquire protection from Earth shielding
1.11a	Earth-produced propellant delivered to STN by generic HLV
1.11a	64 m tons of LO_2 per HLV delivery (75 m tons total propellant)
1.11a	11 m tons of LH_2 per HLV delivery
2.02	Propellant transfer by tank-to-tank pumping, not tank exchange
2.07	Use Space Station technology and systems where possible
2.08a	Use built in test and automatic checkout in space vehicles to be supported
2.08b	All space vehicles flight hardware to be under continuous self check monitoring
2.08c	All space vehicles to have automatic fault detection/isolation to ORU level
2.08d	Design standard interfaces between space vehicles
2.08e	Provide enough access to remove/replace ORU's
2.08f	Require space vehicle maintenance accessibility without necessity to remove healthy equipment
2.10	OMV used for space tug
3.02a	Earth-based mission control provides management of prelaunch, launch, and STN rendezvous operations
3.02b	STN provides operations control of STN approach and proximity operations
3.03	Vehicle service tasks required in space environment performed by teleoperations if possible

Table 4.2-1, Summary of Assumed LEO STN Design Criteria (Continued)

Assumption ID Number	Abbreviated Description of Design Criteria Assumption
3.04a	Maximum EVA duration is eight (8) hours
3.04b	Maximum scheduled EVA operations per crewman is 8 hours/week
3.04c	Maximum simultaneous EVA crew is four (4)
3.04d	Minimum number of crew required during EVA operation is three (3); 2 EVA and 1 IVA monitor
3.05	Lunar crew delivery Orbiter required at STN through translunar injection
3.06	No launch of lunar crew pickup Orbiter until trans-Earth injection
3.07	Two (2) STN crew required on EVA or EVA readiness alert while vehicle servicing in progress
3.08	Two (2) STN crew required to support HLV tanker rendezvous through berthing
3.11	STN crew works seven (7) days/week

5.0 Derived Requirements

Based on the preceding statement of design criteria assumptions, estimates of more specific performance requirements can be derived. This requirements derivation analysis was developed in Eagle, 1988 and is reviewed and summarized here in this section. Where a vehicle has evolved into a somewhat different configuration, the initial requirement has been converted to be consistent with the current vehicle definition. For example, a requirement that previously stated a need for two OTV stages has been converted to a statement for one OTV stage using the current vehicle baseline. In the context of the following discussion, the OTV is one of the STN and mission resources. Others include the OMV, the hangar, the lunar lander crew cabin, an EVA astronaut, an RMS, an IVA crewmember, an Orbiter crew, the lunar crew, a lunar cargo, and the lunar lander stage.

5.1 Derivation Process

The STN space vehicle servicing requirements were initially developed in a two step process. The requirements to service vehicles for each of the eight baseline missions were derived based on the assumed design criteria and experience from previous space program activities. The eight individual schedules of requirements were then overlaid on the one-year baseline flight schedule to accumulate the integrated resource load and derive the various resource capacities required.

Figure 5.1-1 is repeated from Eagle, 1988 to illustrate the STN service activity flow identified as typical of the tasks at the STN required to perform the manned lunar mission baseline. The STN and mission resources necessary to participate were derived for each of the activity boxes. For example, the Lunar Flight Preparation activity (item 016ML05 in Eagle, 1988) requires twenty days. During those twenty days, the following resources are required to be exclusively committed to the mission; one RMS, one OTV, one manned lunar lander, one lunar lander crew cabin, adequate hangar volume, and two IVA STN crew. In addition, two EVA crew are needed for three of the twenty days.

The activity flow for each of the eight baseline missions was aligned on a schedule according to the mission launch dates. This process was automated on a commonly used commercial critical-path-method project planning tool. Using this process, the individual resources were analyzed to determine the peak capability requirement and when the peak occurred. The process can also be used to analyze an individual resource or a desired combination of resources. The derived requirements that follow come from this process.

5.2 Summary of Requirements

The requirements from Eagle, 1988 provide the derived requirement guidelines for this conceptual design task. The requirements most directly affecting the design effort are summarized in Table 5.2-1.

Figure 5.1-1, Service Activities Flow

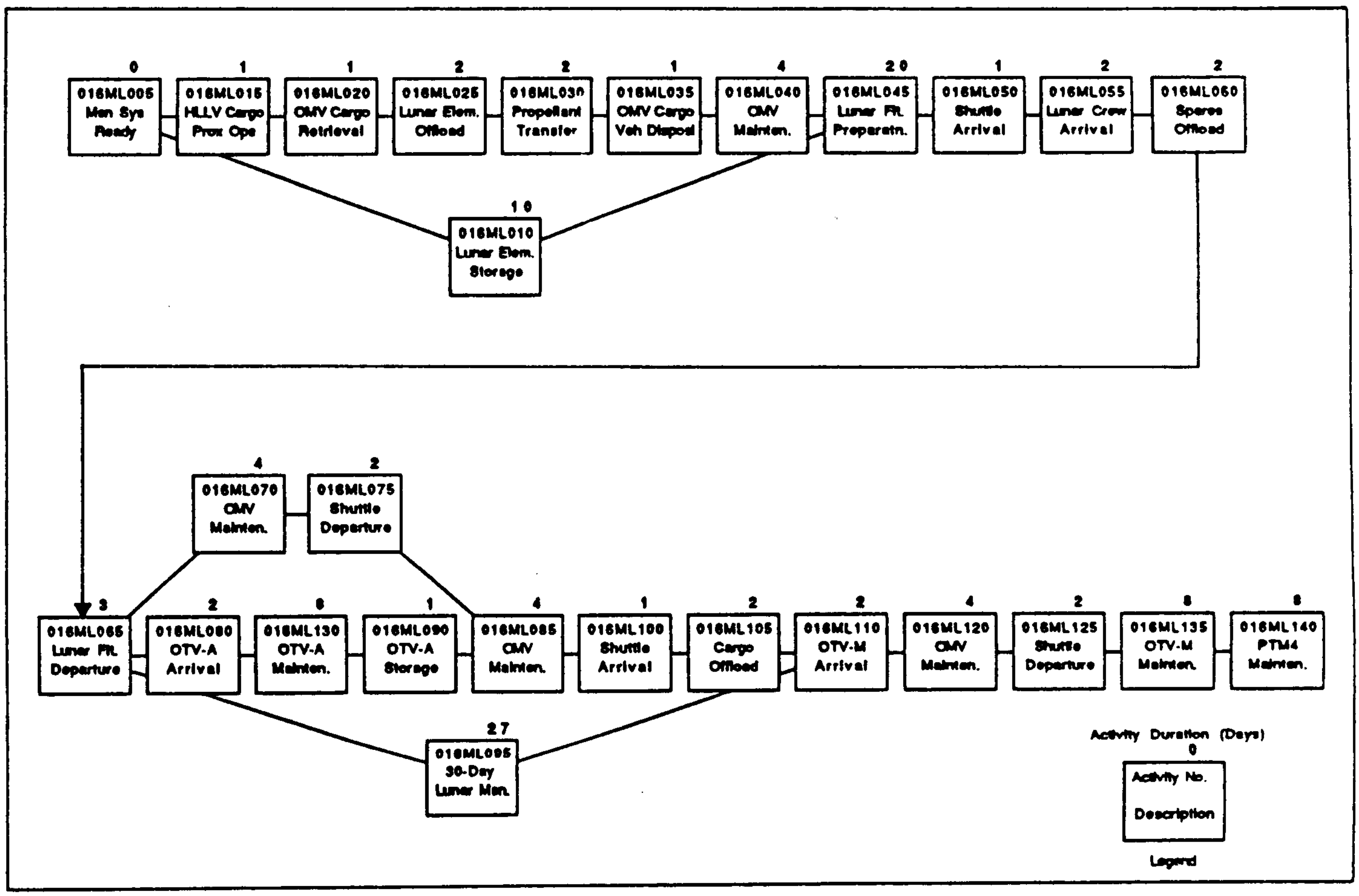


Table 5.2-1, Summary of Derived STN Performance Requirements

Requirement ID Number	Abbreviated Description of Performance Requirement	
1.01.2	STN orbit parameters guideline • enable efficient logistics from Earth • enable efficient transportation to Moon • ensure no other space vehicle collision • avoid Space Station applications view interference Nominal choice - 28.5° inclination to allow in-plane departures when the lunar orbit plane is at its maximum angle (28°) relative to the Earth's equatorial plane and to allow due East launches from KSC. Altitude - approx. 250 nm, or such that the regression rate allows one optimum arrival or departure per month from high inclination lunar orbit. See Eagle, 1988, and section 12 for more discussion.	
1.01.3	Docked Orbiters allowed during lunar departure/arrival	One (1)
1.01.4	STN mission pointing/orientation requirements	None
1.01.5	Space Station technology/design commonality	Maximize
1.01.6	Space exposed maintenance by automated equipment	Maximize
1.03.1	Remote manipulator systems (RMS's) required	Two (2)
1.03.1	RMS mass handling capacity	Lunar stack mass (200 m tons)
1.03.2	RMS robotic end effector capability	Reach and manipulation comparable to or better than EVA astronaut
1.05.1	Number of Orbiter berthings per 30 days	Two (2)
1.05.1	Number of EVA airlock passages per 30 days	Nine (9)
1.05.1	Maximum EVA-person events per 24-hour period	Four (4)
1.19.1	Space required for pressurized logistics/warehouse	Yes
1.22.1	Permanent crew size accommodated	Six (6)
1.22.2	Visitor (14-day) crew size accommodated	Seven (7)

Table 5.2-1, Summary of Derived STN Performance Requirements, Continued

Requirement ID Number	Abbreviated Description of Performance Requirement	
1.24.1	STN Transportation Operations Centers required	One (1)
1.24.2	System health monitoring frequency for stored vehicles	Sufficient to prevent degradation
1.25.1	LO_2 propellant mass storage requirement	Sufficient for one (1) lunar stack + 10 %
1.25.1	LH_2 propellant mass storage requirement	Sufficient for one (1) lunar stack + 10 %
1.25.1	OMV propellant mass storage requirement	Eight (8) proximity ops flights
1.25.2	EVA crew required for nominal propellant transfer	None
1.25.4	Explosion control systems required	Yes
1.26.1	Enclosed hangar vehicle containment capability	
	• RMS	Two (2)
	• OMV	Two (2)
	• Lunar stack (integrated or components)	Two (2)
1.26.4	Rotating vehicle-servicing fixtures in hangar	Two (2)
1.26.6	Access method to Lunar Stack crew cabin	IVA
1.26.7	Protection from radiation, orbital debris, and micro-meteoroids required during vehicle servicing	Yes

6.0 History of Space Station Configurations

In order to design a new Space Station, some understanding of the recent history of space station design and why certain choices were made is required. The following discussion briefly reviews this history.

During the past decade and a half, NASA has conducted a number of Space Station studies which included the types of transportation systems needed to accomplish program objectives. The early studies emphasized the use of Space Stations as scientific laboratories for performing "experiments" in orbit (Rockwell, 1969-70 and McDonnell, 1969-70). The Space Station consisted of a large 33 foot diameter configuration with four vertical decks, a power boom and solar arrays. The Station was placed in low earth orbit utilizing the Saturn V Launch Vehicle. During the early phase of Space Shuttle design, the concept of a large Space Station was discarded and the concept of a "Modular" Space Station evolved (Rockwell, 1971 and McDonnell, 1971). The ensuing configurations included modules which could be launched in the Shuttle 15 foot diameter by 60 foot long cargo bay and assembled in orbit using docking mechanisms which were attached at the ends of the modules. Later studies provided design concepts that included the more science-oriented laboratory as well as a capability to construct systems on-orbit (McDonnell, 1976). During this period, NASA began development of the Shuttle-based Space Transportation System (STS) as a desirable precursor to a permanent orbital facility. As the Shuttle design matured the European Space Agency (ESA) began development of Spacelab modules and pallets that can be flown in the Orbiter 60-foot-long by 15-foot-diameter cargo bay. Spacelab would provide the additional facilities needed to conduct manned operations and also carry payloads and experiments in the Orbiter cargo bay.

Additional concept studies of large orbiting systems, including vehicles operating in geosynchronous orbit with large payloads, indicated a requirement for construction and assembly of systems in space (Rockwell, 1977 and Kraft, 1977). During this time, the Space Shuttle was viewed as the only mode of space transportation for a number of years and would most likely be the initial space base for a number of planned relatively short-duration missions. Further studies (Boeing, 1981) showed that operational support capabilities beyond those of the Shuttle would be required to accommodate long-term mission payload concepts which were being proposed by NASA. Additionally, NASA conducted in-house Space Station conceptual designs which included the "Delta-Truss" (Figure 6.0-1), the "Big T-Truss"(Figure 6.0-2), and the "Power Tower" (Figure 6.0-3) (NASA, 1983) configurations.

As a result of these studies and the mandate from the President of the United States to establish a permanent manned presence in space in this decade, NASA began an in-house concept design study in 1984 to define a Reference Space Station Configuration (NASA, 1984). The Space Station was conceived as a Shuttle serviced permanently occupied facility in low Earth orbit, accommodating a crew of four to twelve, with reduced dependance on Earth for control and resupply. The facility was to accomplish current and future planned programs such as deployment and assembly of large orbiting systems, science applications, materials processing, technology development, satellite servicing/repair and flight support/ maintenance for manned and unmanned propulsion stages. The in-house four-month study evaluated five Space Station concepts which included the Concept Development Group "CDG Planer", the "Delta-Truss", the "Big T-Truss", the "Power Tower", and a configuration called the "Spinner". The "Power Tower" was finally selected as the reference configuration to allow its inclusion in the definition phase B Request for Proposal (RFP). The "Power Tower" was selected because it was seen as maximizing the accommodation of current user and growth requirements while demonstrating acceptable design and operations characteristics.

6.1 Configuration Design Requirements

During the period of this 1984 study, design requirements were derived from established program objectives, both in terms of an Initial Operational Capability (IOC) and a growth capability. Configuration and system requirements were derived from customer requirements and operations requirements. The mission and customer (user) requirements dictated the need for three separate spacecraft: 1) a permanently - manned Space Station with an inclination of 28.5° in low Earth orbit; 2) an unmanned co-orbiting platform which can rendezvous with and dock to the Space Station; and 3) a polar orbiting platform. Transportation systems were needed, in addition to the Space Shuttle, such as an Orbital Transfer Vehicle (OTV) and an Orbital Maneuvering Vehicle (OMV) for transferring personnel and equipment from low Earth orbit to higher energy orbits.

Top level design requirements for the Space Station given in NASA, 1984 are as follows:

1. Pressurized volumes for crew habitation and laboratories to service all users.

2. Provide the necessary systems for Space Station house-keeping, orbital operations, and customer or user operations.

3. Provide mounting locations, and pointing capability for celestial and Earth viewing payloads/instruments.

4. The capability to accommodate, service, and operate with the Space Shuttle, free-flying satellites, co-orbiting platform, and space transportation vehicles such as the OTV and OMV.

5. Provide continuous power for commercial and scientific functions.

6. The ability to store, maintain, service, assemble, and reconfigure vehicles and payloads.

7. The capability to service and refuel free-flying spacecraft, platforms, and attached payloads.

8. The ability to provide micro-gravity operations for long periods of time for scientific and materials processing functions.

Inherent in the above listed design requirements is the necessity for phased buildup and evolutionary growth and the capability to incorporate advances in technology as they occur. This implies the possibility of an initial configuration which may be "man-tended" in which the Space Station is visited at intervals by the Orbiter and operates unmanned the rest of the time. The Space Station then evolves into a permanent manned facility with incremental growth capability.

Figure 6.1-1 shows the early reference IOC Space Station. Key features of the configuration are a 396 foot vertical "keel" with a perpendicular "transverse boom" which supports solar arrays for electrical power generation.

An "upper boom" is provided for attaching stellar viewing payloads and a "lower boom" provides mounting for Earth viewing payloads. Five pressurized modules which consist of two Habitation Modules, two Laboratory Modules, and a Logistics Module are mounted in a "racetrack" configuration where the keel is divided at the bottom to allow installation on the centerline of the keel. A Satellite Servicing Bay and Satellite Storage Bay is located on the keel above the Transverse Boom. A large Mobile Manipulator is located on the structure in the X-Z plane.

Figure 6.0-1, Delta Truss (NASA S84-25886)

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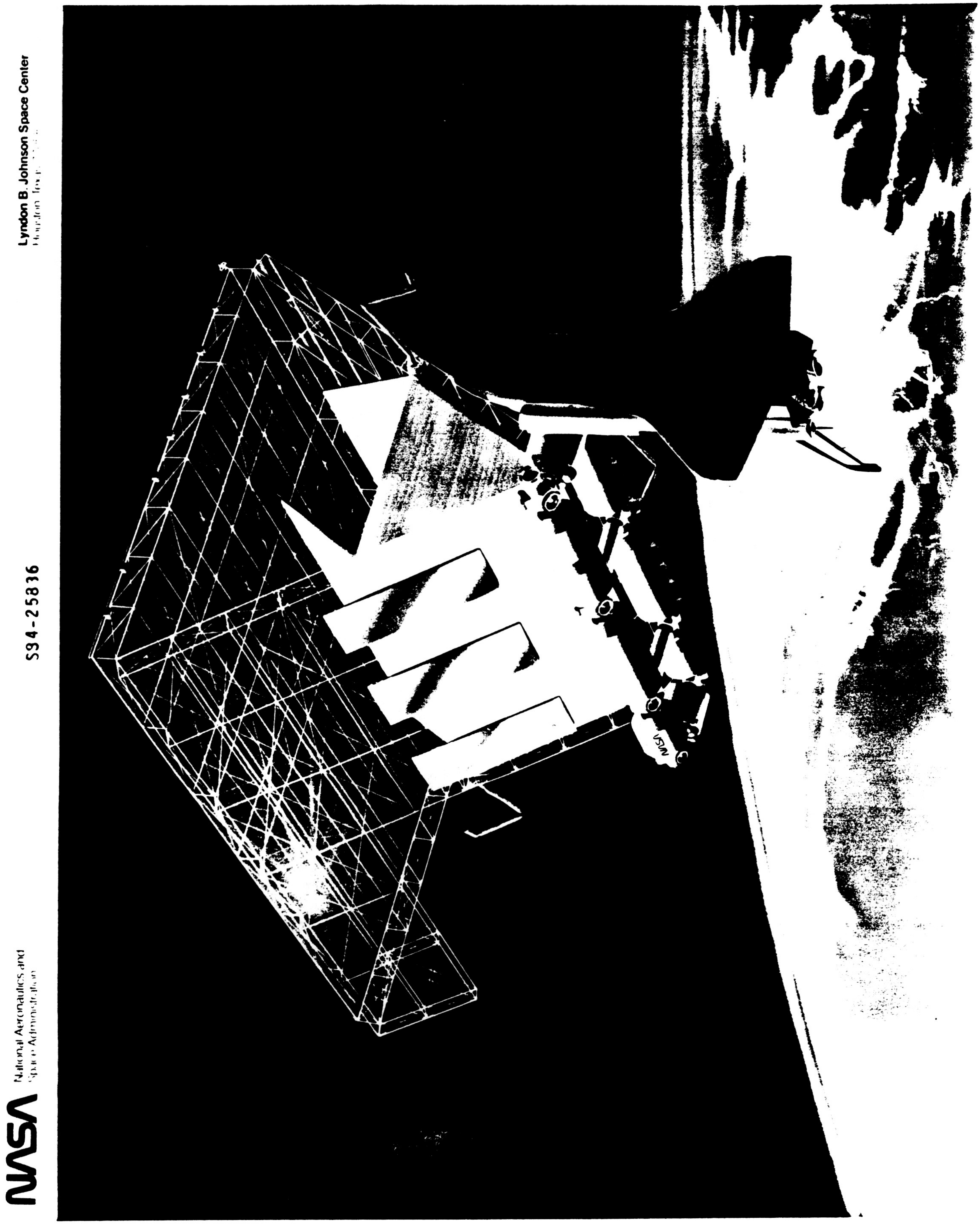

Figure 6.0-2. Big T-Truss (NASA S84-25337)

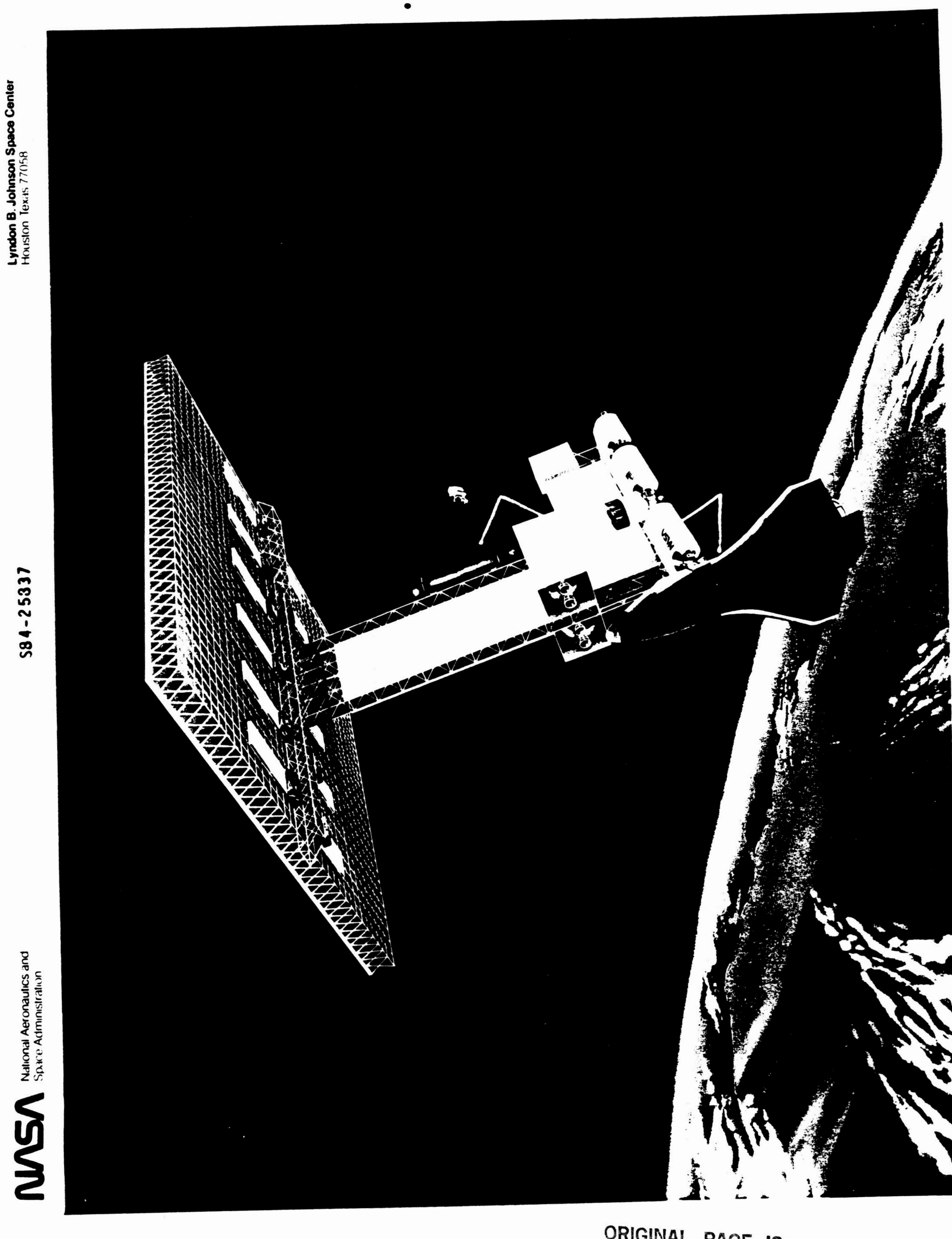

Figure 6.0-3. Power Tower (NASA S84-05434)

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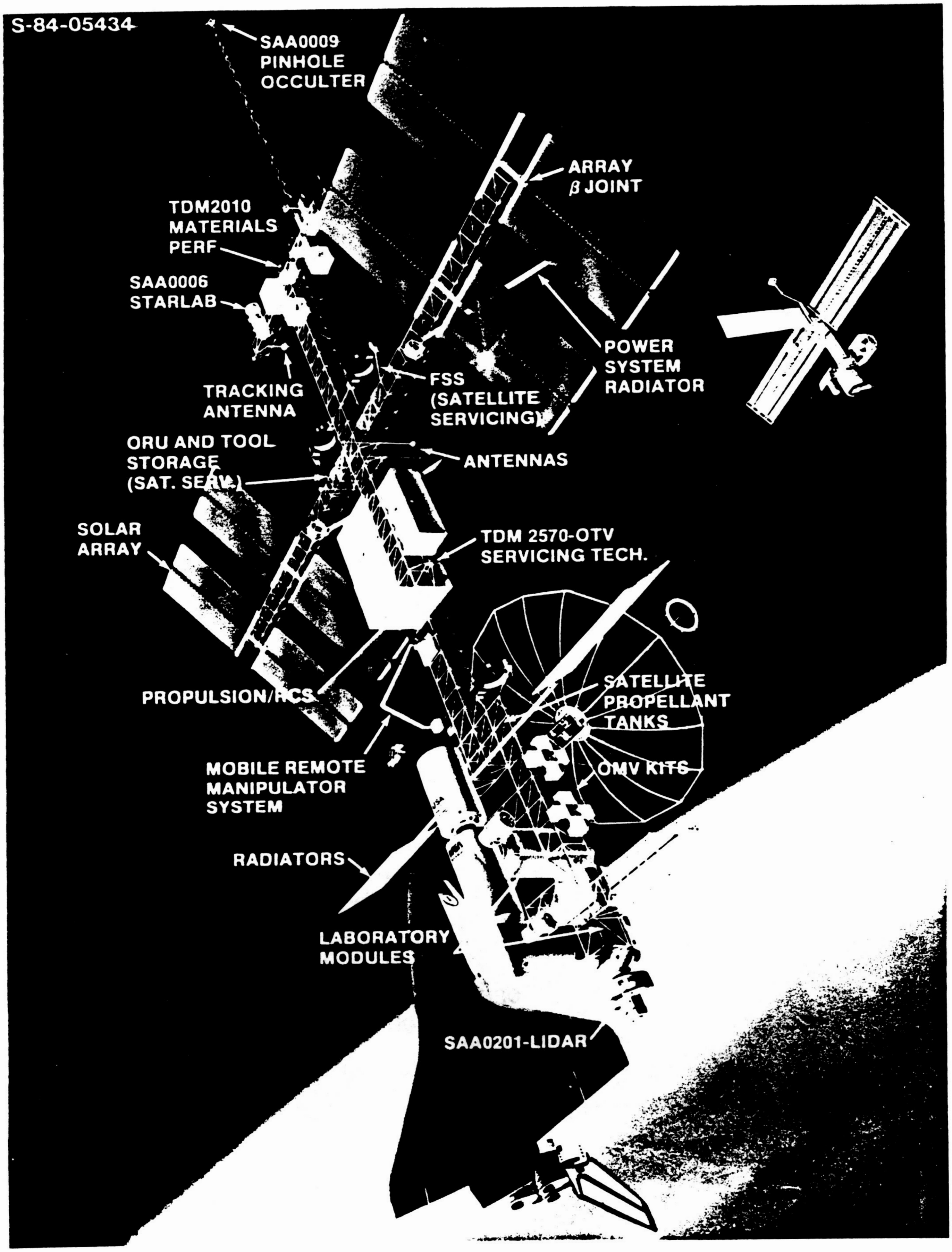
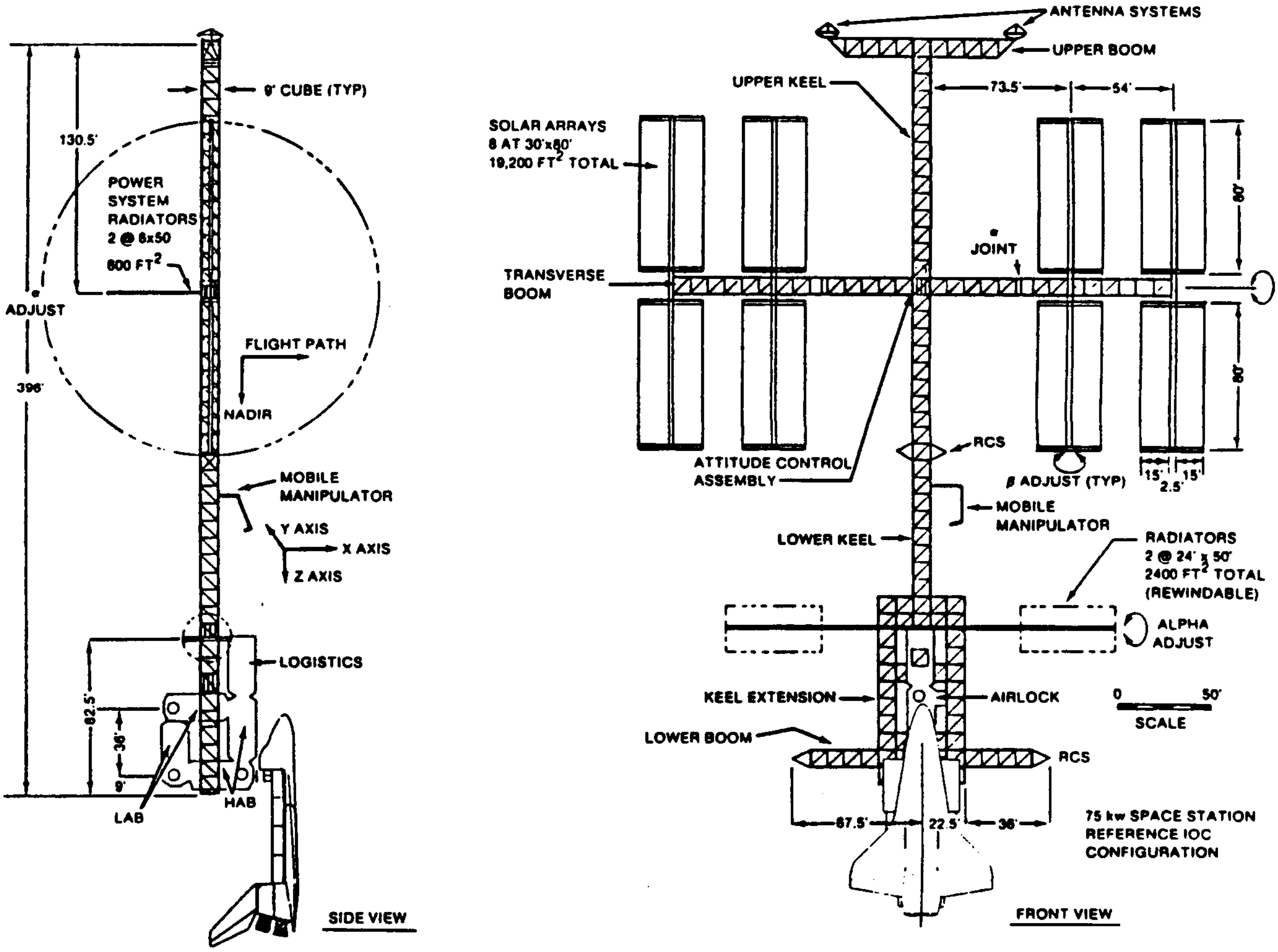


NASA
National Aeronautics and
Space Administration

Lyndon B. Johnson Space Center
Houston Texas 77058

Figure 6.1-1, Early Space Station Reference Configuration (Power Tower)



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6.2 Evolvement to the Dual Keel Configuration

Prior to release of formal Phase C/D requests for proposals (RFPs) to the aerospace industry, NASA performed additional configuration design studies to reduce cost and risk and to ensure that the reference configuration satisfied user requirements. It was determined that the "Power Tower" reference configuration did not provide enough space for locating external mounted payloads and that the laboratory modules needed to be mounted at the composite C.G. to allow for micro-gravity experiment operations. In order to satisfy these requirements, the basic Space Station structure was increased from a 2.74 meter bay box beam to a 5 meter bay box beam. A closed-rectangular configuration box beam was provided for symmetry and stiffness. The transverse boom was centered and located at a right angle to the two vertical keels. The boom supported both gimballed solar arrays and solar dynamic electrical power generation configurations. All modules were located as close to the C.G. as possible for micro-gravity operations. A structural "back porch" was used for module attachment and support. Figure 6.2-1 shows this configuration. Later configuration structural analysis showed that the back porch was not required and it was discarded. The Satellite Servicing Facility and OTV Hangar were each located on the inside of both vertical keels above the transverse boom. The construction area was located at the opposite end (Earth pointing) of the rectangular box beam structure. This configuration not only provided more "real estate" for attaching celestial and earth viewing payloads but provided improvements in stabilization and control and shuttle docking and operations. Payloads could now be located on any face of the 5 meter bay structure, except for the front face which was reserved for translation of the Canadian supplied Manipulator System. The revised reference configuration was then referred to as the "Dual Keel". Figure 6.2-2 shows the Dual Keel configuration, without the back porch.

To clear the way for final Phase C/D RFPs, a review group known as the "Critical Evaluation Task Force (CETF) was formed to further evaluate and recommend changes to the reference configuration. A number of changes were recommended by the CETF and implemented by direction of the NASA Administration. The major configuration design changes that were implemented included: 1) replacing the nodes and tunnels with larger "resource" nodes for connecting the pressurized modules; 2) a command and control station to be located in the appropriate node(s); 3) move equipment that was attached externally and required EVA for maintenance into pressurized modules; and 4) revise the assembly sequence to provide early scientific return and reduce extravehicular activity on early Station assembly flights.

The CETF also recommended a phased I and II buildup of the Space Station, based on costing considerations. The baseline configuration design was revised to provide an initial power capability of 37.5 kw. Figure 6.2-3 shows the Phase I IOC configuration with habitability and experiment modules, nodes, logistics module, airlocks and solar arrays located on the main structural boom. The Phase II configuration is also shown in Figure 6.2-3 and consists of an "enhanced capability" which increases the power level to 75 kw and above by adding solar dynamic power generating capability to the main structural boom. Other features include the addition of the two vertical keels, externally attached stellar and Earth pointing payloads, and fixed servicing capabilities which are located closer to the modules. Figures 6.2-4 and 6.2-5 show more recent versions of the Phase I and Phase II Freedom Space Station with an additional 37.5 kw of solar array power modules.

6.3 Satellite and Propulsion System Servicing

Satellite services studies were performed by NASA (NASA, 1984) with emphasis placed on Orbiter near term operations to the time period of the year 2000. Satellite mission models were developed to identify on-orbit service concepts which were compatible with the satellite user community needs. The study analyzed servicing scenarios associated with selected satellite mission models to derive the appropriate service equipment requirements. Generic types of service equipment were identified which are essential for the majority of anticipated service functions. Four equipment categories were identified which included: 1) inherent equipment, 2) generic equipment, 3) unique equipment, and 4) advanced equipment. A Satellite Services Catalog of tools and equipment was developed under the NASA contract.

Spacecraft servicing operations cover a spectrum of in-space support activities such as refueling, repairing, and maintaining free flyers and co-orbiting satellites. Three spacecraft representing many of the various anticipated servicing operations were selected and analyzed in Rockwell, 1982. The spacecraft consisted of a space-based OTV, a large deployable communications satellite, and a space processing facility. The analysis determined the unique equipment required for each servicing operation, the number of man-hours required to perform the servicing, and the number of crew required for each servicing function. Design of spacecraft to be serviced in orbit should minimize the skills required to perform the space operations. This may be accomplished by increased automation in check-out procedures. Commonality of subsystems and installation designs minimize the amount of unique equipment required for servicing at the Space Station. The establishment of appropriate design criteria should be imposed on all spacecraft requiring space servicing at the Space Station.

6.4 Dual Keel Configuration

The current NASA Space Station Phase C/D contractual effort for design and operation of a low Earth orbit Space Station includes facilities for OMV and OTV servicing and maintenance in the Station enhanced or growth configuration. Weidman (1988) is a "Study of the Use of the Space Station to accommodate Lunar Base Missions". The version of the Space Station used in the study was the Initial Operational Capability (IOC) Dual Keel Configuration as shown in Figure 6.4-1. The vehicle hangar/servicing facility required to house the lunar vehicles is shown attached to the upper keels with the propellant tanks attached below the transverse boom. Since the Dual Keel Configuration was developed to accommodate planned scientific experiments as specified in the Mission Requirements Data Base, additional Station infrastructure was determined essential to accomplishing the lunar base goals.

The additional infrastructure identified, included fuel storage facilities, on-orbit transportation capabilities, and servicing facilities. A fuel depot facility was recognized as a need in the lunar base program. It was also pointed out that the location for this facility, either co-orbiting or station-based, has not been determined.

Figure 6.2-1, Early Dual Keel Station With Back Porch

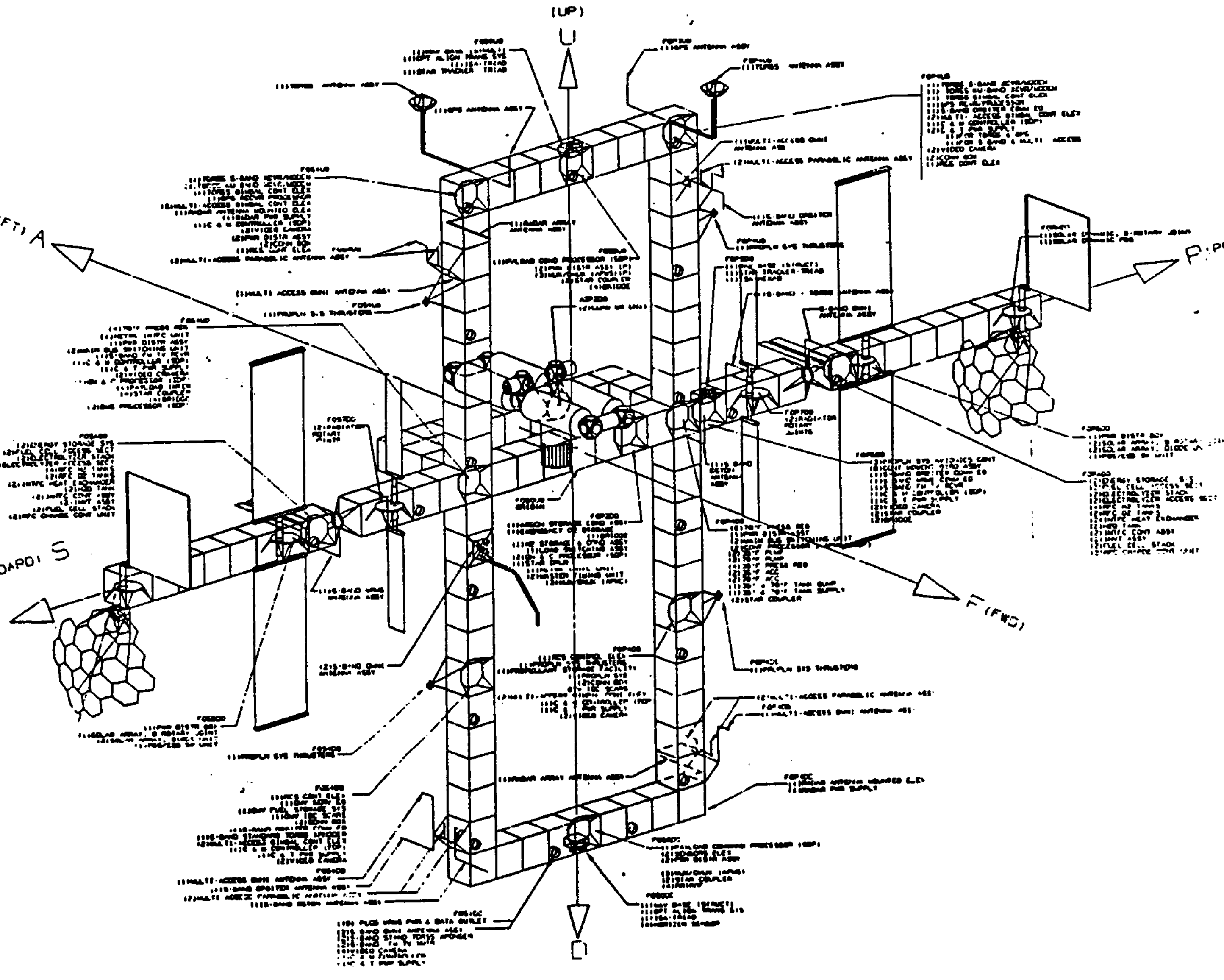


Figure 6.2-2, Dual Keel Without Back Porch (McDonnell Douglas)

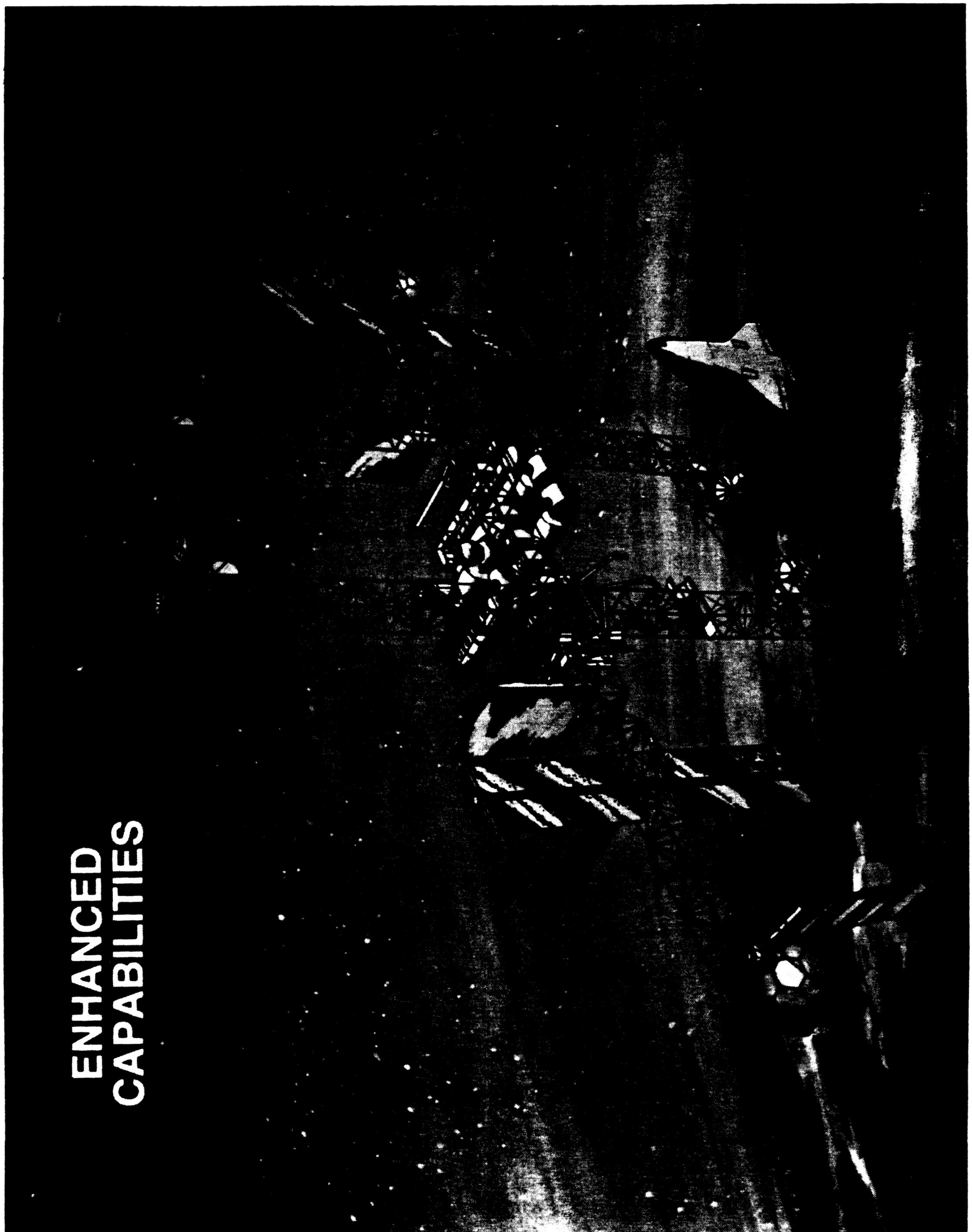


Figure 6.2-3, Early Phase I, and II Configurations (McDonnell Douglas)

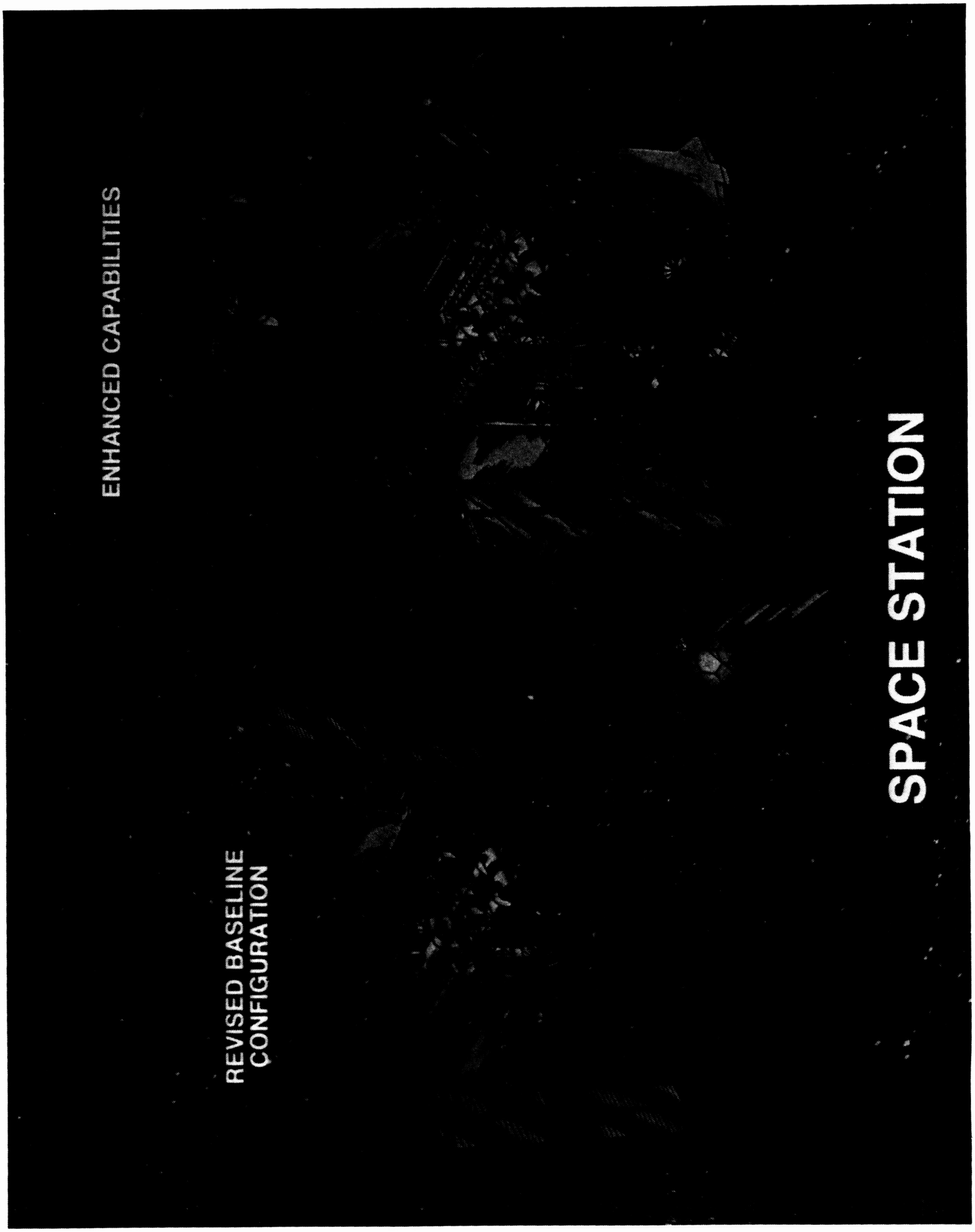


Figure 6.2-4, Phase I Freedom Space Station (NASA)

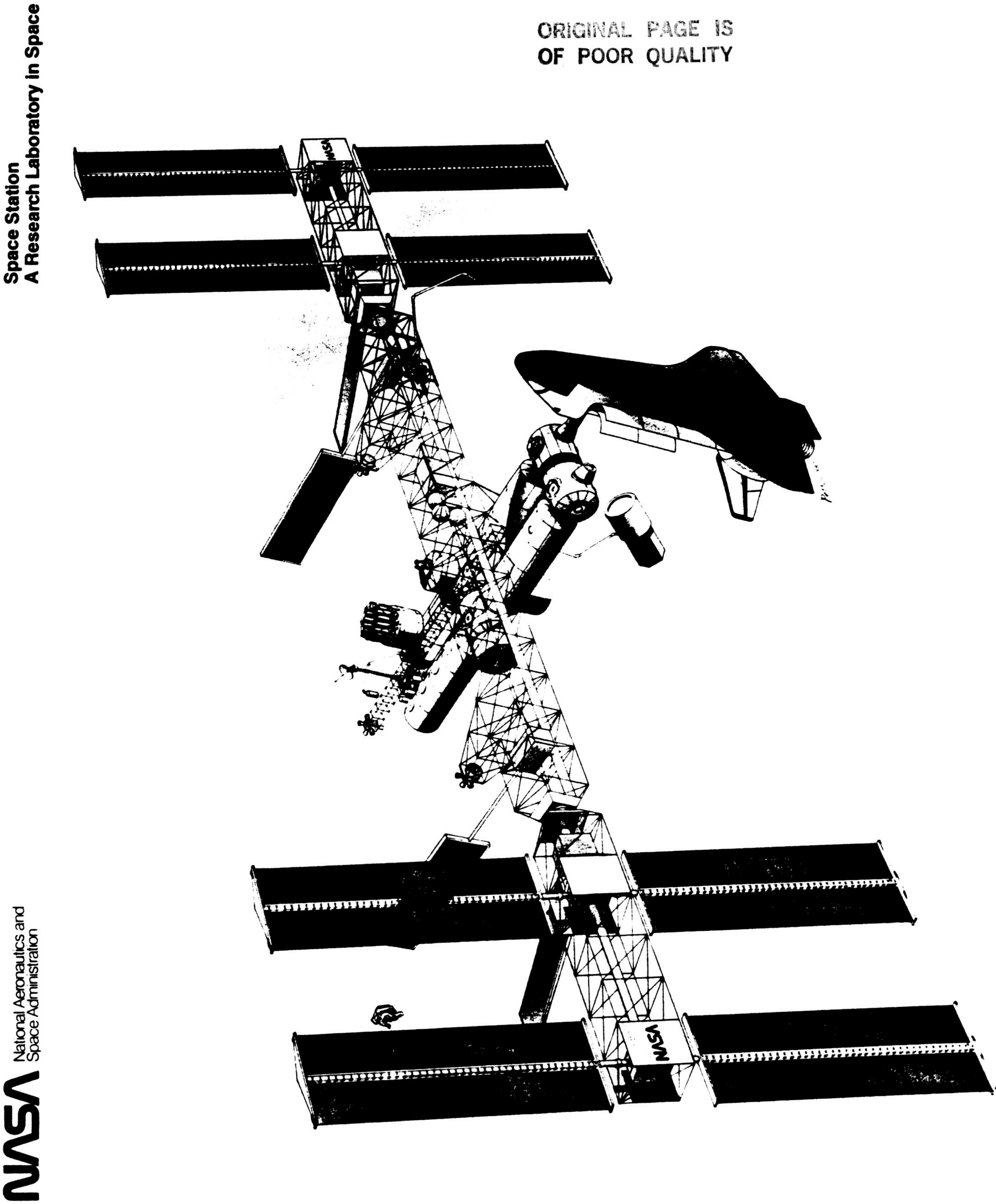


Figure 6.2-5, Phase II Freedom Space Station (NASA S87 38366)

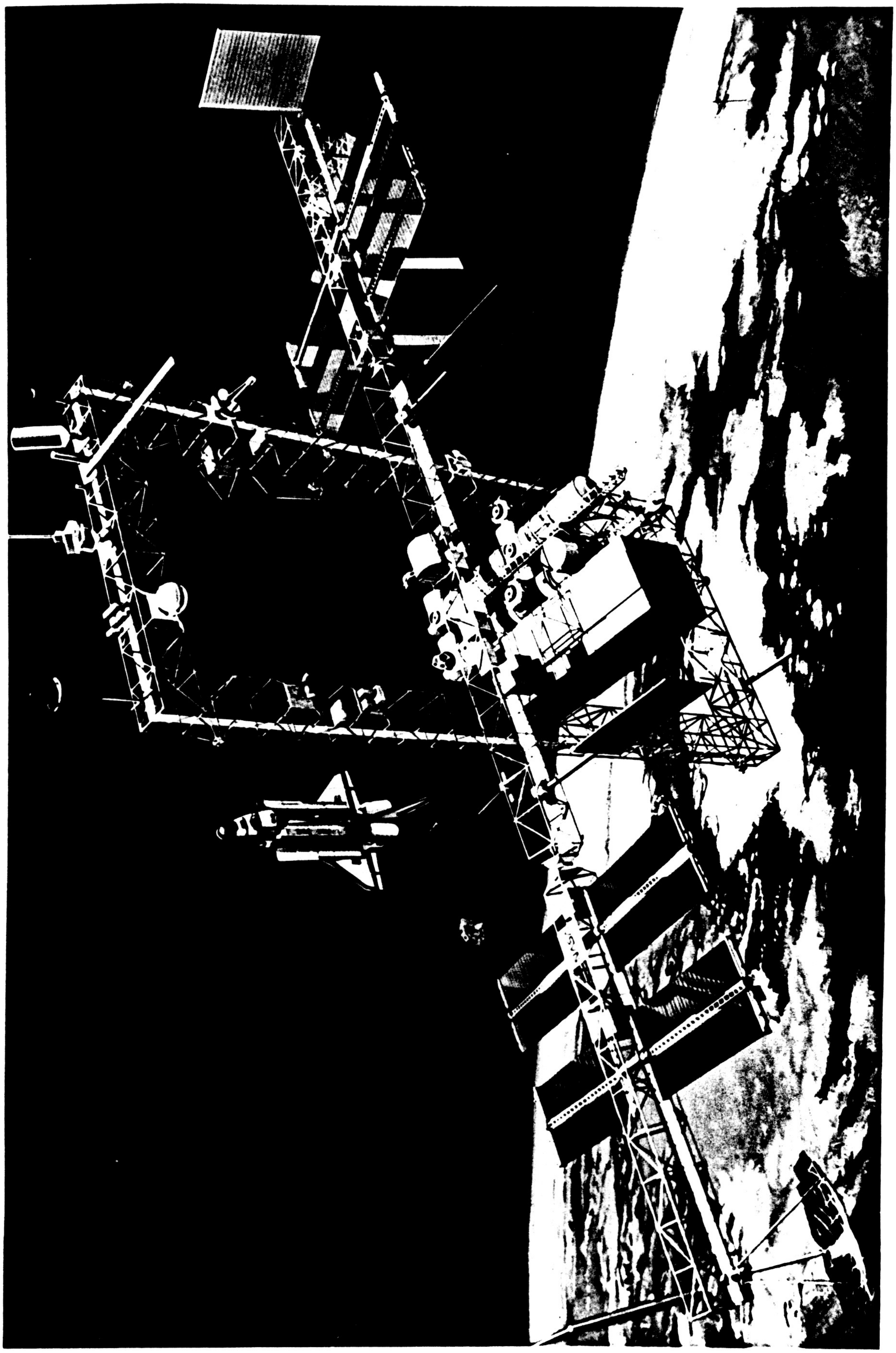
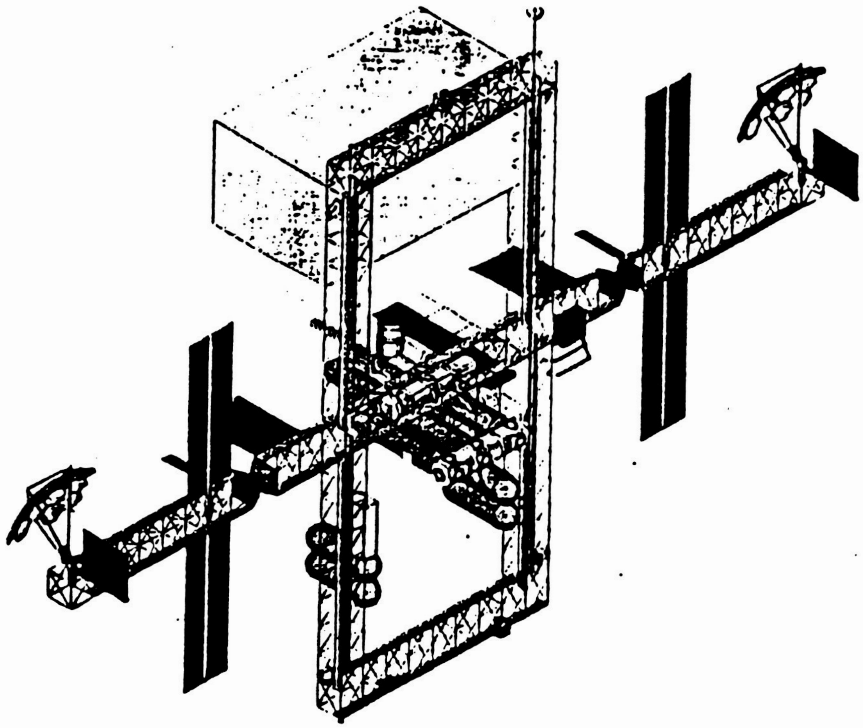


Figure 6.4-1, Space Station Configuration Used by Weidman (1988) for Lunar Support



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A significant amount of the Space Station's already short supply of resources will be required to support lunar missions. The total Space Station support for lunar element assembly, refurbishment, checkout, and verification testing was estimated to exceed 6 crew years/year, use 30 kw of power, and require an additional lab for checkout. The large power requirement was associated with cryogenic space systems management and operations. The station must also provide additional space for assembly, crew time, internal pressurized volume, and crew accommodations and utilities to the lunar support personnel while they are located at the Space Station.

The study concluded that the Dual Keel Space Station could accommodate the lunar base mission activity with a yearly mass to orbit of 1.5 million lbs (682 m tons). HLVs are essential to handle the launch loads with attendant KSC expansion required to handle the HLVs. Because of the large crew requirements, a new crew transport vehicle would be required. It was also concluded that the lunar elements must be designed with modular, self-testing components, with increased reliability, and that automation and robotics must be applied throughout all operations activities for productivity and efficiency.

6.5 Configurations for Lunar and Mars Mission Support

Three different concepts were evaluated in Kaszubowski (1988) as an orbiting vehicle support facility or transportation depot for supporting lunar and Mars missions. The concepts were the Triangular Prism, the Open Box, and the Open Platform concepts.

Since these stations are designed to support Mars missions as well as lunar, they are driven by the Mars requirements, with stack sizes in the range of 1,000 m tons as opposed to 200 m tons and less for lunar. The precise configurations are therefore oversized and not as appropriate for the lunar stacks which require continual launches rather one launch every two years. The configurations are discussed here because they show different overall architectural options, however.

The three depot configuration concepts make use of current Space Station hardware design, i.e., the truss bays are 5 meters square, the solar dynamic, alpha joints, and RCS systems are the same, and the command center and docking ports were taken directly from Station nodes and modules. It is pointed out that the experience gained building the Station is directly applicable to assembly and maintenance of the orbiting depot.

As mentioned previously, current Space Station planning and design must accommodate onboard activities which require a quiescent environment for science, materials processing, and micro-gravity research. The level of support and duration needed at the Space Station for vehicle assembly, fueling, docking, processing, and maintenance/repair activities can potentially produce large dynamic disturbances which is in conflict with a quiescent environment. These two types of activities on the Space Station would have to be scheduled such that one is not compromised over the other. One other possible solution would be to separate the two conflicting activities by moving the lunar support facility to a co-orbiting location with respect to the Space Station. Another solution is a completely separate station. This study in general assumes a separate station.

6.5.1 Triangular Prism Configuration

Figure 6.5.1-1 shows the Triangular Prism configuration with a large manned Mars vehicle enclosed by an equilateral triangular prism truss structure. This configuration has five faces thus allowing hardware to enter or leave through all five faces. The truss structure can be completely covered with thermal and debris protection material. A command center and a docking port are located at the apex of the triangle. The propellant tanks are distributed around the top of the Prism and away from the command center. The triangular section is eleven bays (55m) on each side, and the structure is eleven bays long making a volume of approximately 2,000,000 cubic feet (59,000 cubic meters). Large robotic arms are attached to the structure in support of vehicle operations. Solar dynamic collectors are used for power generation which allows the Triangular Prism to fly earth pointing - the apex always pointing toward the Earth.

6.5.2 Open Box Configuration

The Open Box concept shown in Figure 6.5.2-1 features truss sections arranged in a rectangular box which completely encloses the Mars vehicle during all stages of assembly. The command center is located at the top of the Box and the attached docking port and airlock extend out into the flight path direction. The configuration is open on the front, rear, and top faces, but blocked by a truss structure piece on each side and bottom. Robotic arm access to the vehicle is via the cross pieces, while the vehicle and associated hardware enter or leave the front, rear, or top. The Open Box is twelve truss bays long, nine bays high and nine bays wide. The outside dimensions of the Open Box are 60m x 45m x 45m. The inside dimensions are 50m x 35m x 35m, and the total volume is 2,163,000 cubic feet (61,250 cubic meters). The entire box configuration would be enclosed with thermal and debris protection material which would be opened to provide space for vehicle egress.

6.5.3 Open Platform Configuration

The Open Platform concept shown in Figure 6.5.3-1 is a modification from the Dual Keel Space Station, in that the transverse boom was removed and the keels were rearranged to provide access to the Mars vehicle. The command center and docking port are located on the lower Earth pointing boom structure to facilitate maximum visual viewing to the vehicle. For an approaching vehicle to dock at the docking port, it must travel under the lunar or Mars vehicle and surrounding structure. For balancing purposes, an LO_2 tank is located on the lower boom near the command center.

The rectangle, or platform, which surrounds the vehicle is 60m long and 45m wide. The lower keels, which connect the platform with the lower boom are 40m long. Disadvantages of this configuration are that Robotic arm access to the vehicle is reduced and the vehicle cannot be completely enclosed for thermal and micrometeoroid debris impact protection.

Figure 6.5.1-1, Triangular Prism Configuration, (Kaszubowski, 1988)

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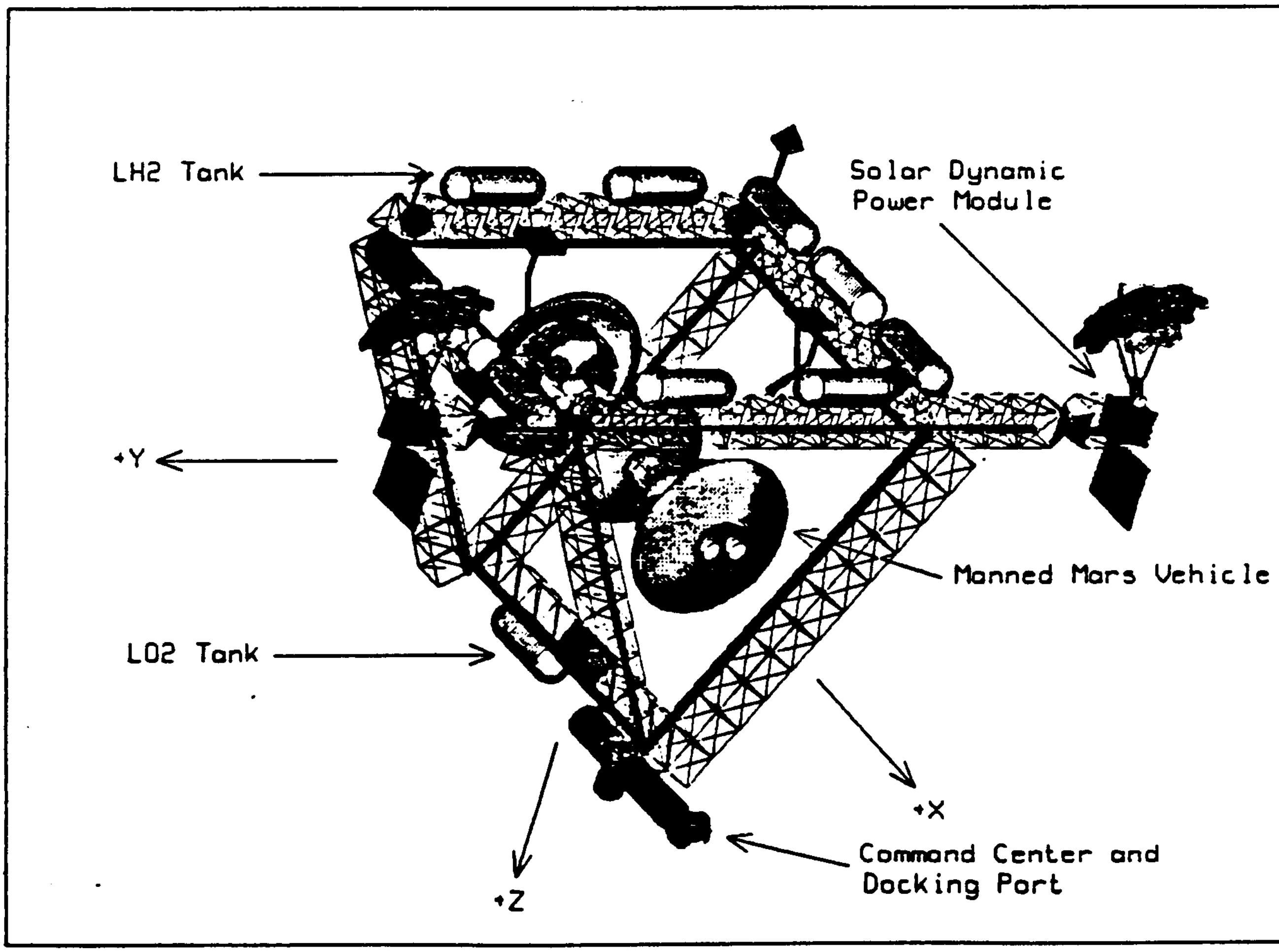

Figure 6.5.2-1, Open Box Configuration (Kaszubowski, 1988)

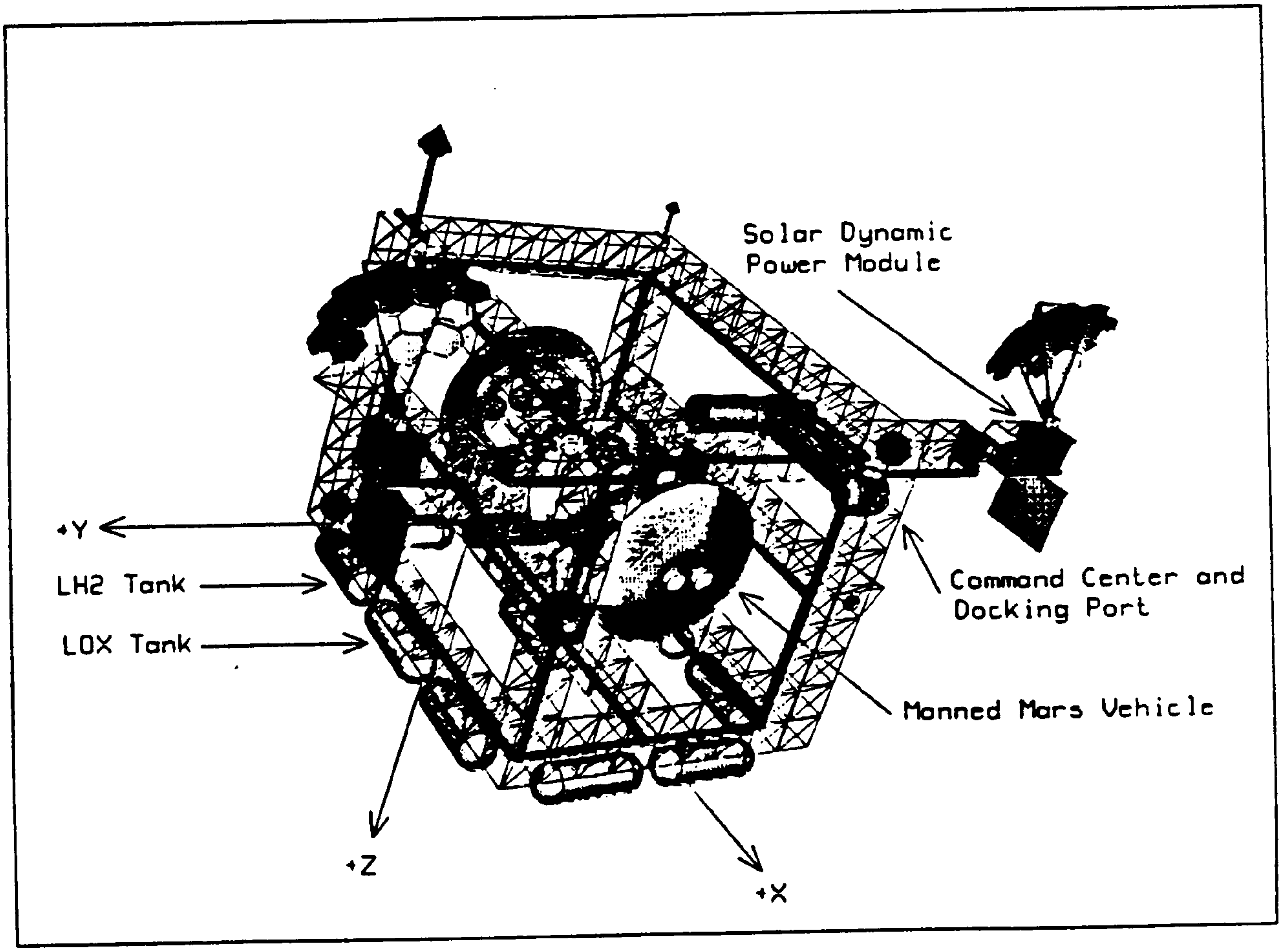

Figure 6.5.3-1, Open Platform Configuration (Kaszubowski, 1988)

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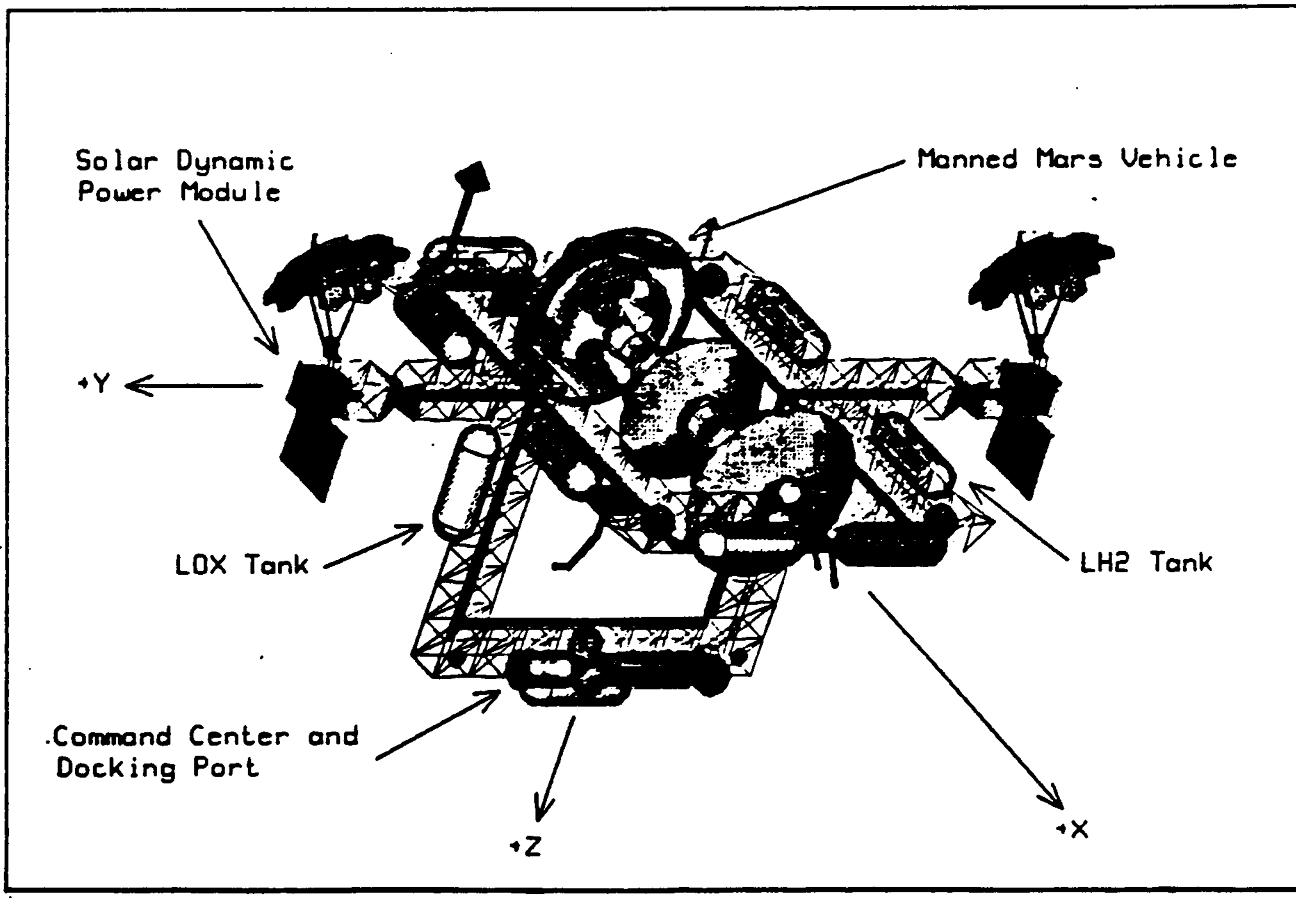


7.0 Conceptual Design of a Space Transportation Node (STN)

The purpose of this effort is to develop a point design configuration for a low earth orbit Space Transportation Node (STN) to support lunar base missions. The assumptions and requirements developed previously in Eagle, 1988 will be used where applicable. The objectives of this effort are:

a. Define a candidate overall configuration concept which meets the revised assumptions and requirements of Eagle, 1988.
b. Define configurations for the flight elements and their locations.
c. Document recommendations and rationale for each flight element.

7.1 Similarities Between the Space Station and the STN

The current NASA Space Station Phase C/D contractual effort for design and operation of a low Earth orbit Space Station includes facilities to service and maintain transportation vehicles such as the OMV and OTV. However, this capability is not envisioned until the Space Station achieves a growth or enhanced capability. During Space Station buildup and subsequent operations the same or very similar functions will be performed that are required for the STN to support Lunar missions. Table 7.1-1 gives a comparison of the major functions required for the Space Station and STN. The functions are listed under five areas: structure, modules, systems, operations, and servicing/maintenance. The Space Station structure serves essentially the same function for the STN except for accommodating external mounted payloads/experiments.

The same type of modules will be needed for the STN excluding the International Modules. The systems for the STN will be similar except for the user community operated commercial and scientific functions. The Space Station provides +5 degrees pointing for Earth and celestial pointing payloads and very low 10^{4} micro-gravity operations. These types of operations are not required for the STN. However, the STN is required to accommodate and operate with a very large HLV. Servicing and maintenance functions will be required for the STN above and beyond the capability of the Space Station to support Lunar missions. In addition to servicing and maintaining the OMV and OTV, the STN will need to accommodate and in some cases provide servicing functions for lunar landers, aerobrakes, HLVs, Space Shuttle Orbiters, cargo, crew modules, and additional crew. The STN must also store large amounts of cryogenic propellants.

Space Station design includes emphasis on commonality and interchangeability of parts and equipment, standardized interfaces, logistics and refurbishment, on-board maintenance and repair, system redundancy, automation and robotics, and an on-going advanced technology program to eliminate system obsolescence. Therefore, there are compelling reasons to utilize Space Station elements, components, and systems where applicable for the STN configuration to keep cost within reasonable limits and reduce crew training, procedures, and operations requirements.

Table 7.1-1, Space Station and STN Functions

Hardware/Functions	Freedom Station	STN
<u>Structure</u>		
• Provide structure to accommodate Modules and Mobile Manipulators	X	X
• Provide structure for mounting external experiments and payloads	X	
• Provide attachments for docking/berthing	X	X
• Provide attachments for modules	X	X
• Provide micro-meteoroid impact protection for vehicles, crew, and propellant	X	X
<u>Modules</u>		
• Pressurized volumes for crew habitation	X	X
• Docking interfaces	X	X
• Airlocks for EVA	X	X
• Logistics resupply modules	X	X
• Accommodate international modules	X	
• Pressurized command center	X	X
<u>Systems</u>		
• Provide continuous power for commercial and scientific functions	X	
• Provide thermal control, computer control, etc. for customer or user payloads	X	
• Provide house-keeping and orbital operations systems	X	X
• Provide thermal protection for EVA crew and propellant	X	X

Table 7.1-1, Space Station and STN Functions

Hardware/Functions	Freedom Station	STN
Operations		
• Capability to operate with the Shuttle, OMV, OTV	X	X
• Pointing capability for celestial and Earth viewing payloads	X	
• Provide micro-gravity operations for scientific and materials processing functions	X	
• Capability to dock and operate with HLV	X	X
• Provide corridor for vehicle arrival/departure	X	X
• Contamination control for payloads/experiments	X	
Servicing/Maintenance		
• Accommodations to service and maintain OMV, OTV	X	X
• Accommodation to service and maintain reusable lunar lander		X
• Store, maintain, service, assemble, and reconfigure payloads	X	
• Capability to service free-flyer platforms and attached payloads	X	
• Provide for very large propellant storage, and transfer capability		X
• Assemble large lunar stacks in a protected environment and enter crew module IVA		X

7.2 Configuration

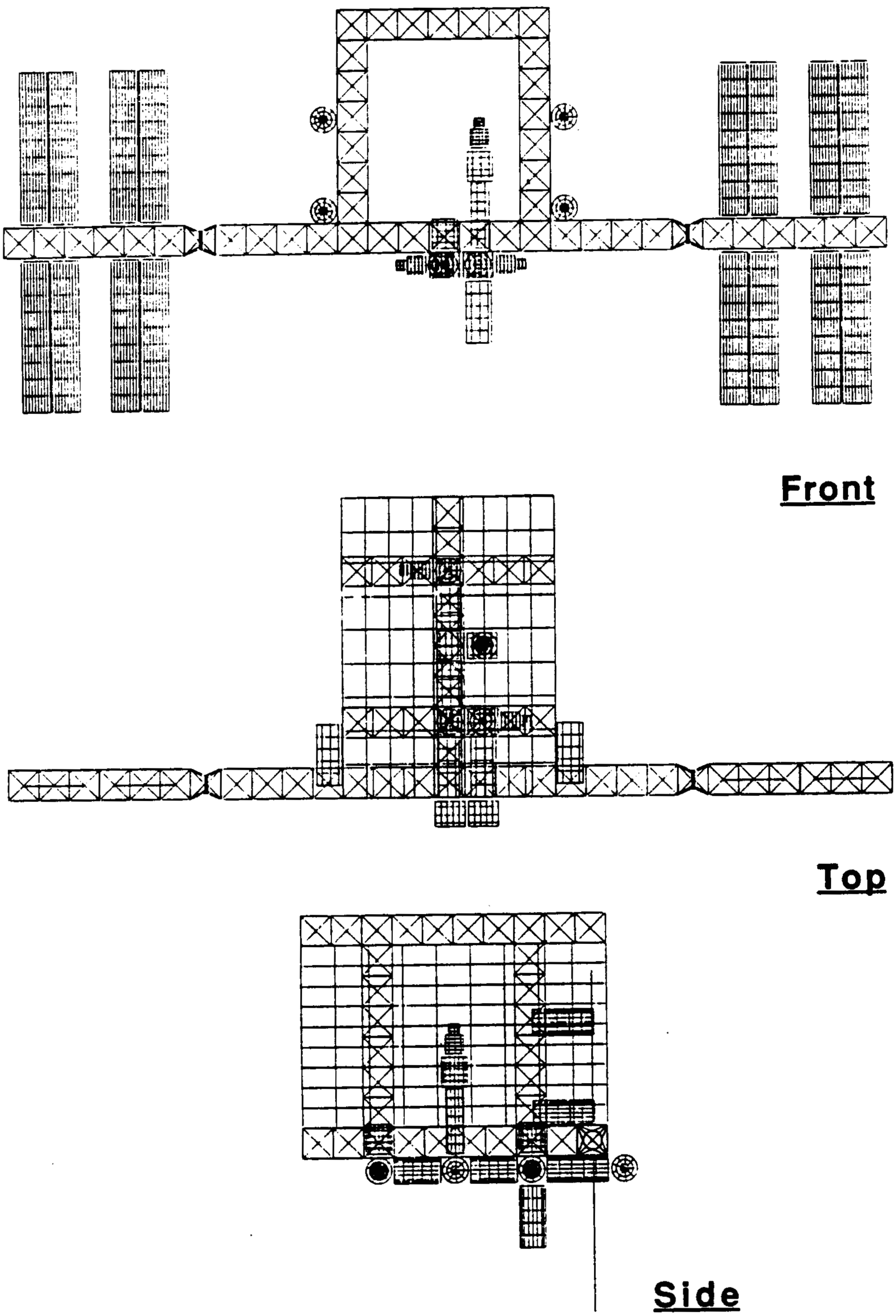
The station must accommodate two stacks, with pressurized access to each. It must protect these stacks from micrometeoroids and orbital debris, and maintain thermal control. In a failure situation a stack might conceivably have to sit at the station, ready to go, for as long as six months. The requirement for pressurized access to both vehicles means they must be located near the pressurized modules. As explained in section 7.4, protection from micrometeoroids and orbital debris will be costly in terms of mass and drag and the size of the hangar must be minimized. Because of these factors, a configuration with a minimum frontal area box on top of a somewhat revised phase 1 Freedom Station was chosen. Figure 7.2-1 illustrates this configuration. The rectangular box structure minimizes the frontal area and makes construction simple.

Flight schedules, numbers of vehicles located at the STN at any one time, propellant storage, assembly, servicing, maintenance, operations, and crew size all must be considered to provide adequate facilities including growth capabilities. Recent Eagle studies indicate that the maximum number of space vehicles which will be located at the STN at any one time are: A Space Shuttle Orbiter, 2 Orbital Maneuvering Vehicles (OMV), 2 single-stage Orbital Transfer Vehicles (OTVs), 2 reusable single-stage lunar lander/launchers and two crew modules. 158 tons of LO_2 and LH_2 propellant is needed for a piloted round trip lunar mission. 151 tons of propellant is needed for a 25 m ton cargo landing. 182 tons will be stored in four special tanks at the STN. Additional storage capacity is available in an OTV/lander stack (158 m tons) and HLV tankers (75 m tons). In an optimum situation, launch of one stack with a crew on board would not occur unless sufficient propellant was also on hand to launch the second stack on a rescue mission.

An HLV will be required to deliver large amounts of propellant from Earth to the STN. The HLV will rendezvous and station-keep with the STN. An STN-based OMV will rendezvous with the HLV and aid in hard docking it to the STN. The HLV propellant will then be transferred directly to the vehicles or to the storage tanks. The empty HLV upper stage and propellant tank set will then be deorbited by the OMV. The ability to dock and undock HLVs with the orbiter docked to the STN at the same time is an advantage. For this reason, HLV docking on top of the hanger is proposed. More than one HLV cargo may conceivably be docked at once in this location, increasing short term propellant storage capacity.

The requirement to house up to 13 crew at once leads to two habitation modules. Two workshop/storage modules are also proposed. The workshops are 2/3 normal module length in order to position the rotating fixtures and the hangar node/cupola in the proper position to work two stacks with the hangar cupola as close to both as possible.

Figure 7.2-1, Space Transportation Node Configuration



7.3 Hangar

The major new element of the STN is the hangar facility. The STN is designed principally to support stacking, refurbishment, and propellant loading operations within this hangar.

7.3.1 Hangar Description

The hangar element is unpressurized and is covered to provide protection to the spacecraft from orbital debris, micrometeoroids, and solar flux. Two hangar doors allow easy access to the interior. The hangar measurements are 164 feet (50 m) long by 115 feet (35 m) wide by 82 feet (25 m) high. This allows sufficient volume (1,546,520 ft^3 or 43,750 m^3 to contain 2 RMS's, 2 single stage OTV's, 2 OMV's, a crew module, one cargo, two lunar landers, and servicing fixtures. Figure 7.3.1-1 shows the hardware envelope area inside the hangar. Hardware arrangement within the hangar allows a minimum of six feet around each piece of hardware for a suited EVA astronaut to perform maintenance and operations functions. The length and width of the hangar is determined by the area required to rotate the two stacks 360° and clearance to move parts. The height is determined by the length of the stack and clearance required to pick it up.

The hangar control station location is shown in the figures. Sufficient space is also needed to allow RMS interface with the spacecraft. Figure 7.3.1-2 shows a front view of the hangar and pressurized module interface. The height of the hangar was dictated by the requirement to transfer the lunar crew IVA between the lander crew module and the STN pressurized flight elements. The crew module is shown located on top of the rotating fixture for crew transfer. Sufficient space is allowed for the RMS to move the whole stack from above or to position the crew module on the rotating fixture or mate it to an OTV.

Figure 7.3.1-3 shows the pressurized module interface with the hangar facility. The pressurized control module penetrates the hangar such that the control section is located within the hangar, between the two stacks, with equal access to both.

Figure 7.3.1-4 shows the standard 5 meter bay truss structural arrangement and RMS movement paths. Numerous translation paths are provided within the hangar for the RMS. The structural arrangement also provides mounting support for the hangar and the pressurized modules.

The propellant storage system consists of four heavily insulated, independent, and micrometeoroid/debris protected tank sets. Each contains an oxygen and hydrogen tank. The storage systems are located on the upper and lower booms on each side as shown in figures. The hangar and its associated support equipment are located as close as possible to the STN composite C.G. to minimize C.G. movement when propellant is transferred and OTV's and payloads are mated and moved during mission operations. Propellant can be pumped to control c.g. location, at the expense of boil-off which increases dramatically during pumping operations. Locating multiple, independent shielded tanks outside the hangar reduces the hangar frontal area and the risk of explosion or puncture losses.

Figure 7.3.1-1, Hardware Envelope, STN, Top View

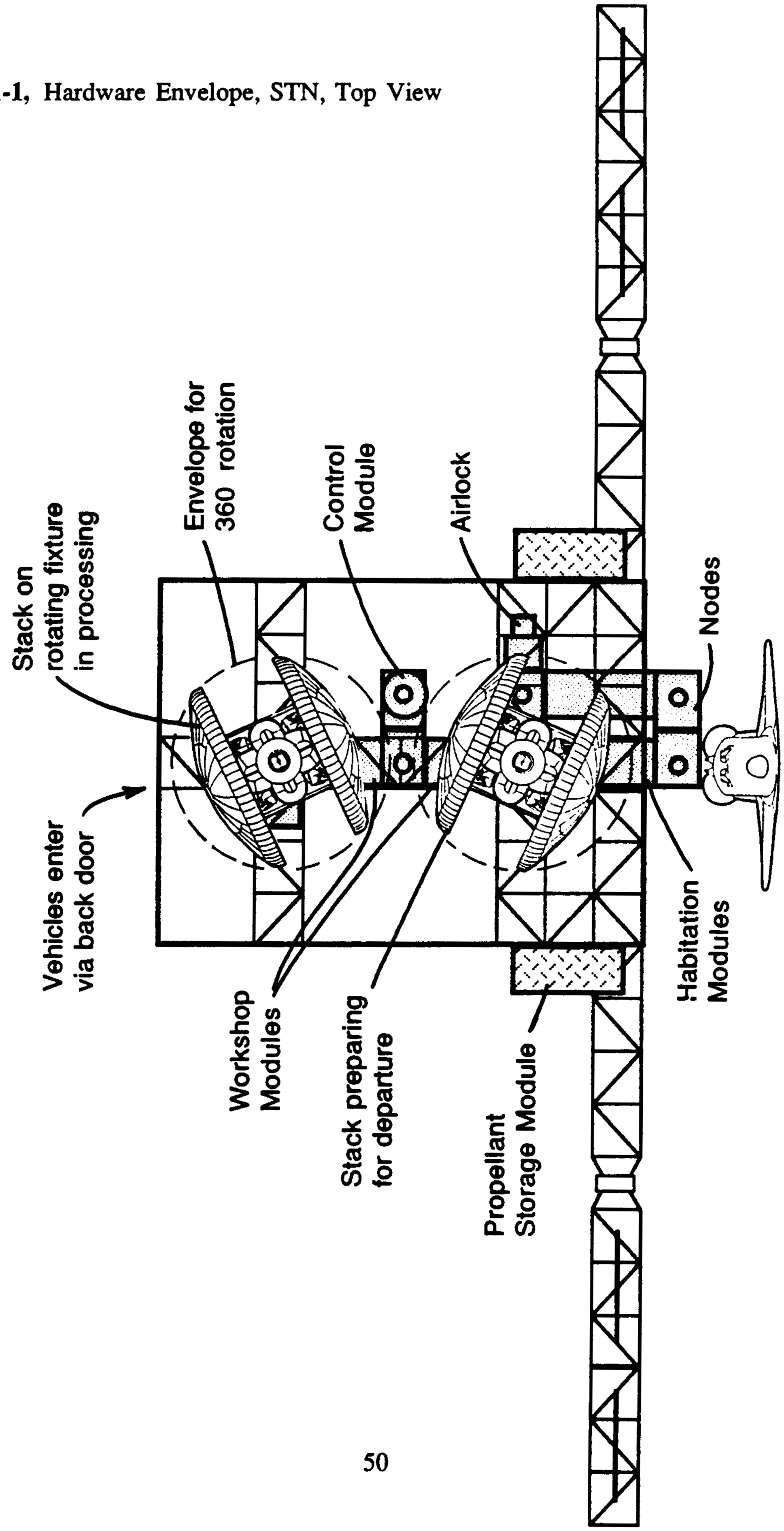


Figure 7.3.1-2, Hardware Envelope, STN, Front View

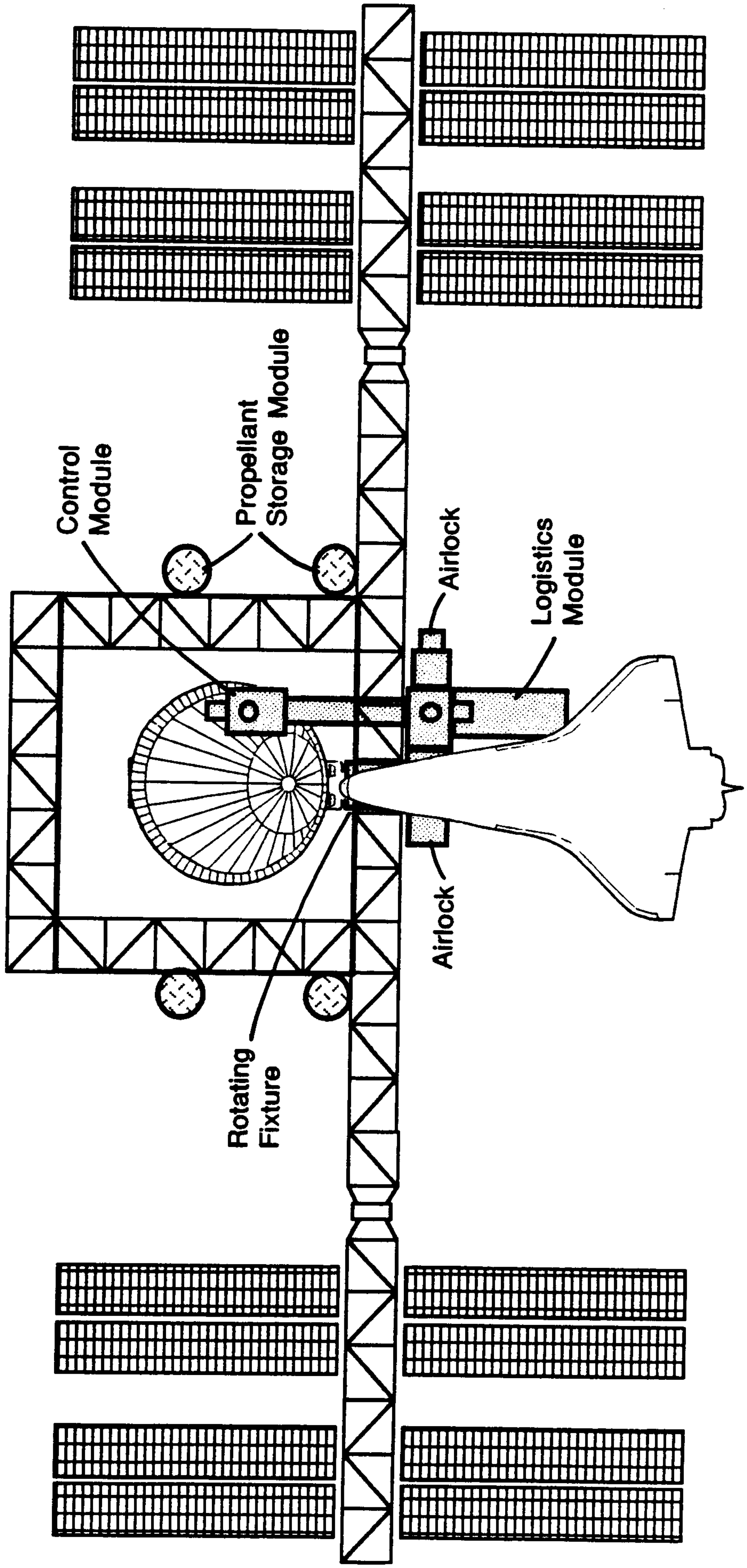


Figure 7.3.1-3, STN, Side View

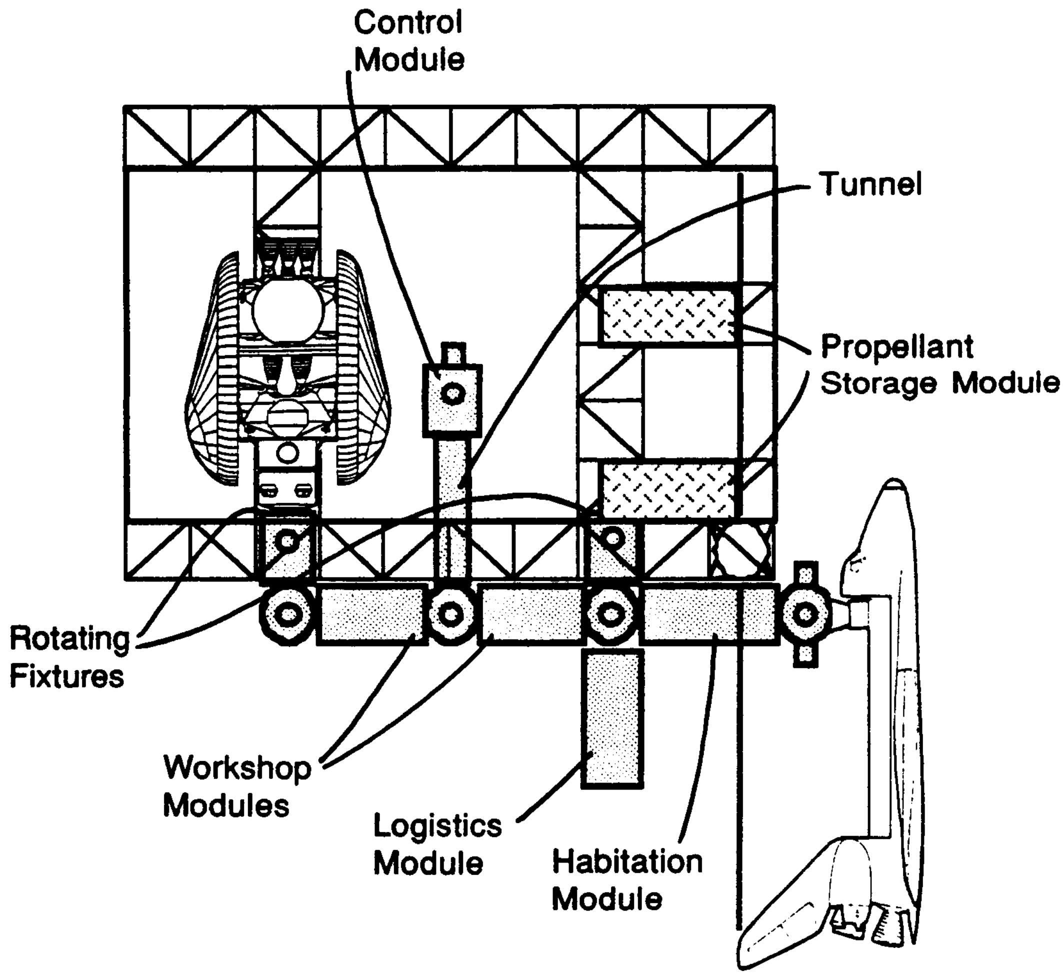
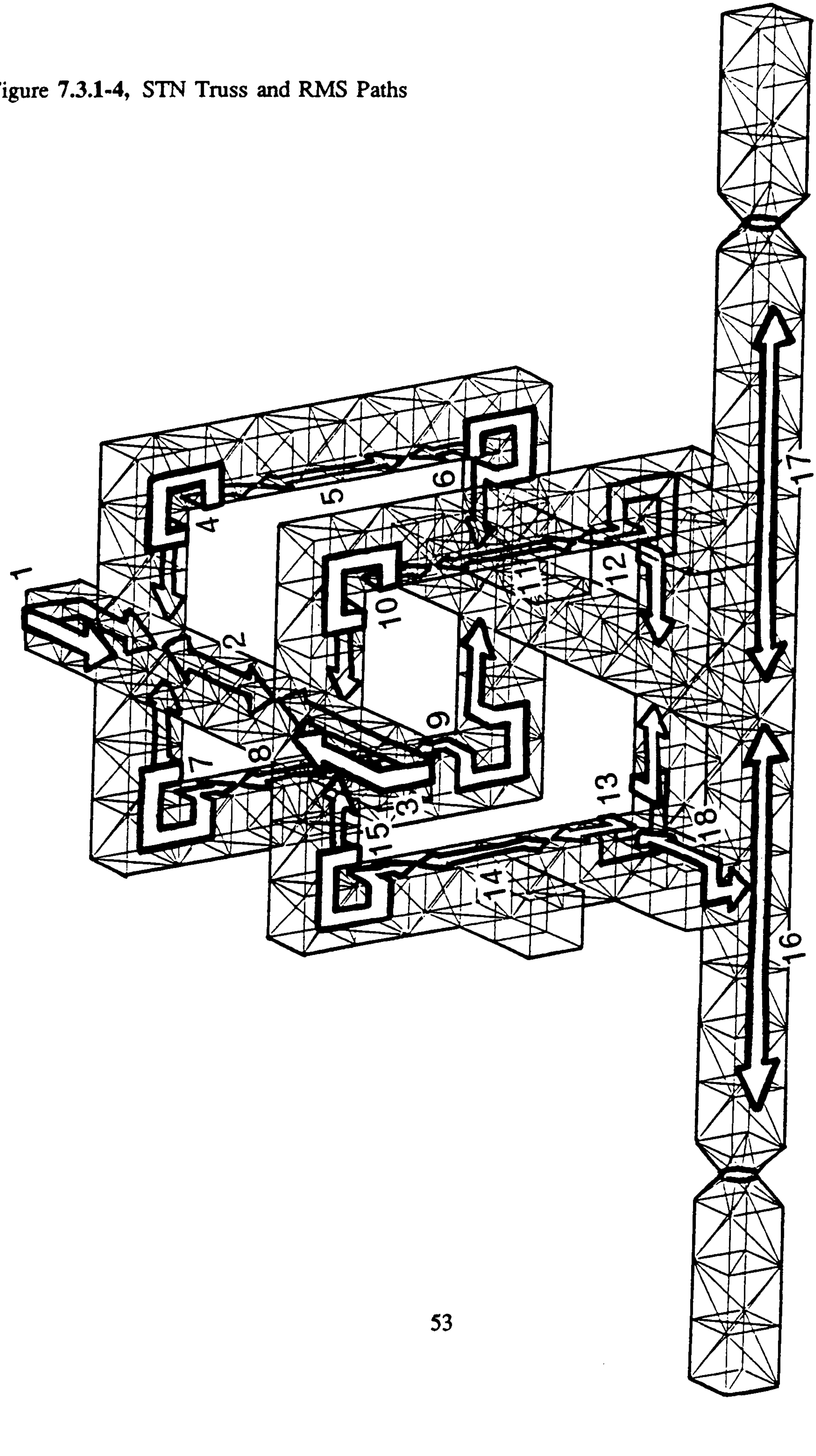


Figure 7.3.1-4, STN Truss and RMS Paths



7.3.2 Hangar Operations

This STN is designed primarily to make operations on-orbit as easy as possible. Other concepts may have different first priorities, such as minimum cost, weight, etc. The following sequences describe the retrieval/stacking/deployment process in the hangar. Other operations such as orbiter docking and departure, or HLV tanker, or cargo docking and disposal are independent of the operations in the hangar (except for obvious dependencies, such as having a crew and propellant). HLV and orbiter docking locations, propellant storage capacity and general hangar configuration were chosen to aid this independence. Operational flexibility is therefore increased, and the probability of one failure bringing down the whole system decreased somewhat.

The lander and OTV arrive from a mission independently, have their aerobrakes removed, and are stacked on the aft rotating fixture. The aerobrakes are hung on the port and starboard interior walls, reworked if required, and then placed on the stack. In a nominal sequence, the ready, but empty stack is then moved to the forward rotating fixture and loaded with propellants, though, it may also be loaded with propellants on the aft fixture and depart out the aft door as well, if for some reason another stack must remain on the forward rotating fixture. The nominal throughput is therefore entry of the returning vehicles aft and departure of the stack through the forward door. This has two advantages: 1) The mass of the fully loaded stack is kept closer to the plane of the forward door, reducing c.g. travel, and 2) departure of a single loaded, checked-out stack by RMS release is more acceptable in the close vicinity of the orbiter on the forward end than arrival of two, perhaps unmanned vehicles for RMS capture after a mission.

A significant docking operations issue concerns RMS or OMV usage to achieve docking. Three options often discussed include: 1) The OMV is deployed and docks with a vehicle in the vicinity of the STN, such as an HLV tanker, OTV, or lander, and acts as a tug to fly or help fly it to a hard dock with a station docking fixture, 2) The vehicle flies itself to within RMS range and the RMS grabs it and docks it as required, or 3) The vehicle flies itself to a hard dock with an STN docking fixture. The correct general procedure is not clear to the authors at present, but it is assumed that 1) above will be used for the HLV tanker, and 2) will be used for the lander and OTV in the following sequences.

RMS travel throughout the hangar raises several issues. Current mobile RMS concepts cannot turn an inside corner as shown on paths 4, 7, 9, 6, 10, 12, 13, and 15. To turn the inside corners the current concepts must turn two outside corners and make three 90° sideways turns as shown in detail in Figure 7.3.1-4. Long RMS movements are therefore complicated. The RMS in this scenario may be much more frequently used than in the Phase 1 and 2 Freedom Station and the translation mechanism proposed for the Freedom Station may therefore be too complicated to be practical for this application. Other concepts using rails or tracks may prove necessary for this type of usage.

The number of RMS units required is another issue. A specially arranged path such as path 18 in Figure 7.3.1-4 will allow the transverse truss RMS to enter the hangar. A minimum of two RMS units seems to be required for easy operation. Other optional techniques using lines and pulleys can be used to suspend or fix vehicles in the hangar while things are being taken off. These techniques can also serve as back-ups in failure situations. A third RMS reduces the length of traverses and adds redundancy. The sequence in section 7.3.2.3 assumes three RMS units though it could also be done with two if path 18 is available.

The optimum propellant loading sequence occurs when the vehicle on the forward rotating fixture is loaded with propellant and sufficient propellant is also available in the HLV tankers to load a second stack should a rescue mission be required. The forward stack would not depart until this propellant was on-hand. This can be achieved by filling the STN storage tanks (enough for one stack plus some) and then waiting until two HLV tankers are docked to the forward and aft upper docking fixtures. Two HLV tankers carry almost enough for one stack given the vehicles assumed. Both stacks can then be filled and launched in rapid sequence if required. Various other propellant loading sequences are possible using lander or OTV tankage for storage or adding additional storage capacity to the STN. This is discussed in section 7.5.

Numerous off-nominal situations must be addressed as well as the nominal sequences described in the following sections. Major failures involve inability to launch a stack for some reason or equipment failure. All these cases have not been examined in detail. The hangar concept uses room to maneuver and redundancy (1 additional unit) for most major items or paths to cover these failures.

7.3.2.1 Nominal Piloted Mission Sequence

1) Lander with crew module and aerobrake flies within RMS range near aft door. RMS A is located on path 1 (see Figure 7.3.1-3).

2) RMS A grapples vehicle and brings it in aft door on translation path 1.

3) RMS B on path 5 grapples lander aerobrake. Lander releases aerobrake. The lander is required to attach and detach automatically from its aerobrake in LLO.

4) RMS B moves aerobrake to attachment fixture on port wall via path 5. Aerobrake is remotely attached and RMS B releases it.

5) RMS A docks lander and crew module to aft rotating fixture. Flight crew departs. Refurbishment crew enters.

6) Scheduled inspection and maintenance, and required repair activities on the lander and aerobrake begin. Inspection and repair can be performed using dexterous manipulators from the hangar cupola. RMS B can access all the lander via path 5.

7) RMS B moves via path 5, 4, and 7 to path 8. OTV with aerobrake arrives within RMS range.

8) RMS A grapples vehicle and brings it in aft door on translation path 1.

9) RMS B on path 8 grapples the OTV aerobrake. The OTV releases the aerobrake or it is released by EVA or teleoperation.

10) RMS B moves the aerobrake to an attachment fixture on the starboard wall via path 8. The aerobrake is remotely attached and RMS B releases it.

11) RMS A stacks OTV on lander. OTV is automatically attached to lander. Note- lander and OTV must separate, attach, and then separate again in LLO.

12) Scheduled inspection and maintenance and repair activities on the OTV and aerobrake begin.

13) When inspection, maintenance, and repair activities for the lander, OTV, and two aerobrakes are complete, RMS B on path 8 grapples the OTV aerobrake on the starboard wall and places it on the stack. It is connected to the stack.

14) RMS B then translates via paths 8, 7, and 4 to path 5. RMS B grapples the lander aerobrake on the port wall and places it on the stack where it is connected.

15) Some integrated testing is performed and the stack is ready for propellant loading.

16) The stack on the forward rotating fixture is assumed to have departed. RMS A grapples the aft stack, takes it off the aft rotating fixture, and moves it along translation path 2 to the forward rotating fixture.

17) Remotely operated propellant loading lines attach to the lander and then the OTV. Propellant is loaded into both.

18) The flight crew comes on board. The forward hangar doors are opened. RMS A moving along translation path 2 and 3, grapples the stack, and deploys it out the front door.

7.3.2.2 Nominal Cargo Mission Sequence

1) Forward hangar doors are opened and OMV A is deployed from its storage position on the forward port vertical truss. The OMV may deploy itself or be deployed by an RMS using paths 11, 10, and 3.

2) The OMV remotely docks with a cargo/HLV upper stage in the vicinity of the STN and brings the vehicle to the forward docking fixture on top of the hangar. The cargo is then docked to this fixture.

3) The HLV upper stage is then detached from the cargo and deorbited by the OMV. The OMV then returns to the hangar and is placed in its dock by an RMS.

4) RMS A travels via path 2 and 3 to the forward upper docking fixture and grapples the cargo.

5) RMS A returns via paths 3, 2, 4, and 5 to the aft rotating fixture and docks the cargo to this fixture.

6) A lander and aerobrake without crew module flies within RMS range near the aft door. RMS B on path 1 grapples the vehicle and brings it in the aft door.

7) RMS A on path 5 grapples the lander aerobrake. The aerobrake detaches from the lander.

8) RMS B then docks the lander to the cargo on the aft rotating fixture.

From this point on, processing of this stack is similar to that described in section 7.3.2.1. Another option for this sequence would be to use an aft upper docking fixture to dock the cargo.

7.3.2.3 Cargo from the Shuttle Payload Bay

1) Shuttle docks with STN at a forward node docking fixture. Shuttle RMS removes the cargo from the payload bay and hands it to RMS C on path 16 or 17.

2) RMS C hands cargo to RMS A on path 12 or 13.

3) RMS A moves cargo to aft fixture via paths 13, 14, 15, 2, 7, and 8 or 12, 11, 10, 2, 4, and 5 or other routes.

4) RMS A docks cargo to aft rotating fixture.

From this point on processing is similar to that described in the previous sequences.

7.4 Transportation Node Meteoroid and Orbital Debris Shield Design

The large frontal and surface area of the hangar makes collisions with relatively large pieces of orbital debris much more probable. Meteoroid and debris shielding therefore becomes a major issue in the overall design. Numerous items on the STN will require shielding work: the modules, propellant storage, and the vehicles. Shielding weights for the modules and storage tanks are accounted for by using space station type designs as discussed in later sections. The hangar that protects the vehicles is potentially the most massive shield and is therefore addressed in detail here.

7.4.1 Summary of Initial Work

The following points summarize the results from initial work on meteoroid and debris shielding for the hangar.

1) Mass of a meteoroid/debris shield for the transportation node hangar depends on: (1) Lifetime, (2) Surface area, (3) Failure criteria, and (4) Acceptable impact reliability. Baseline parameters for this study are: (1) 10 year lifetime, (2) 7,750 m^2 exposed area (25 m x 35 m x 50 m hangar), (3) Failure is defined as a penetration of the one

or more bumpers by a projectile that has not been completely disrupted (shocked) by the shield, and (4) 50% reliability from impact failure (1 chance in 2 of receiving an impact over 10 years from a large enough particle that will not be completely disrupted by the shield). Completely disrupted typically means the projectile is vaporized or melted for hypervelocities above 7 km/sec.

2) Required shield mass to satisfy these requirements is estimated as 22 metric tons (single wall bumper only). The shield will protect 5 sides of the hangar and will vary in impact resistance depending on the hangar surface. Because orbital debris is highly directional, the heaviest shielding is required on the forward doors and sides of the hangar. Protection for the top and back surfaces can be less because primarily only meteoroid impacts are received on these surfaces. No shielding is required on the bottom surface because the Earth shields the hangar bottom surface from most meteoroid impacts (>80% of the meteoroid flux at 500 km) and the flux of out-of-horizontal-plane orbital debris is negligible. It is assumed that the transportation node orientation is controlled and fixed in an Earth-pointing mode.

3) A dual-wall shield or bumper is proposed, utilizing a flexible ceramic material or metallic mesh outer bumper separated by 5 cm from a rigid graphite/epoxy second bumper. Superior performance (in terms of breaking up impacting projectiles and protecting underlying structures) of this concept was demonstrated in experimental hypervelocity impact tests compared with a single-plate aluminum bumper (Christiansen, 1987). All mass estimates for shields or bumpers in this report are conservatively based on an aluminum bumper however (Al 6061-T6 as baselined for Space Station pressurized modules). Software was available to design aluminum bumpers and the process is well understood. Initial work indicates the flexible ceramic or metallic mesh outer bumper and rigid graphite/epoxy inner bumper design will weigh on the order of 25% less than the aluminum counterpart. This 25% has not been removed from the aluminum numbers in this report however.

4) It is recommended that the hangar/shield area be reduced as much as possible to reduce shield mass, exposed area to impacts, and atmospheric drag (or to lower operating altitude which will reduce the debris flux). Smaller dedicated shields for particularly vulnerable elements may be substituted for a large hangar structure protecting the entire spacecraft. To avoid penalizing spacecraft payload, the dedicated shields should be attached to the transportation node structure, not to the spacecraft, or should be removed prior to spacecraft departure.

5) For the baseline hangar configuration, to reduce mass it is recommended: (1) That lower impact damage reliability be accepted than that used for Space Station hardware (to achieve a 0.95 reliability, as used for Space Station hardware design, of completely shocking all impacting particles over 10 years, single wall aluminum shield mass would increase to 187 metric tons for all 6-sides), (2) Consider eliminating the Nadir facing shield (saves 6.9 metric tons, 23% over 6-sided shield), (3) Use variable thickness shields on the different sides of the hangar (saves 1.8 metric tons, 8%, for 5-sided, 50% reliable shield), (4) Use flexible shield materials to ease launch vehicle manifesting and on-orbit deployment.

6) It should be noted that a one or two wall bumper or shield alone only fragments and disperses an impacting particle. Although this reduces the chance that the impacting particle can penetrate underlying structures, the resulting debris cloud could present hazards to particularly vulnerable objects in the hangar bay (such as EVA personnel). If it is decided to eliminate this hazard, additional shielding must be added to completely stop all fragments. An aluminum backwall structure, with a >20 cm standoff from the bumper or shield, having a 50% reliability factor for stopping all meteoroids/debris over 10 years, is estimated to mass 111 metric tons (3.1 mm thick A12219-T87) in addition to the shield mass.

7) Additional experimental hypervelocity impact tests are recommended to further define and develop the optimum shielding/backwall concept. Additional analytical studies are recommended to better define hypervelocity impact failure modes and required impact reliability.

7.4.2 Meteoroid Environment

The NASA recommended meteoroid model (NASA, 1987 and Vaughan, 1985) was used in the impact assessment. The average near-Earth meteoroid flux, F_{met} (impacts/year/m^2 surface area), with mass M_{met} (g) and larger is given by:

for $M_{met} >= 10^{-6}$ g,

$$\text{Log}(F_{met}) = -1.22\ \text{Log}(M_{met}) - 6.911 \qquad \text{(Eqn. 7.4-1)}$$

for $M_{met} < 10^{-6}$ g,

$$\text{Log}(F_{met}) = -0.063\ (\text{Log}(M_{met}))^2 - 1.58\ \text{Log}(M_{met}) - 6.841 \qquad \text{(Eqn. 7.4-2)}$$

The meteoroid flux is assumed omnidirectional although recent work (Zook, 1986) indicates that directional dependence exists for meteoroid impacts on an orbiting object. A higher impact flux is expected on the forward surface (in direction of flight) as viewed from the object. However, because meteoroid directionality has not been incorporated in the flux models, omnidirectionality in the local horizontal plane was assumed. Earth provides partial shielding from meteoroids, and a multiplicative factor expressed by the shielding factor, SF, is used to compensate the meteoroid flux for this effect:

$$SF = (1 + \cos(\arcsin(R/(R+H))))/2 \qquad \text{(Eqn. 7.4-3)}$$

The shielding factor (0.67 @ 500 km) depends on the distance, R, from the Earth's center to the top of the atmosphere, and the altitude, H, of the orbiting object above the atmosphere. Since the atmosphere is defined as 100 km above the surface, $R = R_e + 100$ and $H = h - 100$, where R_e is the Earth's radius (6,378 km) and h is the orbital altitude above the Earth's surface. Because meteoroids are attracted by the Earth's gravity field, the meteoroid flux is also factored by an Earth defocusing factor, DF (0.97 @ 500 km), which depends on the ratio, r, of the distance from the orbiting object to the Earth's center in units of Earth's radius, ie.: $r = (R_e + h)/R_e$.

$$DF = 0.568 + (0.432/r) \qquad \text{(Eqn. 7.4-4)}$$

Meteoroids are assumed spherical with a typical density of 0.5 g/cc for particles greater than 50 microns in diameter and 2 g/cc for particles less than 50 microns in diameter. (Vaughan, 1985). Average collisional velocity for meteoroids is 20 km/sec. The number of impacts, N_{met}, from particles with a given mass and greater is related to the impact flux, surface area, A (m^2), and time, t (yrs), by:

$$N_{met} = F_{met} * SF * DF * A * t \qquad \text{(Eqn. 7.4-5)}$$

7.4.3 Orbital Debris Environment

The 1990's predicted orbital debris environment (NASA, 1987 and Kessler, 1984) was used in the impact assessment. The orbital debris flux, F_d, is defined as the number of impacts from particles with diameter, D (cm), and greater per surface area, A (m^2), per year on a randomly oriented surface. The flux of debris particles with diameter less than 1 cm on spacecraft at 500 km altitude and 30° inclination is given by:

$$\text{Log } F_d = -2.52 \text{ Log } D - 5.46 \qquad \text{(Eqn. 7.4-6)}$$

The debris flux with diameter greater than 1 cm is:

$$\text{Log } F_d = 0.352 (\text{Log } D)^2 - 1.358 \text{ Log } D - 5.46 \qquad \text{(Eqn. 7.4-7)}$$

The flux of less than 1 cm particles at 400 km is:

$$\text{Log } F_d = -2.42 \text{ Log } D - 5.82 \qquad \text{(Eqn. 7.4-8)}$$

For altitudes between 400-500 km, a logarithmic interpolation is used:

$$\text{Log } F_d = \text{Log } F_{d(500)} - (500 - h) * [\text{Log } F_{d(500)} - \text{Log } F_{d(400)}]/100 \qquad \text{(Eqn. 7.4-9)}$$

The total number of debris impacts, N_d, is calculated from:

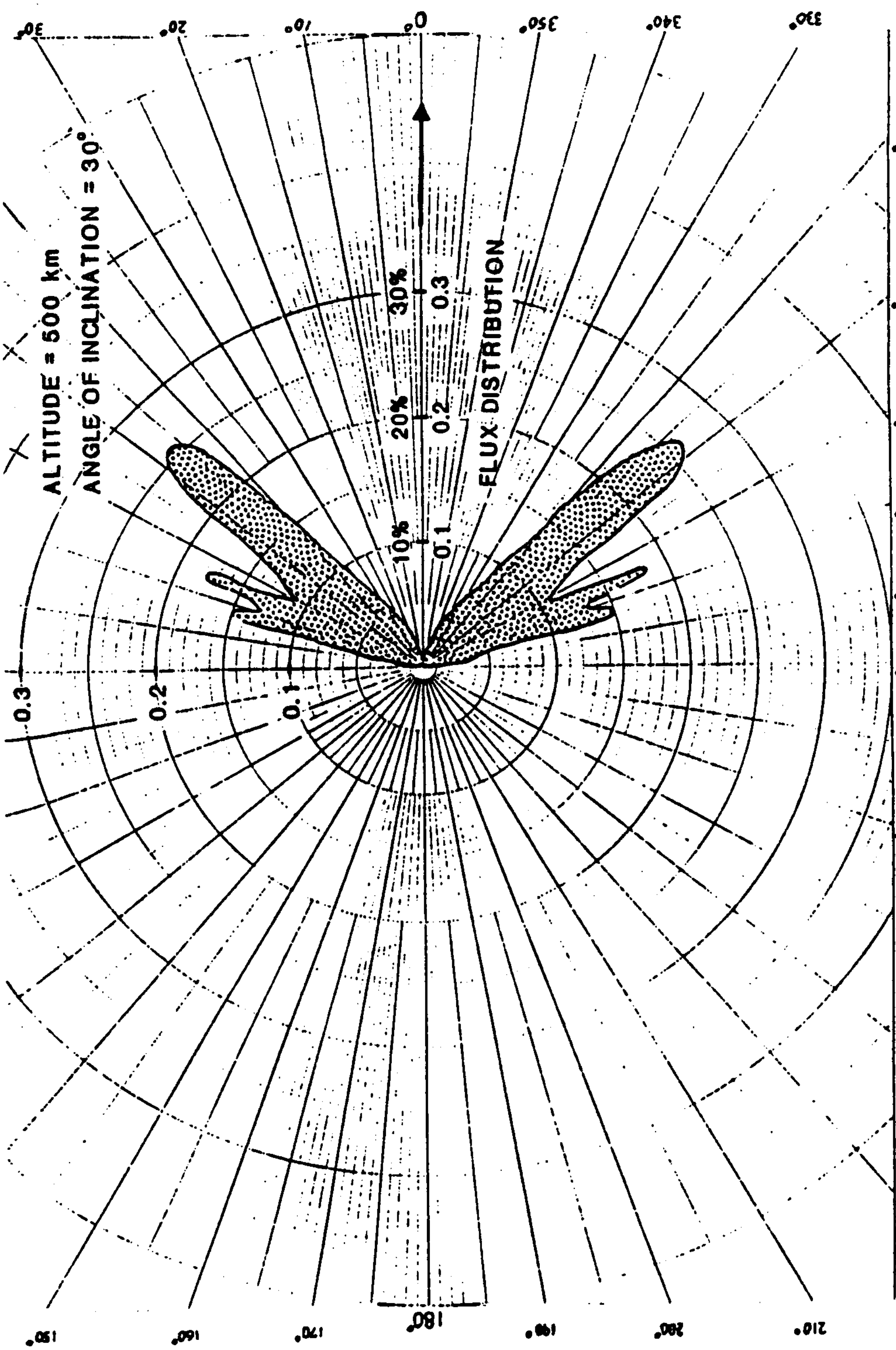
$$N_d = F_d * A * t \qquad \text{(Eqn. 7.4-10)}$$

where A is the surface area (m^2), and t is the exposure time (yrs).

Debris particles smaller than 1 cm are assumed spherical with an average mass density defined as 2.8 g/cc (expected to be the same as aluminum). Orbital debris velocity ranges from 0-16 km/sec with an average collisional encounter speed of 10 km/sec. Debris particles are highly directional, appearing to an orbiting object to approach from directions in a 180° arc centered on the spacecraft's velocity vector (forward direction), with most concentrated in a region extending 30°-70° right and left of the direction of flight as given in Figure 7.4.3-1. Impacting debris objects are almost entirely confined in a plane parallel to the Earth (typically +/-3° from local horizontal), since debris objects intersecting the flight path with elevation angles greater than ~10° to local horizontal will enter the atmosphere. If the position of the orbiting facility is fixed, the directional nature of debris will produce a greater impact density on forward and side surfaces of the facility.

Figure 7.4.3-1, Spatial Distribution of Orbital Debris Flux (Kessler, 1984)
(30° inclination circular orbit, 500 km altitude)

0° is the direction of the velocity vector. The STN is at the center of the plot. The plot looks down on the STN toward the center of the Earth. The flux distribution is in units of tenths of the entire flux.



7.4.4 Meteoroid and Debris Impact Probability

The combined number of impacts from meteoroid and debris particles, N, is:

$$N = N_{met} + N_d = A * t * (F_{met} * SF * DF + F_d) \qquad \text{(Eqn. 7.4-11)}$$

where A is the surface area (m^2), t is the exposure time (yrs), F_{met} is the meteoroid flux (#/m^2-yr), SF is the Earth shielding factor, DF is the Earth defocusing factor, and F_d is the debris flux (#/m^2-yr). The probability of exactly n impacts is described by the Poisson distribution:

$$P = N^n/n! \exp(-N) \qquad \text{(Eqn. 7.4-12)}$$

which for no impacts becomes

$$P = \exp(-N) \qquad \text{(Eqn. 7.4-13)}$$

For an Earth-fixed transportation node, the directionality of debris means that the forward and sides will sustain the most impacts while the top and aft walls of the hangar will be impacted by meteoroids only. In addition, the hangar bottom will receive few meteoroid impacts because of the shielding from the Earth and self-shielding from other node components.

7.4.5 Transportation Node Meteoroid/Debris Shield Design

To properly assess hypervelocity impact shielding requirements for the transportation node, the following is required: (1) definition of the failure criteria, and (2) specification of the lifetime reliability from impact induced failure.

The following baseline specifications are proposed:

- Failure is defined as an impact from a particle too large to be completely shocked by the shield. It is assumed that critical damage to internal components within the hangar can occur if an impacting particle is not completely disrupted (shocked) by the shield.

- A 10 year lifetime for the transportation node.

- A 50% probability that failure (a non-disrupted particle gets through the bumper or bumpers) will not occur over the node lifetime (1 chance in 2 of a critical impact). This reliability was selected as a baseline since the full spacecraft stack will not be hangared continuously over the lifetime, and because the stack does not occupy the entire interior of the hangar. Sensitivity to this reliability is discussed in section 7.4.8. A better design criteria, such as probability of no penetration of any critical item in the hangar over the design lifetime of the STN can be devised when more detailed work is possible.

7.4.6 Meteoroid/Debris Shield Sizing

The concept of a shield is to break up impacting projectiles into a multitude of smaller debris that is dispersed over a wide area of the underlying structure as illustrated in Figure 7.4.6-1. Experimental studies have demonstrated that two-wall bumper/backwall structures save as much as 80 percent of the mass of a single-wall with equivalent impact resistance.

The most important factors determining the success of a bumper/backwall system are the state of the particles in the debris cloud (governed by bumper and impact conditions), the spacing or standoff distance, and backwall properties (primarily thickness and yield stress). Intense shock waves generated by the impact propagate at supersonic speeds forward into the bumper and backward into the oncoming projectile, compressing these materials beyond their original density and increasing temperatures and pressures by many orders of magnitude. When these compressional shock waves encounter free surfaces, they are reflected as tensile or rarefaction waves that relieve the pressure back toward zero and reduce temperatures. The initial compressive shock wave adds entropy to the material in an amount almost proportional to the peak shock pressure and the material's shock compressibility. The release from the shock-compressed state is nearly isentropic, thus, entropy is transferred to the material by transit of the shock waves. This entropy increase appears as internal energy or heat (Swift, 1982; Kinslow, 1970). If the added heat is less than the material's heat of fusion, the shocked material releases into a solid but massively disrupted state. The shocked material becomes liquid if the added internal energy exceeds its heat of fusion, and a gas if the material's vaporization energy is exceeded. For aluminum-on-aluminum impacts, such as orbital debris impacts on the node hangar/shield, shock heating causes incipient melting of the projectile at approximately 5 km/sec and completely melts it above 7 km/sec.

Impact parameters, bumper thickness, and material properties determine the peak shock pressure and state of the debris plume. An optimal thickness bumper will cause the rarefaction wave from the bumper to overtake the compressive shock wave in the projectile at the instant it has swept through the entire projectile, i.e., at the back of the projectile. This results in the greatest projectile heating and greatest likelihood of projectile melting or vaporization. In addition, the rarefaction from the bumper imparts particle velocities with the greatest dispersive effect on the projectile. If complete shock compression and rarefaction of the projectile has been accomplished with the thinnest bumper, the mass of bumper and projectile material in the debris plume which subsequently impacts underlying structures will be minimized (minimizing damage to these structures).

An impact on too thin a bumper causes the rarefaction wave from the bumper to overtake the compressive shock wave in the projectile and sharply attenuate it before it completely traverses the projectile. This means that a portion of the projectile is only lightly shocked and will likely strike underlying structures as an intact solid fragment, with far greater destructive potential than the rest of the debris plume.

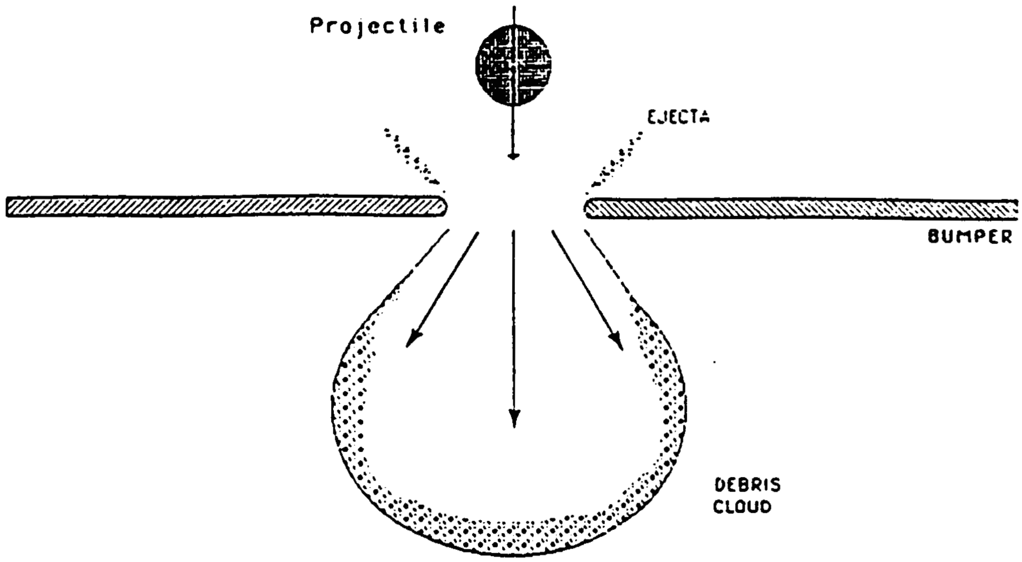
An analytical model developed in a previous study (Christiansen, 1987) was adapted to calculate the peak shock pressure and optimal bumper thickness for the impact conditions expected for the transportation node. A one-dimensional calculational approach is used with Hugoniot-Rankine relationships and simplified equations-of-state. It is described in more detail elsewhere (Christiansen, 1987 and 1988). Calculations from the model are shown in Figure 7.4.6-2 which indicate that for the average debris velocity of 10 km/sec, the thickness of an aluminum shield should be approximately 0.2 x the diameter of the maximum size impacting particle.

7.4.7 Meteoroid/Debris Shield Mass

Given a 7,750 m^2 surface area of a 50 m long x 35 m wide x 25 m high hangar/shield structure, there is a 50% chance that the structure will receive a single impact from a 7 mm or larger particle over a 10 year lifetime (see Figure 7.4.7-1). A 1.4 mm thick aluminum shield is required. Mass of a shield on all 6 sides of the hangar is 30.4 metric tons while mass of a shield protecting all but the nadir oriented side (Earth-facing surface) is 23.5 metric tons. The Earth-facing shield will probably not be required because the meteoroid and debris flux is so low on this surface. Variable thickness, due to a greater flux on the front and sides allows a reduction of another approx. 1.8 metric tons for the single wall aluminum bumper or shield.

Figure 7.4.6-1, Hypervelocity Impact

a. Impacts by hypervelocity projectiles will result in a debris plume of solid fragments, liquid, or vapor particles.



b. The structure inside the bumper or shield must then survive the fragments and blast loading. It could rupture from the blast loading, or fail due to spall or complete perforation from individual fragments.

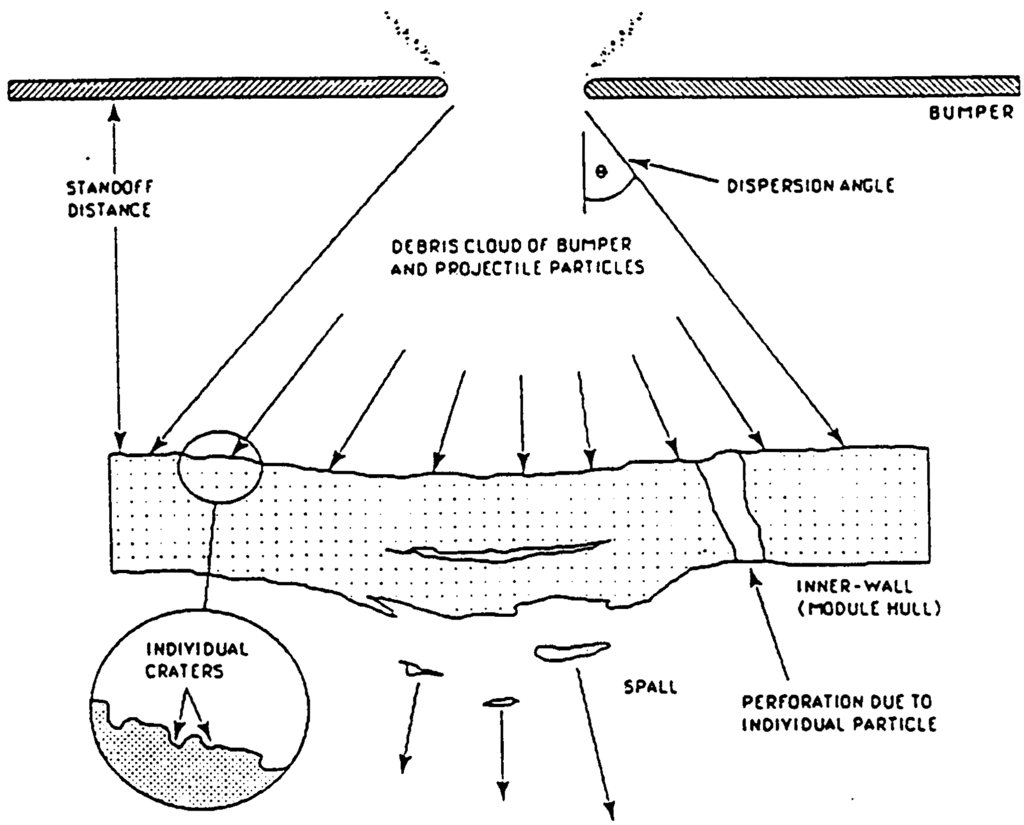


Figure 7.4.6-2, Aluminum Shield Thickness vs. Orbital Debris Particle Velocity

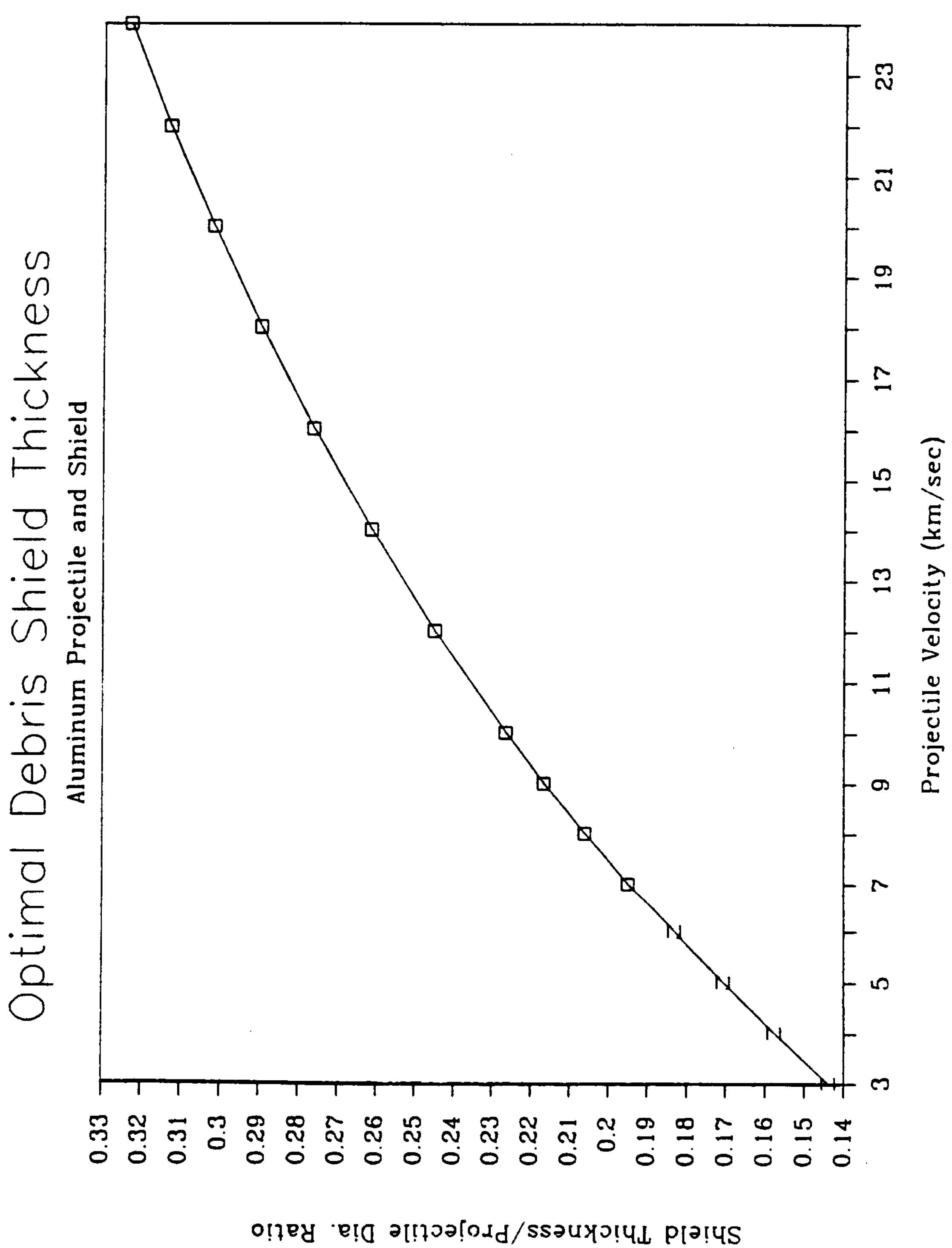
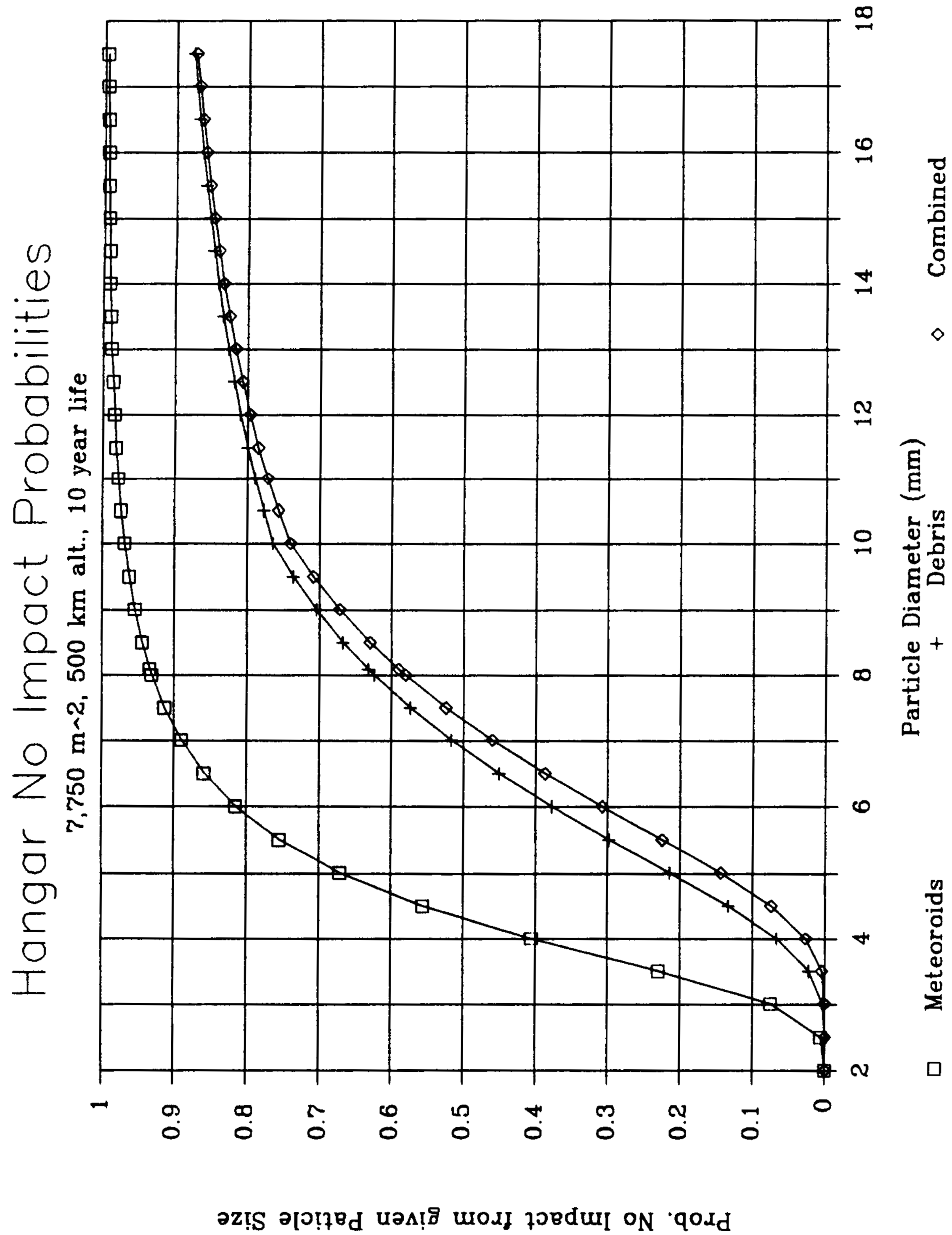


Figure 7.4.7-1, Probability of No Meteoroid and Orbital Debris Impacts on STN Hangar Versus Particle Diameter



7.4.8 Effect of Reliability Requirements

The 50% reliability requirement from impact damage used for baseline shielding sizing/mass estimates could very well be established at a higher value. For instance, from the Space Station Project Requirements document, the pressurized module meteoroid/debris protection requirements are 0.9955 for 10 years (NASA, April 22, 1987):

> "The design goal for each SSCE (Space Station Core Equipment) classified as being critical is to have a minimum probability value of 0.9955 of experiencing no failure due to meteoroid or debris impact that would endanger the crew or Space Station survivability for the 30-year life of the Space Station. However, due to uncertainties, both in the meteoroid and debris environments, and the behavior of materials in this environment, the initial Space Station design requirement shall use a 10-year exposure time period with the minimum probability of 0.9955. It is anticipated that a significant increase in the content of data bases covering both environments and material behavior will occur during the design and development of the Space Station. Therefore, each SSCE's protection must be capable of being improved incrementally in order to provide the required protection. In addition, the design requirements will probably become more severe as the various data bases mature."
>
> Each individual pressurized module is considered a critical SSCE covered by this requirement (NASA, April 22, 1987). A penetration of the pressure vessel is deemed a "critical" failure (NASA, April 22, 1987).

For non-critical space station equipment, a 0.95 reliability against meteoroid and debris impact damage has been proposed for a typical 10 year design lifetime (NASA, January 15, 1987).

Thus, the 0.5 reliability used in the baseline assessment is lower than that used in Space Station design. However, higher reliabilities require thicker shielding which will impose severe mass penalties as given in Figures 7.4.8-1 and 7.4.8-2. In addition, the actual surface areas requiring protection inside the hangar are poorly defined and more detailed work is likely to show that a 50% chance of a non-disrupted projectile getting through corresponds to a significantly lower probability of critical or non-critical equipment damage in the hangar.

Figure 7.4.8-1, Shield Thickness as a function of Reliability

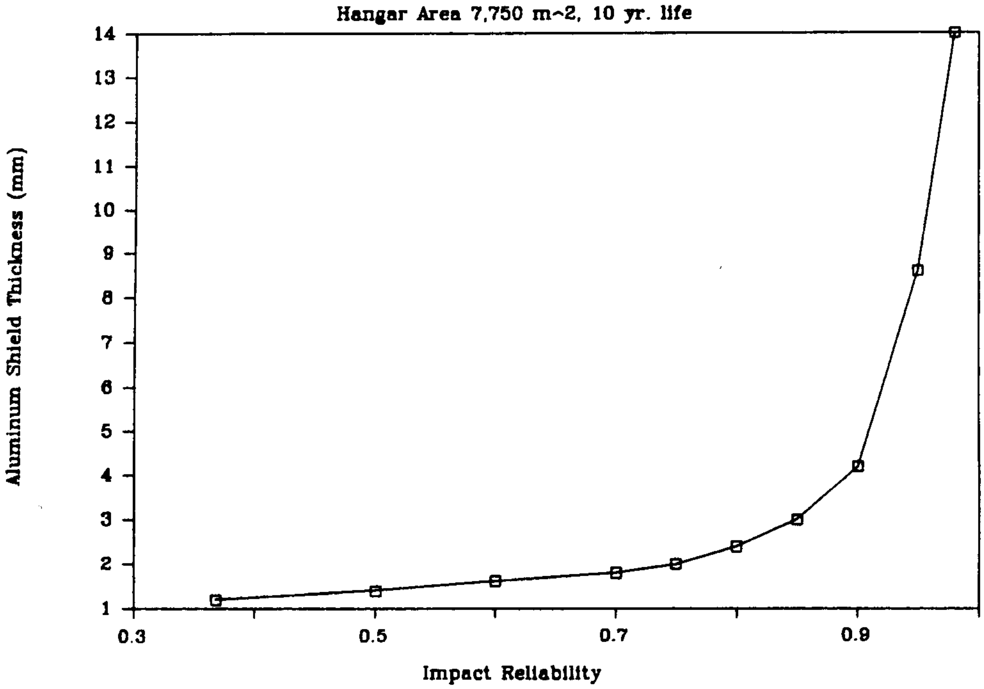
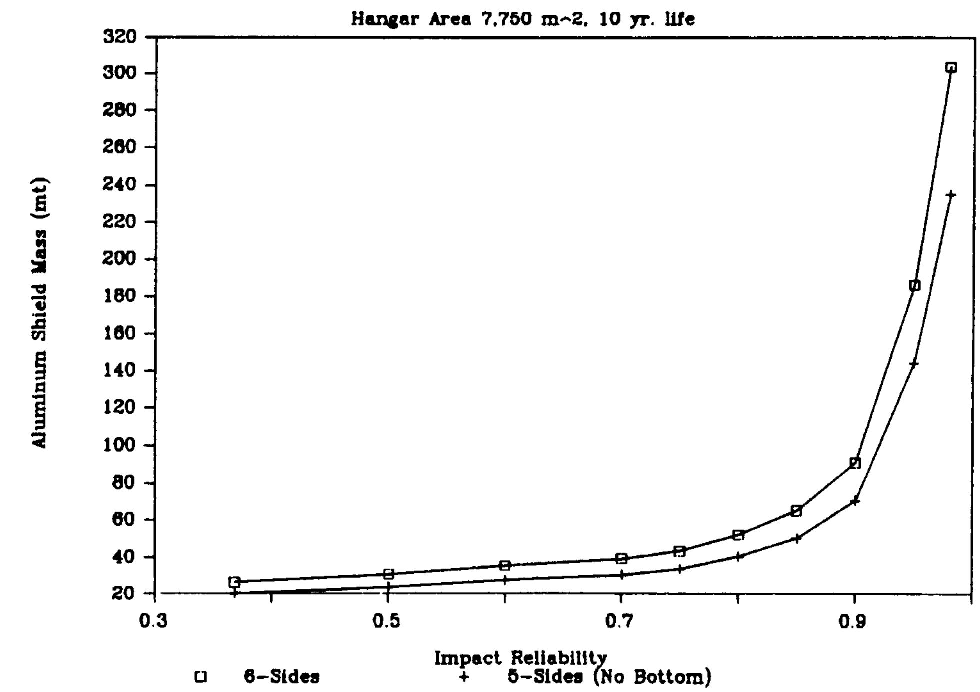


Figure 7.4.8-2, Shield Mass as a Function of Impact Reliability
(For 5-Sided Shielding: Leaving Bottom Open)



7.4.9 Variable Thickness Shielding

Debris impacts will be concentrated on the forward facing hangar wall and on the sides. Little or no debris will impact on the top, bottom, or aft surfaces of the hangar. In addition, because of Earth shielding, meteoroid impacts will be confined mainly to top, front, sides and back hangar surfaces (in that order). The bottom surface shield can be eliminated since there will be essentially no debris impacts, and because the meteoroid flux on the Earth facing bottom surface will be quite low, with only glancing meteoroid impacts possible (impacting at <19.6° for 500 km orbits assuming a 100 km atmosphere). Thus, the thickness of the shielding can vary to match the impact flux. The forward wall will receive a debris flux 2.3 times the debris flux on a randomly oriented surface and will need the thickest shields, while the debris flux on the starboard and port walls will be 1.5 times the flux on a random surface and will need the next thickest shields. Top and rear shields will be the thinnest. For the same overall 0.5 reliability, the wall thicknesses and mass are:

Surface	Area (m^2)	Thickness (mm)	Mass (kg)	Areal Density (g/cm^2)
Forward	875	1.9	4,655	0.53
Port Side	1,250	1.6	5,600	0.45
Starboard Side	1,250	1.6	5,600	0.45
Aft Wall	875	0.8	1,960	0.22
Top	1,750	0.8	3,920	0.22
Bottom	1,750	0	0	0
Total	**7,750**		**21,735**	

7.4.10 Effect of Alternative Failure Criteria

As shown in Figure 7.4.6-1, a multitude of smaller, high speed debris will result from an impact on the shield. In certain cases, this may present a significant hazard to objects within the shield enclosure (i.e. for EVA personnel, high pressure tanks, or sensitive optic or thermal surfaces). Another alternative criterion for success of the shielding could be stated as a certain reliability (say 50% over 10 years) that the inner wall of a dual wall structure will not be penetrated. This shielding requirement would require an additional wall beyond the bumper walls.

Multiple-wall shield mass for a completely shielded hangar (all 6 sides) having a 50% reliability of stopping all particles over a 10 year period is (based on constant thickness walls):

	Mass (kg)
Bumper	30,380
Backwall	68,315
Total	98,695

For shielding on 5-sides, with no bottom:

	Mass (kg)
Bumper	23,520
Backwall	52,889
Total	76,409

If the backwall is used just on the forward and sides, its mass is 37.5 metric tons. For a dual-wall on the forward surface only, backwall mass is 15.4 metric tons.

These calculations were based on an aluminum backwall sizing equation (Cour-Palais, 1979)

$$t_b = f\,(p_m\, p_t)^{1/6}\, M^{1/3}\, V/S^{0.5}\, [70{,}000/\sigma]^{0.5}$$

where,

t_b = second wall thickness (cm)
f = factor = 0.05 for particles < 0.32 cm, 0.14 for particles > 1 cm, and linearly interpolated between these sizes.
p_m = impacting particle density (g/cc)
p_t = target (backwall) density (g/cc)
M = impacting particle mass (g)
V = impact speed (km/s)
S = spacing between bumper and backwall (cm) - maximum of 25x impacting particle diameter.
σ = backwall yield stress (psi) (51,000 psi for Al 2219-T87).

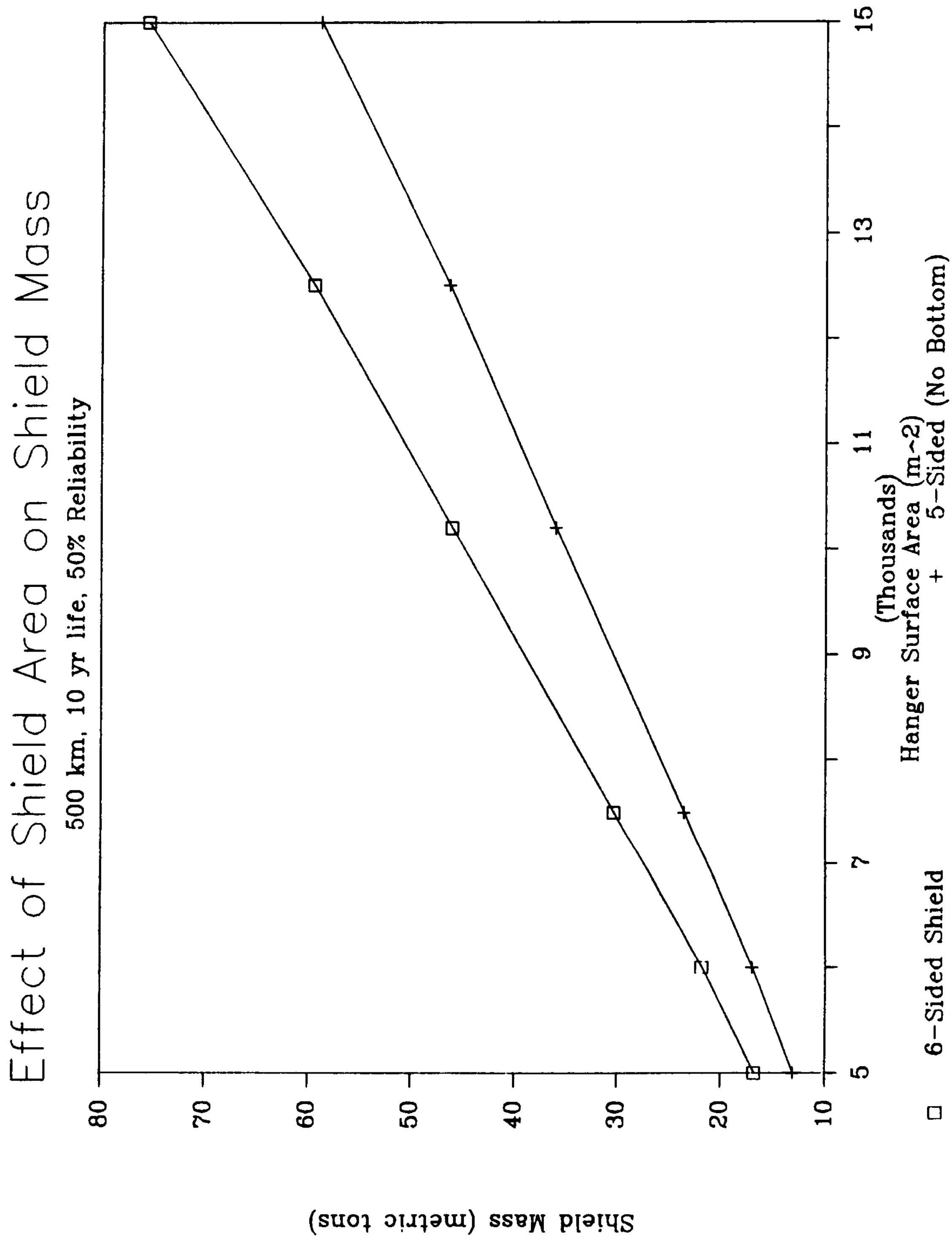
The calculations were based on aluminum bumpers (Al 6061-T6) and backwalls (Al 2219-T87) as baselined for Space Station pressurized modules. Backwall thickness is defined by debris impact parameters (p_m = 2.8 g/cc, V = 10 km/s).

7.4.11 Hangar Area Effects on Shielding Mass

The most effective way of reducing shielding mass is to reduce the shield area. This directly reduces the volume (and mass of shielding) as well as reduces the exposed area to debris/meteoroid impacts which decreases the maximum particle size expected to impact the shield over the life of the transportation node, and thus the required shield thickness. Reducing shield area also has the advantage of decreasing drag, thus decreasing reboost propellant requirements or allowing the nominal operational altitude to decrease. Since the debris population is less at lower altitudes, a secondary result of a smaller exposed area if the operating altitude of the space station is reduced would be a lower debris flux, thus allowing even greater shielding mass reductions.

The effect of reduced shield area on shield mass is shown in Figure 7.4.11-1 (assuming constant orbital altitude).

Figure 7.4.11-1,Effect of Shield Area on Shield Mass
(Basis: 500 km orbit, 10 year life, 50% Reliability)



7.4.12 Alternative Shielding Materials

Certain shielding materials and concepts other than a single, rigid, aluminum plate have demonstrated greater penetration resistance and could ease deployment (Christiansen, 1988, Cour-Palais, 1988, and Stump and Crews, 1984). Some of these are classified and cannot be discussed here. Unclassified results of a recent study (Christiansen, 1988) showed that dual bumper systems, consisting of a front flexible, metallic mesh or ceramic bumper followed (at 1/4 of the total front bumper to backwall standoff distance) by a second bumper sheet of aluminum or graphite/epoxy, protected the backwall significantly better than a single aluminum bumper. In these comparison tests (Christiansen, 1988), for equal areal density bumper systems (0.22 g/cm^2), standoff distances, and projectile parameters, less measured damage occurred to a 0.05" backwall with the dual bumper systems than occurred to a 0.063" backwall with a Al 6061-T6 bumper. Thus, if backwalls are required for achieving shielding requirements, 20% reductions in backwall mass may be possible by using dual bumpers.

If the failure criteria remains fully shocked particles (thus, not needing a second wall), then flexible metallic mesh or ceramic bumpers would still be desirable as a method to ease deployment. Constructing a hangar out of rigid aluminum plates would probably be more time consuming than using a flexible material attached to a low-mass supporting structure.

Non-metallic backwall materials have also shown more resilience to penetration by the debris from the bumper shield (Stump and Crews, 1984). A graphite/epoxy balsa-wood sandwich shows particular promise for non-pressurized backwall applications (Christiansen, 1988).

Additional experimental testing is required to precisely quantify mass savings and assess the optimum shielding/backwall materials and configurations.

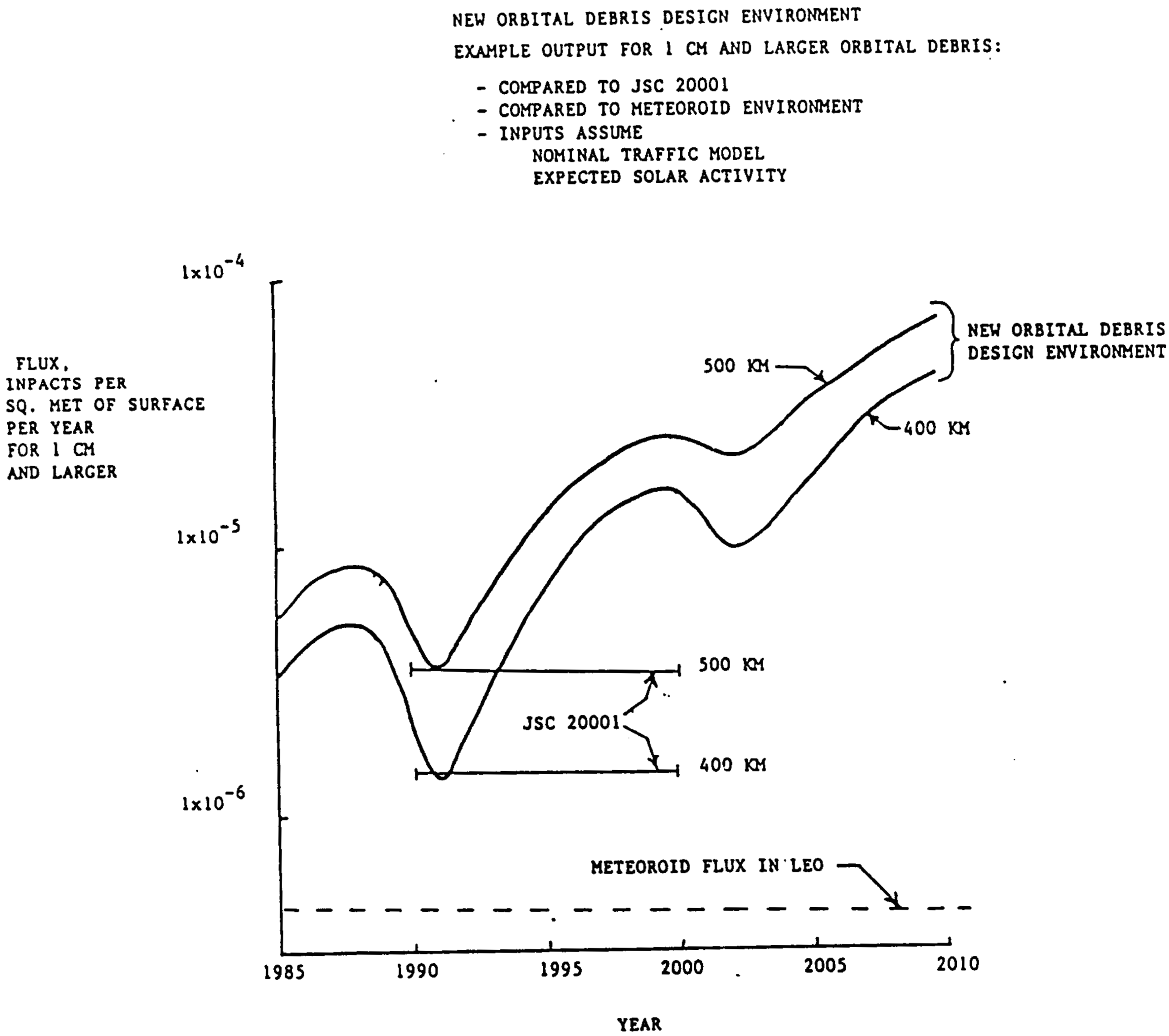
7.4.13 Baseline Shield Design

A 50% reliability for a 10 year lifetime is proposed. The failure criterion of a non-disrupted projectile is selected (instead of completely stopping all debris). A variable thickness shield is proposed as the baseline. No shield on the Earth-facing side is proposed. Total mass of the shield is estimated to be 22,000 kg. A dual-bumper system is proposed consisting of a low-mass ceramic fabric outer bumper (SiC or other ceramic), a 5 cm spacing, and a second graphite/epoxy inner bumper. The outer fabric can be easily deployed over a low mass graphite/epoxy frame. The second rigid graphite/epoxy bumper will require large hinged panels to reduce deployment time. Experimental hypervelocity impact studies should be conducted to determine if a tight weaved fabric would sufficiently contain fragments from the outer bumper. If so, a fabric layer (perhaps of kevlar, ceramic, or graphite cloth) could be substituted for the rigid panels of the second bumper and deployment will be easier.

7.4.14 Effect of Changes in the Debris Environment Definition

Orbital debris personnel at JSC (Kessler, 1988) have been working on an update to the space debris environment model defined in JSC-30425 (NASA, January 15, 1987) and JSC-20001 (Kessler, 1984). The new model accounts for atmospheric density changes due to the solar cycle and also incorporates a predicted debris growth term. A comparison of the proposed new environmental model with the current one is reflected in Figure 7.3.14-1. In the early 1990's, there is little difference between the current and new model debris flux. However, within 10-15 years the flux grows by an order of magnitude for particles 1 cm and larger. A 10 x change in flux would decrease impact reliability. If adopted, this change in the debris environment will probably result in unacceptably low reliability. A re-evaluation of transportation node impact damage vulnerability and shielding options is recommended if the new model is adopted.

Figure 7.4.14-1,Effect of Proposed New Orbital Debris Environment (Kessler, 1988)



7.5 Propellant Storage

Transfer of large amounts of cryogenic propellant from Earth to the STN will be accomplished by a heavy lift vehicle (HLV). The HLV will rendezvous and station-keep with the STN. A STN based OMV will depart from the STN and rendezvous with the HLV, capture the propellant tanker, and return to the STN and berth at the upper boom of the STN on top of the hangar. The liquid oxygen and liquid hydrogen propellant storage tanks are located on the truss on the sides of the hangar. The propellant will then be transferred to the storage tanks or directly to the vehicles. The OMV will then separate from the STN and place the propellant tanker in a deorbit path.

The currently planned OMV may not be capable of handling the 85 m ton (187,000 lbm) arriving HLV. The largest payloads currently discussed for the planned OMV are in the range of 34 m tons (75,000 lbm). A larger OMV will probably be required. The currently planned OMV uses storable hypergolic main propellants. A new OMV is assumed to use all cryogenic propellants.

The STN is oriented toward maintenance and refurbishment of vehicles in space. In order to further this objective, hypergolic propellants are assumed to be replaced with easier to live with cryogenic oxygen and hydrogen. No provisions are made for hypergolics on the STN.

The propellant storage system consists of four liquid oxygen and liquid hydrogen storage tanks with appropriate structure, meteoroid and debris shielding, insulation, thermal control, and transfer systems built into each tank set. A typical tank set is shown in Figure 7.5-1 with its multilayer insulation (MLI) blanket and vapor-cooled shields (VCS) surrounding both the hydrogen and oxygen tanks. The propellant transfer equipment is located at the end of each module. Interconnecting the four tank sets are insulated propellant distribution lines which are joined at the HLV docking station and the OTV/OMV refueling station. The overall configuration of the propellant storage system is illustrated in Figure 7.5-2.

7.5.1 Sizing

The four propellant storage modules are sized to store 182 metric tons total (400 klb) of propellants, which should be sufficient to load the worst case single lunar departure spacecraft (single stage OTV plus lunar lander with crew module), eight OMV prox. ops flights, and a 10% contingency. This choice concerning on-orbit propellant storage capacity was based on considerations of design options ranging from direct propellant transfer from the HLV tanker to the lunar departure stack, to options to store sufficient propellants at the STN to support two lunar departure stacks plus attendant OMV and STN needs. Table 7.5-1 is a matrix of the possible propellant storage options.

From the matrix, we can see that some minimal requirement for storage of propellants aboard the STN exist to support the STN systems and the OMV operations. Since the maximum HLV payload capability is approximately 75 m tons (165,000 lbs.) of propellant, it will require three HLV flights to support the STN and the lunar spacecraft stack. If the HLV propellant resupply tanker is to be used as the principal on-orbit storage container, it must be designed for long-term thermal control, debris protection, and micrometeoroid protection. Such design requirements would have a significant cost impact to the tanker design which is presently expended and deorbited after use. Over the life of the program, this delta cost may significantly exceed the cost of developing and maintaining an STN on-board propellant storage system. Another important factor is that the use of the HLV resupply tanker for on-orbit storage leads to the dependence of all propellant transfer operations on the HLV launch schedule and capabilities.

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Figure 7.5-1, Typical Tank Set (General Dynamics, 1987)

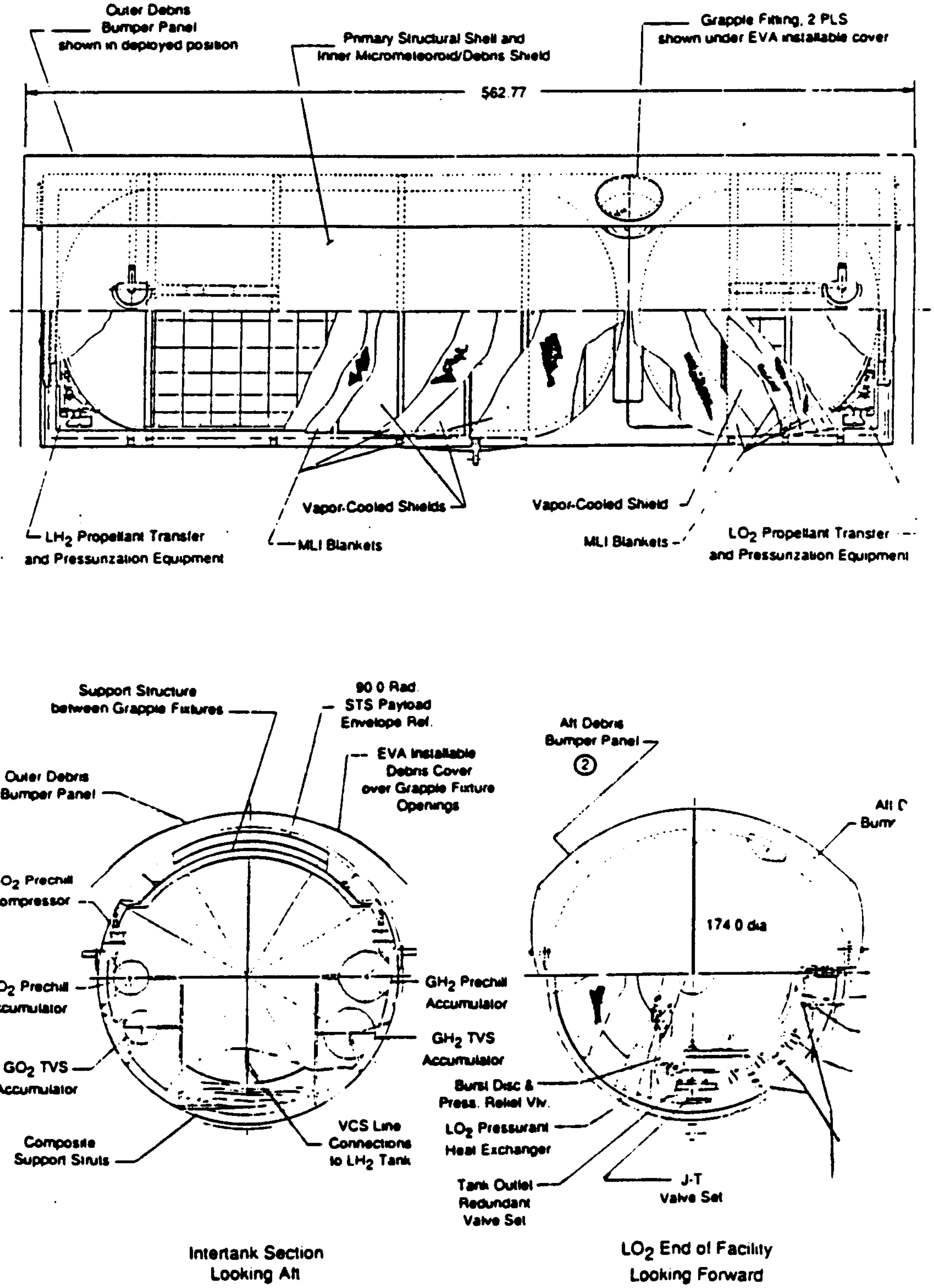

Figure 7.5-2, Overall Propellant Tank Configuration

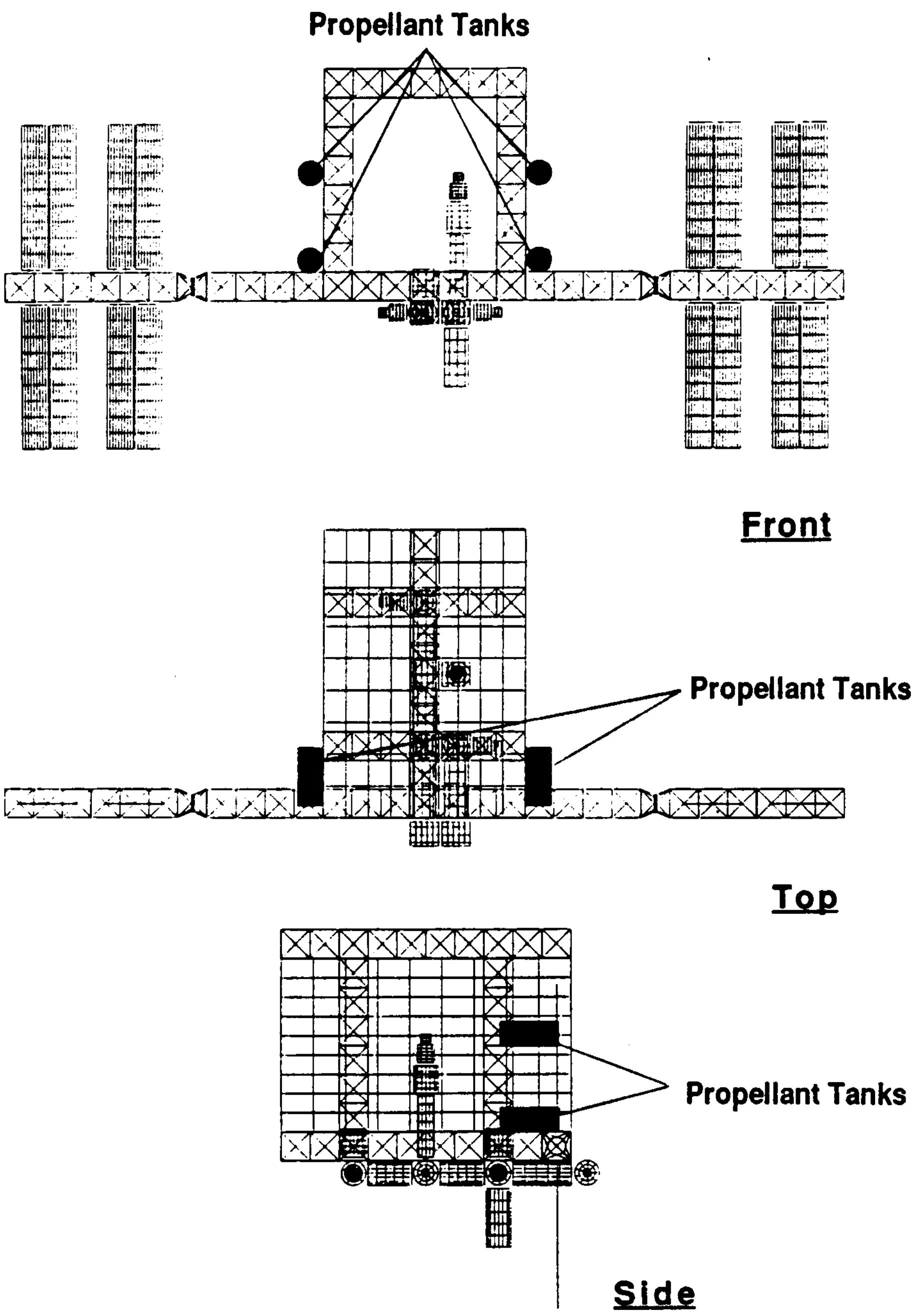

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Table 7.5-1, Propellant Storage Options

	Direct Transfer from Tanker (Storage in tanker or in stacks only)	182 Metric Ton (400,000 lbm) Storage Capacity (Storage for 1 stack)	364 Metric Ton (800,000 lbm) Storage Capacity (Storage for 2 stacks)
STN Propulsion	On-board storage for RCS + Reboost required	Use storage tanks	Use storage tanks
OMV Resupply	On-board resupply storage required	Use storage tanks*	Use storage tanks*
HLV Tanker Design	Requires: -Debris Protection -Micrometeoroid Protection -Thermal Control	Minimum insulation and debris protection req.	Minimum insulation and debris protection req.
Lunar Stack Dep. Schedule	Highly dependent on HLV sched.	Partially dependent on HLV sched.	Minimum impact from HLV sched.
STN Dry Mass	Minimum impact	+130,000 lbm (59 m tons)	+260,000 lbm (118 m tons)
STN Cost	Minimum impact	Moderate impact	Largest impact
STN Assembly	Minimum impact	+5 STS Flights**	+9 STS Flights**
STN Operations	Minimum impact	+80 IVA MH/yr +60 EVA MH/yr	+160 IVA MH/yr +120 EVA MH/yr

* Assumes the upgraded OMV will use cryogens. The planned version uses space storables, and is probably too small to handle the 200 m ton stacks or 90 m ton HLV.

** Delivery of storage modules.

The 182 metric ton propellant storage system avoids the necessity of designing the HLV resupply tanker as a long-term on-orbit propellant storage system, thus reducing the cost of the expendable component of the STN propellant resupply system. In addition, it partially decouples the propellant storage for the STN, OMV's, and one lunar departure spacecraft stack from the HLV launch schedule. The cost to provide this capability on the STN is the development and production cost of the propellant storage system, approximately 59 m tons of additional mass on the STN, five launches to deliver the components to the STN, and operations/maintenance costs for the duration of the program. The major drawback of having only enough propellant on-orbit at one time to support one lunar departure spacecraft stack is the partial dependence on HLV launch schedules to provide propellants for the second stack. This disadvantage may be partially alleviated by the fact that lunar flight preparation time for one stack may be as long as twenty days. This may be sufficient time to support additional HLLV propellant resupply flights and have adequate propellants on-orbit in time to support the loading of propellants in the second stack.

A 364 metric ton propellant storage system would essentially decouple the HLV resupply tanker schedule from the STN propellant loading of both the lunar departure spacecraft stacks. The penalty of this approach is the increased cost of production of the propellant storage system tanks, increased inert mass on-orbit of approximately 60 m tons (four more tanks), the addition of four launches to deliver the components to the STN, and additional operations/ maintenance for the duration of the life of the STN.

Based on the evaluation of the three approaches to STN propellant storage, the 182 metric ton configuration was selected as a conservative middle option. More detailed study is required of vehicle schedules and propellant handling in the next iteration study.

7.5.2 Cryogenic Storage System

The method proposed for long-term storage of cryogenics on the STN is an all-passive storage system with propellant boil-off fed to the STN Attitude Control/Reboost System and Environmental Control and Life Support System. Passive control of cryogenics was selected after consideration of the needs of the STN subsystems. This approach avoided the equipment, power, cost, and complexity of providing a reliquification capability for the cryogenic boil-off gasses. If boil-off in excess of the STN subsystem needs are generated in the refueling process, they will be dispersed out of non-propulsive vents located at separated points on the keel of the STN.

The tank sets are oriented horizontally along the velocity vector and symmetrically with respect to the X axis to aid propellant settling and c.g. control. This orientation takes advantage of the aerodynamic induced drag to localize the vapor pocket near the vent of a partially filled tank. The small drag force causes the propellants to move forward in the tanks. Figures 7.5-2, 7.3.1-1, 7.3.1-2, and 7.3.1-3 show the tank locations. The original idea was to bracket the c.g. with the tanks in all three dimensions. As the design process progresses the tanks may need to be moved to achieve this.

For the purpose of this study, the passive vent tank design described in the December 1987 report on the "Long Term Cryogenic Storage Facility Systems Study" performed by General

Dynamics was selected as a point design with reasonable data for mass, size, and component details. Each tank set has a 100,000 lb (45 m ton) storage capacity of oxygen and hydrogen at a 6:1 ratio. Four tanks provide a total combined storage capacity of 400,000 lb (182 m tons). Each tank set is surrounded by four layers of Multilayer Insulation (MLI) for a total thickness of 4 inches. Combined with the MLI are vapor cooled shields to reduce hydrogen boil-off and eliminate oxygen boil-off. A Thermodynamic Vent System (TVS) is employed with tank wall-mounted heat exchangers to control tank pressure. Provisions for fluid mixing are included to assure thermal equilibrium within the tanks in the micro-gravity environment. Table 7.5-2 shows estimated weights for the system, transfer lines, and tanker interface. Each tank set is estimated to require 500 watts of power when active (pumping) and 160 watts when passive.

7.5.3 Cryogenic Transfer System

Transfer of propellants to and from the storage tank will be pump assisted with autogenous, cold vapor pressurization of the ullage of the supply tank. This minimizes the post transfer boil-off of the cryogens in the supply tank. The receiver tank will employ a thermodynamic fill technique to avoid the problem of vent vapor separation in the micro-gravity environment. With this approach, the initial subcooled fluid is introduced to prechill the receiver tank and cause a pressure rise in the receiver tank. When the receiver tank is cooled to the condensation point of the fluid, the fill process assisted by pumped transfer will continue the process until the transfer is complete. Details on this process and other aspects of on-orbit transfer of cryogenic propellants are included in Appendix A of this report. Acquisition of liquids from the supply tank is accomplished by a capillary Liquid Acquisition Device (LAD). These channels are total communication screened surface devices which direct the propellant flow to the tank outlet. To avoid vapor breakthrough of the screen LAD, the pressure assisted pump-fed transfer keeps fluid within the capabilities of the LAD. Excessive vapor boil-off from the prechill of the propellant transfer lines and receiver tank is fed to the high pressure gaseous propellant storage tanks for the Attitude Control/Reboost System and ECLSS. Figure 7.5.3-1 and 7.5.3-2 are typical schematics representing the components involved in the cryogenic transfer process.

A mobile propellant supply boom will be located in the hangar between the vehicle stacks to fuel each stage of the stack. This remotely controlled boom will operate in a manner similar to the manipulators and be controlled from the centralized operator station inside the hangar. Engagement and disengagement of the propellant and electrical interfaces will also be controlled from this station. EVA will not be required to hook up or disconnect these lines.

7.5.4 Cryogenic Boil-Off Recovery System

The STN Propellant Storage System uses passive thermal control to maintain propellants in a cryogenic liquid state while awaiting transfer to the lunar departure spacecraft or the OMV. Some boil-off will occur to compensate for heat gain while the propellants are stored on-orbit. This boil-off is directed through the vapor-cooled shields which surround the cryogenic tanks. Most of the heat leak penetrating the MLI is intercepted by the vapor-cooled shield and carried away with the boil-off gasses. These gasses are collected from the tanks as well as the propellant lines and compressed for storage in high pressure gaseous storage containers.

Table 7.5-2, Cryogenic Storage System Weight Statement
kg (lb)

ELEMENTS	kg	lb
TANK SET (EACH)*		
Structure	8,702	(19,145)
Active Thermal Control	1,091	(2,401)
Passive Thermal Control	1,963	(4,319)
Fluid Systems	1,474	(3,243)
Data Management	93	(205)
Electrical	91	(200)
ELV Attachment	205	(450)
Wire Harness	182	(400)
RMS Attachment	136	(300)
Subtotal	18,938	(30,663)
TOTAL TANKS SETS (4)	55,751	(122,652)
PROPELLANT TRANSFER LINES*		
Fill and Drain		
Pressurization		
Vent		
Vapor Cooled Shield		
MLI		
Subtotal	1,864	(4,100)
HLLV TANKER RESUPPLY INTERFACE		
Berthing Mechanism	167	(371)
Structural Frame	92	(203)
Latch Mechanism	41	(90)
Misc.	35	(78)
Subtotal	337	(742)
LANDER/OTV PROP BOOM AND INTERFACE	909	(2,000)
TOTAL SYSTEM MASS (DRY)	58,861	(129,494)

* Scaled from General Dynamics concept.

The expected boil-off rate from one 45 m ton (100,000 lbm) capacity tank set is on the order of 0.2 percent per month (90 kg). Even with all of the tank sets completely full, this amount of boil-off may be less than the average demand for gaseous propellants by the STN attitude control system and the ECLSS. At 0.2% per month, all four modules produce 364 kg/month or 4,368 kg/year total if they are always full. Depending on the solar flux, resistojets described in section 8.5.3 would use from 1 to 12 m tons of hydrogen per year for drag makeup. It is more realistic to consider the boil-off losses from the prechill of the propellant transfer lines and fluid interfaces. During any transfer process, the prechill of the propellant transfer lines is expected to generate the majority of the boil-off gasses. In an evaluation to store all of the boil-off gasses from 91 m tons (200,000 lb) of stored propellants for a period of 90 days, the equipment to capture, pressurize, and totally store 545 kg (1,200 lbm) of gases at 3000 psi was estimated to have a mass of 7,000 lb and occupy a volume of 600 cubic feet. The tankage required for the gases is anticipated to require four 5 ft diameter spheres for GH_2 storage and one 2.5 ft diameter sphere for GO_2 storage.

With the STN using the boil-off gasses for attitude control, reboost, and environmental control atmosphere makeup, it is estimated that the actual required tankage will be much less than that to store the boil-off. The actual sizing of the high-pressure gaseous propellant storage should be based on anticipated periodic usage with sufficient reserves for contingencies.

7.5.5 Handling of HLV Cryogenic Tanker

The HLV Cryogenic Tanker is basically an expendable LO_2 and LH_2 tank set capable of delivering 75 m tons (165,000 lb) of propellants to the STN. It is a simple insulated tank set designed for delivery by an expanded ALS. Thermal control of the cryogenic propellants is primarily handled by supercooled propellants and passive insulation similar to that used on the external tank of the STS. The tanker is filled on the pad by conventional ground service connectors. The forward end of the tanker has a docking adapter for handling by the OMV while the aft end contains the docking mechanism and fluid interface connectors for connection of the tanker to the STN/Tanker interface. Table 7.5.4-1 is a summary of the weights for the components and propellants for the tanker.

The operation for resupply of the STN begins with the launch of the tanker and delivery of the tanker to a parking orbit near the STN with the shroud staged away. An OMV is then deployed to rendezvous with the tanker and dock with the forward end of the tanker. The OMV then returns the tanker to the STN where it docks with the STN/tanker interface. Once all of the interface connections are confirmed, the transfer of propellant is begun. Upon completion of the propellant transfer, the interfaces are disconnected and the OMV departs with the tanker to the STN departure zone where the OMV deorbits the empty tanker into its planned deorbit trajectory. Upon completion of that task, the OMV returns to the STN for refueling and appointment to the next mission. Two docking interfaces will be available on top of the hangar to provide redundancy and allow short term storage in tankers if needed.

Figure 7.5.3-1, Hydrogen Tank Fluid Line Schematic (Gen. Dynamics, 1987)

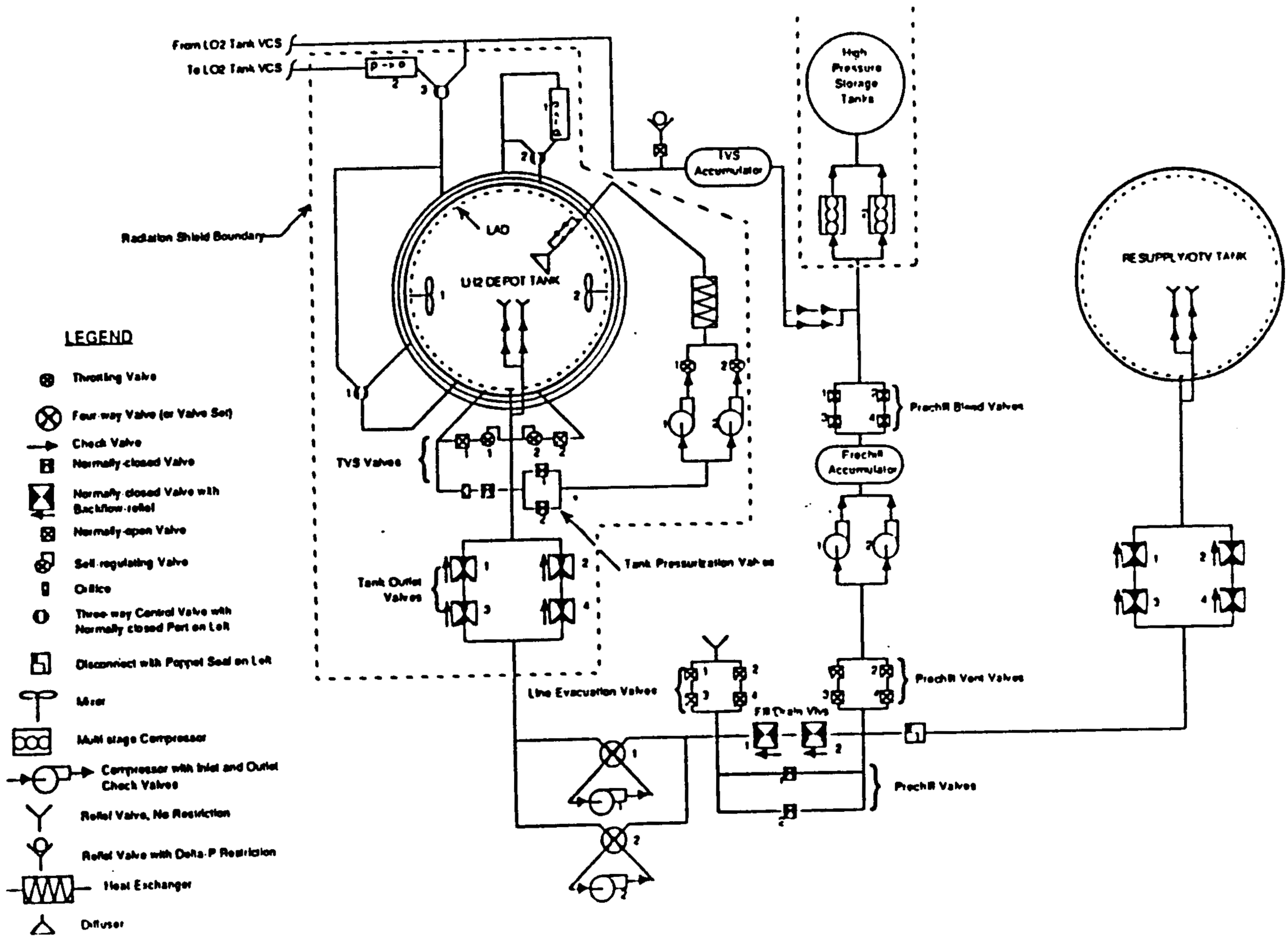

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Figure 7.5.3-2, Oxygen Tank Fluid Line Schematic (Gen. Dynamics, 1987)

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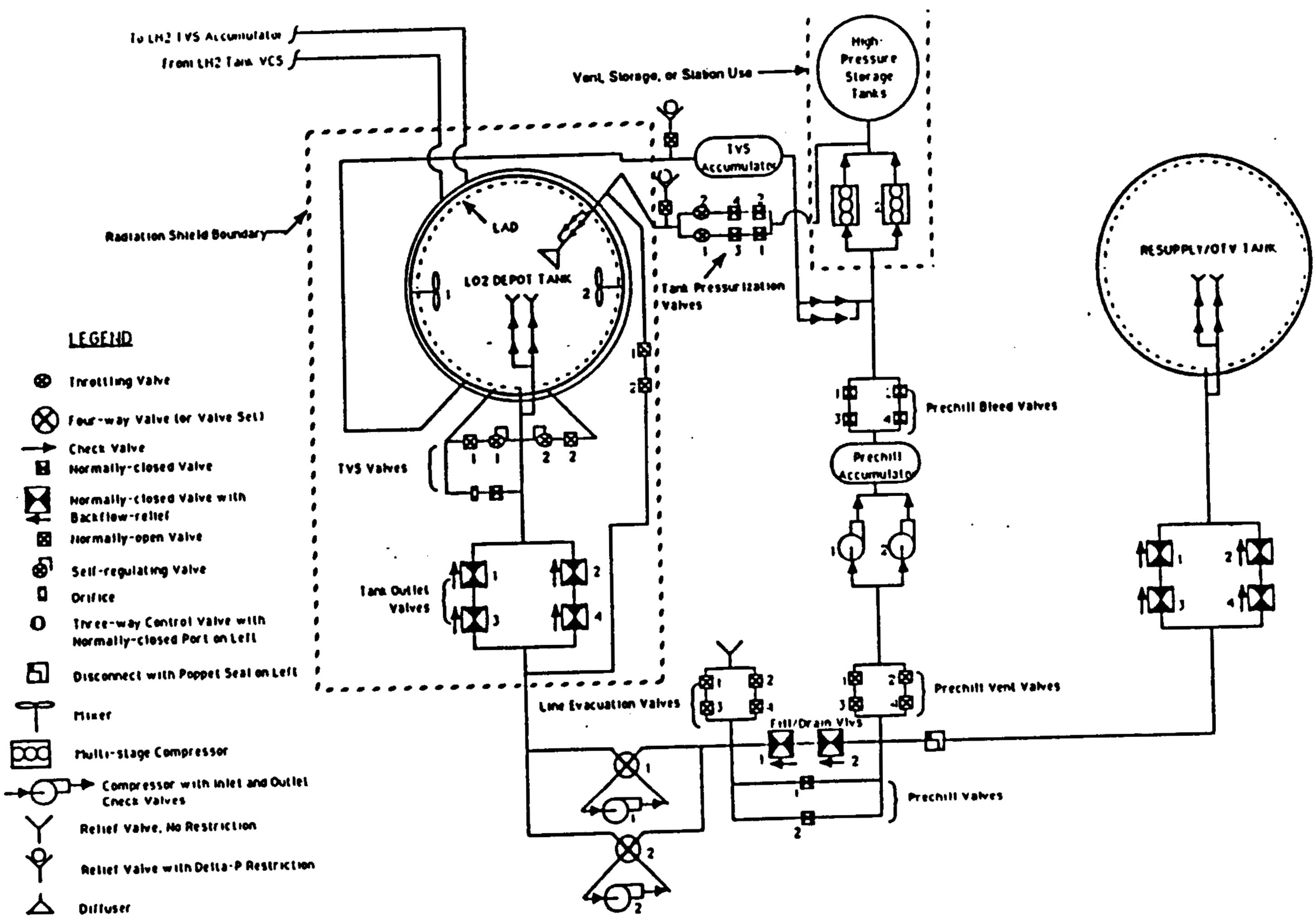

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Table 7.5.4-1, HLV Tanker Weight Statement
kg (lbm)

ELEMENT	kg	lbm
Structures	6,375	(14,024)
Primary Structure	3,873	(8,520)
Support Structure	72	(158)
LO_2 Tank	861	(1,894)
LH_2 Tank	1,569	(3,452)
Passive Thermal Control	1,205	(2,652)
LO_2 Tank	318	(699)
LH_2 Tank	888	(1,953)
Fluid Systems	348	(765)
Mass Gauging	13	(28)
Liquid Acquisition	235	(518)
Plumbing	100	(219)
Data Management	66	(145)
Control & Checkout	36	(80)
Instrumentation	20	(45)
STN Interface	9	(20)
Electrical	91	(200)
STN Interface	68	(150)
Distribution	23	(50)
Total System Weight	85,580	(188,277)
Propellant	75,000	(165,000)
Residual Weight	2,392	(5,264)
Payload Adapter	103	(227)
Tanker Inert Weight	8,058	(17,786)

7.6 Habitation Modules

Figure 7.6-1 shows the overall pressurized volume configuration. A permanent crew of 6 is required to perform the various STN house-keeping, systems, and vehicle servicing activities. In addition to the permanent crew, habitation facilities for 7 visitors are needed for a period of 14 days. The seven visitors (3 Orbiter and 4 lunar) are derived from an operations scenario which requires that an Orbiter remain docked to the STN until the lunar flight is ready for departure so that the lunar crew could be returned to Earth in the Orbiter in the event of a failure to launch the lunar vehicle.

The Space Station habitation module can accommodate a crew of up to 8 with periodic logistics resupply. The ECLSS and waste management can be sized to handle a maximum of 8 people with full redundancy. Since the Orbiter will be docked to the STN during departing and returning lunar missions, one habitation facility would be sufficient for the STN permanent crew if the seven visiting crew could use the Orbiter as a habitation facility. On the other hand, the Orbiter is not designed to support crews for stays longer than a week or so. From a safety standpoint and schedule flexibility, two habitation modules with facilities for 13 or more personnel are therefore needed.

The Space Station Habitation Module (HAB) will serve as the point of departure for this design. Two HAB modules will be needed to handle the 13 occupants. It may be possible to expand the number of sleeping quarters in one module to twelve and keep a crew member on the Shuttle and thus eliminate the need for two modules. A single HAB module arrangement, however, would significantly reduce availability of stowage space and schedule flexibility and would inhibit growth should more permanent or transient crew accommodations be needed.

Each HAB can be nominally configured with facilities for 8 crew members. The HAB module contains sleeping quarters, an environmental control and life support system (ECLSS), hygiene facilities, an integrated galley/wardroom, as well as stowage of various types. Under current plans, a health maintenance facility is also contained in the HAB. While some of the facilities will need to be duplicated, some will be extraneous if two are provided. The ECLSS will be needed in each module just as it is in the Space Station, but a single health facility may be adequate for the 13 person crew. The health maintenance facility includes exercise facilities. If each person is required to use the facility two hours per day and there is an eight hour sleep period for all, then two sets of exercise equipment will be needed for 13 people. The Freedom Station is currently planning only one exercise facility for 8 people.

The galley/wardroom contains two table arrangements and is sized to handle twelve people. This may be adequate for the full crew but an additional galley/wardroom may be appropriate. Two full personal hygiene arrangements will be provided to accommodate the expanded crew. Each of these will contain a personal hygiene station, a waste management system, and a full body shower.

Two HAB modules will provide more living space than is needed, but will allow for easy growth of up to 3 more people. Because two modules are dedicated to habitation, some rearrangement is possible. Several options are available for this rearrangement depending on how the crew quarters are distributed.

Quarters may be distributed in both modules so that each module could essentially stand alone, or they may be grouped so one module serves as a quiet module and one serves as an active module. Figures 7.6-2 and 7.6-3 show one possible layout with all crew sleeping quarters in one module and crew activity centers in the other. The quiet module has 8 sets of crew quarters on each end separated by a buffer zone with ELCSS equipment, a hygiene system, and stowage areas. The active module contains the galley/wardroom on one end with the crew health facilities and DMS control station on the other. An additional two table arrangement has been provided to handle up to 16 crew members. Again, in the active module, the two ends are separated by a buffer zone containing the ECLSS and hygiene equipment. The overall arrangement does not represent an extensive functional analysis, of course, and optimum layouts will result from further studies. Table 7.6-1 contains weight summaries of the two modules.

Figure 7.6-1, Pressurized Volume Configuration

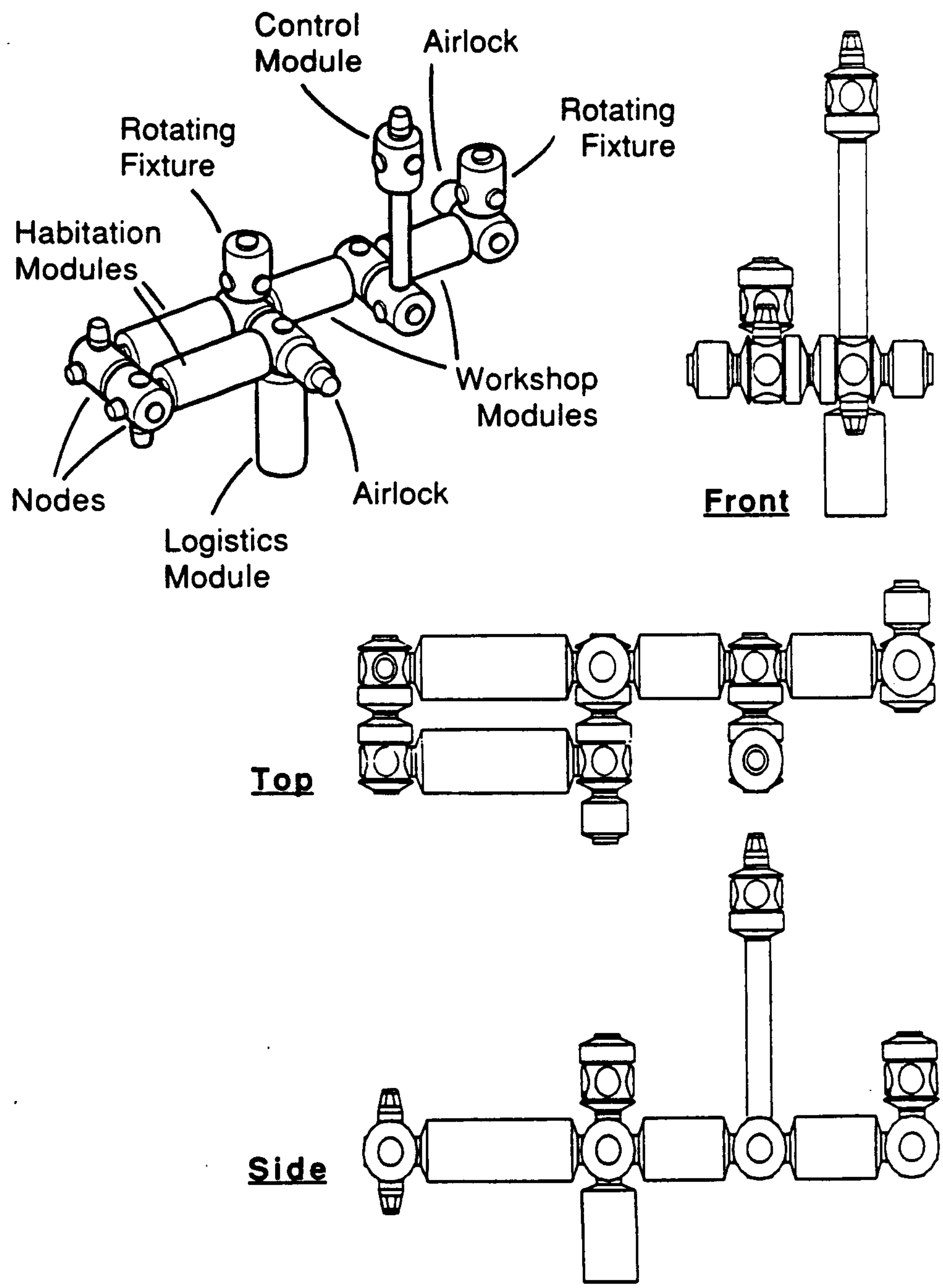


Figure 7.6-2, Proposed Habitat Layout, Active Module

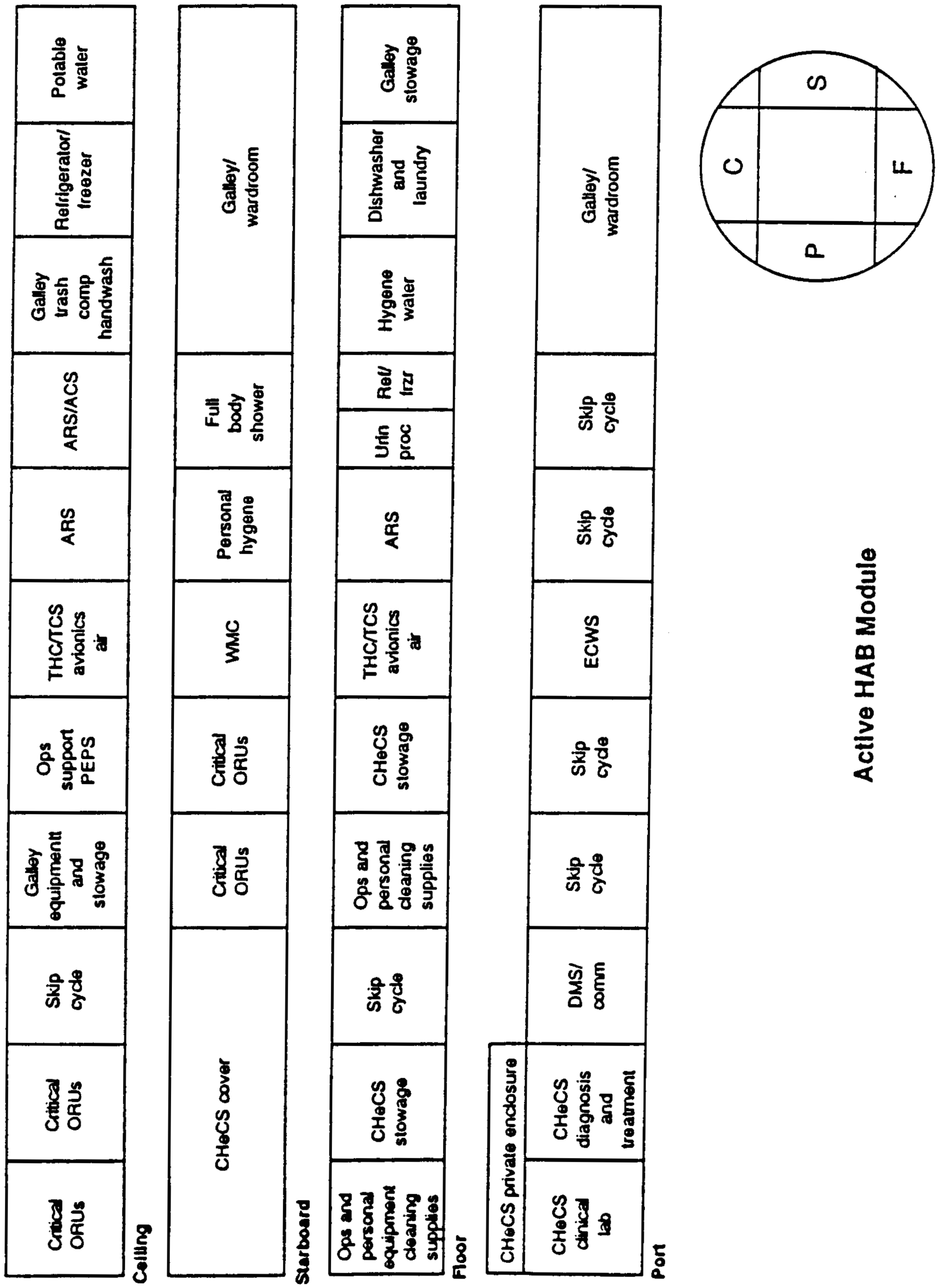


Figure 7.6-3, Proposed Habitat Layout, Quiet Module

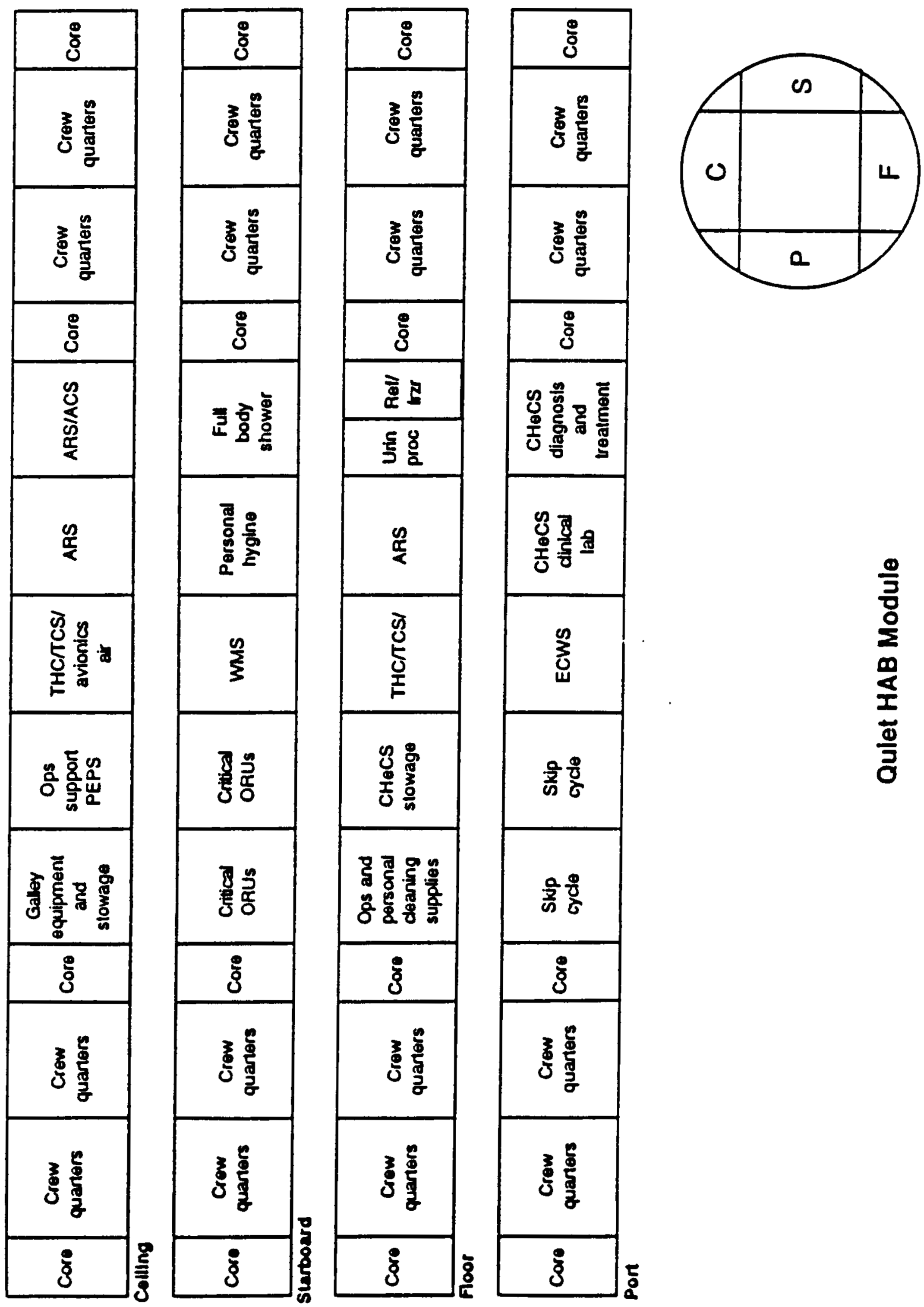


Table 7.6-1, Weight Statement for Habitation Modules

Unit	Quiet Module Weight kg	Quiet Module Weight (lb)	Quiet Module Quantity	Quiet Module Avg Pwr	Active Module Weight kg	Active Module Weight (lb)	Active Module Quantity	Active Module Avg Pwr
Non-rack Stor.	796	(1,752)	N/A	-	796	(1,752)	N/A	-
Safe Haven	382	(840)	N/A	-	382	(840)	N/A	-
Skip Cycle Stor.	97	(214)	1	-	97	(214)	1	-
Skip Cycle Stor.	455	(1,001)	1	-	455	(1,001)	1	-
Skip Cycle Stor.	640	(1,407)	1	-	640	(1,407)	1	-
CRT ORU 2 Stor.	668	(1,469)	1	-	668	(1,469)	1	-
CRT ORU 1 Stor.	675	(1,484)	1	-	675	(1,484)	1	-
Non-rack Stor.	947	(2,083)	N/A	-	947	(2,083)	N/A	-
Ops Equip.	177	(390)	1	-	177	(390)	1	-
Galley eq. & Stor.					97	(214)	1	-
Ops. and pers. eq. stor.					269	(591)	1	-
Galley Stor.					99	(217)	1	-
CHC Stor.					177	(390)	1	-
Galley/word rack					98	(215)	1	-
Crew qtrs.	2,330	(5,125)	16	560				
CHC exercise					137	(301)	1	
CHC medical					826	(1,818)	2	134
Ref/frez.					375	(824)	2	229
Laundry/Dishwasher					376	(828)	1	191
Galley/wardroom					1,018	(2,240)	6	160
Galley					221	(487)	1	61
Whole body shower	186	(410)	1	14	186	(410)	1	14
Pers. hygiene	219	(482)	1	35	210	(462)	1	35
Waste mgt	246	(542)	1	81	219	(482)	1	35
EPDS	330	(727)	N/A	460	330	(727)	N/A	460
ECWS	441	(970)	1	537	441	(970)	1	537
DMS/Comm					239	(526)	1	534
Urine processor	292	(643)	1	221	292	(643)	1	221
Hygiene water	267	(588)	1	201	267	(588)	1	201
Potable water					306	(674)	1	206
ARS/ACS	261	(574)	1	490	261	(574)	1	490
ARS	887	(1,952)	2	1,972	887	(1,952)	2	1,972
THC/TCS/AV air	696	(1,532)	2	1,830	696	(1,532)	2	1,830
Standoffs	1,064	(2,341)	N/A	105	1,064	(2,341)	N/A	105
Struc. & mech.	6,788	(14,934)	N/A		6,788	(14,934)	N/A	
Total module	**18,755**	(41,260)		**6,503**	**20,835**	(45,838)		**7,461**

7.7 Pressurized Workshops

The STN has two workshops in proximity to the pressurized access to each of the vehicle stacks. Each of these are contained in a two-thirds length Space Station module. The station module length is reduced in order to control the position of the three pressurized structures in the hangar, the two rotating fixtures and the control node. The two rotating fixtures each must handle a vehicle stack and be as close as possible to the center control node from which manipulators and propellant loading are controlled.

While some maintenance and servicing activities must be done EVA, the workshops provide a shirt-sleeve environment to accommodate as many tasks as possible. The functions to be accommodated in these workshops include Line Replaceable Unit (LRU) diagnosis, LRU storage, maintenance of vehicle subsystems, mechanisms, fluid, mechanical, and electrical interfaces, and controls and displays. Storage of LRUs and other parts and tools will also require space in these modules.

The real number of functions that must be performed in this workspace is not well understood. The goal will be to design the space-based vehicles such that on-orbit work is minimized. The true practicality of on-orbit maintenance and refurbishment remains to be determined however. The modules are therefore sized by the desire to get the two stacks as close together as possible rather than a clear understanding of the functions that must go in them.

A two-thirds length module contains space for 28 standard space station racks. Each module contains several STN core systems including ECLSS and hygiene systems. These core systems occupy 12 to 14 racks leaving the remaining half of the module for working and storage areas. The makeup of the work areas has not been examined in detail. Based on function, one module might be used for mechanical/propulsion work and one for electrical work. Table 7.7-1 is a pressurized workshop weight statement. Figure 7.7-1 is a possible layout of the workshop.

7.8 Nodes and Control Stations

The STN makes extensive use of Space Station Resource Nodes. A total of 10 are included in this configuration. One is used as a hangar control station and is mounted within the hangar itself. Two additional nodes are used to provided pressurized access to the two vehicle stacks. One node is used as an STN control station and is fitted with a cupola to allow direct viewing during Shuttle proximity operations. The remainder of the nodes are used for the connection of the various STN modules and for the storage of STN consumables, ORU's, as well as logistics supplies and spares for the lunar vehicles. Figure 7.8-1 is a general illustration of a Space Station Resource Node. Table 7.8-1 is a summary of the weight of a basic node with only core systems. The weight of stowage items, control stations etc. have been excluded. Figure 7.6-1 shows the location of all the nodes.

Figure 7.7-1, Pressurized Workshop Layout

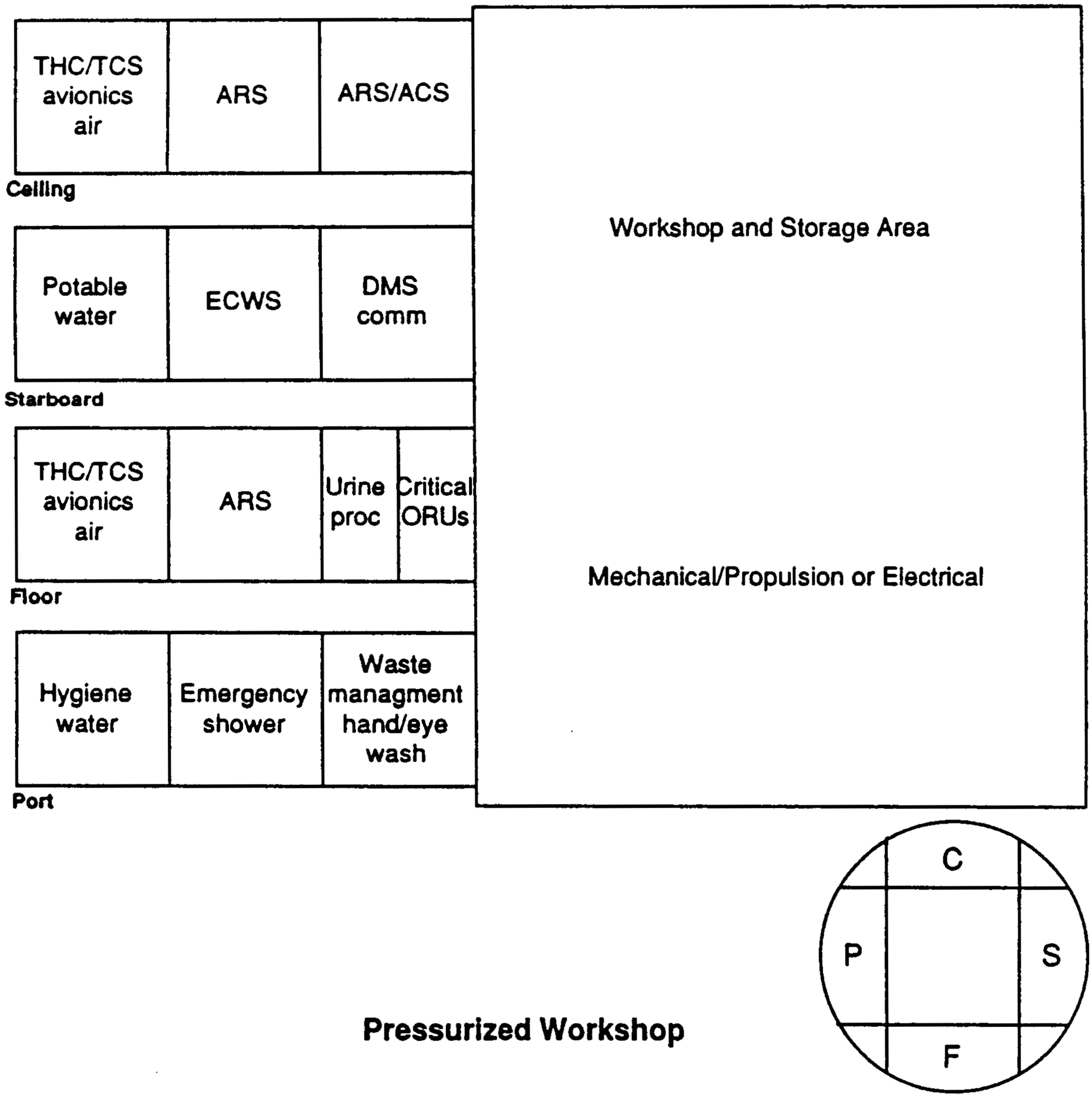


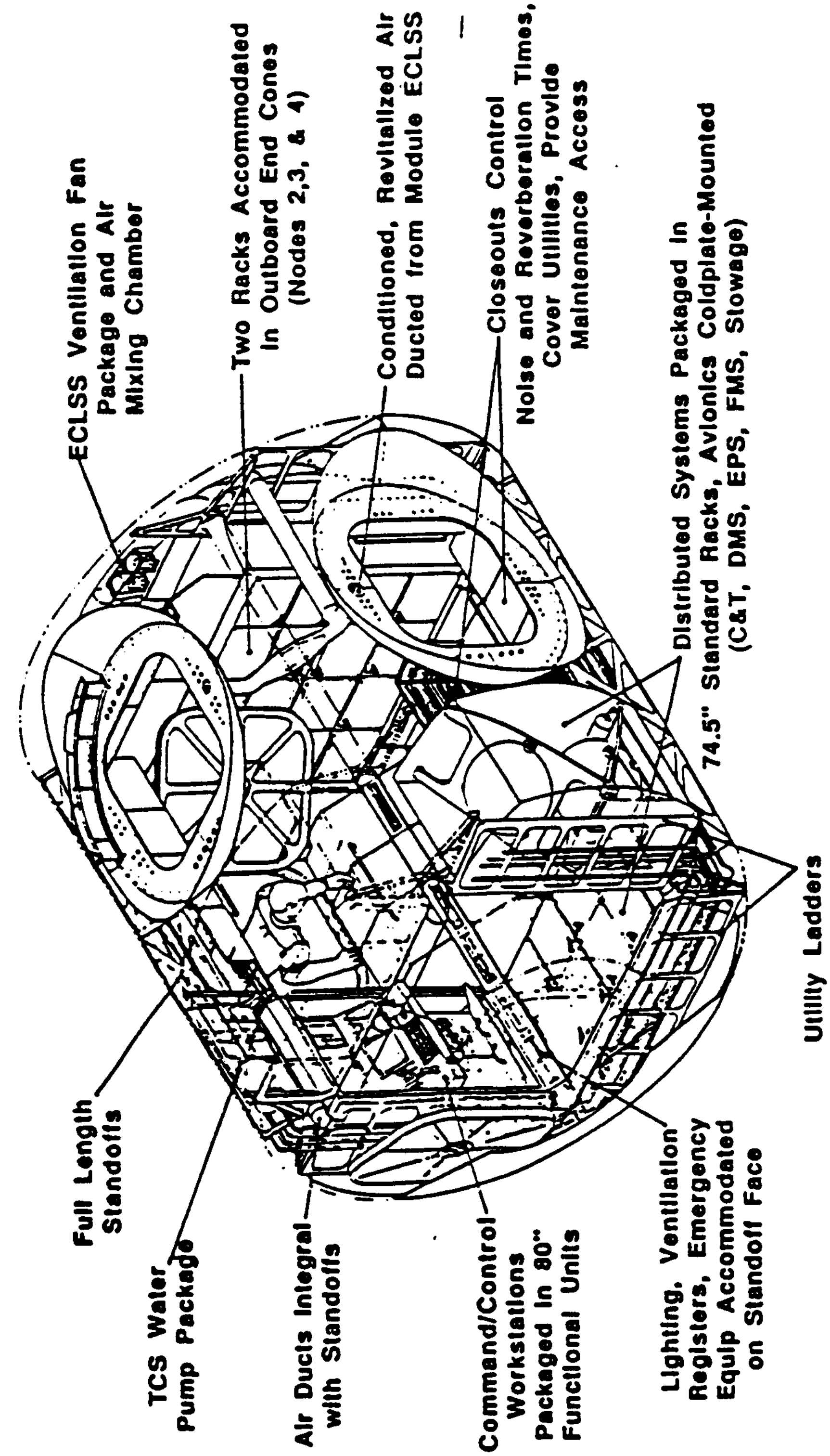
Table 7.7-1, Pressurized Workshop Weight Statement

Unit	Weight kg	Workshop Module Weight (lb)	Quantity
Workshop	3,182	(7,000)	N/A
Whole body shower	186	(410)	1
Waste mgt.	246	(542)	1
EPDS	330	(727)	N/A
ECWS	441	(970)	1
DMS/communication	240	(528)	1
Urine processing	292	(643)	1
Hygiene water	266	(586)	1
Potable water	306	(674)	1
ARS/ACS	261	(574)	1
ARS	887	(1,952)	2
THC/TCS/AV Air	696	(1,532)	2
Standoffs	682	(1,500)	N/A
Structural and mechanical	4,545	(10,000)	N/A
Total	**12,563**	(27,640)	

Table 7.8-1, Node Weight Statement (McDonnell Douglas WP2 Proposal)
(Weight of stowage items and control station excluded)

Equipment Description	**Weight kg**	**Weight (lb)**	**Power W**
Structures	9,629	(21,184)	-
ECLSS	536	(1,179)	712
Mechanisms	3,139	(6,906)	200
Racks and mounting	574	(1,263)	-
TCS	1,375	(3,025)	538
Audio-video	169	(371)	538
Otherman systems	125	(275)	12
Total	**15,583**	(34,283)	**2,000**

Figure 7.8-1, Resource Node (McDonnell Douglas WP2 Proposal)



7.9 Airlocks and EVA Systems (EVAS)

7.9.1 Airlocks

Two Space Station type airlocks (A/L) will be required for the STN configuration. The Airlocks serve as the crew/equipment passageway into and out of the pressurized modules of the STN. One of the airlocks will also serve as the hyperbaric treatment facility for the STN. The airlocks are outfitted with standard docking adapters to connect to the Space Shuttle as well as the STN nodes. The Phase 1 Freedom Space Station also has two airlocks of these types.

The basic configuration of each airlock is a two chamber (cylindrical chamber) design, one chamber being larger than the other. The two chambers are separated by an oval hatch. A similar hatch exits to space. The smaller chamber (sometimes called the crew lock) is used primarily as the ingress and egress port. This smaller chamber is also used as the hyperbaric pressure chamber in the event of an EVA accident. It should be noted that only one of the two airlocks has the hyperbaric capability. The larger chamber (sometimes called the equipment lock) is used primarily for normal EVA preparation such as donning and doffing the EMU and servicing the EMU's and EVA tools.

Trade studies completed for Space Station have concluded this design will minimize the amount of consumables used during EVA activity by using the smaller chamber as the primary ingress/egress port and thereby minimizing the amount of air lost per EVA. The other major advantage of this configuration is a weight savings by minimizing the size of the hyperbaric pressure chamber. If a single chamber is used for the airlock then the entire outer shell would have to be much thicker to accommodate the higher hyperbaric pressures (approximately 6 atmospheres). Only the smaller chamber must be capable of withstanding the hyperbaric pressures in a two chamber design. Another advantage of the two chambers is a built-in redundancy of ingress and egress ports.

The two chambers are structurally similar in that both have an outer primary pressure shell with an internal secondary structure which supports the outer shell and internal outfitting equipment. At the time of this writing the Space Station Program has yet to determine if the two cylindrical chambers are connected in a tee or an in-line fashion (See Figure 7.9.1-1). The two options are being studied to determine which configuration will have a weight and consumable (air and power) advantage.

The larger chamber is also used to house the EMU's (2) as well as the EMU service and performance checkout equipment. This service equipment recharges the EMU and diagnoses the performance of the suit and life support system within the EMU after each EVA. Other equipment within the equipment chamber is the pressurization/depressurization pump and its associated valves, piping, safety and relief devices, and miscellaneous hardware. Stowage of EVA tools and accessories will also be housed in each equipment chamber.

Table 7.9-1 provides weight estimates for the airlocks and other smaller devices.

Figure 7.9-1, Airlock

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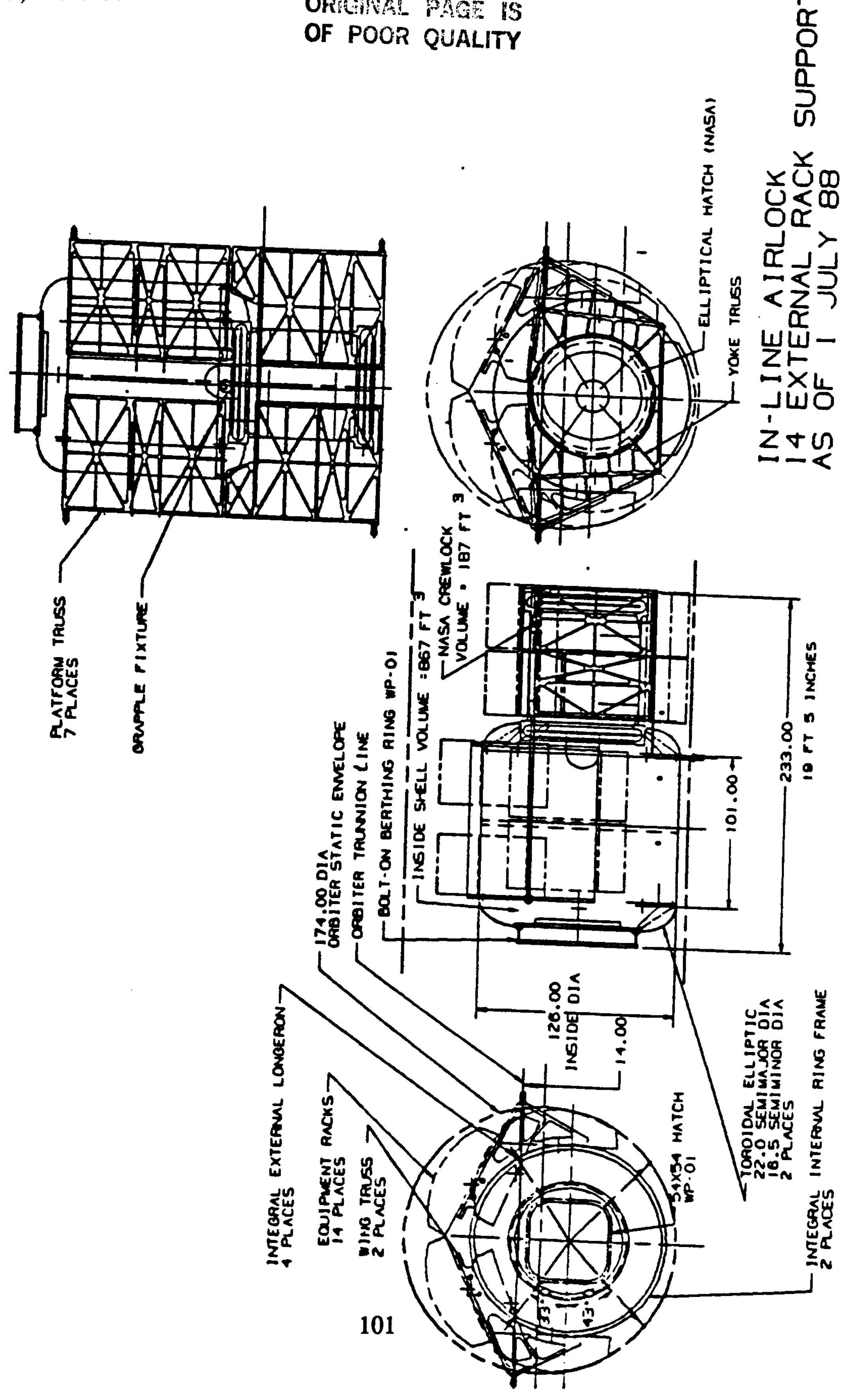


Table 7.9-1, Weight Statement for Airlock/EVA/RMS Equipment kg (lb)

Element	Weight kg	(lb)	Reference
Airlock and EVAS	6,800	(15,000)	WP-01
RMS	900	(2,000)	
Mobile Transporter	1,400	(3,000)	WP-01
Crew and Equipment Translation Assembly (CETA)	700	(1,500)	WP-01

7.9.2 Extra-Vehicular Activity Systems (EVAS)

As a minimum, two extra-vehicular mobility units (EMU's) or space suits will be housed in each equipment lock to accommodate the standard EVA safety and maintenance protocols which require two crew members. Thus up to 4 EVA crew can operate at one time. A fifth EMU, must be stored on board the STN in case of a failure of one of the primary EMU's. It is stowed as components.

The proposed Space Station EMU consists of the space suit assembly (SSA) and the life support system (LSS). The STN EMU's functional design requirements must be similar to the Station suit's in that it must be capable of at least a four to eight hour EVA. This is due to the lengthy spacecraft assembly and maintenance protocols that will probably be required. The suit is made up of an integral helmet, and hard upper torso, a hard brief and hip assembly and fabric arms, legs, and gloves. The LSS consists of the oxygen supply system, CO_2 and trace gas removal system, the communication and tracking module, the fans/humidity control and heat exchanger system, and a self-diagnostic module, all of which is located in the back pack. This back pack is also used as the mounting fixture for the proposed upgraded Space Shuttle Manned Maneuvering Unit (MMU).

Included in the EVAS are the many specialized tools that will be used for the assembly and repair of spacecraft. Translational aids for EVA crew members will be part of the external structure of the pressurized modules as well as the trusses and hangar structure. These aids include work platforms, foot restraints, tethers, umbilicals, and possibly a monorail for translating across a truss or the hangar structure. Commonality with similar proposed Space Station structures and aids will minimize design costs.

Another EVAS requirement will be a decontamination system for crew and/or equipment from any fuel or waste leak. The system must be capable of detection and decontamination of the crew or equipment prior to re-entering any pressurized module, airlock or other vehicle. This system should be designed to be portable to accommodate stationary structures.

7.10 Mobile Remote Manipulator System (MRMS)

The STN requires a minimum of two remote manipulator systems to maneuver large payloads and/or spacecraft. Each RMS is mounted on its own mobile transporter. The RMS must be capable of maneuvering a 200 metric ton departure stack from the rotating fixture to a safe deployment position. The RMS's are also required to manipulate spacecraft throughout the hangar with a variety of other spacecraft or payloads simultaneously docked within the hangar. This requires the translation path/track to be configured to allow the RMS's to travel from one wall to another. The mobile RMS's must allow travel from the interior hangar walls to exterior walls while handling a large spacecraft. This maneuver requires that the hangar doors be open and the translation path/track make a turn around the hangar ceiling or floor (assuming that the hangar doors are hinged from the side walls, which is the current plan). Figure 7.3.1-4 shows the proposed MRMS paths and section 7.3.2 discusses operations within the hangar.

7.10.1 STN RMS

The Space Shuttle and Space Station RMS technology can be used to develop the RMS for the STN with changes in the mass maneuvering capability. The Shuttle's RMS is only capable of maneuvering a payload of approx. 30 m tons. However, the structural, controls and end effector technology which was developed for the Shuttle RMS and will be developed for the Space Station RMS can be transferred to the STN RMS design. Structural strength, rigidity, and power improvements will allow for the added weight capabilities required. New end effector technology is rapidly advancing for many terrestrial applications such as the nuclear industry, underwater robots, hazardous chemical industries, and the Space Station, and can be modified for use in the STN RMS. Advances in Expert System software control systems may greatly improve the handling of large spacecraft in the hangar. Vision systems coupled with Expert Systems may increase the autonomous operation of the STN RMS. The Space Station RMS (Canadian design) and Mobile Transporter (WP-02 design) proposed should offer comparable capabilities to the STN requirements. The Freedom Space Station RMS design is expected to use the Shuttle RMS technology. The STN RMS may not be constrained to the same low velocity of the Freedom Space Station RMS because the STN does not have tight vibration and acceleration requirements. The STN RMS may also be used and moved much more than the Freedom Station RMS, requiring a more rugged design.

7.10.2 Mobile Transporter

The mobile transporter proposed for Space Station relies on maneuvering down the length of a truss by an inch worm motion. The mobile transporter features a dual base made up of an upper base and a lower base. Each base has four corner latches to secure it to the truss members. During translation the upper base slides to the next truss section and lowers its four latches into the truss mating holes. The lower base is then unlatched from the previous truss section and is pulled to the next truss section as the upper base holds tight. The sequence is then repeated. The transporter is also designed to change planes or faces of the truss. This operation is similar to the above with a hinge being activated between the two bases. Inner corners cannot be turned however.

The truss design for the STN is anticipated to be very similar to that of the Freedom Space Station and therefore a direct transfer of mechanical design could be utilized. The STN may require considerably more motion of the transporter than the Freedom Space Station however, and therefore other mechanisms for the transporter such as tracks with rollers require serious consideration. The time frame in which the STN is anticipated to be operational is such that a significant amount of autonomous (vision and expert systems) operation could be incorporated as compared to the baseline configuration of the Space Station.

Electrical power would be supplied by a retractable umbilical cord. Studies completed for Space Station concluded that approximately 600 to 1200 watts of power are required to operate the mobile transporter during various maneuvers. To enhance the autonomous operation of the mobile transporter and the RMS, a portable power system could be added. This system would consist of rechargeable batteries. There would be significant weight penalty (1,500 to 3,000 lbs) for this autonomous operation however.

7.11 Free Flyers and Other Robotic and Telerobotic Devices

7.11.1 Crew and Equipment Retrieval System

The requirement for an emergency retrieval device for the STN is very similar to that of the Space Station. In the event a crew member becomes detached from a tether, there is a malfunction of an MMU, or even a spacecraft coming loose from its mooring, during a reboost or other maneuver some type of retrieval device is needed.

The current safety requirements for Space Station are to not risk another crew member for one lost crew or equipment. This implies the retriever be operated either remotely or be an autonomously operating vehicle. Some communications with the retriever would be necessary to deploy the device and possibly direct it in the general direction of the tumbling crew or equipment. It will also require a sophisticated grapple or end effector mechanism to capture the tumbling object. Technology currently being developed for the Station's retriever can be used for the STN retriever. This technology includes vision systems, communications (voice, video, and data) and tracking systems, propulsion and guidance systems and grappling devices.

A trade study is required to determine if a small retrieval system such as the Space Station's proposed retriever or a Shuttle type of vehicle will be needed. Some of the considerations for this trade include delta-V capability, type of range/rate device, remote controlled or autonomous operation and type of capture mechanism for an undamaged grapple. This device would require enough propellant for the search and return trips and have to be capable of determining the proper rate of departure and approach velocities. The capture mechanism must satisfy safety issues concerning the grappling of a crew member without damaging the suit, stopping the crew from tumbling, and redirect both itself and the crew back towards the STN.

7.11.2 Crew and Equipment Translation Aid (CETA)

To rapidly and safely transport crew and equipment around the hangar, a transporter may be required. A variety of options exist, ranging from a manually controlled and powered system similar to the Freedom Station design to a fully automated transporter capable of autonomous operations.

An automated transporter should be capable of being programmed from the STN control stations as to pickup points and destination points for equipment without the need for an EVA crew. The translation path and mobility method could be the same used for the RMS mobile transporter and thus reduce design and manufacturing costs. This may cause some operational interference with the RMS.

The above functional requirements dictate that the CETA be either driven by a portable power source or a retractable umbilical. There will also be a need for communication with the control center via the umbilical or an antenna. An on-board vision system coupled with an expert system would allow the CETA to locate objects and propel itself to and from various work stations. The CETA also requires a crew carrier/work platform and an equipment mounting platform.

7.12 Truss Structure

To keep development and production costs at a minimum, many components for this transportation node are identical to, or derived from, parts developed for previous programs. The truss assembly is no exception and is primarily based on the Integrated Truss Assembly (ITA) concept conceived by the McDonnell Douglas Astronautics Company - Space Station Division's winning Work Package 2 Technical Proposal. Other options, such as the Lockheed aluminum clad cast and cured strut must also be considered in later work.

The ITA provides a lightweight structure for the physical integration of the Transportation Node systems and elements. The ITA further provides integration of the distributed systems, distribution of station resources, solar power and thermal radiator pointing, mobile transporter roadbed, crew/equipment movement provisions, external lighting, and fluid systems. The design features a deployable utility distribution system and a user-friendly, efficient equipment-packaging concept. These features minimize EVA and reduce assembly risks.

The truss structure provides a stiff, thermally stable framework for attachment and support of other systems, elements, and payloads. It consists primarily of a 16.4 ft. (5 m) erectable truss with graphite-epoxy struts and aluminum nodes. A 246 ft. transverse boom forms the forward section of the assembly and supports an alpha joint and radiator panels on either end (see Fig. 7.12-1). The transverse boom attaches to a 164 ft. lower keel truss which acts as the backbone of the station, providing support for the attached modules. Finally, two 150 ft. x 131 ft. rectangular assemblies and a 164 ft. upper keel truss provide stability for the hangar and lunar support operations which take place within it. The structure is pre-assembled as much as possible to minimize on-orbit assembly operations, and each member may be replaced individually to maximize maintainability.

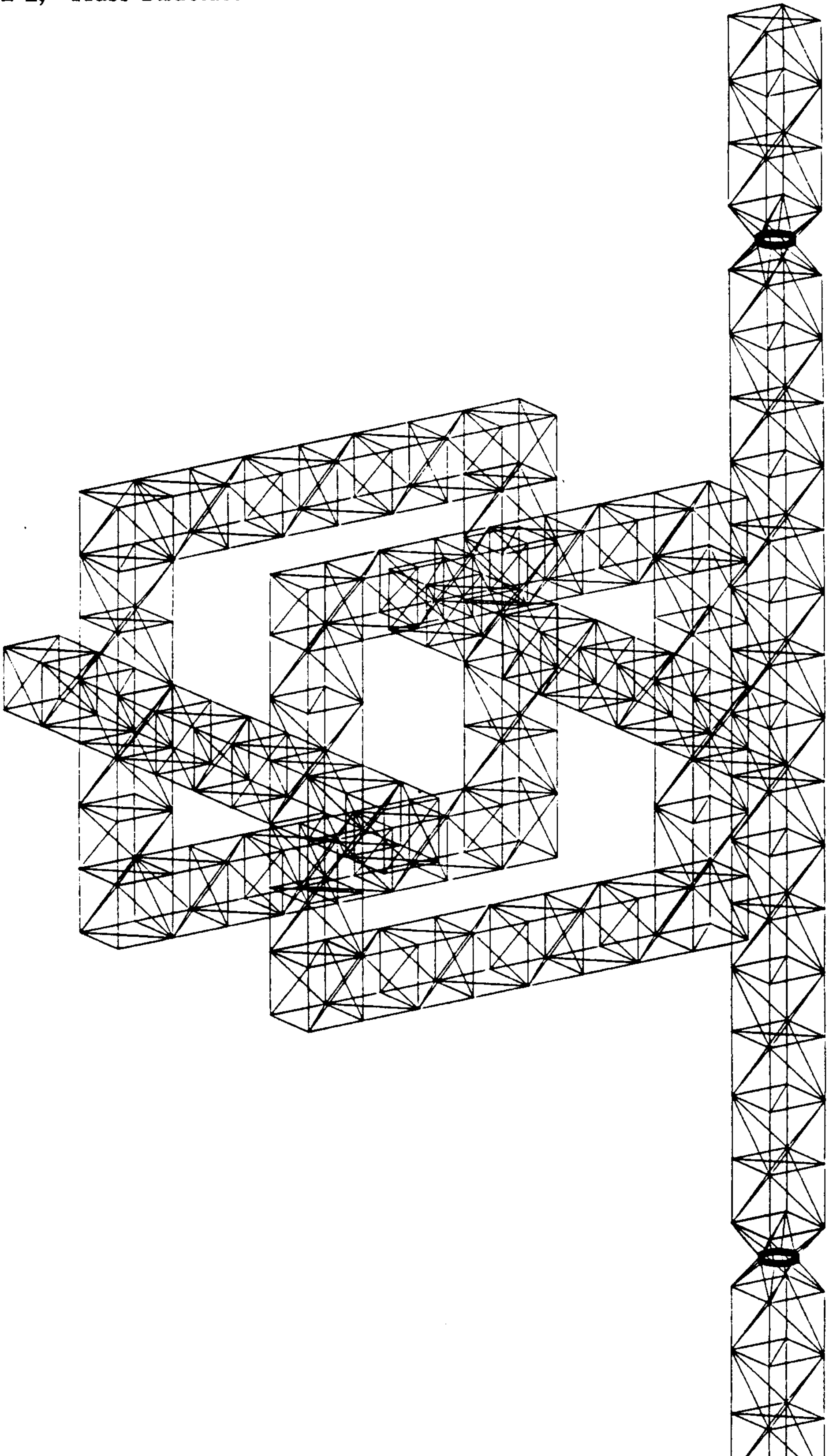
The strut material is forty million lb/in^2 (msi) modulus filament-wound graphite epoxy (T-40) selected for its cost, weight, temperature and expansion characteristics, and fabrication maturity. Graphite epoxy has a near zero thermal coefficient of expansion over a wide temperature range. An aluminum foil covering is bonded to the outside of the tubes to protect the material from atomic oxygen and UV radiation. These struts are assembled into a four longeron truss with alternating face and batten diagonals. This design provides a factor of safety of 1.0 with one strut out. This configuration also provides a 50 x 72 in. EVA corridor inside of the truss.

The alpha joints are a 12-joint transition structure with a 120 in. diameter rotary joint designed by Lockheed. The solar alpha rotary joint supports the transverse boom and provides controlled rotation to point the power generation equipment toward the sun, while transferring power and data across this rotating interface.

The thermal radiator rotary joint supports the central radiator panels and provides controlled rotation for aligning the panel edges to the sun. It transfers liquid/gaseous ammonia between the station and the panels.

Finally, aluminum utility trays run throughout the truss assembly to distribute station resources and fluids. They provide protection to cabling and piping from UV radiation, atomic oxygen, and meteoroid-debris impact, and provide numerous utility ports for electrical and fluid interfaces. External lights are integrated into the utility trays and installed on the truss nodes. Systems are needed for illuminating EVA traverse routes for crew safety and for lighting payloads, hazardous areas, worksites, and other exterior surfaces and equipment. This integration approach and the use of quick disconnects minimize EVA time for assembly and maintenance.

Figure 7.12-1, Truss Structure



7.13 Hangar Tunnel

To supervise and control activities taking place in the hangar bay, a tunnel/cupola structure has been defined. The tunnel attaches to common nodes at both ends using standard hatch and berthing mechanisms (50 in. square hatches); the cupola would then connect to the upper node (see Figure 7.6-1).

The primary purpose of the tunnel is to allow pressurized passage from the main module grouping to the hangar/node and cupola, therefore the inside diameter is equal in width to the interior diameter of a full entry cupola, approximately seven ft. Provisions must also be made for allowing data transmission/communications cabling and ventilation ducts between the cupola and module grouping. To facilitate movement through the tunnel and provide support within it if needed, standard handholds are located at two foot intervals on both sides of the tunnel for its entire length.

It is evident from previous studies that the primary driver in space pressure vessel wall design at internal pressures of one atmosphere is the ability to resist puncture by small scale space debris and micrometeoroids. Although the hangar walls will provide a certain measure of protection, the hangar doors will be opened regularly, therefore the tunnel has been defined as if it were fully exposed to the space environment. The vessel is constructed as a dual shelled tube with an inner wall of 0.125 in. aluminum and an outer aluminum bumper wall of 0.063 in. The shells are held together with an aluminum waffle construction I-beam web. The vessel has a total thickness of 4.5 in. and it is hollow between shells.

8.0 Distributed Systems Conceptual Design

Distributed systems are those systems that are spread out, throughout the STN and not confined to any one element. Data Management, communications and tracking, GN&C, electrical power, propulsion, thermal control and ECLS systems all fall within this definition.

8.1 Data Management System (DMS)

8.1.1 Architecture

The DMS architecture is a large scale distributed processing network for long term use in space. It provides a growth oriented base for automation to increase crew productivity, thus enhancing the station's operational capabilities. The DMS uses common hardware resources and software services to achieve an architecture that is physically distributed yet functionally integrated.

The system consists of a distributed network of smaller hybrid processors that are capable of both numeric and symbolic processing. These processors are connected with a fiber optic network that utilizes a dual counter-rotating ring configuration.

The DMS software is physically distributed, but functionally integrated into a cohesive operational environment and command and control framework. Interface details and the physical location of resources are transparent to the user. The key DMS software-to-user interfaces interconnect people, applications, and databases to form an architecture that is an integral part of the overall information system. This connectivity with ground elements allows transparent command and control data base exchanges.

8.1.2 Automation and Robotics and the DMS

The DMS of the STN will interface with and control numerous automated systems and robotic devices. Some automated systems will be part of the DMS and other systems will only interact with the DMS. Automated signal acquisition for downlink/uplink data, automated diagnostic systems for the health and care of the DMS, automated resource allocation and monitoring, automated switch over to redundant systems, automated data capture and storage of data, automated dump sequences, and automated reconfiguration for mission scenarios, will all be included within or connected to the DMS.

The DMS of the lunar transfer vehicles will be highly automated. This however, does not preclude the system being overridden by ground controllers, other vehicle operators, and EVA crew working on or near the transfer vehicle.

The STN DMS will have to interact with other automated systems that will be distributed throughout the transfer vehicles and STN. These systems will use the STN DMS as the backbone of their own network to distribute data and command sequences, store data, store and execute time tagged commands, distribute sensory data, and in some cases the DMS will be used as the network for switching to redundant systems. These automated systems may be thought of as a subset of the DMS.

Robotic and teleoperated devices will be an essential part of large spacecraft maintenance and propellant loading in space. Therefore, it is important that the DMS be able to support these devices. To do so, the DMS, will have to be able to handle high data rates that will be generated by these devices (both digital and video). The robotic devices will obtain from the DMS such information as vehicle systems status, resource availability information, guidance information, tracking information, and command/control information.

The robotic devices will rely on the DMS to receive instructions from a multitude of sources and report to the same. The DMS will be the primary interface for the control and monitoring of such devices. It is important to note that the DMS in its role of command and control of the robotic device will have to route inputs to the device from the vehicle, the ground, the EVA crew, and other vehicles.

8.2 Communication and Tracking (C&T) System

The C&T system provides transparent transmission and reception of audio, video, telemetry, commands, text and graphics, and user data. It also provides tracking data and onboard audio and video services.

8.2.1 Space to Space Radio

The space to space radio provides communication between the Space Station and an EVA astronaut, Free Flyers, Space Shuttle, OMV, and the Flight Telerobotic Service (FTS). It uses a Ku-band, frequency division multiple access system, providing continuous proximity operations coverage using automatic antenna switchover. The parabolic antennas are sized for 2,000 km range, and may be expanded by modular additions. The design supports the video/data requirements for the FTS and teleoperation functions.

8.2.2 Video

The video subsystem uses wideband distribution to support both standard and high resolution TV. It employs fiber-optic point to point video links and a distributed solid state digital-switch network to interconnect video cameras, monitors, recorder, and communication links. Six simultaneous downlink video channels and space to space channels provide video in each direction. The video system supports the assembly, surveillance, EVA tracking, docking and berthing, teleconferencing, public affairs, experiments, etc.

8.2.3 Radar

The radar system will be used as a rendezvous and docking aid. When a vehicle is at a distance, the radar will be used for range, rate, and position determination. When the vehicle gets closer, the radar will only be used for range and rate determination. A proximity operations system using a laser or other technique may also be required for actual docking of large unmanned vehicles.

8.3 Guidance Navigation and Control System

8.3.1 System Description

The Guidance Navigation and Control (GN&C) system will provide attitude control for the STN as well as pointing of the power system and thermal radiators. In addition, it will provide STN state and attitude information to other systems, and it is responsible for controlling incoming, outgoing, and station keeping traffic within the Command and Control Zone (CCZ) of the STN. The GN&C must also control docking and berthing operations and monitor the trajectories of vehicles and objects which may intersect the orbit of the STN to predict potential collisions. The GN&C system consists of Inertial Sensor Assemblies (ISA)'s, star trackers, Control Moment Gyros (CMG's) which are located at the Attitude Control Assemblies (ACA's); and the Standard Data Processors (SDP's). The most important interface of the GN&C system is with the Reaction Control System (RCS) which provides part of the attitude control of the STN. This interface is through the propulsion system electronics located at each Reaction Control Module (RCM).

8.3.2 GN&C Subsystems Location and Selection Criteria

The STN GN&C subsystems will be similar to those of the Space Station except for the number of subassemblies and their location. The ACA's are located primarily on the transverse boom at the first truss bay inboard of the port and starboard alpha joints. Since the CMG's and ISA's functions are independent of location, the locations of the ACA's were chosen primarily by maximizing the look angles of the star trackers. The SDP's will be located in two of the resource nodes.

8.3.3 System Failure Criteria

The overall GN&C subsystem function is considered both safety and time critical. Therefore, the system must be configured to provide double redundancy to support a fail-safe capability with the failure of any two GN&C components.

8.3.4 GN&C Functions

The functions of the GN&C are similar to the Space Station GN&C functions outlined in JSC 30259. The following functions should be considered in all GN&C design phases.

- Reboost and collision avoidance targeting
- STN translation maneuvers guidance
- STN pointing commands
- Maintenance of star catalog
- STN state and attitude determination
- STN state propagation
- STN state prediction for STN maneuvers
- Momentum management control laws
- Translation and attitude control laws during translation burns
- RCS control laws and jet commands

- System moding and reconfiguration

- Fault detection, isolation and reconfiguration for GN&C subassembly components such as star trackers, ISA's, CMG's, power system drive electronics and RCM electronics
- Mass properties extraction
- Closed-loop control of the power system alpha joints
- Pointing information to the power system for their beta gimbal control and to the thermal systems for their radiator control
- Issuance of constraints on hangar activities such as berthed OTV angular rotation and angular rates, RMS displacements, rotations and rates, fuel loading management and overall fluid systems management

Other GN&C functions related to traffic management within the CCZ are:

- Constellation state determination, propagation and prediction
- Relative state and attitude determination between the STN and incoming vehicles
- Relative maneuver coordination
- Collision monitoring
- Flight planning for remote vehicles
- Backup navigation for manned vehicles
- Translational and rotational commands for unmanned remote vehicles which are performing proximity operations with the STN
- Override commands for remote vehicle docking and berthing

Most of these functions are software related and will be taken care of by the SDP's. The actual translation and rotation commands will be issued by the SDP's to the CMG and RCS electronics to correct for environmental and operations disturbances.

The primary concern of this section is the conceptual design and sizing of the primary control hardware, namely CMG's and RCM's. The software and electronics necessary to take care of all functions are assumed to be available.

8.3.5 Control Moment Gyros and Reaction Control Modules

The hardware to be used for the attitude control of the STN will be double gimbal control moment gyros (DGCMG)'s and reaction control modules (RCM)'s. These were selected because they both are mature subsystems in the Space Station and they both may be utilized as complete, isolated modules that can be selected according to the number required for the attitude control of the STN. Table 8.3.5-1 shows the design and performance characteristics of the DGCMG's and the RCM's. Six DGCMG's are located on each ACA with their spin axes pointed in the same direction.

Table 8.3.5-1, ACA/DGCMG and RCM Performance

ACA/DGCMG	
Total angular momentum stored per ACA:	28,479 N-m-sec
Total torque per gimbal per ACA:	1,627 N-m
Gimbal travel angle:	$\pm$ 20°
RCM	
Thrust per axis:	334 N
Isp:	380 sec

8.3.6 Attitude Control System Design

8.3.6.1 Background

The primary concern in the attitude control system design is to ensure that the system will be able to handle attitude disturbances caused either by the environment or by operations within the STN. Environmental attitude disturbances are due primarily to upper atmospheric drag forces, solar pressure forces and gravity gradient forces. Operations disturbances are caused primarily by rotating masses such as solar panels, radiators, berthed OTV's, rotating machinery, and by mass transfers within the STN such as propellants, other fluids, RMS's and crew.

8.3.6.2 Requirements

Design requirement WBS no. 1.01 calls for no mission pointing or orientation requirements; therefore, the requirements on the attitude control system design will be established on the basis of maintaining the STN attitude within acceptable limits of all angles, angular rates and angular accelerations for all operations within the STN and the CCZ.

8.3.6.3 System Design

As the design attitude limits have not yet been established, the design approach may be other than the classical inverse design approach. In this case, the design approach of the attitude control system was to use existing Space Station hardware (DGCMG's and RCM's) together with known physical properties of the STN to estimate the angular rates and accelerations attainable with single hardware units. As the requirements become available, these unit rates and accelerations may be multiplied times a number of units or sets of units until the requirements are met, yielding a number of hardware units.

The design approach is as follows:

Attitude disturbances, whether caused by the environment or by operations, will translate directly into moments about the CG of the STN. These moments will cause the STN to attain angular accelerations and angular rates that, over a period of time, will result in changes to its attitude that will be unacceptable given a set of attitude requirements. Furthermore, the disturbances are time variant such that the attitude control system must always be prepared to take care of any change in attitude within an acceptable time limit. A better representation of the attitude changes of a rigid body may be made by referring to the changes in torques that result in a rate of angular momentum about its center of mass, or in general,

$$\mathbf{M} = d\mathbf{h}/dt = [d\mathbf{h}/dt]_b + \mathbf{w} \times \mathbf{h} \qquad \text{(Eqn. 8.3-1)}$$

where:

- M = moment or torque vector
- **h** = angular momentum about the CG
- d/dt = rate of change with respect to time
- **w** = angular rate
- **x** = vector (cross) product

Expanding the R.H.S., the sum of all torques acting on the body, in cartesian coordinates, is expressed as:

$$M_x = h_x + w_y h_z - w_z h_y \qquad \text{(Eqns. 8.3-2)}$$
$$M_y = h_y + w_z h_x - w_x h_z$$
$$M_z = h_z + w_x h_y - w_y h_x$$

Furthermore, the rate of angular momentum as well as the angular momentum may be expressed in terms of the body moments and products of inertia in the following form:

$$h_x = I_x \alpha_x \qquad \text{(Eqns. 8.3-3)}$$
$$h_y = I_y \alpha_y$$
$$h_z = I_z \alpha_z$$

and

$$h_x = I_x w_x - I_{xy} w_y - I_{xz} w_z \qquad \text{(Eqns. 8.3-4)}$$
$$h_y = I_y w_y - I_{xy} w_x - I_{yz} w_z$$
$$h_z = I_z w_z - I_{xz} w_x - I_{yz} w_y$$

where : $I_{x,y,z}$ = moments of inertia about x,y,z axes
$I_{xy,xz,yz}$ =products of inertia about x,y,z axes
$\alpha_{x,y,z} = w_{x,y,z}$ = angular accelerations about x,y,z axes
$w_{x,y,z}$ = angular rates about x,y,z axes

For the purpose of this study only uncoupled rotations were considered; therefore, it is assumed that only single axis torques are applied to the STN at any one time and that the moments of inertia lie on the principal axes (i.e. $I_{xy}=I_{xz}=I_{yz}=0$). Applying these assumptions, equation (8.3-2) may be written in terms of the body moments of inertia as follows:

$$M_x = I_x \alpha_x \qquad \text{(Eqns. 8.3-5)}$$
$$M_y = I_y \alpha_y$$
$$M_z = I_z \alpha_z$$

and equations (8.3-4) may be expressed as:

$$h_x = I_x w_x \qquad \text{(eqns. 8.3-6)}$$
$$h_y = I_y w_y$$
$$h_z = I_z w_z$$

Equations (8.3-5) are the simplest form of equation (8.3-1), and together with equations (8.3-6), represent the heart of the attitude control system design. Using these equations, angular rates and accelerations may be obtained if the inertia characteristics of the body are known. This is true whether the moments are due to external or internal forces, as long as the same assumptions for uncoupled rotations are made. Integration with respect to time using these accelerations and rates will yield the particular attitude change for any or all three axis.

In the case of the STN, all that is required is that the mass properties be known to estimate all angular rates and accelerations, since the torques applied by the DGCMG's are known a priori and the torques applied by the RCM's can be estimated once their relative location is known. The mass properties are given in section 9.0. These rates and accelerations will be the maximum attainable for any given set of mass properties and will serve as a guide and basis for estimating the required sets of DGCMG pods and RCM's for a given set of attitude requirements.

8.3.7 Attitude Control System Performance

As mentioned in section 8.3.2, the locations of the ACA's were selected primarily by maximizing the star tracker's look angles. The performance of the DGCMG's is independent of location; therefore, the ACA's may be located anywhere on the truss structure and point in any direction from the standpoint of performance. However, the locations of the RCM's were chosen based on two criteria; to minimize the possibility of plume impingement and to maximize the moment arm with respect to the location of the CG. Figure 8.3.7-1 shows these locations.

The symmetry of the STN configuration about the X-Z plane is such that the RCM's may be located in sets of two, one on either side of this plane. This is convenient because, independent of the location of the CG, RCM torques about any one axis can be applied as pure couples, cancelling any forces associated with firing RCM thrusters. This facilitates the analysis as well as the overall design. These sets are also shown in Figure 8.3.7-1.

With the location of the RCM's and their performance known, the torques applied by each set of RCM's is substituted in equations (8.3-5) to obtain the maximum attainable angular accelerations for each axis. Table 8.3.7-1 shows the results for each of the RCM sets for all axes (pitch, roll and yaw) and for every STN loading case. With the performance characteristics of the DGCMG's known, the torques applied by the gimbal motors were substituted in equations (8.3-5) and in the same fashion as for the RCM's, the maximum attainable angular accelerations were obtained. Then, the total angular momentum stored per ACA was multiplied times the sine of the maximum gimbal travel angle of the DGCMG's and was substituted in equations (8.3-6) to obtain the maximum attainable angular rates for each axis. This was also done for all three STN load conditions. These results are presented in Table 8.3.7-2.

When the attitude control requirements of the STN are established, Tables 8.3.7-1 and 8.3.7-2 will serve as a comparison against the desired rates and accelerations. If more control power is desired, a decision can be made as to the extra number of ACA's and/or RCM's based on these results, however, other factors must be taken into account, such as RCS duty cycles, CMG desaturation, failure modes, redundancy, etc to arrive at a definitive design.

Table 8.3.7-1, RCM Performance Per Set

	Pitch							Yaw			Roll
RCS Set #	1	1 + 2	1 + 3	2	2 + 3	3	1 + 2 + 3	1	3	1 + 3	2
Dry	+.00323 -.00064	+.00696 -.00437	+.00905 -.00645	±.00373	±.00955	±.00582	±.01277 -.01018	±.01168	±.00568	±.01737	±.00381
Wet	+.00376 -.00137	+.00814 -.00574	+.00932 -.00556	±.00438	±.00994	±.00555	±.0137 -.00040	±.0072	±.00351	±.01072	±.00237
Gross	+.00298 -.00110	+.00518 -.004334	+.00598 -.00413	±.00220	±.00520	±.00299	+.00817 -.00632	±.00485	±.00237	±.00721	±.00243

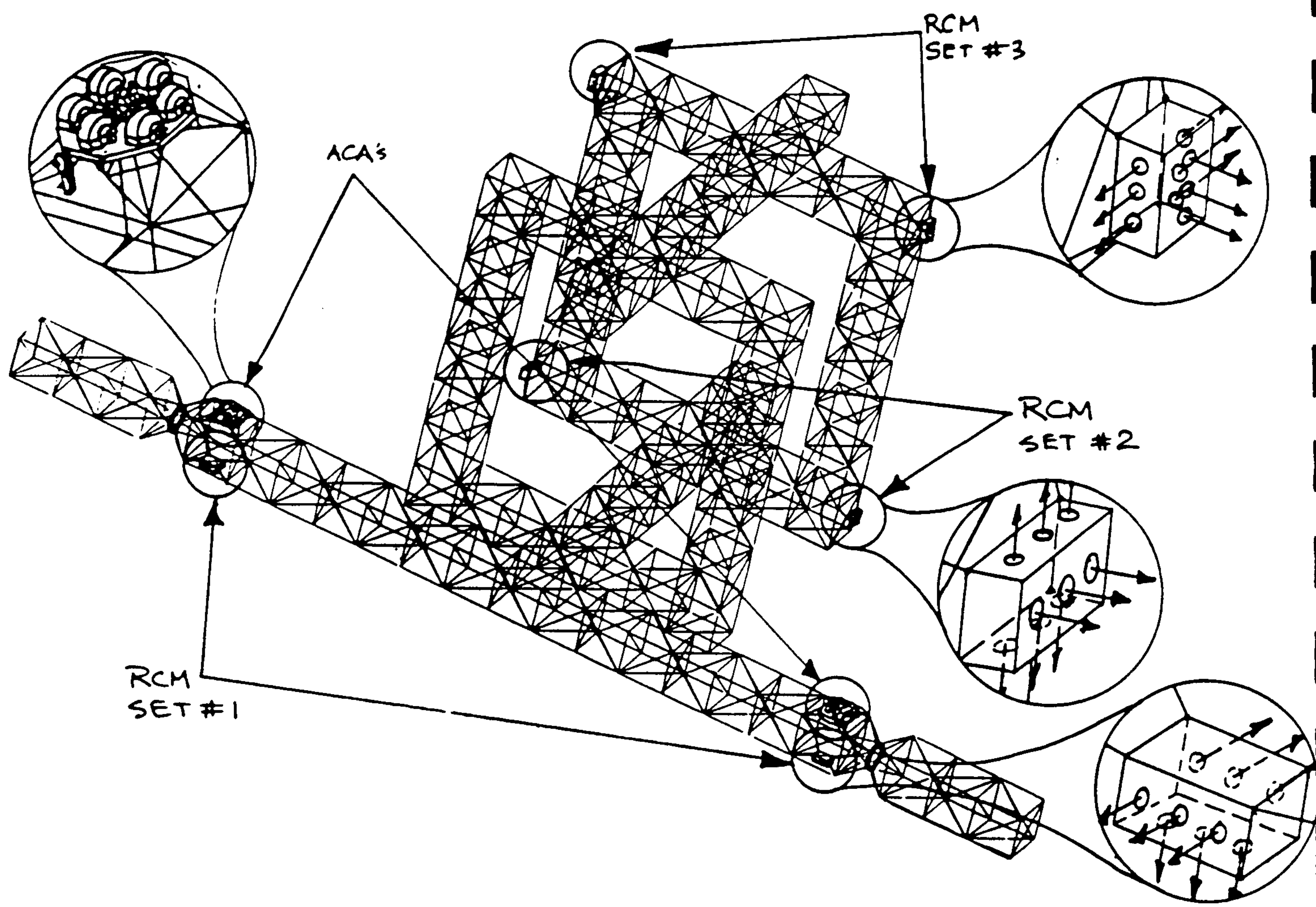
RCS (uncoupled motion) angular accelerations, deg/sec^2

Table 8.3.7-2, ACA/DGCMG Pod Performance (ACA pod contains 6 DGCMGs)

	Pitch		Yaw		Roll	
	w	α	w	α	w	α
Dry	±.00236	±.00039	±.00463	±.00078	±.00311	±.00052
Wet	±.00259	±.00042	±.00285	±.00047	±.00190	±.00032
Gross	±.00151	±.00026	±.00192	±.00032	±.00198	±.00033

w = Angular rate (°/sec)
α = Angular acceleration (°sec^2)

Figure 8.3.7-1, RCM Location



8.4 Electrical Power System

Table 8.4-1 provides a first estimate of the power requirements for the STN. More detailed study may show that all these items are unlikely to all function at once, but more study will probably also uncover other items that were forgotten or underestimates for named items. It therefore appears that the 75 kW continuous power (Phase I Freedom Station) configuration of arrays and batteries such as is shown in Figure 1.0-3 will be required. Table 8.4-2 is a weight statement representative of this system.

Some of the illustrations of the STN in this book, such as shown on the cover, only show one set of arrays on each side. This is a 37.5 kW continuous configuration and will probably not be adequate. Two sets on each side appear to be required.

Each photo-voltaic power increment contains two solar array wings and associated equipment, and allows it to operate independently of the other increments. Nickel hydrogen batteries are used for energy storage. Power converters convert the output to 20 KHz single phase AC power, which is distributed throughout the STN.

Power is distributed throughout the LAB, HAB, and LOG modules by locating the WP04-supplied power distribution and control units in the module end cones. The electrical power distribution system design within each element preserves the two failure tolerance inherent in the dual-ring bus architecture and has the ability for inter-module power transfer of 50kW. Redundant housekeeping subsystems located in separate racks require only a single power feed to any rack location. The ring bus allows at least two ways to feed any rack through the power distribution and control units. There is a 30% design margin in specifying the current ratings of all wiring and converters unique to WP01. Rack feeder ratings (3, 6, and 15 kW) are approximately 50% higher than most racks require, allowing future growth and rack function interchangeability.

A common rack electrical design for the HAB, LAB, and LOG modules reduces development and recurring costs and simplifies rack maintenance. Racks have a common power protection assembly that houses the manual circuit breakers and distributes power to individual loads.

Table 8.4-1, STN First Guess Power Requirements

Item	Avg. Power, Watts	Ref.
Habitation Module 1 (Active)	7,461	Table 7.8-1
Habitation Module 2 (Quiet)	6,503	Table 7.8-1
Workshop Module 1	4,974	2/3 x 7,461
Workshop Module 2	4,974	2/3 x 7,461
Logistics Module (Pressurized)	1,010	(McDonnell, 1988)
Node 1 (Control Node)	3,260	(McDonnell, 1988)
Node 2	2,280	(McDonnell, 1988)
Node 3	2,280	(McDonnell, 1988)
Node 4	2,280	(McDonnell, 1988)
Node 5	2,280	(McDonnell, 1988)
Node 6	2,280	(McDonnell, 1988)
Node 7	2,280	(McDonnell, 1988)
Airlock 1	180	(McDonnell, 1988)
Airlock 2 (Hyperbaric)	130	(McDonnell, 1988)
Rotating Fixture 1	1,000	Estimate
Rotating Fixture 2	1,000	Estimate
Truss Structure	6,007	(McDonnell, 1988)
Tunnel	2,000	Estimate
Hangar Control Node	3,260	(McDonnell, 1988)
Hangar Interior	2,000	Estimate
OTV 1 housekeeping	1,150	Estimate
OTV 2 housekeeping	1,150	Estimate
Lander 1	2,000	(Eagle, March 30, 1988)
Lander 2	2,000	(Eagle, March 30, 1988)
Cryo Module 1-4	1,000	Section 7.5.2
RMS 1	1,000	Section 7.10.2
RMS 2	1,000	Section 7.10.2
OMV 1 housekeeping	100	Estimate
OMV 2 housekeeping	100	Estimate
Propulsion System	3,430	(McDonnell, 1988)
Total	**70,369**	

Table 8.4-2, Power System Weight Statement

Top Level Weight Statement (75 kW system)

	lb	kg
Power System* (breakdown below)	33,910	15,414
Structures	6,225	2,830
Mechanisms	4,848	2,200
Thermal Control	11,192	5,087
Data Management	248	113
EVA Systems	636	289
Total	57,059	22,936

* Composed of four photo-voltaic power modules, two that weighted 8,829 lbs and two that weighed 8,127 lbs. See lower level of detail representative weight statement below.

Inboard Photo-Voltaic Power Module, Starboard

Item	No. of Units	Unit lb	lb	kg
Solar Array Assembly				
Photo-Volt. Blanket and Box (L)	2	488.5	977.0	444
Photo-Volt. Blanket and Box (R)	2	488.5	977.0	444
Mast and Canister	2	230.0	460.0	209
Sequential Shunt Unit	2	37.5	75.0	34
Energy Storage Assembly				
Battery Assembly	15	242.5	3,637.5	1,653
Charge/Discharge	5	160.0	800.0	364
Electrical Equipment				
DC Switch	2	257.0	514.0	234
Main Invertor Unit	2	205.0	410.0	186
PV Controller	2	111.0	222.0	101
MBSU (Less 2BIA & 2EDP)	2	138.0	276.0	125
PDCU (Less 2BIA & 1EDP)	2	213.0	426.0	194
Total			8,829.0	4,013

8.5 Propulsion System

In the case of the STN the propulsion system has a dual role: 1) To provide attitude control in orbit for rotational and translational maneuvers and 2) To provide propulsion in the direction of the flight path either to make up for Δv losses due primarily to atmospheric drag or to boost to a higher orbit. The first role has already been described in the GN&C section. In this section, orbit boost requirements are established, the type of propulsion selected, and its performance presented.

8.5.1 Propulsion Types

Two types of propulsion systems are available for Space Station. RCS thrusters and resistojets. To assess the feasibility of utilizing either or both systems, it was necessary to estimate the effects of the upper atmosphere on the STN. Using an orbital decay code, the average drag force and average orbit decay times were estimated for altitudes from 280 km to 500 km in steps of 1 km and for all three STN ballistic numbers. In this case, average refers to the standard 1976 atmosphere without diurnal effect or other corrections. These results were obtained for the following densities:

- Lowest at sunspot minimum
- Average at sunspot minimum
- Average at sunspot maximum
- Highest at sunspot maximum

8.5.2 Drag Force and Orbital Decay

The average drag force is independent of mass and therefore is the same for all STN loadings. Table 8.5.2-1 shows the results for all four densities, and Figure 8.5.2-1 shows their relative trends. Table 8.5.2-2 shows the orbit decay times for all three STN load conditions respectively and each for all densities up to 2000 days. Also, Figures 8.5.2-2, 8.5.2-3, and 8.5.2-4 show the relative differences between densities and ballistic numbers. In these figures and tables the "dry weight" is the weight less propellants and spacecraft and is assumed to be approximately 347 m tons. The "wet weight" adds stored propellants and is 528 m tons. The "gross weight" adds two fully loaded stacks and is 954 m tons. Note that these are first iteration weights and differ somewhat from later heavier weights found in the weight statements in section 9.0. The STN drag coefficient used is an average free molecular flow coefficient from available flat plate data. The reference area was estimated using the maximum truss, hangar, modules and solar panel frontal area in the +X direction. The center of pressure was also estimated using these areas (hangar doors closed) and its location is at Y = - 32.61, Z = - 994.25 cm.

8.5.3 Resistojets

The function of resistojets is to provide thrust by heating a variety of either single or mixed non-reactive gases and by expanding the hot gas through a high area ratio nozzle. Heating the gas to sufficiently high temperatures provides molecular dissociation of the gas and prevents recondensation, which minimizes contamination of adjacent surfaces. The specific impulse of resistojets is usually high, however, they have thrust levels that are only in the order of 50 to 100 millipounds. This is reasonable when compared to the maximum drag levels of one or two pounds that are experienced in orbit since only a few resistojets would be necessary to overcome the force due to drag. The resistojets can be firing constantly and thus prevent any orbit decay. The number of resistojets baselined for Space Station is six, which are placed in a cluster as a single module. Table 8.5.3-1 shows resistojet performance parameters for several gases. Based on this performance and on the average drag force acting on the STN for the given solar activity densities, as shown in Table 8.5.2-1, one module of six resistojets with a maximum total force of 1.87 N would be sufficient to maintain the STN above 413 km for the worst solar activity case. This could be achieved by burning any of the gases in Table 8.5.3-1. H_2 or O_2 from the main tanks could be used; however, a trade-off analysis between power required and mass flow rate required must be done to assess its impact on propellant storage requirements and overall subsystems power requirements. It is recommended that two resistojet modules be baselined for the STN for redundancy. Also, a more detailed analysis is necessary to choose the location of the modules to minimize any adverse effects on the attitude control system. If six resistojets, using hydrogen were burned continuously, they would require roughly 1.36 kg/hr or 32.6 kg/day or 11,931 kg/year of hydrogen. The six would also need roughly 9,400 watts of power. Much of this propellant may be supplied by boil-off from pumping and storage.

Table 8.5.2-1, Average STN Drag
(Frontal Area = 2,496 m², C_D = 2.25, 1976 Standard Atmosphere)

Orbit Alt. (km)	Solar Activity (sunspot) Lowest Minimum (N)	Average Minimum (N)	Average Maximum (N)	Highest Maximum (N)
277.8	0.854	2.626	5.936	15.488
279.7	0.784	2.444	5.580	14.715
281.5	0.719	2.275	5.246	13.980
283.4	0.659	2.117	4.932	13.282
285.2	0.605	1.970	4.637	12.619
287.1	0.555	1.834	4.360	11.989
288.9	0.509	1.707	4.099	11.390
290.8	0.467	1.588	3.853	10.822
292.6	0.428	1.479	3.623	10.281
294.5	0.393	1.376	3.406	9.768
296.3	0.360	1.281	3.202	9.280
298.2	0.331	1.192	3.011	8.817
300.0	0.303	1.109	2.830	8.377
301.9	0.279	1.033	2.661	7.958
303.7	0.255	0.961	2.502	7.561
305.6	0.234	0.894	2.352	7.183
307.4	0.215	0.832	2.211	6.825
309.3	0.197	0.775	2.079	6.484
311.1	0.181	0.721	1.955	6.160
313.0	0.166	0.671	1.838	5.852
314.8	0.152	0.625	1.728	5.560
316.7	0.140	0.582	1.624	5.282
318.5	0.128	0.541	1.527	5.019
320.4	0.117	0.504	1.435	4.768
322.2	0.108	0.469	1.350	4.530
324.1	0.099	0.436	1.269	4.304
326.0	0.091	0.406	1.193	4.089
327.8	0.083	0.378	1.122	3.885
329.7	0.078	0.361	1.083	3.789
331.5	0.074	0.348	1.054	3.729
333.4	0.071	0.335	1.027	3.670
335.2	0.067	0.323	1.000	3.612
337.1	0.064	0.312	0.974	3.555
338.9	0.061	0.301	0.949	3.499
340.8	0.058	0.290	0.924	3.444
342.6	0.055	0.280	0.900	3.390
344.5	0.052	0.270	0.877	3.336
346.3	0.049	0.260	0.853	3.283
348.2	0.047	0.251	0.831	3.231
350.0	0.044	0.242	0.810	3.180
351.9	0.042	0.233	0.788	3.130
353.7	0.040	0.225	0.768	3.081
355.6	0.038	0.217	0.748	3.032
357.4	0.036	0.209	0.728	2.984
359.3	0.035	0.202	0.709	2.937
361.1	0.033	0.194	0.691	2.891
363.0	0.031	0.187	0.673	2.845
364.8	0.030	0.181	0.655	2.800
366.7	0.028	0.174	0.639	2.756
368.5	0.027	0.168	0.622	2.712
370.4	0.025	0.162	0.606	2.670
372.3	0.024	0.156	0.590	2.627
374.1	0.023	0.150	0.574	2.586
376.0	0.022	0.145	0.559	2.545
377.8	0.021	0.140	0.545	2.505
379.7	0.020	0.135	0.531	2.465
381.5	0.019	0.130	0.517	2.426
383.4	0.018	0.125	0.503	2.388
385.2	0.017	0.121	0.490	2.351
387.1	0.016	0.117	0.477	2.313
388.9	0.015	0.113	0.465	2.277
390.8	0.015	0.108	0.453	2.241
392.6	0.014	0.105	0.441	2.206

Table 8.5.2-1, Average STN Drag, Continued

Orbit Alt. (km)	Solar Activity (sunspot) Lowest Minimum (N)	Average Minimum (N)	Average Maximum (N)	Highest Maximum (N)
394.5	0.013	0.101	0.430	2.171
396.3	0.012	0.097	0.418	2.137
398.2	0.012	0.093	0.408	2.103
400.0	0.011	0.090	0.397	2.069
401.9	0.011	0.087	0.387	2.037
403.7	0.010	0.084	0.376	2.004
405.6	0.010	0.081	0.367	1.973
407.4	0.010	0.078	0.357	1.942
409.3	0.009	0.075	0.348	1.911
411.1	0.009	0.073	0.339	1.881
413.0	0.009	0.070	0.330	1.851
414.8		0.068	0.321	1.822
416.7		0.065	0.313	1.793
418.6		0.063	0.305	1.765
420.4		0.061	0.297	1.737
422.3		0.058	0.289	1.710
424.1		0.056	0.282	1.683
426.0		0.054	0.274	1.656
427.8		0.052	0.267	1.630
429.7		0.050	0.260	1.604
431.5		0.048	0.253	1.579
433.4		0.047	0.247	1.554
435.2		0.045	0.240	1.530
437.1		0.044	0.234	1.505
438.9		0.042	0.228	1.482
440.8		0.040	0.222	1.458
442.6		0.039	0.216	1.435
444.5		0.038	0.210	1.413
446.3		0.036	0.205	1.390
448.2		0.035	0.200	1.368
450.0		0.034	0.194	1.347
451.9		0.032	0.190	1.326
453.7		0.031	0.185	1.305
455.6		0.030	0.180	1.284
457.4		0.029	0.175	1.264
459.3		0.028	0.170	1.244
461.1		0.027	0.166	1.224
463.0		0.026	0.162	1.205
464.9		0.025	0.158	1.186
466.7		0.024	0.154	1.167
468.6		0.024	0.150	1.149
470.4		0.023	0.145	1.131
472.3		0.022	0.142	1.113
474.1		0.021	0.138	1.095
476.0		0.020	0.134	1.078
477.8		0.020	0.131	1.061
479.7		0.019	0.128	1.044
481.5		0.018	0.124	1.027
483.4		0.017	0.121	1.011
485.2		0.017	0.118	0.995
487.1		0.016	0.115	0.980
488.9		0.016	0.112	0.964
490.8		0.015	0.109	0.949
492.6		0.015	0.106	0.934
494.5		0.014	0.103	0.919
496.3		0.013	0.101	0.905
498.2		0.013	0.098	0.890
500.0		0.012	0.095	0.877

Figure 8.5.2-1, Average Drag Force

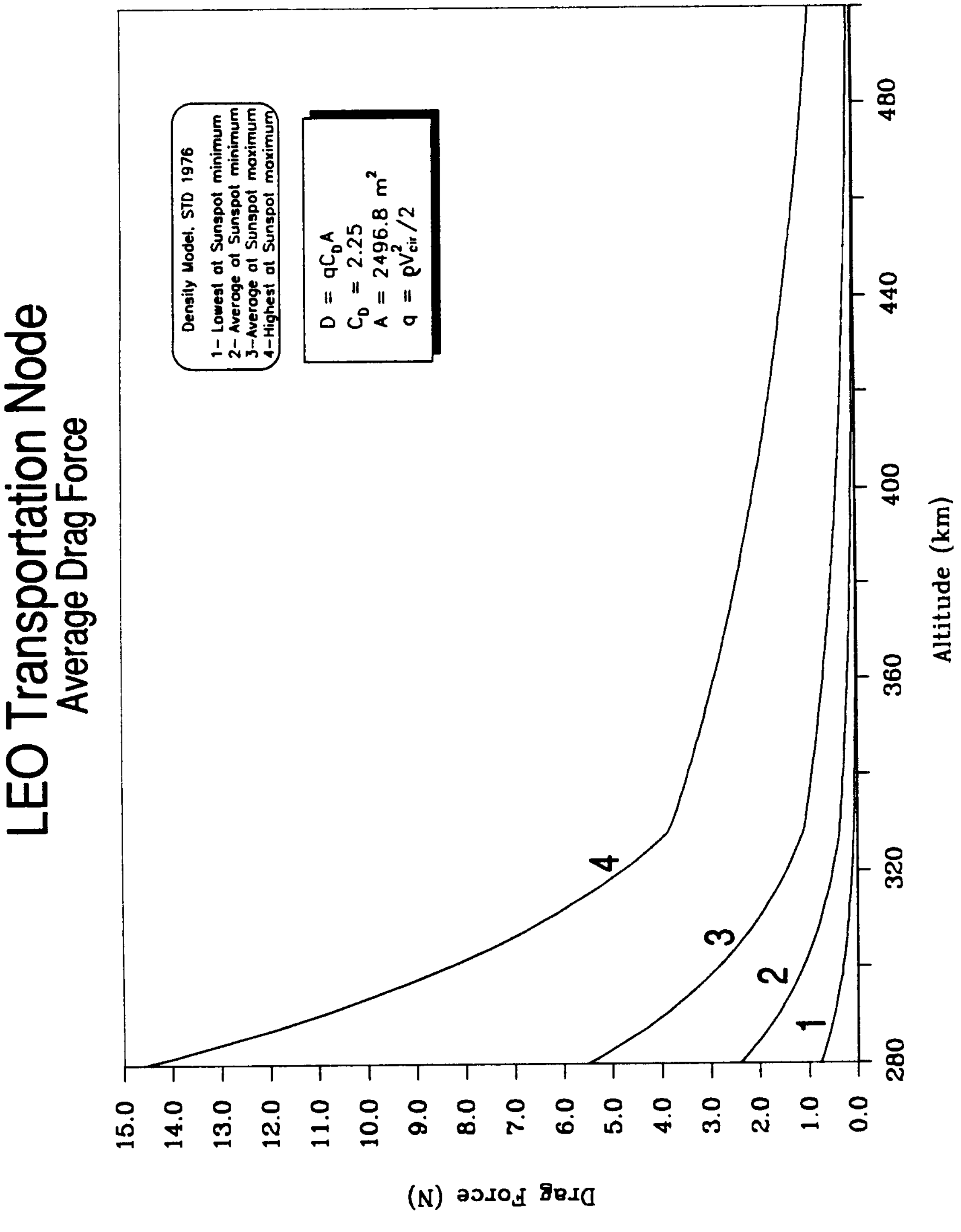


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Table 8.5.2-2, Orbital Decay Time

Dry Weight = 347 m tons
Wet Weight = 528 m tons
Gross Weight = 954 m tons
Frontal Area = 2,497 m^2, C_D = 2.25
1976 Standard Atmosphere

Orbital Altitude		*Dry Weight* Solar Activity (sunspot)				*Wet Weight* Solar Activity (sunspot)				*Gross Weight* Solar Activity (sunspot)			
		Lowest Minimum	Average Minimum	Average Maximum	Highest Maximum	Lowest Minimum	Average Minimum	Average Maximum	Highest Maximum	Lowest Minimum	Average Minimum	Average Maximum	Highest Maximum
nm	km	days	days	days	days	days	days	days	days	days	days	days	days
150	277.8	5.06	1.65	0.73	0.43	7.70	2.51	1.11	0.43	13.92	4.53	2.00	0.77
151	279.6	10.57	3.41	1.50	0.87	16.10	5.20	2.29	0.87	29.08	9.39	4.13	1.58
152	281.5	16.57	5.31	2.33	1.34	25.24	8.09	3.54	1.34	45.61	14.62	6.40	2.43
153	283.3	23.12	7.35	3.20	1.84	35.21	11.20	4.87	1.84	63.61	20.23	8.81	3.32
154	285.2	30.25	9.54	4.13	2.36	46.07	14.53	6.29	2.36	83.23	26.25	11.37	4.26
155	287.0	38.01	11.89	5.12	2.91	57.91	18.11	7.80	2.91	104.61	32.72	14.09	5.25
156	288.9	46.48	14.42	6.17	3.48	70.80	21.96	9.40	3.48	127.91	39.67	16.98	6.29
157	290.7	55.71	17.13	7.29	4.09	84.85	26.09	11.10	4.09	153.29	47.14	20.06	7.39
158	292.6	65.76	20.04	8.48	4.73	100.16	30.53	12.91	4.73	180.95	55.15	23.33	8.54
159	294.4	76.71	23.17	9.74	5.40	116.85	35.30	14.84	5.40	211.10	63.76	26.81	9.75
160	296.3	88.65	26.53	11.09	6.11	135.03	40.41	16.89	6.11	243.94	73.01	30.51	11.03
161	298.1	101.65	30.14	12.52	6.85	154.84	45.91	19.06	6.85	279.73	82.94	34.44	12.37
162	300.0	115.82	34.02	14.03	7.63	176.43	51.81	21.38	7.63	318.73	93.61	38.62	13.78
163	301.9	131.27	38.18	15.65	8.45	199.95	58.15	23.84	8.45	361.22	105.06	43.06	15.27
164	303.7	148.09	42.65	17.37	9.32	225.58	64.96	26.45	9.32	407.53	117.36	47.79	16.83
165	305.6	166.43	47.45	19.19	10.23	253.51	72.28	29.23	10.23	457.99	130.57	52.82	18.48
166	307.4	186.41	52.61	21.13	11.19	283.94	80.13	32.19	11.19	512.97	144.76	58.16	20.21
167	309.3	208.18	58.14	23.20	12.20	317.11	88.56	35.34	12.20	572.88	160.00	63.84	22.03
168	311.1	231.90	64.09	25.39	13.26	353.24	97.62	38.68	13.26	638.16	176.37	69.88	23.95
169	313.0	257.75	70.48	27.73	14.37	392.62	107.35	42.23	14.37	709.30	193.94	76.30	25.96
170	314.8	285.92	77.34	30.21	15.55	435.53	117.80	46.01	15.55	786.82	212.82	83.12	28.08
171	316.7	316.62	84.70	32.84	16.78	482.28	129.03	50.03	16.78	871.28	233.09	90.38	30.32
172	318.5	350.06	92.62	35.65	18.08	533.23	141.08	54.30	18.08	963.32	254.87	98.10	32.66
173	320.4	386.51	101.12	38.63	19.45	588.74	154.02	58.84	19.45	1063.61	278.25	106.30	35.13
174	322.2	426.22	110.24	41.80	20.89	649.23	167.92	63.67	20.89	1172.90	303.37	115.03	37.73
175	324.1	469.50	120.04	45.17	22.40	715.15	182.85	68.80	22.40	1291.98	330.34	124.30	40.47
176	325.9	516.65	130.57	48.75	23.99	786.98	198.89	74.26	23.99	1421.75	359.31	134.16	43.34
177	327.8	568.03	141.88	52.56	25.67	865.25	216.11	80.07	25.67	1563.14	390.42	144.64	46.37
178	329.6	622.58	153.70	56.51	27.38	948.33	234.13	86.08	27.38	1713.23	422.97	155.50	49.47
179	331.5	679.95	165.97	60.56	29.13	1035.72	252.81	92.24	29.13	1871.11	456.72	166.65	52.62
180	333.3	740.29	178.68	64.72	30.90	1127.64	272.17	98.58	30.90	2037.18	491.70	178.09	55.82
181	335.2	803.77	191.86	68.98	32.70	1224.33	292.25	105.07	32.70		527.97	189.83	59.07
182	337.0	870.54	205.52	73.36	34.53	1326.03	313.06	111.74	34.53		565.57	201.87	62.37
183	338.9	940.77	219.69	77.85	36.38	1433.01	334.64	118.59	36.38		604.56	214.24	65.73
184	340.7	1014.65	234.38	82.46	38.27	1545.54	357.01	125.61	38.27		644.97	226.93	69.13
185	342.6	1092.35	249.60	87.20	40.18	1663.91	380.20	132.82	40.18		686.87	239.96	72.59
186	344.4	1174.09	265.39	92.06	42.12	1788.41	404.25	140.22	42.12		730.31	253.33	76.10
187	346.3	1260.07	281.75	97.04	44.10	1919.37	429.17	147.82	44.10		775.34	267.05	79.67
188	348.1	1350.50	298.72	102.16	46.10	2057.13	455.02	155.61	46.10		822.03	281.13	83.29
189	350.0	1445.63	316.31	107.41	48.14		481.81	163.61	48.14		870.43	295.58	86.97
190	351.9	1545.69	334.54	112.80	50.21		509.59	171.82	50.21		920.61	310.41	90.71
191	353.7	1650.94	353.45	118.33	52.31		538.38	180.25	52.31		972.63	325.64	94.50
192	355.6	1761.65	373.05	124.01	54.44		568.23	188.90	54.44		1026.57	341.26	98.36
193	357.4	1878.10	393.36	129.84	56.61		599.19	197.78	56.61		1082.48	357.30	102.27
194	359.3	2000.59	414.43	135.82	58.81		631.27	206.89	58.81		1140.45	373.76	106.25
195	361.1		436.27	141.96	61.05		664.54	216.24	61.05		1200.55	390.65	110.28

Table 8.5.2-2, Orbital Decay Time, Continued
Dry Weight = 347 m tons
Wet Weight = 528 m tons
Gross Weight = 954 m tons
Frontal Area = 2,497 m^2, C_D = 2.25
1976 Standard Atmosphere

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Orbital Altitude		*Dry Weight* Solar Activity (sunspot)				*Wet Weight* Solar Activity (sunspot)				*Gross Weight* Solar Activity (sunspot)			
		Lowest Minimum	Average Minimum	Average Maximum	Highest Maximum	Lowest Minimum	Average Minimum	Average Maximum	Highest Maximum	Lowest Minimum	Average Minimum	Average Maximum	Highest Maximum
nm	km	days	days	days	days	days	days	days	days	days	days	days	days
196	363.0		458.91	148.26	63.32		699.03	225.83	63.32		1262.85	407.98	114.38
197	364.8		482.38	154.72	65.62		734.78	235.68	65.62		1327.44	425.78	118.55
198	366.7		506.72	161.36	67.96		771.85	245.79	67.96		1394.41	444.04	122.78
199	368.5		531.94	168.17	70.34		810.28	256.16	70.34		1463.83	462.78	127.07
200	370.4		558.10	175.16	72.75		850.12	266.81	72.75		1535.81	482.02	131.44
201	372.2		585.22	182.34	75.21		891.42	277.74	75.21		1610.43	501.76	135.87
202	374.1		613.33	189.70	77.70		934.24	288.96	77.70		1687.78	522.02	140.37
203	375.9		642.47	197.26	80.23		978.63	300.47	80.23		1767.98	542.82	144.94
204	377.8		672.69	205.01	82.80		1024.66	312.28	82.80		1851.13	564.16	149.58
205	379.6		704.01	212.97	85.41		1072.37	324.41	85.41		1937.33	586.07	154.30
206	381.5		736.48	221.14	88.06		1121.84	336.85	88.06		2026.70	608.55	159.09
207	383.3		770.15	229.53	90.75		1173.12	349.62	90.75			631.63	163.95
208	385.2		805.06	238.13	93.48		1226.29	362.73	93.48			655.31	168.89
209	387.0		841.24	246.97	96.26		1281.41	376.19	96.26			679.61	173.91
210	388.9		878.76	256.03	99.08		1338.55	389.99	99.08			704.56	179.00
211	390.7		917.65	265.33	101.95		1397.80	404.17	101.95			730.16	184.17
212	392.6		957.97	274.88	104.85		1459.22	418.71	104.85			756.44	189.43
213	394.4		999.78	284.68	107.81		1522.89	433.64	107.81			783.41	194.77
214	396.3		1043.11	294.74	110.81		1588.90	448.96	110.81			811.09	200.19
215	398.1		1088.04	305.07	113.86		1657.34	464.69	113.86			839.49	205.69
216	400.0		1134.62	315.66	116.95		1728.30	480.82	116.95			868.65	211.28
217	401.9		1182.91	326.53	120.10		1801.85	497.39	120.10			898.57	216.96
218	403.7		1232.98	337.70	123.29		1878.11	514.39	123.29			929.29	222.73
219	405.6		1284.88	349.15	126.53		1957.18	531.84	126.53			960.81	228.59
220	407.4		1338.69	360.91	129.82		2039.14	549.74	129.82			993.16	234.54
221	409.3		1394.48	372.97	133.17			568.12	133.17			1026.36	240.58
222	411.1		1452.31	385.35	136.56			586.98	136.56			1060.44	246.72
223	413.0		1512.27	398.06	140.01			606.34	140.01			1095.41	252.95
224	414.8		1574.44	411.11	143.52			626.21	143.52			1131.31	259.28
225	416.7		1638.88	424.49	147.08			646.60	147.08			1168.15	265.71
226	418.5		1705.69	438.23	150.69			667.53	150.69			1205.96	272.24
227	420.4		1774.96	452.34	154.36			689.01	154.36			1244.76	278.87
228	422.2		1846.77	466.81	158.09			711.06	158.09			1284.59	285.60
229	424.1		1921.22	481.66	161.88			733.69	161.88			1325.47	292.44
230	425.9		1998.40	496.91	165.72			756.91	165.72			1367.42	299.39
231	427.8		2078.42	512.56	169.63			780.74	169.63			1410.48	306.44
232	429.6			528.62	173.59			805.20	173.59			1454.67	313.61
233	431.5			545.10	177.62			830.31	177.62			1500.03	320.89
234	433.3			562.01	181.71			856.08	181.71			1546.58	328.28
235	435.2			579.38	185.87			882.52	185.87			1594.35	335.79
236	437.0			597.19	190.09			909.66	190.09			1643.39	343.41
237	438.9			615.48	194.37			937.52	194.37			1693.71	351.15
238	440.7			634.25	198.73			966.11	198.73			1745.36	359.02
239	442.6			653.51	203.15			995.45	203.15			1798.38	367.00
240	444.4			673.29	207.64			1025.57	207.64			1852.78	375.12
241	446.3			693.58	212.20			1056.48	212.20			1908.62	383.36
242	448.1			714.40	216.83			1088.20	216.83			1965.93	391.72
243	450.0			735.78	221.53			1120.76	221.53			2024.75	400.22
244	451.9			757.71	226.31			1154.18	226.31				408.85
245	453.7			780.23	231.16			1188.47	231.16				417.62
246	455.6			803.34	236.09			1223.67	236.09				426.52

Table 8.5.2-2, Orbital Decay Time, Continued

Dry Weight = 347 m tons
Wet Weight = 528 m tons
Gross Weight = 954 m tons
Frontal Area = 2,497 m², C_D = 2.25
1976 Standard Atmosphere

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Orbital Altitude		Dry Weight — Solar Activity (sunspot) Lowest Minimum	Dry Weight Average Minimum	Dry Weight Average Maximum	Dry Weight Highest Maximum	Wet Weight — Solar Activity (sunspot) Lowest Minimum	Wet Weight Average Minimum	Wet Weight Average Maximum	Wet Weight Highest Maximum	Gross Weight — Solar Activity (sunspot) Lowest Minimum	Gross Weight Average Minimum	Gross Weight Average Maximum	Gross Weight Highest Maximum
nm	km	days	days	days	days	days	days	days	days	days	days	days	days
247	457.4			827.05	241.10			1259.80	241.10				435.56
248	459.3			851.40	246.18			1296.87	246.18				444.75
249	461.1			876.38	251.34			1334.93	251.34				454.07
250	463.0			902.02	256.59			1373.98	256.59				463.54
251	464.8			928.33	261.91			1414.07	261.91				473.16
252	466.7			955.34	267.32			1455.21	267.32				482.94
253	468.5			983.06	272.81			1497.43	272.81				492.86
254	470.4			1011.51	278.39			1540.77	278.39				502.94
255	472.2			1040.71	284.06			1585.24	284.06				513.17
256	474.1			1070.68	289.81			1630.89	289.81				523.57
257	475.9			1101.43	295.66			1677.74	295.66				534.13
258	477.8			1133.00	301.59			1725.82	301.59				544.85
259	479.6			1165.40	307.62			1775.17	307.62				555.74
260	481.5			1198.65	313.74			1825.82	313.74				566.80
261	483.3			1232.78	319.96			1877.80	319.96				578.04
262	485.2			1267.80	326.28			1931.16	326.28				589.45
263	487.0			1303.75	332.69			1985.91	332.69				601.03
264	488.9			1340.64	339.20			2042.11	339.20				612.80
265	490.7			1378.51	345.82				345.82				624.75
266	492.6			1417.37	352.54				352.54				636.89
267	494.4			1457.26	359.36				359.36				649.22
268	496.3			1498.20	366.30				366.30				661.74
269	498.1			1540.21	373.33				373.33				674.46
270	500.0			1583.34	380.48				380.48				687.38

Figure 8.5.2-2, Orbital Decay Time, Dry Weight (347 m tons)

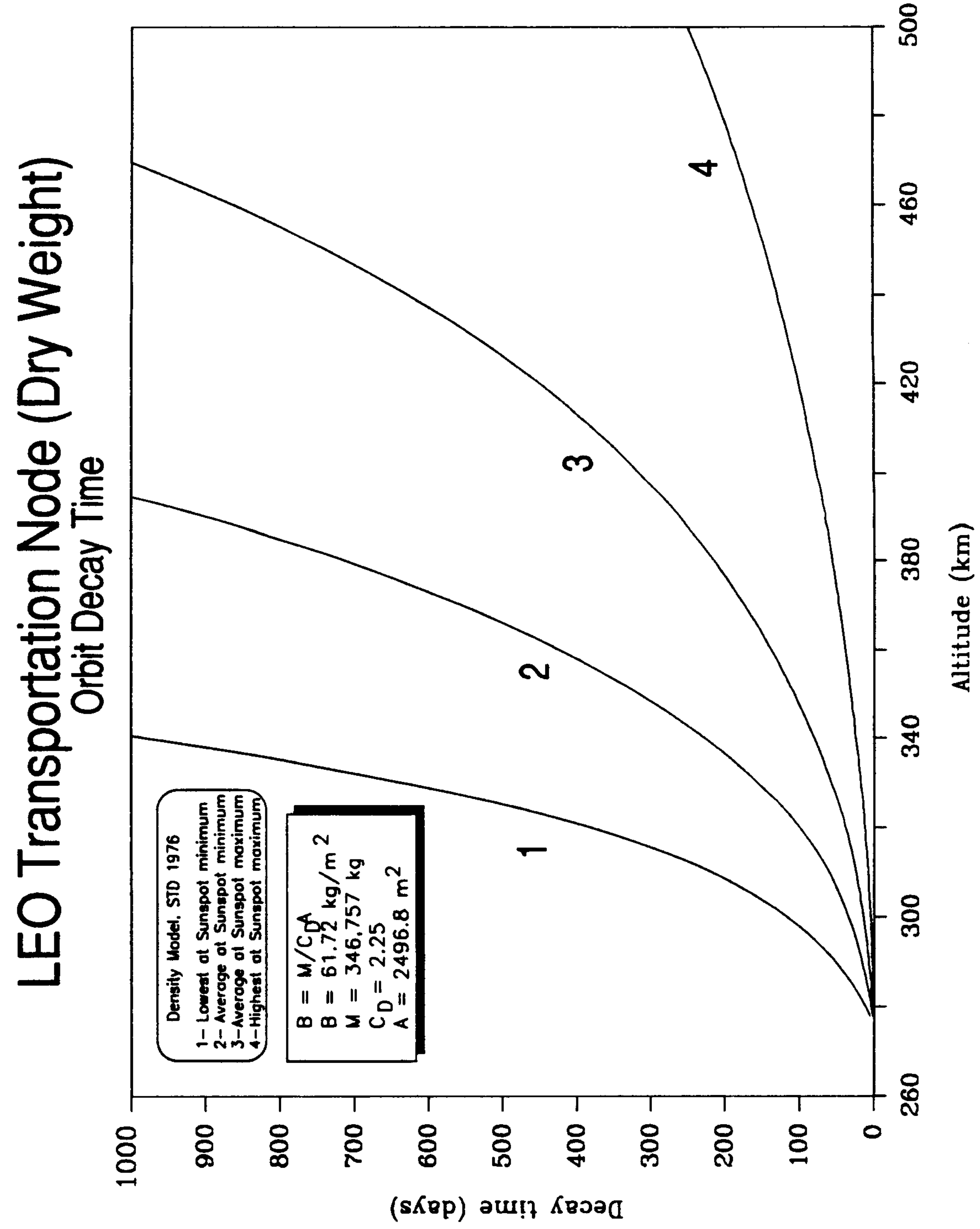

Figure 8.5.2-3, Orbital Decay Time, Wet Weight (528 m tons)

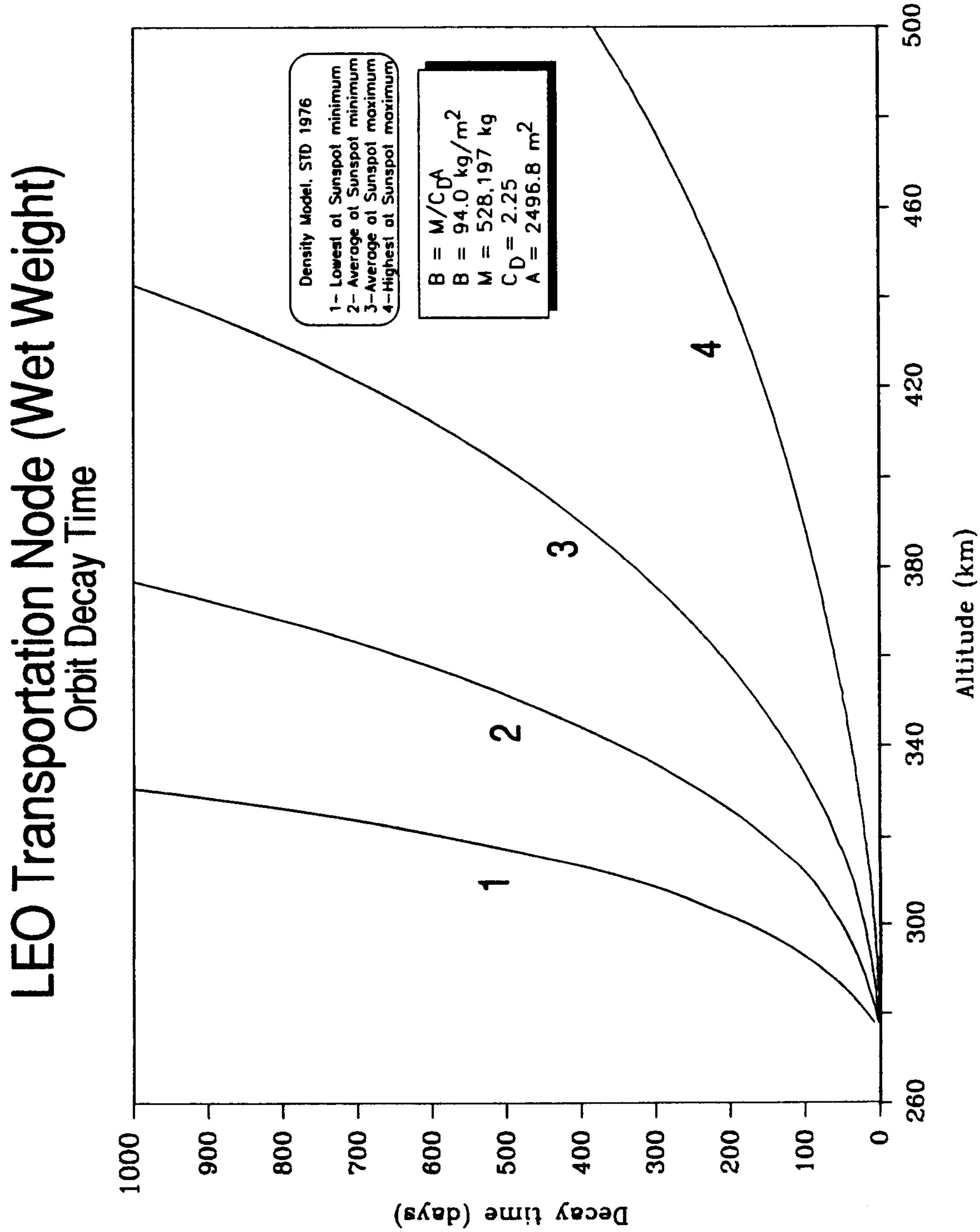

Figure 8.5.2-4, Orbital Decay Time, Gross Weight (954 m tons)

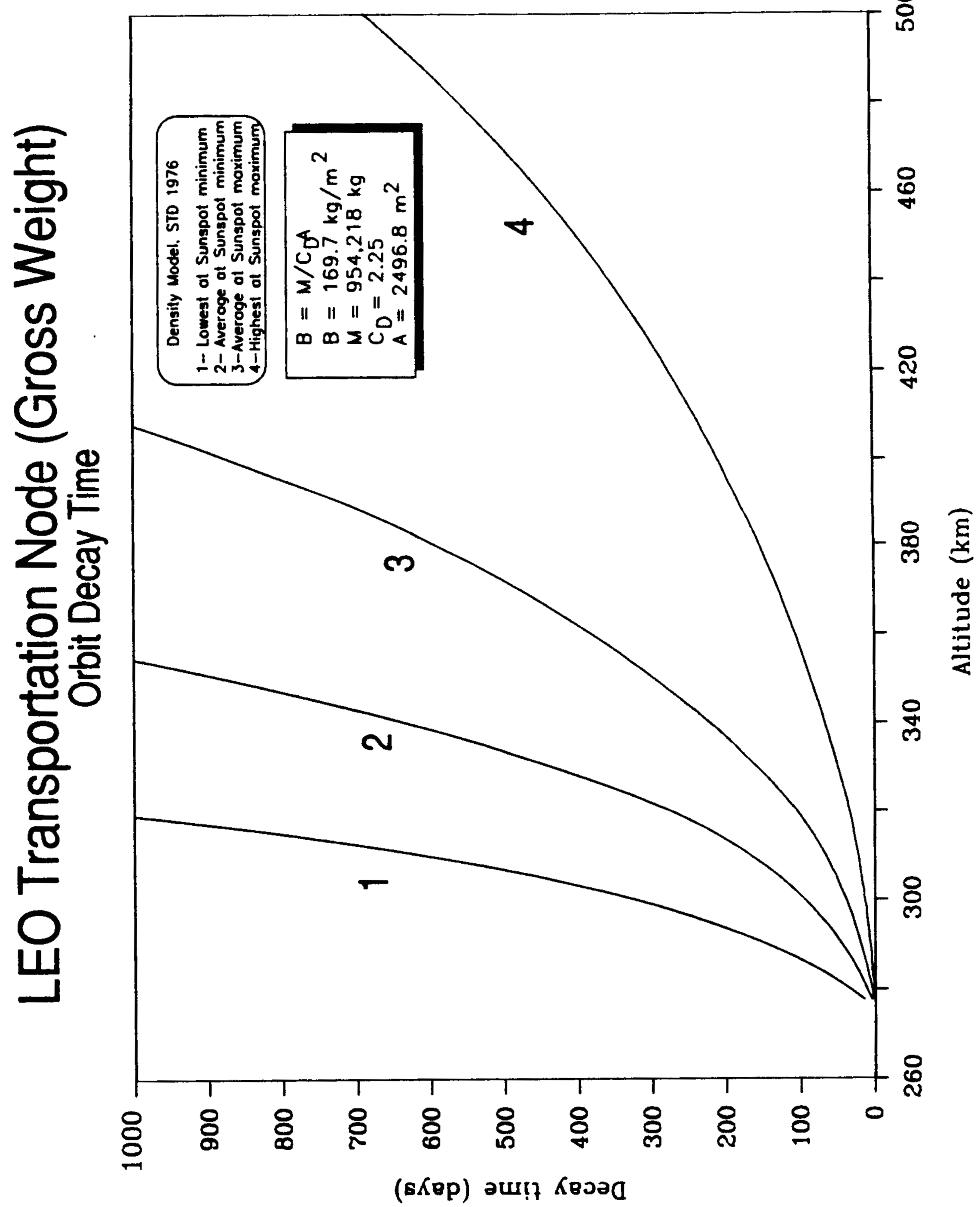

Table 8.5.3-1, Resistojet Performance Parameters for Variable Power (Heckert, 1987)

P_{inlet} = 50 psia

$T_{gas,out}$ = 1400°C

Fluid	Isp(s)	$\dot{m}$ (kg/h)	Req. Power (w)	Thrust (N)	(lb)
H_2	500	.227	1564	0.3115	0.07
O_2	150	.907	436	0.3693	0.08
Steam	200	.6577	726	0.3604	0.08
Cabin air	157	.8618	449	0.3693	0.08
Mixed gases	235	.5897	587	0.3782	0.08

8.5.4 Reboost Scenarios

Since the STN configuration studied is a final assembly configuration, the conditions for reboost scenarios were baselined upon the orbit decay data of Table 8.5.2-2. Two approaches were taken to estimate STN propellant and orbit boost time requirements. The first was to estimate these requirements on a propellant mass per km of altitude and a boost time per km of altitude basis, for each of the STN loading conditions. With these results a quick method for estimating propellant mass and RCS burn time is given for any delta altitude between 280 km and 500 km. The second approach was to use these results and apply them to a 90 day decay time limit for an altitude of 500 km. Table 8.5.4-1 shows the propellant mass and burn time requirements for the dry, wet and gross weights per km of altitude. Table 8.5.4-2 shows the propellant mass and burn time requirements for each STN loading and for maximum solar activity densities. The requirements for minimum solar activity and those not shown on this table were not estimated since the orbit life time was greater than 2000 days. All orbit boost and orbit correction maneuvers will be done using the RCM's. Independent of the number of sets of RCM's used, the following equations apply,

$$M_p = M_o\ (1 - e^{-a}\ )$$

$$a = \Delta v / g_o\ I_{sp}$$

$$\Delta t = I_{sp}\ g_o\ M_p / T$$

where:

M_p = propellant mass, kg
M_o = STN total mass, including propellant mass, kg
Δv = delta velocity required to raise orbit, m/s
 = .575 m/s/km (280 km < h < 500 km)
g_o = gravity constant at sea level = 9.809 m/s^2
I_{sp} = specific impulse of motors, s
Δt = burn time to raise orbit, s
T = total thrust, N

If any of these propellant mass and burn time requirements are unacceptable for the STN, changes will have to be made to the number or type of thrusters. If a lower burn time is required then the number of RCM's will have to increase. If the baseline configuration is to consist of single set RCM's as shown in Figure 8.3.7-1, then only 1,332 N of thrust are available in the +X direction (333x4); however, this thrust would double if double RCM sets were used and the burn time would be cut by half. To decrease the propellant mass requirements, different RCS thrusters with a higher specific impulse would have to be used. A possible candidate could be the Bell H_2/O_2 thruster that was developed as an alternate for Space Station. This thruster has a thrust of 222 N (50 lbs) and an I_{sp} of 410. One set of RCM's using these thrusters would have the same thrust as two sets of the current Space Station baseline RCM's, but they would burn 7% less propellant.

Table 8.5.4-1, STN Propellant Mass and Burn Time Requirements

STN Load Condition	Propellant mass per km (kg)	RCM Burn time per km, (s)
Dry (347 m tons)	53.5	150
Wet (582 m tons)	81.47	228
Gross (954 m tons)	147.24	412

Table 8.5.4-2, Propellant Required for Reboost to h=500 km After 90 Day Decay

STN Load Condition	Propellant mass, kg		Burn time, s	
	Average max Solar Flux	Highest max Solar Flux	Average max Solar Flux	Highest max Solar Flux
Dry (347 mt)	267.5	1,337.5	750	3,750
Wet* (528 mt)	---	18,330.75	----	51,300
Gross (954 mt)	---	2,016.4	----	5,798

* The intermediate weight requires the greatest amount of reboost propellant. The highest weight has a significantly greater ballistic number. It does not decay as much in 90 days.

8.6 Thermal Control System

The STN Thermal Control System (TCS) consists of both an active system and a passive system. The active system consists of a heat collection, transport, and rejection system, to remove equipment, metabolic, and environmental loads from various points on the STN. The passive system utilizes insulation and surface coatings to protect external equipment from the space environment.

8.6.1 Active Thermal Control System

The active thermal control system (ATCS) has a two phase exterior system for transporting heat from equipment and modules and has single phase internal systems for transporting heat within the pressurized modules.

The two-phase external system uses ammonia circulating in two loops to provide 35° F and 70° F cooling. Figure 8.6-1 is a functional schematic of one loop of the ATCS. Heat is collected in evaporators located at either an external equipment source or at a module heat exchanger. The evaporators are supplied with liquid ammonia at the appropriate pressure for the ATCS loop temperature set point. In the evaporator, enough of the ammonia is boiled to accommodate the load. The ammonia then travels to a condenser coupled with the prominent radiators mounted to the transverse boom.

Several ACTS configurations are now being evaluated for use on the Freedom Space Station. The differences between them relates basically to the nature of the evaporator outlet flow. In one system, the flow is two-phase and the pumping system must handle this type of fluid. The pump acts as a separator and a regenerator as well as a pump. Liquid in the evaporator return line is centrifugally removed from the flow and added to the fluid returning to the pump from the condenser. Vapor-phase ammonia passes on to the condenser after separation in the pump. A valve on the vapor outlet of the pump regulates the pressure in the pump and thereby the temperature of the loop. In another configuration, the liquid and vapor phases are separated by a special evaporator design and the pump only has to handle liquid return from the condenser. This system uses a series of valves to control the pressures of the evaporator inlet and outlets. Table 8.6-1 shows a mass and power summary for the second type of system as it is currently the baseline for the Space Station. The Phase 1 system has a heat rejection capacity of 75 kW, using eight radiator panels, two pair on each end of the transverse boom.

An internal ACTS using single-phase liquid water is used to transport cooling loads from within the pressurized volumes to the exterior ammonia system. Since ammonia is toxic, and the potential for leaks does exist, the interior loop to exterior loop heat exchangers are located outside the pressure shell. Chilled water may be supplied at 40°F or at about 75°F since two external loops will be available. To be sure, most of the loads internal to pressurized volumes will use the lower temperature water, but any high temperature equipment that may be used in a workshop could use higher temperature water. In some locations, body mounted radiators (BMR's) have been utilized to provide an external thermal control system during station buildup while the full STN system has not been completed. In addition, the BMR's provide some measure of redundancy should the external system fail. Note that the external system is required to fail operational/fail safe/recoverable, so a complete failure of the external system will be extremely rare.

Figure 8.6-1, Active Thermal Control System

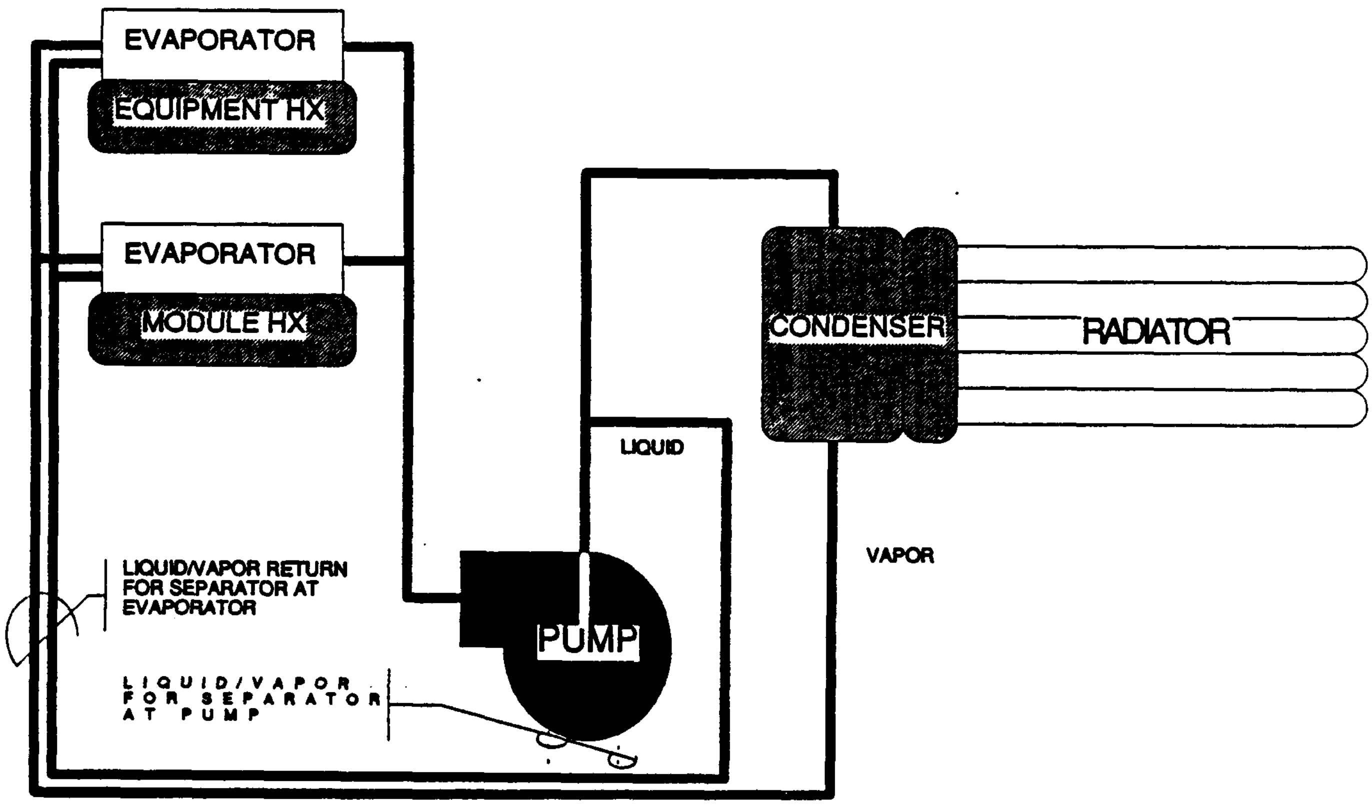

Table 8.6-1, Phase 1 Space Station Thermal Control System Mass and Power Summary (McDonnell Douglas WP2)

Physical Characteristics

Component	**Qty**	**Mass** kg	(lbm)
Payload Heat Exchanger (HAD)	4	127	(280)
Module Heat Exchanger (HAD)	14	770	(1,694)
Node Heat Exchanger (HAD)	4	93	(204)
Service Fac Heat Exchanger	0	0	(0)
Cold Plates (HAD)	0	0	(0)
Condenser Assy	6	502	(1,104)
Subcooler Assy	4	182	(400)
Radiator Panel	62	2,795	(6,150)
Pumps	8	12	(26)
Filters	16	16	(35)
Accumulators	4	80	(176)
Tanks	2	160	(353)
Control Valves	36	53	(117)
Isolation Valves	388	441	(970)
Monitor Sensors	1,866	118	(260)
Total		5,350	(11,769)

Power Requirements (watts)

Pumps	355
Valves	90
Total	445

8.6.2 Passive Thermal Control System

The passive thermal control system is simply a system of coatings and insulations for the STN hangar, exterior equipment, and the exterior portions of the pressurized volumes. In general, the passive control requirement is to isolate equipment from the space environment.

Multilayer Insulation (MLI) made of double-aluminized kapton is used selectively where needed. The number of MLI layers on external equipment may vary depending on the application. MLI is used between the pressure shell and the meteoroid shield of the modules and nodes.

Surface coatings can include black anodize, silver teflon, chromic acid anodize, and white paint. Modules are painted with white paint and other coatings can be used depending on the particular needs of the application. The black anodize has an α/ε ratio 0.8/0.8 and so is basically an absorbing coating. Silver teflon has an α/ε of 0.2/0.8 and as such is a good reflector. The chromic acid anodize has an α/ε of 0.3/0.6 and is a moderate reflector. The paint used for the modules will have properties similar to the teflon, although, the specific values are not known. The truss structure itself will be coated with an aluminum foil to protect it from ultraviolet radiation and atomic oxygen degradation as well as for thermal protection.

The largest structure requiring passive thermal protection will be the STN vehicle hangar. The wall construction may be a single aluminum panel, a double-wall construction with an aluminum bumper and graphite back-wall, or it may utilize a ceramic or metal fabric to allow flexibility. Since the optimum material has not been selected, the thermal characteristics of the hangar walls are not known. In addition, for meteoroid and debris protection, a hangar wall on the nadir side is not needed. The need for this wall from a thermal standpoint has not yet been established.

A thermal analysis of the hangar enclosure is needed to resolve the issues at-hand and to establish criteria for the thermal characteristics of the hangar itself. First, the desired thermal environment within the hangar itself must be defined. Next, an analysis of the enclosure itself including vehicles and propellant storage tanks must be completed. This analysis, coupled with the desired thermal radiation environmental characteristics, will allow the definition of the required thermal characteristics of the hangar walls as well as affirm or deny the need for a nadir hangar wall. Another area for significant passive and possibly active thermal control will be the propellant storage tanks themselves. An analysis of the effects of the tanks on the hangar environment and vice versa is needed. If the tanks are maintained outside the hangar enclosure, they will still affect the hangar environment by presenting the hangar exterior with a possibly low temperature sink. This thermal analysis is well beyond the scope of this conceptual design but must be completed before the material selections and design configurations are finalized.

8.7 Environmental Control and Life Support System

The ECLSS is a regenerative system that minimizes the crew consumable expendables required for resupply. All metabolic water and oxygen requirements are generated by the closed-cycle system to avoid large resupply requirements. The only resupplied consumables are food and hydrazine, the latter being dissociated to make up both atmospheric nitrogen leakage and the hydrogen requirement of the reactor that reduces the carbon dioxide generated by the crew. Hydrazine can also be used as a monopropellant in the reaction-control system. Major functions of the ECLSS are cabin air revitalization, radiator heat rejection, ventilation/temperature control, water processing, carbon dioxide reduction, atmosphere makeup supply, hygiene/health, and EVA support.

The Environmental Control and Life Support System provides a shirt-sleeve environment in all of the pressurized volumes of the STN. The system is partially regenerative with the oxygen and water loops closed. Fluid makeup is limited to the water contained in food and a modest amount of nitrogen. The system has a common set of hardware in the habitation modules as well as in the workshops. The nodes and airlocks contain a subset of the core systems. The major or core functions of the ECLSS include the air revitalization (AR), water recovery management (WRM), temperature and humidity control (THC), atmosphere control and supply (ACS), fire detection and suppression (FDS), and waste management (WM).

The THC subsystem utilizes ventilation fans coupled with condensing heat exchangers and ducted air supply and return to accommodate metabolic cooling loads and rack mounted equipment loads. The core THC located in the HAB and Workshop modules also provides intermodule air to the nodes and airlocks as revitalized air. As a result, the ECLSS in the nodes and airlocks does not need to perform CO_2 and contaminant removal functions allocated to the AR subsystem.

The AR system uses a molecular sieve to remove CO_2 from the cabin air. The Bosch process is used to reduce the CO_2 to carbon and water. The water is then electrolyzed with potassium hydroxide electrolyte. The oxygen is reused in the cabin air to revitalize breathing air and to make up leakage and airlock losses. Trace contaminants are controlled by a nonregenerable carbon and lithium hydroxide sorbent system. A gas chromatograph is used to monitor the contaminant levels. In the nodes and airlocks, temperature, humidity, and O_2/N_2 partial pressure control as well as CO_2 and contaminants removal functions are handled in the core systems of the HAB and workshop modules via an intermodule supply system. Supplemental temperature control is provided within each node and airlock.

The ACS maintains the STN total pressure and oxygen/nitrogen partial pressures. The oxygen is supplied by the AR subsystem electrolysis process and nitrogen from supercritical storage in a logistics carrier via external lines. Oxygen and nitrogen supply lines at 3,000 psia are provided on the outside of the modules for pressurized element repressurization and for hyperbaric operations. Total pressure is maintained by positive/negative pressure relief valves in each element. If the total pressure rises above a certain setpoint, cabin air is vented to space.

The WRM system recovers and distributes water used in the STN. Potable water used by the crew for drinking and food preparation is handled separately from hygiene water used for cleaning. Hygiene from showers, hand washing, and from dish washing operations is in a multifiltration process before recycling, and urine and flush water undergoes thermoelectric evaporation and distillation. Potable water is recovered from the cabin humidity control systems and from CO_2 reduction processes. Water quality is monitored by culture plate counts and controlled in the drinking water by the addition of two PPM of iodine at dispensing points. Processed waters are monitored for total organic carbon, conductivity, pH, and residual biocides.

Fire sensing, alarm activation and extinguishing are provided by the common fire detection system (FDS) in the STN. Fires in open areas such as aisles and workstations are detected by ultraviolet flame detectors mounted on the module end-cones, and ionization sensors are located in the ductwork. Portable CO_2 extinguishers are provided for fire suppression in open areas.

The WM system separately collects and stores solid human wastes and urines. Solid wastes are compacted, allowed to biodegrade, and stored for return to Earth.

9.0 Weight Statement

Table 9.0-1 gives an upper level weight statement. Tables 9.0-2 and 9.0-3 show more detailed weight statements with an estimate of all mass properties.

For the purpose of obtaining a detailed mass properties statement for the STN, a reference point was chosen at the geometric center of the transverse boom of the truss structure. Figure 9.0-1 shows the reference point and axes for all CG and inertia computations. The STN velocity vector is in the +X direction. A detailed weight statement was prepared using weights and locations for items found in this report. The rest were estimated from other references to the Freedom Space Station. Since design requirements call for commonality of systems with Space Station the weights and estimated inertias for all other subsystems were considered valid for this design. Tables 9.0-3 shows the weight statement for the STN in Metric units and Table 9.0-2 shows the weight statement in English units. The mass of all components was known; however, the moments of inertia had to be estimated for each. Some of the moments and products of inertia were assumed to be zero, since their relative size compared to the total mass was so small; however, their mass moment was included in the overall moments of inertia. The weight statement was separated into three different cases, one for the dry weight, one for the wet weight, and one for the gross weight. The dry weight includes only STN subsystems and no propellants, the wet weight includes propellant in all four tanks, and the gross weight includes two OTV's and landers fully loaded in the hangar. This was done primarily to estimate the relative performance of the current double gimbal control moment gyros and reaction control modules, but it also establishes a basis for relative sizing of attitude control systems for the detailed phases of the STN design.

Table 9.0-4 shows a Phase I Freedom Space Station summary weight statement for comparison purposes.

Table 9.0-1, Summary Weight Statement

Elements	**lbs**	**kgms**
Hangar	48,257	(21,935)
Propellant Storage (4 Tanks, dry)	122,652	(55,751)
Transfer lines, Interfaces, Other prop. related (wet)	37,975	(17,222)
Remote Manipulator System with Transporter (2)	7,200	(3,273)
Truss	14,963	(6,801)
Power Supply	57,059	(25,936)
Habitation Module 1 (Active)	45,838	(20,835)
Habitation Module 2 (Quiet)	41,260	(18,755)
Workshop Module	127,640	(12,564)
Workshop Module	227,640	(12,564)
Pressurized Logistics Module	16,845	(7,657)
Node 1 (forward starboard)	34,283	(15,551)
Node 2 (forward port)	34,283	(15,551)
Node 3 (starboard)	34,283	(15,551)
Node 4 (port)	34,283	(15,551)
Node 5 (for vert. with rotating fixture)	34,283	(15,551)
Node 6 (starboard)	34,283	(15,551)
Node 7 (port)	34,283	(15,551)
Node 8 (Hangar Control)	34,283	(15,551)
Node 9 (Aft.)	34,283	(15,551)
Node 10 (Aft Vert. with rotating fixture)	34,283	(15,551)
Airlock 1 (Hyperbaric)	8,254	(3,744)
Airlock 2	8,254	(3,744)
CETAs	4,209	(1,913)
Thermal Control (Rad. & Pallets)	8,092	(3,678)
Cupola 1	3,000	(1,364)
Cupola 2	3,000	(1,364)
Cupola 3	3,000	(1,364)
Tunnel	3,058	(1,390)
GN&C Pallets (2)	9,648	(4,385)
RCS Tank Pallets (2)	8,922	(4,055)
Utility Trays	29,878	(13,550)
Antennas	1,348	(613)
Lander OTV Propellant Boom	2,000	(909)
RCMs (6)	1,782	(810)
Total (Dry)	**884,604**	**(401,686)**
Stored Cryogenic Propellant	400,000	(182,000)
Total (Wet)	**1,284,604**	**(583,686)**
Loaded OTV Stacks (2)	877,684	(398,947)
Total (Gross)	**2,162,288**	**(982,633)**

Figure 9.0-1, Axis and Origin Definition

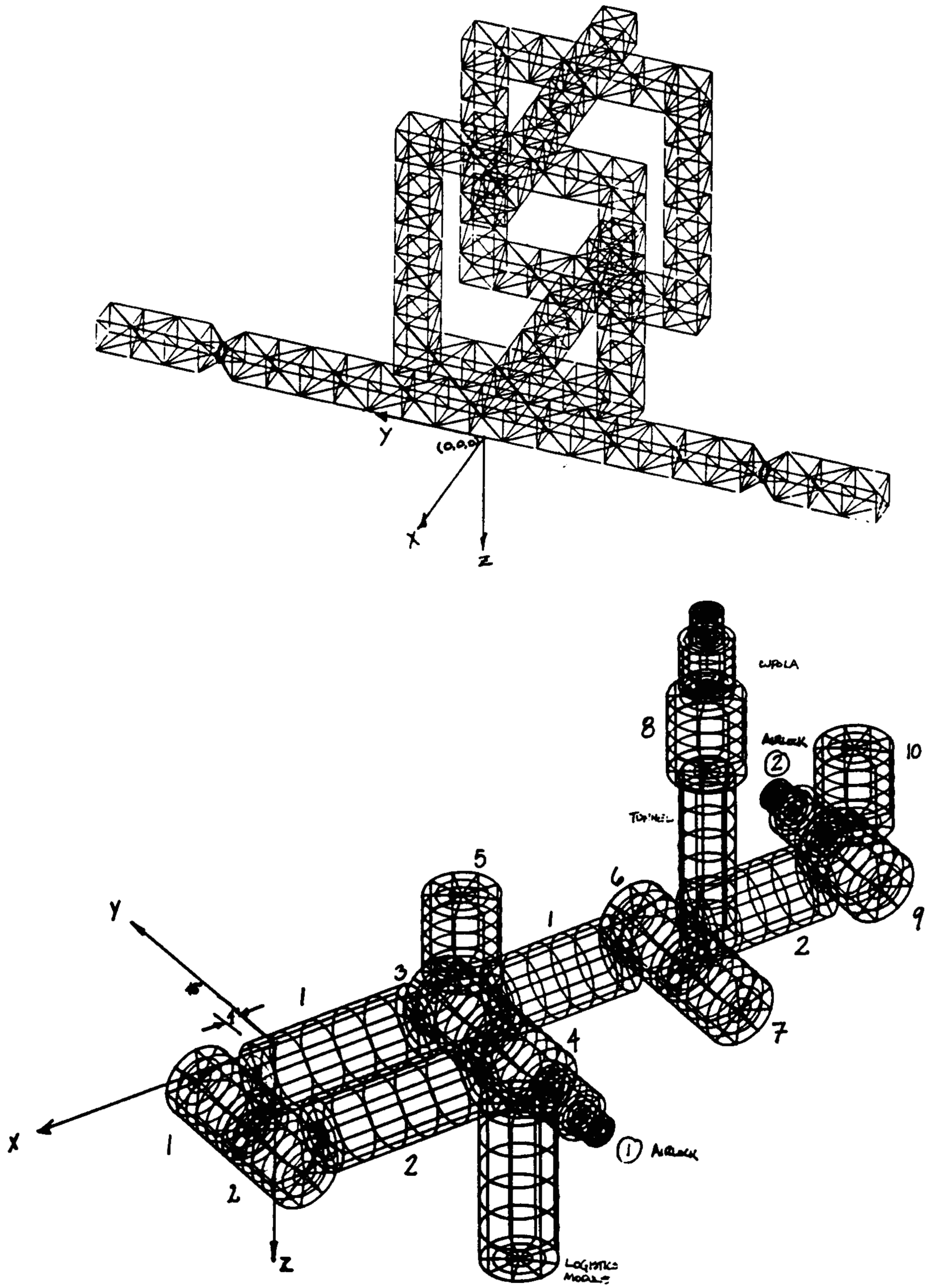

Table 9.0-2, Mass Properties, Metric

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LEO Transportation Node Weight Statement (Metric units)

Item:	Weight (kg)	x (m)	y (m)	z (m)	Ixx (kg-m^2)	Iyy (kg-m^2)	Izz (kg-m^2)	Ixy (kg-m^2)	Ixz (kg-m^2)	Iyz (kg-m^2)
S wing truss	1857	-2.44	40.00	-2.50	353651	42860	82449	-661	-6722	-13709
P wing truss	1857	-2.44	-40.00	-2.50	353651	42860	353593	-661	-6722	-13709
CF "T" truss	197	-3.46	0.00	-2.50	3656	2639	3817	-441	441	-789
F square truss	1643	-12.41	0.00	-19.92	604211	340043	277256	-8172	-7824	1927
A square truss	1643	-37.50	0.00	-19.92	604211	340043	277256	-8172	-7825	1927
UC longeron truss	225	-25.08	0.00	-37.50	2371	7397	6848	-1184	-662	662
LC longeron truss	225	-25.08	0.00	-2.50	2371	7397	6848	-1184	-662	662
UCF truss	126	-4.00	0.00	37.50	1344	1827	1490	-789	-441	441
UCA truss	126	-46.00	0.00	-37.50	1344	1827	1490	-789	-441	441
LCA truss	126	-46.00	0.00	-2.50	1344	1827	1490	-789	-441	441
F wall	4655	0.00	0.00	-15.00	824489	349195	475294	0	0	0
A wall	1960	-50.00	0.00	-15.00	347153	147029	200123	0	0	0
P wall	5600	-25.00	17.50	-15.00	420084	1586983	1166899	0	0	0
S wall	5600	-25.00	-17.50	-15.00	420084	1586983	1166899	0	0	0
T wall	3920	-25.00	0.00	-35.00	400246	816829	1217075	0	0	0
B wall	200	-25.00	-5.00	0.00	20421	41675	62096	0	0	0
USI PV wing	1129	-2.50	50.00	-21.53	93590	84780	8810	0	0	0
USO PV wing	1129	-2.50	70.00	-21.53	93590	84780	8810	0	0	0
LSI PV wing	1129	-2.50	50.00	16.53	93590	84780	8810	0	0	0
LSO PV wing	1129	-2.50	70.00	16.53	93590	84780	8810	0	0	0
UPI PV wing	1129	-2.50	-50.00	-21.53	93590	84780	8810	0	0	0
UPO PV wing	1129	-2.50	-70.00	-21.53	93590	84780	8810	0	0	0
LPI PV wing	1129	-2.50	-50.00	16.53	93590	84780	8810	0	0	0
LPO PV wing	1129	-2.50	-70.00	16.53	93590	84780	8810	0	0	0
SI EPS radiator	1269	-12.24	50.00	-2.50	1415	22177	23592	0	0	0
SO EPS radiator	1269	-12.24	70.00	-2.50	1415	22177	23592	0	0	0
PI EPS radiator	1269	-12.24	-50.00	-2.50	1415	22177	23592	0	0	0
PO EPS radiator	1269	-12.24	-70.00	-2.50	1415	22177	23592	0	0	0
SI Batteries/Data management	2903	-2.50	50.00	-2.50	2018	1261	1261	0	0	0
SO Batteries/Data management	2585	-2.50	70.00	-2.50	2018	1261	1261	0	0	0
PI Batteries/Data management	2903	-2.50	-50.00	-2.50	2018	1261	1261	0	0	0
PO Batteries/Data management	2585	-2.50	-70.00	-2.50	2018	1261	1261	0	0	0
S Thermal radiator	1835	-14.16	30.00	19.30	54353	21608	32745	0	0	0
P Thermal radiator	1835	-14.16	-30.00	19.30	54353	22	33	0	0	0
Node 1 (FS)	15548	3.72	-0.19	2.47	61194	47214	61194	0	0	0
Node 2 (FP)	15548	3.72	-5.19	2.47	61194	47214	61194	0	0	0
Node 3 (S)	15548	-11.08	-0.19	2.47	61194	47214	61194	0	0	0
Node 4 (P)	15548	-11.08	-5.19	2.47	61194	47214	61194	0	0	0
Node 5 (FV)	15548	-11.08	0.00	-2.69	61194	61194	47214	0	0	0
Node 6 (S)	15548	-24.61	-0.19	2.47	61194	47214	61194	0	0	0
Node 7 (P)	15548	-24.61	-5.19	2.47	61194	47214	61194	0	0	0
Node 8 (V)	15548	-24.61	-8.28	-13.36	61194	61194	47214	0	0	0
Node 9 (A)	15548	-35.00	-0.19	2.47	61194	47214	61194	0	0	0
Node 10 (VA)	15548	-35.00	0.00	-2.69	61194	61194	47214	0	0	0
Hab Module 1 (S/quiet)	20788	-4.04	0.00	2.50	62225	224047	224047	0	0	0
Hab Module 2 (P/active)	18712	-4.04	-8.27	2.50	58451	127166	127166	0	0	0
Pressurized Logistics Module	7639	-11.08	-8.33	7.12	23622	23622	22713	0	0	0
Airlock 1 (F)	3743	-11.76	-12.62	2.47	7374	6260	7374	0	0	0

Abbreviations: P - Port, S - Starboard, F - Forward, A - Aft, T - Top, B - Bottom, U - Upper, L - Lower, I - Inner, O - Outer

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Table 9.0-2, Mass Properties, Metric, Continued

Item:	Weight (kg)	x (m)	y (m)	z (m)	Ixx (kg-m^2)	Iyy (kg-m^2)	Izz (kg-m^2)	Ixy (kg-m^2)	Ixz (kg-m^2)	Iyz (kg-m^2)
Airlock 2 (A)	3743	-35.00	1.84	2.47	7374	6260	7374	0	0	0
Workshop Module 1 (F)	12535	-18.19	0.00	2.50	39156	85188	85188	0	0	0
Workshop Module 2 (A)	12535	-31.04	0.00	2.50	39156	85188	85188	0	0	0
Cupola 1	1361	-24.61	-8.31	-16.94	562	562	422	0	0	0
Cupola 2	1361	3.72	-10.21	2.47	562	562	422	0	0	0
Cupola 3	1361	3.72	4.81	2.47	562	562	422	0	0	0
Tunnel	1387	-24.61	-8.31	-5.33	14633	14633	2956	0	0	0
P GN&C pallet	2188	-2.50	-35.00	-7.00	1333	1333	2462	0	0	0
S GN&C pallet	2188	-2.50	35.00	-7.00	1333	1333	2462	0	0	0
F RCS tanks pallet	2023	-2.50	0.00	-2.50	2165	2165	1332	0	0	0
A RCS tanks pallet	2023	-37.50	0.00	-2.50	2165	2165	1332	0	0	0
P TCS pallet	1290	-2.50	-30.00	3.00	860	1398	1398	0	0	0
S TCS Pallet	789	-2.50	30.00	3.00	526	855	855	0	0	0
F Square Truss U Trays	3670	-12.50	0.00	-20.00	1736984	768495	969128	0	0	0
A Square Truss U Trays	3670	-37.50	0.00	-20.00	1736984	768495	969128	0	0	0
F Transverse U Trays	4093	-2.40	0.00	-2.50	4342253	25940	4316995	0	0	0
U Longitudinal U Trays	1129	-25.00	0.00	-37.50	7156	256450	249483	0	0	0
L Longitudinal U Trays	988	-28.21	0.00	-37.50	6261	173598	167502	55411	-268386	-46511
LOX/LH propellant feed lines (wet)	16885	-13.87	-0.27	-18.44	4084740	2656283	1983431	0	0	0
F RMS mobile transporter	726	-12.50	15.00	-34.50	1016	1016	1814	0	0	0
A RMS mobile transporter	726	-37.50	-15.00	-34.00	1016	1016	1814	0	0	0
S S/G S/S Antenna	306	-12.50	18.49	-40.00	0	0	0	0	0	0
P S/G S/S Antenna	306	-12.50	-18.49	-40.00	0	0	0	0	0	0
HLLV tanker resupply interface	337	-12.50	0.00	-40.00	0	0	0	0	0	0
Lander/OTV propellant boom/interface	907	-24.61	-2.50	-10.67	0	0	0	0	0	0
A RMS	907	-37.50	-15.00	-33.50	17559	5	17559	0	0	0
F RMS	907	-12.50	15.00	-33.50	17559	5	17559	0	0	0
Transverse truss CETA	590	-2.50	0.00	-4.24	276522	171	276522	0	0	0
U Longitudinal CETA	201	-25.00	0.00	-39.24	58	41864	41864	0	0	0
L Longitudinal CETA	201	-25.00	0.00	-4.24	58	41864	41864	0	0	0
F Square truss CETA	603	-12.50	0.00	-20.00	265783	147070	118866	0	0	0
A Square truss CETA	603	-37.50	0.00	-20.00	265783	147070	118866	0	0	0
UAP RCM	135	-39.60	-17.94	-39.36	27	26	16	0	0	0
UAS RCM	135	-39.60	17.94	-39.36	27	26	16	0	0	0
LAP RCM	135	-39.36	-17.94	-0.40	16	26	27	0	0	0
LAS RCM	135	-39.36	17.94	-0.40	16	26	27	0	0	0
FP RCM	135	-0.40	-36.86	0.44	27	16	26	0	0	0
FS RCM	135	-0.40	36.86	0.44	27	16	26	0	0	0
UP Cryogenic tanks	13906	-10.12	-19.79	-22.50	132422	401600	401600	0	0	0
US Cryogenic tanks	13906	-10.12	19.79	-22.50	132422	401600	401600	0	0	0
LP Cryogenic tanks	13906	-10.12	-19.79	-7.29	132422	401600	401600	0	0	0
LS Cryogenic tanks	13906	-10.12	19.79	-7.29	132422	401600	401600	0	0	0
TOTAL DRY WEIGHT	404040	-14.39	-1.72	-4.72	204184191	116828227	214064821	-1041379	7250831	-2904517
UP LOX/LH	45351	-10.12	-19.79	-22.50	89015	606336	606336	0	0	0
US LOX/LH	45351	-10.12	19.79	-22.50	89015	606336	606336	0	0	0
LP LOX/LH	45351	-10.12	-19.79	-7.29	89015	606336	606336	0	0	0
LS LOX/LH	45351	-10.12	19.79	-7.29	89015	606336	606336			
TOTAL WET WEIGHT	585446	-13.07	-1.19	-7.87	299393765	144988667	290174727	-121898	1812882	-5094592
F OTV (fully loaded in hangar)	199021	-11.08	0.00	-14.36	7809601	7809601	4891017	0	0	0
A OTV (fully loaded in hangar)	199021	-35.00	0.00	-14.36	7809601	7809601	4891017	0	0	0
TOTAL GROSS WEIGHT	983489	-17.27	-0.69	-10.61	325037058	250221241	354831538	-3040980	17769598	-6993346

Abbreviations: P - Port, S - Starboard, F - Forward, A - Aft, T - Top, B - Bottom, U - Upper, L - Lower, I - Inner, O - Outer

Table 9.0-3, Mass Properties, English

ORIGINAL PAGE IS OF POOR QUALITY

LEO Transportation Node Weight Statement (English units)

Item:	Weight (lbs)	x (in)	y (in)	z (in)	Ixx (slugs-ft^2)	Iyy (slugs-ft^2)	Izz (slugs-ft^2)	Ixy (slugs-ft^2)	Ixz (slugs-ft^2)	Iyz (slugs-ft^2)
S wing truss	4096	-96	1575	-98	260858	31613.92	60815.5	-487.7	-4958.2	-10112.2
P wing truss	4096	-96	-1575	-98	260858	31613.9	260815.5	-487.7	-4958.2	-10112.2
CF "T" truss	434	-136	0	-98	2696.6	1946.2	2815.4	-325.6	325.5	-582.2
F square truss	3622	-489	0	-784	445675.2	250820.7	204508.2	-6028	-5771.2	1421.7
A square truss	3622	-1476	0	-784	445675.2	250820.7	204508.2	-6028	-5772.2	1421.7
UC longeron truss	496	-988	0	-1476	1748.8	5456.3	5051.5	-873.3	-488.2	488.2
LC longeron truss	496	-988	0	-98	1748.8	5456.3	5051.5	-873.3	-488.2	488.2
UCF truss	279	-157	0	1476	991.3	1347.5	1098.8	-582.2	-325.5	325.5
UCA truss	279	-1811	0	-1476	991.3	1347.5	1098.8	-582.2	-325.5	325.5
LCA truss	279	-1811	0	-98	991.3	1347.5	1098.8	-582.2	-325.5	325.5
F wall	10264	0	0	-591	608155.2	257571.6	350583.6	0	0	0
A wall	4322	-1969	0	-591	256065.1	108451.1	147614	0	0	0
P wall	12348	-984	689	-591	309859.9	1170582	860721.8	0	0	0
S wall	12348	-984	-689	-591	309859.9	1170582	860721.8	0	0	0
T wall	8644	-984	0	-1378	295227.6	602505.3	897732.8	0	0	0
B wall	441	-984	-197	0	15062.63	30740.06	45802.7	0	0	0
USI PV wing	2489	-98	1969	-848	69033.65	62534.95	6498.7	0	0	0
USO PV wing	2489	-98	2756	-848	69033.65	62534.95	6498.7	0	0	0
LSI PV wing	2489	-98	1969	651	69033.65	62534.95	6498.7	0	0	0
LSO PV wing	2489	-98	2756	651	69033.65	62534.95	6498.7	0	0	0
UPI PV wing	2489	-98	-1969	-848	69033.65	62534.95	6498.7	0	0	0
UPO PV wing	2489	-98	-2756	-848	69033.65	62534.95	6498.7	0	0	0
LPI PV wing	2489	-98	-1969	651	69033.65	62534.95	6498.7	0	0	0
LPO PV wing	2489	-98	-2756	651	69033.65	62534.95	6498.7	0	0	0
SI EPS radiator	2798	-482	1969	-98	1043.56	16358.45	17402.02	0	0	0
SO EPS radiator	2798	-482	2756	-98	1043.56	16358.45	17402.02	0	0	0
PI EPS radiator	2798	-482	-1969	-98	1043.56	16358.45	17402.02	0	0	0
PO EPS radiator	2798	-482	-2756	-98	1043.56	16358.45	17402.02	0	0	0
SI Batteries/Data management	6402	-98	1969	-98	1488.16	930.14	930.14	0	0	0
SO Batteries/Data management	5700	-98	2756	-98	1488.16	930.14	930.14	0	0	0
PI Batteries/Data management	6402	-98	-1969	-98	1488.16	930.14	930.14	0	0	0
PO Batteries/Data management	5700	-98	-2756	-98	1488.16	930.14	930.14	0	0	0
S Thermal radiator	4046	-557	1181	760	40091.38	15938.22	24153.16	0	0	0
P Thermal radiator	4046	-557	-1181	760	40091.38	15.938	24.153	0	0	0
Node 1 (FS)	34283	146	-8	97	45137.63	34826.02	45137.63	0	0	0
Node 2 (FP)	34283	146	-204	97	45137.63	34826.02	45137.63	0	0	0
Node 3 (S)	34283	-436	-8	97	45137.63	34826.02	45137.63	0	0	0
Node 4 (P)	34283	-436	-204	97	45137.63	34826.02	45137.63	0	0	0
Node 5 (FV)	34283	-436	0	-106	45137.63	45137.63	34826.02	0	0	0
Node 6 (S)	34283	-969	-8	97	45137.63	34826.02	45137.63	0	0	0
Node 7 (P)	34283	-969	-204	97	45137.63	34826.02	45137.63	0	0	0
Node 8 (V)	34283	-969	-326	-526	45137.63	45137.63	34826.02	0	0	0
Node 9 (A)	34283	-1378	-8	97	45137.63	34826.02	45137.63	0	0	0
Node 10 (VA)	34283	-1378	0	-106	45137.63	45137.63	34826.02	0	0	0
Hab Module 1 (S/quiet)	45838	-159	0	98	45898.1	165260.6	165260.6	0	0	0
Hab Module 2 (P/active)	41260	-159	-326	98	43114.4	93799.2	93799.2	0	0	0
Pressurized Logistics Module	16845	-436	-328	280	17424.1	17424.1	16753.45	0	0	0
Airlock 1 (F)	8254	-463	-497	97	5438.92	4617.77	5438.92	0	0	0

Abbreviations: P - Port, S - Starboard, F - Forward, A - Aft, T - Top, B - Bottom, U - Upper, L - Lower, I - Inner, O - Outer

Table 9.0-3, Mass Properties, English, Continued

Item:	Weight (lbs)	x (in)	y (in)	z (in)	Ixx (slugs-ft^2)	Iyy (slugs-ft^2)	Izz (slugs-ft^2)	Ixy (slugs-ft^2)	Ixz (slugs-ft^2)	Iyz (slugs-ft^2)
Airlock 2 (A)	8254	-1378	73	97	5438.92	4617.77	5438.92	0	0	0
Workshop Module 1 (F)	27640	-716	0	98	28882.2	62835.9	62835.9	0	0	0
Workshop Module 2 (A)	27640	-1222	0	98	28882.2	62835.9	62835.9	0	0	0
Cupola 1	3000	-969	-327	-667	414.46	414.46	311.54	0	0	0
Cupola 2	3000	146	-402	97	414.46	414.46	311.54	0	0	0
Cupola 3	3000	146	189	97	414.46	414.46	311.54	0	0	0
Tunnel	3058	-969	-327	-210	10793.43	10793.43	2180.49	0	0	0
P GN&C pallet	4824	-98	-1378	-276	983.56	983.56	1815.79	0	0	0
S GN&C pallet	4824	-98	1378	-276	983.56	983.56	1815.79	0	0	0
F RCS tanks pallet	4461	-98	0	-98	1597	1597	982.8	0	0	0
A RCS tanks pallet	4461	-1476	0	-98	1597	1597	982.8	0	0	0
P TCS pallet	2845	-98	-1181	118	634.6	1031.2	1031.2	0	0	0
S TCS Pallet	1739	-98	1181	118	387.9	630.3	630.3	0	0	0
F Square Truss U Trays	8092	-492	0	-787	1281225	566853.1	714843	0	0	0
A Square Truss U Trays	8092	-1476	0	-787	1281225	566853.1	714843	0	0	0
F Transverse U Trays	9025	-94	0	-98	3202910.52	19133.78	3184280	0	0	0
U Longitudinal U Trays	2490	-984	0	-1476	5278.1	189161.1	184022.1	0	0	0
L Longitudinal U Trays	2178.4	-1110.85	0	-1476.45	4618.4	128048.7	123551.9	0	0	0
LOX/LH propellant feed lines (wet)	37231	-546	-11	-726	3012965	1959314	1463008	40872.1	-197965.6	-34307.4
F RMS mobile transporter	1600	-492	591	-1358	749.5	749.5	1338.3	0	0	0
A RMS mobile transporter	1600	-1476	-591	-1339	749.5	749.5	1338.3	0	0	0
S S/G S/S Antenna	674	-492.13	728	-1574.8	0	0	0	0	0	0
P S/G S/S Antenna	674	-492.13	-728	-1574.8	0	0	0	0	0	0
HLLV tanker resupply interface	742	-492.15	0	-1574.8	0	0	0	0	0	0
Lander/OTV propellant boom/interface	2000	-969.12	-98.4	-420	0	0	0	0	0	0
A RMS	2000	-1476.38	-590.55	-1318.9	12952.14	3.45	12952.14	0	0	0
F RMS	2000	-492.15	590.5	-1318.9	12952.14	3.45	12952.14	0	0	0
Transverse truss CETA	1300.5	-98.4	0	-166.86	203966.55	126.23	203966.55	0	0	0
U Longitudinal CETA	443	-984.3	0	-1544.88	43.03	30879.47	30879.47	0	0	0
L Longitudinal CETA	443	-984	0	-167	43.03	30879.47	30879.47	0	0	0
F Square truss CETA	1329	-492	0	-787	196045.4	108480.7	87677.6	0	0	0
A Square truss CETA	1329	-1476	0	-787	196045.4	108480.7	87677.6	0	0	0
UAP RCM	297	-1559	-707	-1550	20.25	18.91	11.73	0	0	0
UAS RCM	297	-1559	707	-1550	20.25	18.91	11.73	0	0	0
LAP RCM	297	-1550	-707	-16	11.73	18.91	20.25	0	0	0
LAS RCM	297	-1550	707	-16	11.73	18.91	20.25	0	0	0
FP RCM	297	-16	-1451	18	20.25	11.73	18.91	0	0	0
FS RCM	297	-16	1451	18	20.25	11.73	18.91	0	0	0
UP Cryogenic tanks	30663	-398	-779	-886	97676.72	296226	296226	0	0	0
US Cryogenic tanks	30663	-398	779	-886	97676.72	296226	296226	0	0	0
LP Cryogenic tanks	30663	-398	-779	-287	97676.72	296226	296226	0	0	0
LS Cryogenic tanks	30663	-398	779	-287	97676.72	296226	296226	0	0	0
TOTAL DRY WEIGHT	890908.6	-566.55	-67.71	-185.94	150609300	86174240	157897400	-768136.5	5348321	-2142415
UP LOX/LH	100000	-398	-779	-886	65658.6	447242.6	447242.6	0	0	0
US LOX/LH	100000	-398	779	-886	65658.6	447242.6	447242.6	0	0	0
LP LOX/LH	100000	-398	-779	-287	65658.6	447242.6	447242.6	0	0	0
LS LOX/LH	100000	-398	779	-287	65658.6	447242.6	447242.6	0	0	0
TOTAL WET WEIGHT	1290909	-514	-47	-310	220837300	106945800	214037200	-89913.7	1337209	-3757847
F OTV (fully loaded in hangar)	438842	-436	0	-565	5760478	5760478	3607687	0	0	0
A OTV (fully loaded in hangar)	438842	-1378	0	-565	5760478	5760478	3607687	0	0	0
TOTAL GROSS WEIGHT	2168593	-680	-27	-418	239752175	184566914	261729027	-2243072	13107120	-5158396

Abbreviations: P = Port, S = Starboard, F = Forward, A = Aft, T = Top, B = Bottom, U = Upper, L = Lower, I = Inner, O = Outer

Table 9.0-4, Phase 1 Freedom Space Station Weight Statement

WEIGHT BREAKDOWN OF WORKPACKAGE PROPOSALS

JUNE 6 1988 WEIGHTS

FLIGHT WEIGHT (PH 1) FOR ELEMENT & SYSTEM	A	B	C	D	E	F	G	H	I	J	K	L	M	N	O	P	Q	R	S	T	U	V	W	X	Y	
ELEMENT	US LAB MODULE	COLUMBU: LAB MODULE	JEM LAB: & EIP FAC.	HAB MODULE	MSC & SPDM	MSC MAINT DEPOT	MOBILE TRANS-PORTER		ATT P/L ACCOM EQUIP	PRESS LOG CARRIER	UNPRESS LOG CARRIER	JEM LOG EXPRMNT MODULE	RESOURC NODE 1	RESOURC NODE 2	RESOURC NODE 3	RESOURC NODE 4	AIRLOCK	HYPER-BARIC AIRLOCK	SOLAR POWER MODULE	TRUSS ASSEM.	PROP ASSEM.	US POLAR PLATFOR		COLUMBU POLAR PLATFOR	TELE-ROBOTIC SERVICE	TOTAL
SYSTEM	LB/DRY	LB/DRY	LB/DRY	LB/DRY	LB/DRY	LB/DRY	LB/DRY	LB/DRY	LB/DRY	LB/DRY	LB/DRY	LB/DRY	LB/DRY	LB/DRY	LB/DRY	LB/DRY	LB/DRY	LB/DRY	LB/DRY	LB/DRY	LB/DRY	LB/DRY	LB/DRY	LB/DRY	LB/DRY	LB/DRY
1 ECLSS	7629			7888						1271			401	324	253	215	180	166								18327
2 PROPULSION																					9659					9659
3 THERMAL,INT	3203			2693						642	117		737	576	603	409	51	51								9082
4 THERMAL,EXT	846	202	202	564			26		1327				71	71	71	71	60	60	11192	10004	8					24935
5 FLUIDS													434	194	107	292				3102	205					4334
6 COMM & TRACKING							267						610	610	47	47	68	83		1268	211					3211
7 AUDIO/VIDEO,INT	710			864						51			95	88	310	154	20	20								2344
8 GUIDE, NAV & CNTRL									160											3964	478					4602
9 DATA MANAGEMENT	2530			1778			244		114	150			1086	1020	739	637	46	46	248	1749	633					11020
10 MANNED	4179			6761						122			1510	262	2115	894	6	6								15855
11 EVA	385	45	19	385						64			12	12	38	38	3845	2691	636	893	32	33				9148
12 POWER	3323			2000			429			564	6		1059	1059	1424	1394	442	442	33910	3070	809	2533				53426
13 STRUCTURES	10944			17823			150			12503	2563		6717	6737	10051	9738	3263	3263	6225	15309	2025					116279
14 MECHANISMS	1010			1010			1100			676	138		2440	2302	4215	3956	1178	1178	4048	2362						26517
15 ELEMENT UNIQUE	13523			6006					13564	727	2009		239	239	251	244	943	1090							1983	40813
16																										0
INTERNAT SUB TOTAL		27552	47551		9389	1424						9755														95671
US SUB TOTAL	56300	327	301	48660			2312		15165	16845	4033		15411	13502	20232	10089	10110	9104	57059	41001	14960	2566			1983	349560
P/L US																										
P/L INTERNATIONAL																										
P/L TOTAL																										
TOTAL	56300	27879	47852	48660	9389	1424	2312		15165	16845	4033	9755	15411	13502	20232	10089	10110	9104	57059	41001	14960	2566			1983	445231

NOTES: a) Manned Systems does not include GFE consumables or outfitting.
b) Pressurized and unpressurized logistics carriers are shown as unit weights
c) There are 2 APAE - 1 pointing and 1 non-pointing
d) The P/L total of 77,000 is supplied by SSU
e) Telerobotic servicer includes FTS shelter weight (483 lb) and FTS (1,500 lb)
f) PLC - element unique (15J) - does not include ASTF (1,242 lb) which is not on every flight
g) UPLC - element unique (15K) - includes 1 fluid subcarrier and 1 dry cargo subcarrier
h) Manned systems weights do not include GFE consumables or outfitting

Changes from April 8, 1988 Matrix:

Cell 10 D: Dropped Safe Haven, Skip Cycle & ORU's (not picked up elsewhere
Cell 15 D: SH, SC, ORU went from 10,000 lb to 2,000lb
Cell 15 I: More detail, better estimate
Cell 15 M: Moved SH, SC, ORU to cell 10 M
Cell 10 M: See above
Cell 13 O: Added Berthing/Docking Structure
Cell 10 O: Added SH, SC, ORU Storage
Cell 15 O: Moved SH, SC, ORU to cell 10 O
Cell 15 P: Moved SH, SC, ORU to cell 10 P
Cell 10 P: See above
Cell 10 O: 860 lbs carried in PDRD not identified, Current 6 lbs for handrails, etc.

Cell 15 R: Greater detail, better estimate .
Cell 9 T: Added Avionics Pallet 016
Cell 12 T: Decrease in weight of PMAD Estimate
Cell 13 T: Added 2 propulsion pallets
Cell 2 U: Added 2 propulsion units

10.0 Scaling Factors

This report is a first pass at a space station to support a permanent reusable transportation system. The numbers generated are highly dependent on the basic assumptions used which can easily change. This section provides first pass methods of scaling some of the major masses which may be of use in future works.

10.1 Mass and Dimensions of Tankage as a Function of Storage Capacity

The following graphs and spreadsheets give mass, dimensions, and boil-off of tankage as a function of storage capacity.

The basis of this analysis will be storage of liquid propellants. Other storage schemes are possible. For instance, a solid hydride could be used to store hydrogen for long time periods without boil-off. Heating would release the hydrogen which could be subsequently liquified and used on a short term basis. Another option is to store water, then electrolyze and liquify LO_2/LH_2 when required. These options will not be included in this analysis.

The propellants considered for the analysis are given in Table 10.1-1 and include liquid oxygen and hydrogen, methane, hydrazine, nitrogen tetroxide, UDMH, and MMH. The graphs (Figures 10.1-1 through 10.1-12) show tank shell and insulation mass as a function of usable propellant mass, and as a function of the number of storage tanks, for cryogenic oxygen and hydrogen storage, as well as typical space-storable propellants. The tank shell and insulation mass plots alone may result in unrealistically low estimates if taken alone. Baffles, plumbing, launch loads, structural meteroid protection, quantity measurements, connections, vents, and manufacturing problems were not included or considered in these plots and may result in greater system mass. Cryogenic propellant boil-off rates are calculated from a simplified thermal model that includes the effects of solar, Earth reflected, and Earth infrared radiation fluxes. As expected, the graphs show that it is more efficient, in terms of tank shell/insulation mass and boil-off, to use the minimum number of storage tanks within the constraints of launcher volume. Tank diameters are also given in the graphs. This analysis is based on spherical tanks, other geometries have not been examined.

The following sections show the scaling equations used to produce the graphs. These scaling equations can be used to complete the following trades:

- Cryogenic tank pressure vs. shell mass & change in boil-off rate.
- Insulation thickness & mass vs. boil-off rate.
- Shell material type (aluminum, titanium, etc.) vs. tank shell mass.

10.1.1 Tank Scaling Equations

Basis: Spherical tanks.

$$D = \{6 * M_p * (1 + U)/(\rho_p * \pi * N)\}^{1/3} \qquad \text{(Eqn. 10.1-1)}$$

where,

D = Tank diameter (m)

M_p = Propellant mass (mt) = Usable Propellant (mt)/Recovery Factor, where the usable propellant (recovery factor) = 98%

U = Tank ullage, fraction of tank volume not filled at 100% capacity = 0.05.

p_p = Propellant density (mt/m^3)

N = Number of propellant tanks, variable.

$$t = P * D * FOS/(4 * \sigma) \qquad \text{(Eqn.10.1-2)}$$

where,

t = Tank skin thickness (mm), where the lower limit of skin thickness is defined as 25 mils (0.635 mm)

P = Maximum internal tank pressure (kPa)

FOS = Factor of safety = 1.5 as defined for space station pressure vessels.

σ = Skin material yield stress (MPa), which for Al 2219-T87 (used for cryogenic applications) = 352 MPa.

$$M_s = N * 4/3 * \pi * p_s * [(D/2 + t/1000)^3 - (D/2)^3] \qquad \text{(Eqn. 10.1-3)}$$

where,

M_s = Mass of tank shells (mt)

p_s = Density of tank shell (mt/m^3), for Al 2219-T87 = 2.82 mt/m^3.

If the thickness of the tank shell is not limited by the minimum thickness constraint (0.635 mm), the mass of spherical tank shells is independent of the number of tanks:

$$M_s = p_s * M_p * (1 + U)/p_p * \{(1 + P * FOS/\sigma)^3 - 1\} \qquad \text{(Eqn. 10.1-3b)}$$

The maximum internal tank pressure is defined as:

$$P = 2 * P_{op} * CF \qquad \text{(Eqn. 10.1-4)}$$

where,

P_{op} = Tank operating pressure (atm), which for space-storable propellants was selected as 1 atm, but for cryogenic propellants was baselined at 5 atm. The higher pressure increases the boiling point of the cryogenic propellants and reduces boil-off. Table 10.1-2 lists the expressions that relate pressure to temperature for oxygen, hydrogen, and methane.

CF = Conversion factor for converting atm to kPa = 101.325

The mass of multilayer insulation (MLI) is calculated from:

$$M_{mli} = N * 4/3 * \pi * p_{mli} * [(D/2 + t/1000 + t_{mli}/100)^3 - (D/2 + t/1000)^3] \quad \text{(Eqn. 10.1-5)}$$

where,

M_{mli}	=	Mass of MLI (mt)
p_{mli}	=	Density of MLI (mt/m^3) = 0.12
t_{mli}	=	MLI thickness (cm), baselined as 7.62 cm (3") for cryogenic fluids and 1 cm for storable propellants.

10.1.2 Boil-Off Scaling Equations

The boil-off rate for cryogenic propellants was calculated from a simplified thermal analysis that included the effects of solar, Earth reflected, and Earth infrared radiation fluxes. The steady-state thermal model is:

$$Q_{in} = Q_{out}$$

$$f_s\, \alpha\, G\, A_p + f_s\, \alpha\, E_r\, F_r\, A_p + \varepsilon\, E_{ir}\, F_{ir}\, A_p = \varepsilon\, \sigma\, T_s^4\, A_s + Q_{abs} \quad \text{(Eqn. 10.1-6)}$$

where,

α	=	Absorptivity of the exterior surface (α = 0.04 for aluminized mylar MLI)
ε	=	Emissivity of exterior surface (ε = 0.72 for aluminized mylar MLI)
G	=	Solar insolation = 1399.7 W/m^2
A_p	=	Projected surface area (m^2) = $\pi D^2/4$
A_s	=	Tank surface area (m^2) = πD^2
E_r	=	Solar radiation reflected from Earth = albedo * G = 0.3 * G = 420 W/m^2
F_r	=	Thermal view factor for reflected radiation = $R_e^2/(R_e^2 + h^2)$ where R_e is the Earth's radius = 6378.145 km and h is the orbital altitude, baselined as 500 km.
E_{ir}	=	Earth thermal radiation flux = 237 W/m^2
F_{ir}	=	View factor for infrared radiation = F_r.
σ	=	Stefan-Boltzmann constant = 5.67 10^{-8} W/m^2-K^4
T_s	=	Exterior surface temperature (K)
Q_{abs}	=	Heat rate absorbed by interior cryogenic fluid (W)
f_s	=	Fraction of orbital period exposed to sunlight and reflected radiation.

$$f_s = P_s/P_o \quad \text{(Eqn. 10.1-7)}$$

$$P_s = (1 - \arcsin(R_e/(R_e + h))/\pi) * P_o \quad \text{(Eqn. 10.1-7b)}$$

$$P_o = 2 * \pi * [(R_e + h)^3/\mu_e]^{0.5} \quad \text{(Eqn. 10.1-7c)}$$

where the arcsin angle is determined in radians and where,

P_s	=	Period node is in sunlight (sec)
R_e	=	Earth's radius, 6378.145 km.
h	=	Orbital altitude, 500 km.
P_o	=	Orbital period (sec)
μ_e	=	Mu for Earth, 398601.2 km^3/s^2

Rearranging Eqn. 10.1-6, and neglecting Q_{abs} as a simplification, surface temperature, T is:

$$T_s = \{A_p * [\alpha f_s * (G + F_r E_r) + \varepsilon E_{ir} F_{ir}]/(\varepsilon \sigma A_s)\}^{1/4} \qquad \text{(Eqn. 10.1-7d)}$$

A thermal conduction model is used to determine the heat absorbed by the cryogenic fluid.

$$Q_{abs} = k_{mli} A_{gm} (T_s' - T_i)/t'_{mli} \qquad \text{(Eqn. 10.1-8)}$$

where,

Q_{abs}	=	Heat rate absorbed by cryogenic fluid (W)
k_{mli}	=	Thermal conductivity of multilayer insulation = 3.4 10^{-5} W/m-°K
T_s'	=	Surface Temperature (°C) = T_s - 273.15
T_i	=	Interior Temperature (°C) of tank which is defined as the boiling point of the cryogenic liquid at the operating pressure of the tank. The correlations in Table 10.1-2 are used to determine this temperature as a function of tank operating pressure.
t'_{mli}	=	MLI thickness (m) = $t_{mli}/100$
A_{gm}	=	Geometric mean surface area (m^2), which is used for thermal conduction in spherical geometries. It is found from:

$$A_{gm} = (A_o * A_i)^{0.5} \qquad \text{(Eqn. 10.1-9)}$$

$$A_{gm} = \pi * [(D + 2\ t/1000 + 2\ t'_{mli})^2 * D^2]^{0.5} \qquad \text{(Eqn. 10.1-9b)}$$

where,

A_o	=	Surface area of tank exterior (m^2)
A_i	=	Surface area of tank interior (m^2)

The mass rate of cryogenic liquid boil-off from all tanks is found from:

$$m_{vap} = N * 24 * Q_{abs}/H_{vap} \qquad \text{(Eqn. 10.1-10)}$$

where,

m_{vap}	=	Boil-off rate (kg/day)
N	=	Number of tanks
H_{vap}	=	Heat of vaporization (W-hr/kg) as given in Table 10.1-1. 59 W-hr/kg for O_2.

Table 10.1-1, Physical Properties of Propellants

	LOX	LH_2	N_2O_4	N_2O_4	UDMH	MMH	CH_4
Molecular Weight	31.999	2.016	92.02	32.05	60.099	46.072	16.043
Density (mt/m³)	1.14	0.0709	1.448	1.011	0.786	0.870	0.415
@ temp. (°C)	-183	-252.7	20	15	25	25	-164
Normal Boiling Point (°C)	-183	-252.7	21.3	113.5	62.3	87.7	-161.4
Melting Point (°C)	-218.4	-259.1	-9.3	1.4	-57.2	-52.4	-182.6
Heat of Vaporization @ BP (W-hr/kg)	59.1	124.5	88.9				141.6
Heat of Fusion @ MP (W-hr/kg)	3.85	16.14	69.98				16.31

Propellant Formula	Full Name	Propellant Classification
LO_2	Liquid Oxygen	Cryogenic - Oxidizer
LH_2	Liquid Hydrogen	Cryogenic - Fuel
N_2O_4	Nitrogen Tetroxide	Storable - Oxidizer
N_2H_4	Hydrazine	Storable - Hypergolic Monopropellant or Bipropellant Fuel
$(CH_3)_2N_2H_2$	Unsymmetrical Dimethyl-Hydrazine (UDMH)	Storable - Fuel
$CH_3N_2H_3$	Monomethylhydrazine (MMH)	Storable - Fuel
CH_4	Methane	Cryogenic - Hydrocarbon Fuel

Figure 10-1.1, LO_2 Storage Tank Mass Versus Stored LO_2 Mass

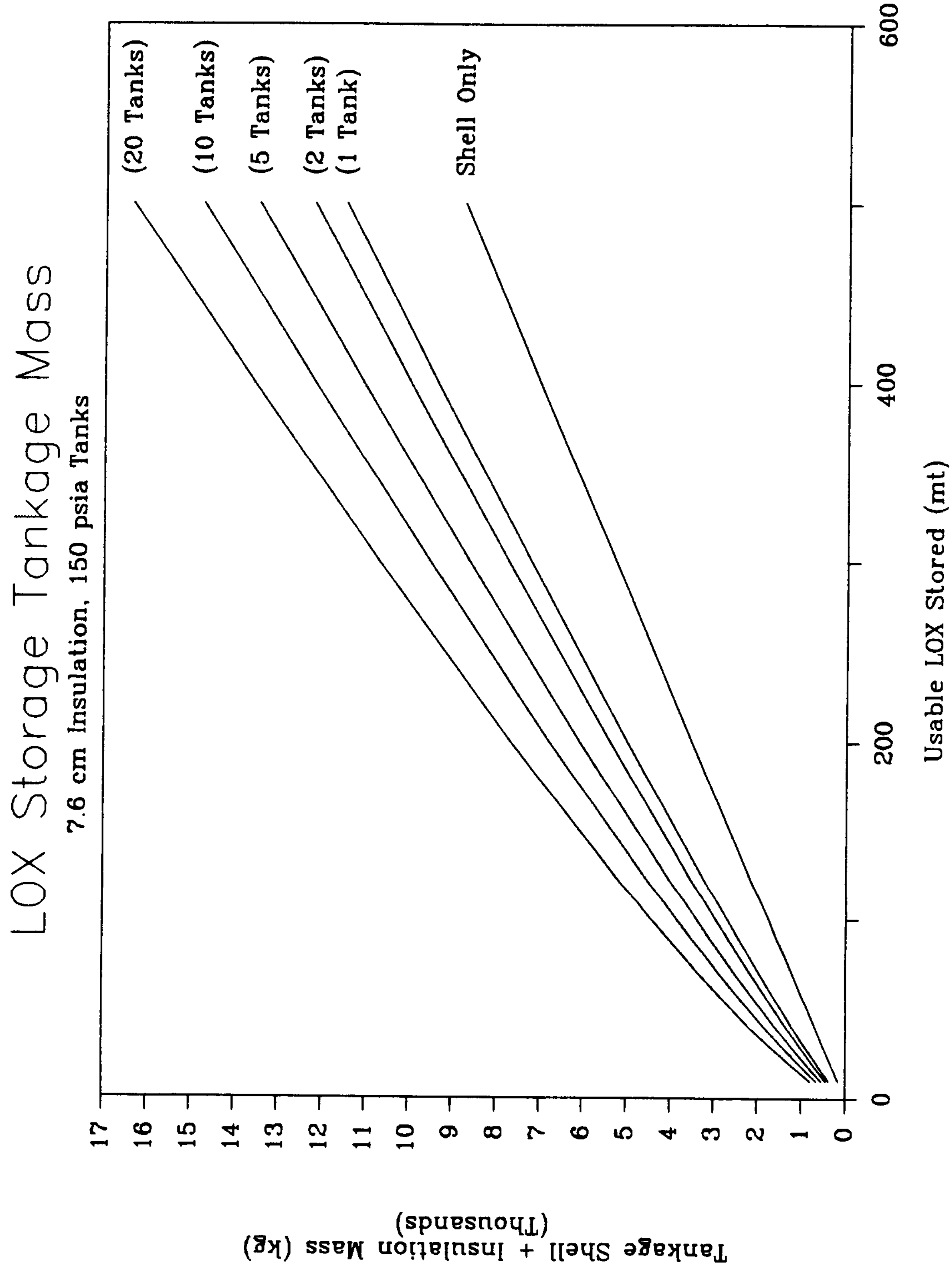


Figure 10.1-2, LO_2 Storage Tank Boil-Off Rate Versus Stored LO_2 Mass

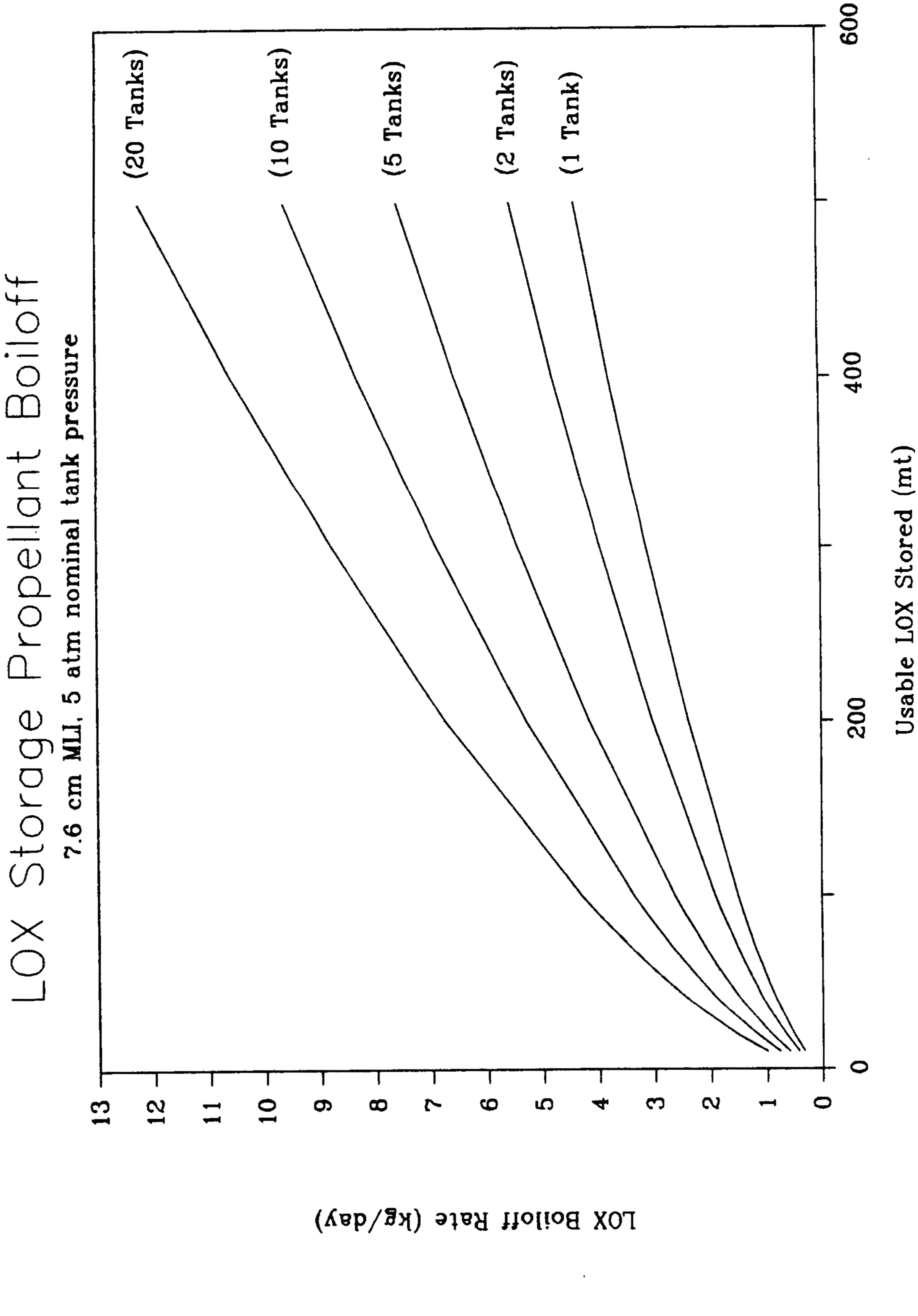


Figure 10.1-3, LO_2 Storage Tank Diameter Versus Stored LO_2 Mass

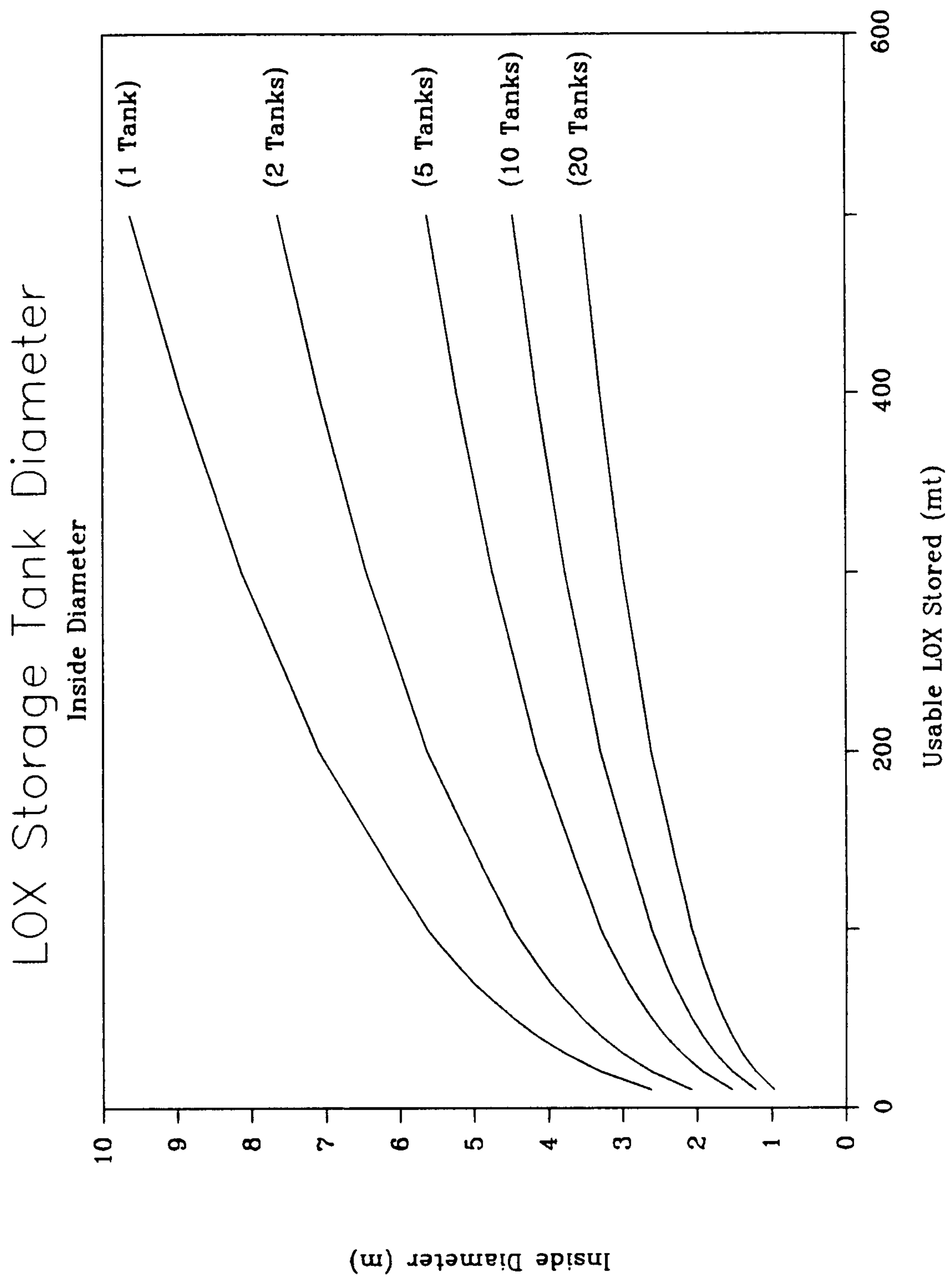


Figure 10.1-4, LH_2 Storage Tank Mass Versus Stored LH_2 Mass

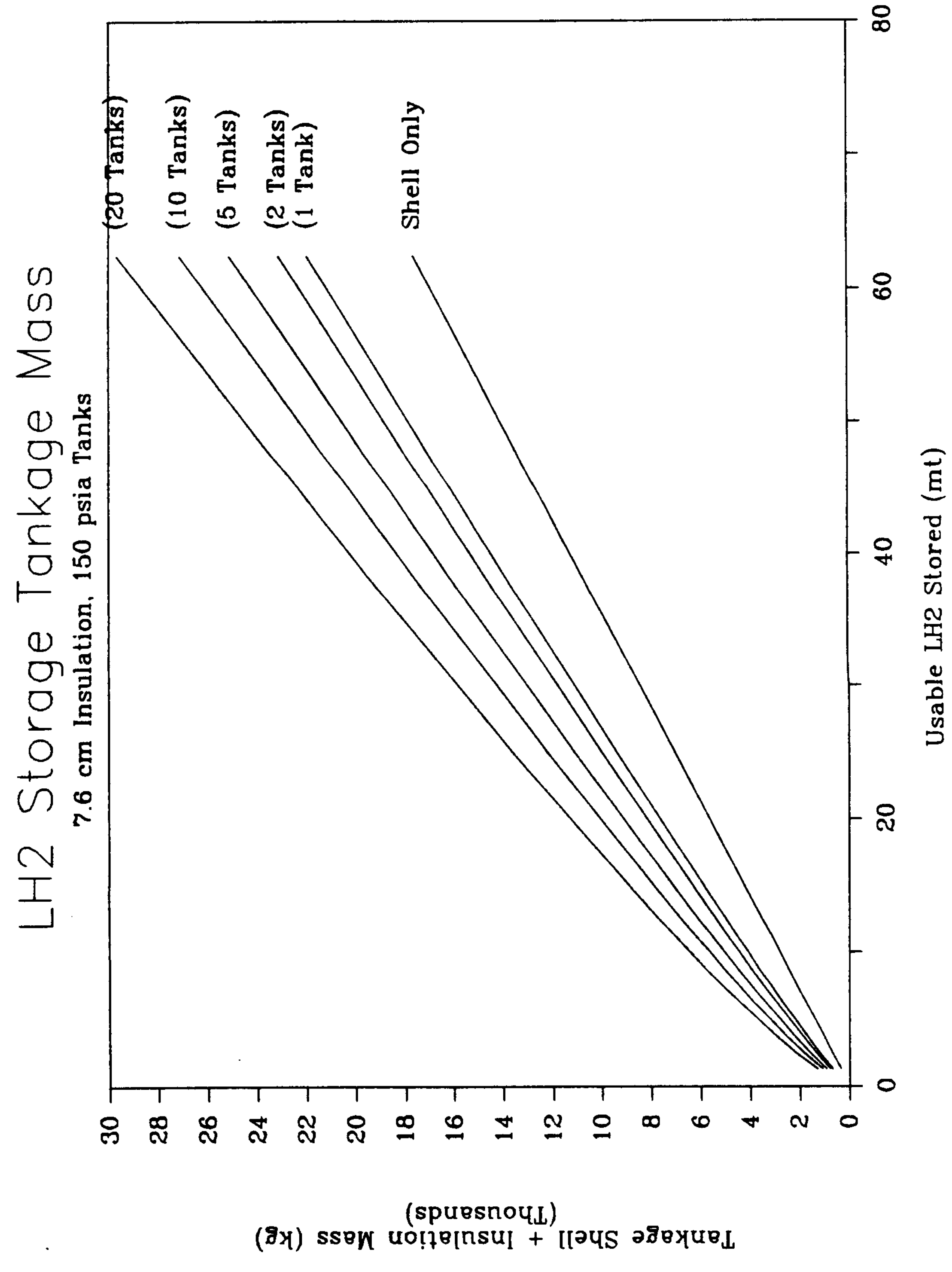


Figure 10.1-5, LH$_2$ Storage Tank Boil-Off Rate Versus Stored LH$_2$ Mass

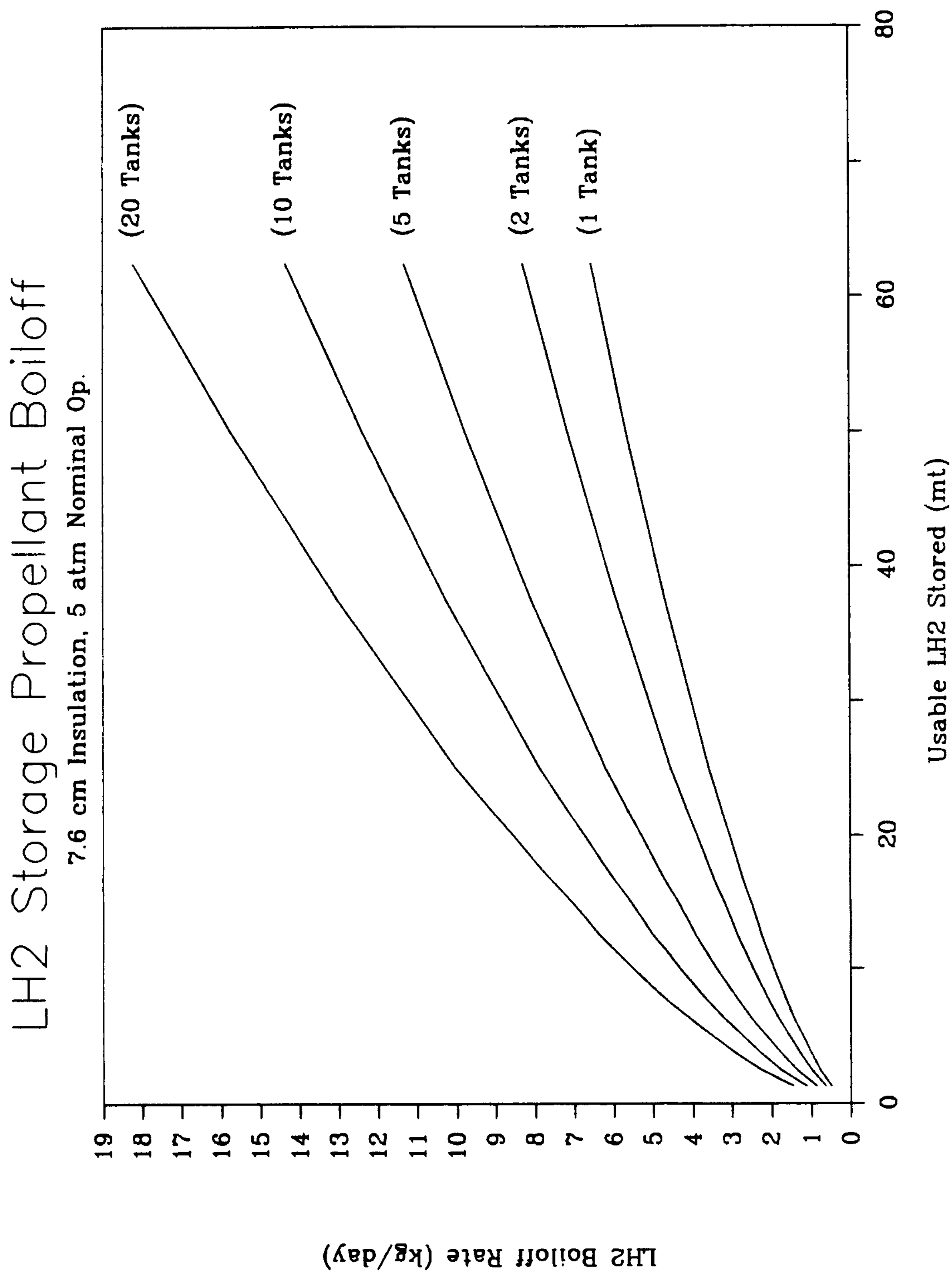


Figure 10.1-6, LH_2 Storage Tank Diameter Versus Stored LH_2 Mass

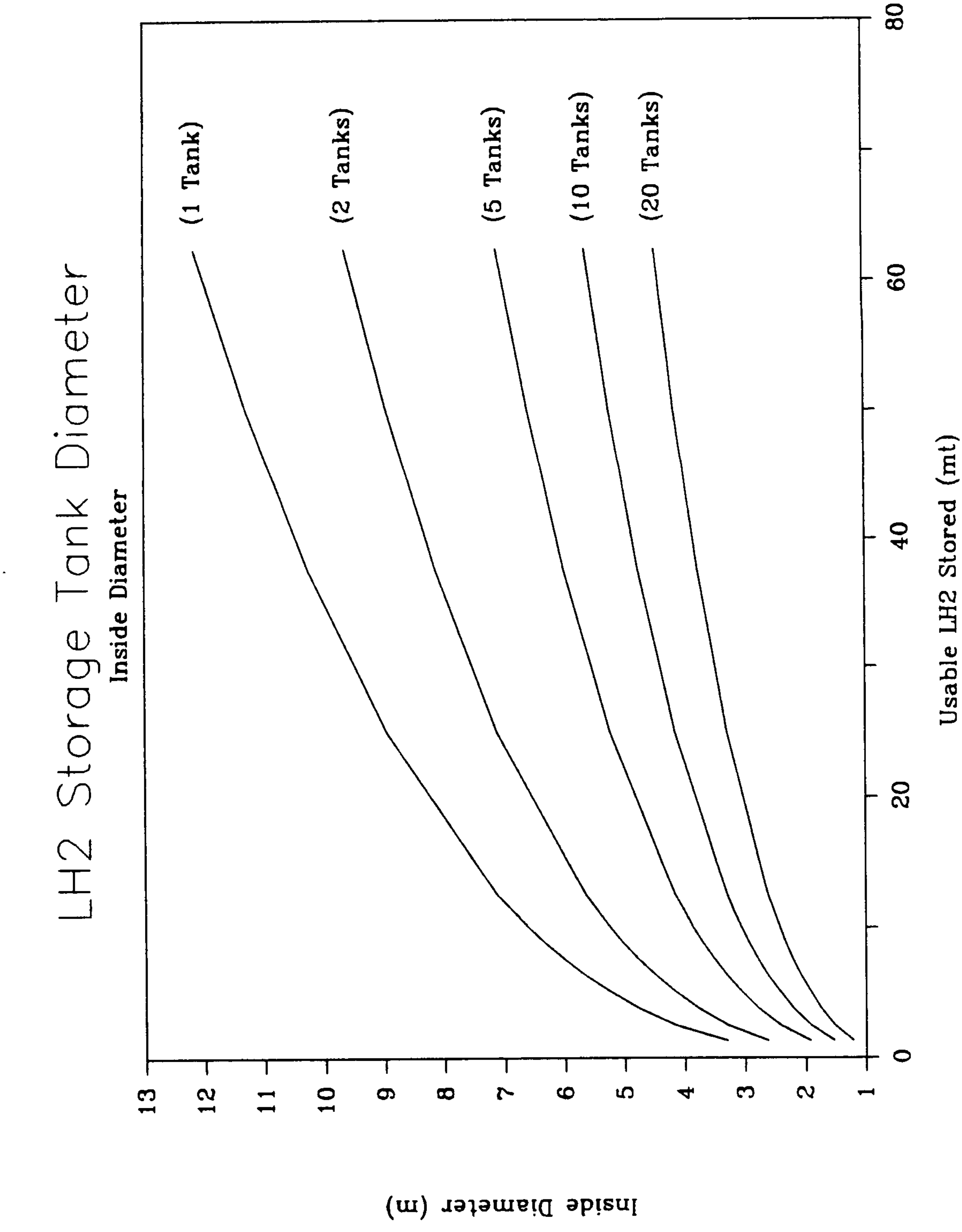


Figure 10.1-7, Hydrazine Tank Mass Versus Stored Hydrazine Mass

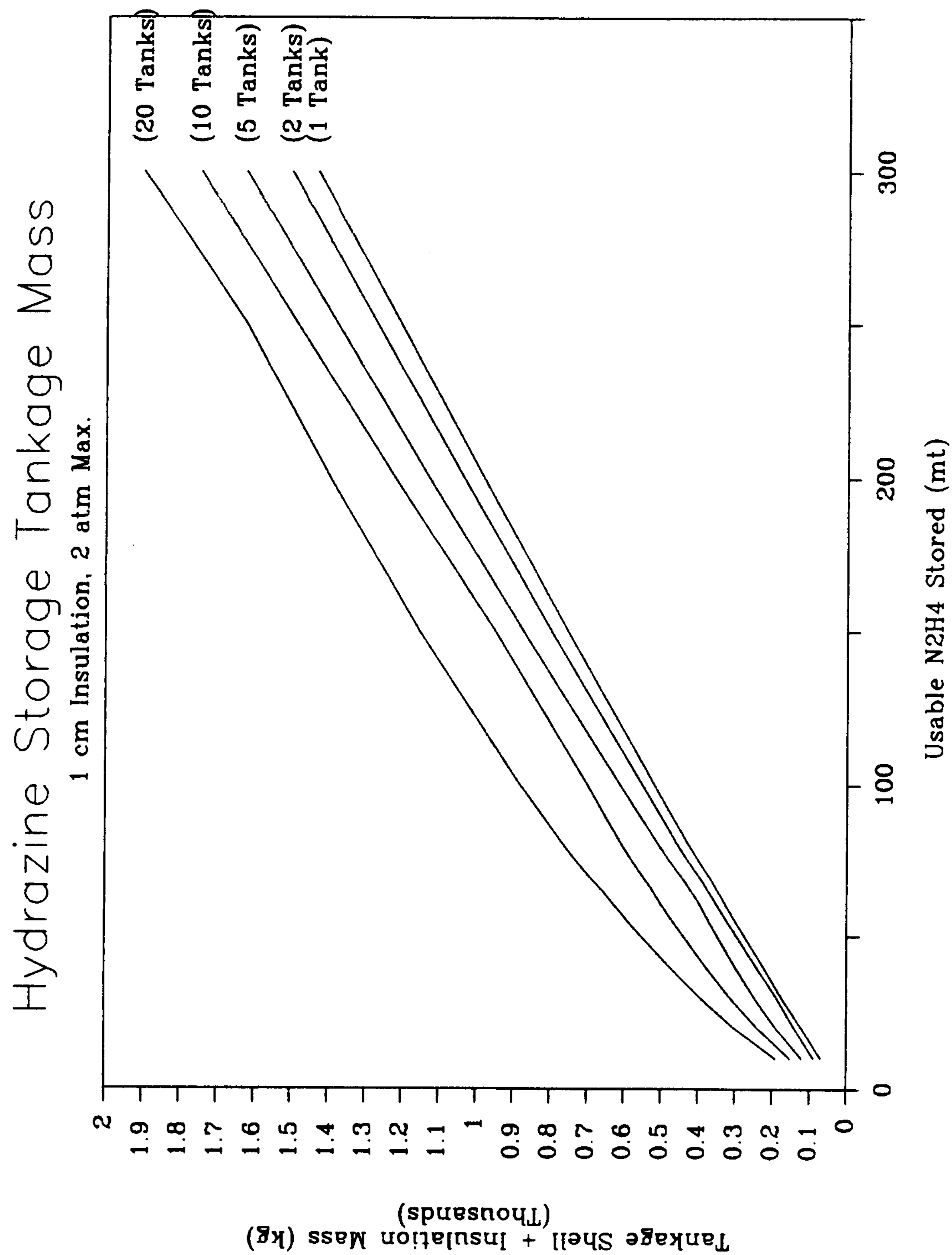


Figure 10.1-8, Hydrazine Tank Diameter Versus Stored Hydrazine Mass

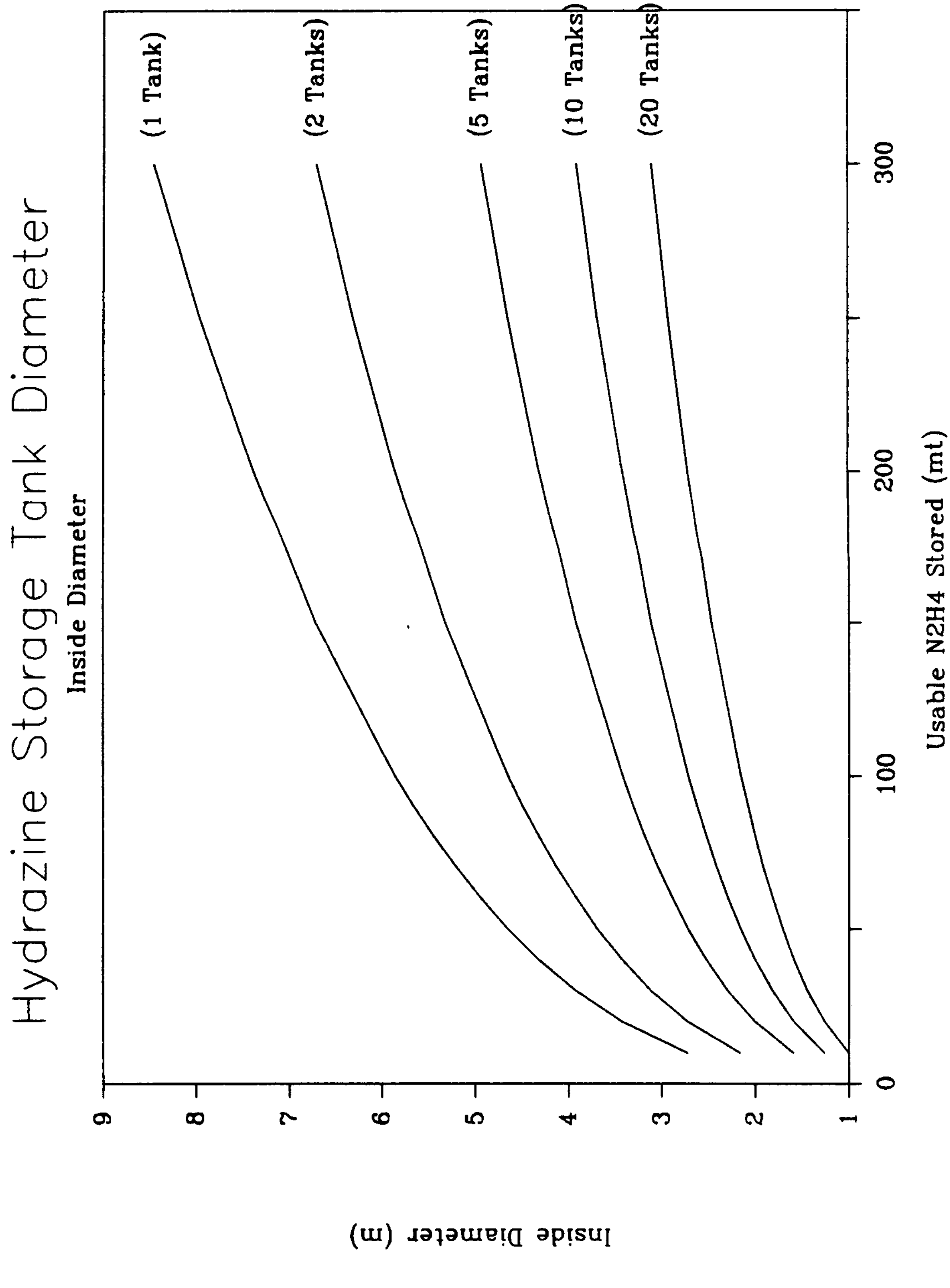


Figure 10.1-9, UDMH Tank Mass Versus Stored UDMH Mass

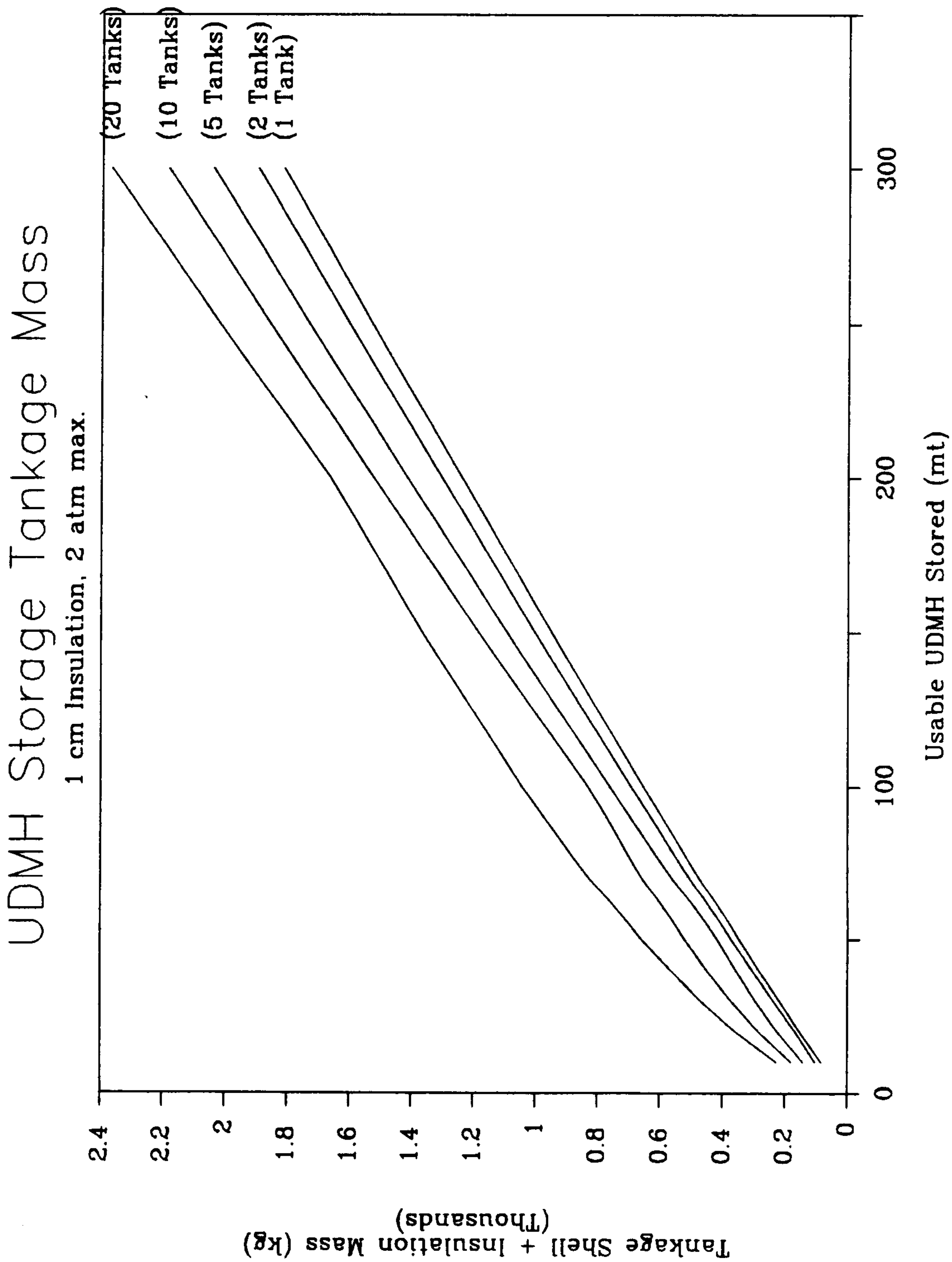


Figure 10.1-10, UDMH Tank Diameter Versus Stored UDMH Mass

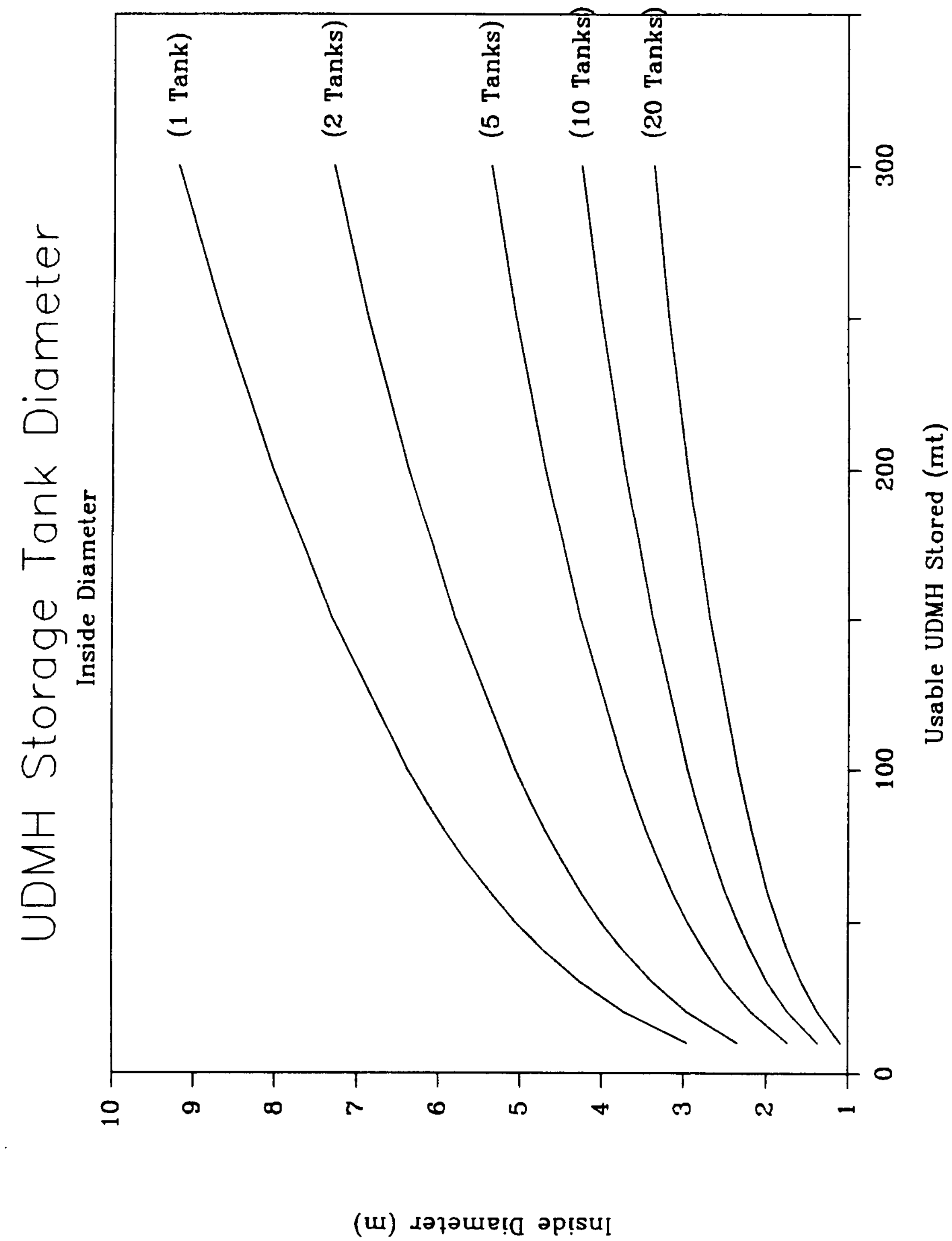


Figure 10.1-11, N_2O_4 Tank Mass Versus Stored N_2O_4 Mass

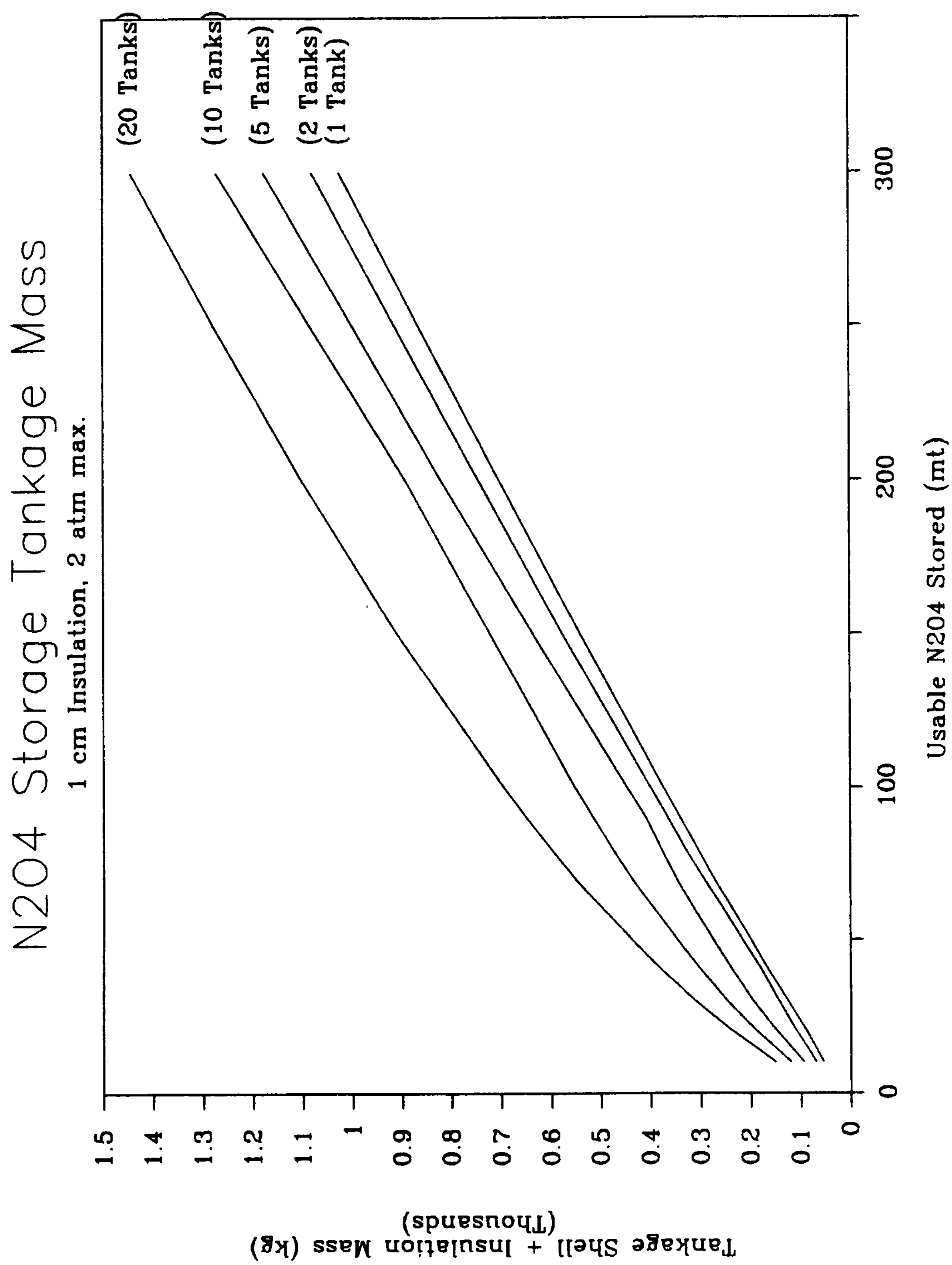


Figure 10.1-12, N_2O_4 Tank Diameter Versus Stored N_2O_4 Mass

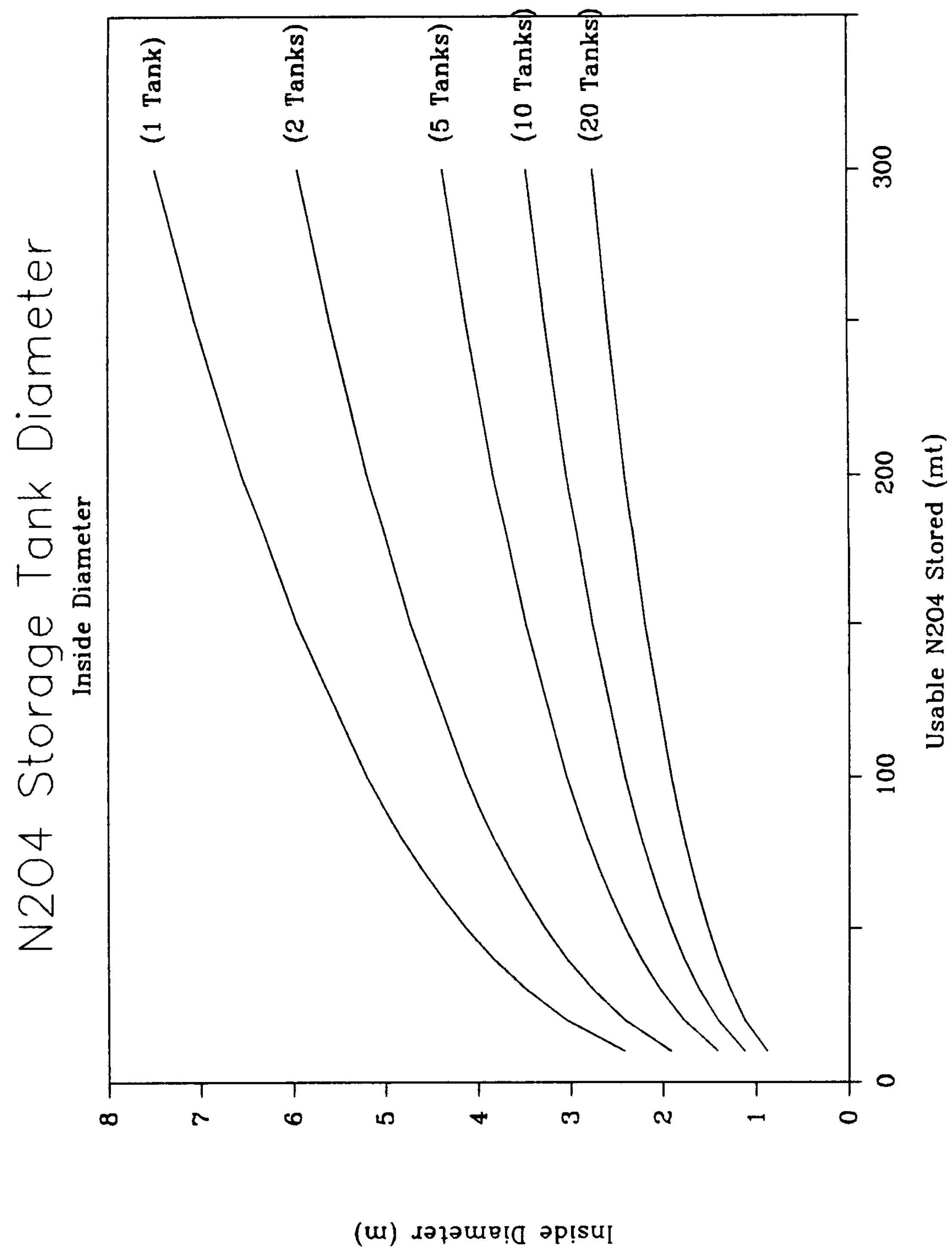


Table 10.1-2, Cryogenic Propellant Vapor Pressure Correlations

Propellant Type	Vapor Pressure (atm)	Temperature (°C)	
OXYGEN	1	-183.1	
	2	-176.0	
	5	-164.5	
	10	-153.2	
	20	-140.0	
	30	-130.7	
	40	-124.1	
	49.7	-118.9	Critical Point

Correlation: $T\ (^\circ C) = 31.4479 \bullet [P\ (atm)]^{0.2857} - 214.268$

$P\ (atm) = (0.31799 \bullet T\ (^\circ C) + 6.8134)^{3.5}$

Propellant Type	Vapor Pressure (atm)	Temperature (°C)	
HYDROGEN	1	-252.5	
	2	-250.2	
	5	-246.0	
	10	-241.8	
	12.8	-240.0	Critical Point

Correlation: $T\ (^\circ C) = 9.3474 \bullet [P\ (atm)]^{1/3} - 261.922$

$P\ (atm) = (0.10698 \bullet T\ (^\circ C) + 28.021)^{3}$

Propellant Type	Vapor Pressure (atm)	Temperature (°C)	
METHANE	1	-161.5	
	2	-152.3	
	5	-138.3	
	10	-124.8	
	20	-108.5	
	30	-96.3	
	40	-86.3	
	45.8	-82.1	Critical Point

Correlation: $T\ (^\circ C) = 30.587\ [P\ (atm)]^{1/3} - 191.19$

$P\ (atm) = (0.03269 \bullet T\ (^\circ C) + 6.2508)^{3}$

10.2 Gross Cryogenic Propellant Storage Scaling

Section 7.5 discusses a cryogenic storage system based on General Dynamics, 1987 work. 100,000 lbm of cryogenic oxygen and hydrogen (6:1) can be stored and protected in a tank weighing 18,938 lbm. See Table 7.5-2 for more details. The tankage/protection is therefore roughly 19% of the oxygen and hydrogen propellant mass.

10.3 Habitation Module Scaling

One Freedom Space Station type habitation module weighs on the order of 40 to 50,000 lbs (18 to 23 m tons including all interior parts). The ECLSS is sized to handle eight people and the module contains crew quarters for eight. The scaling is therefore roughly 2.6 m tons/person or 5,625 lbs/person.

10.4 Power System Mass as a Function of Power Required

Table 8.4-2 provides a weight statement for the power system. For the overall power system, including solar arrays, batteries, associated structures and mechanisms, thermal control, data management, and EVA systems the mass per kw is 22,936 kgms/75 kw = 306 kgms/kw, or 761 lbs/kw.

For the solar array, mast, canister, batteries, and associated electrical equipment only, the scaling is 4,013 kgms/18.75 kw = 214 kgms/kw, or 471 lbs/kw.

10.5 Thermal Control System Mass as a Function of Heat Rejection Required

The thermal control system weight statement, Table 8.6-1, indicates 5,350 kg of equipment total (the whole system) is required to collect, transfer, and radiate 75 kw of heat. The scaling is therefore roughly 5,350 kg/75 kw = 71 kgms/kw.

11.0 Conclusions and Recommendations

The following conclusions and recommendations resulted from this study.

1. Partially reusable or totally expended systems will change the nature of the STN considerably. These should be examined. This study assumes a completely reusable trans-lunar transportation system; a reusable, space maintainable OTV and lander. The design of the STN is driven by the poorly defined requirements for servicing these vehicles. The true feasibility of space-based reusable vehicles must be established to bring these concepts closer to reality. At present, there are no fully reusable space transportation systems and the true operational feasibility of similar reusable systems on Earth has only been established after long experience with vehicles in service.

2. The hangar micrometeoroid and orbital debris protection requirements can result in major weight additions. These need to be determined in detail. The requirement to protect the OTVs and landers, drive the hangar requirement and must therefore also be determined.

3. Technology required for effective space-basing and maintenance of OTVs and landers should be identified. Technology areas requiring work identified in this report include:

    a. Micro-g cryogenic storage and transfer
    b. Space maintainable propulsion, main engines and RCS
    c. Quick connect/No EVA cryogenic fill/drain lines
    d. Space maintainable, removable aerobrakes
    e. Easily deployable fabric hangar walls

4. Experiments currently planned in cryogenic storage and transfer (see Appendix A) are required precursors to the design and operation of an STN for reusable vehicle servicing as envisioned in this report.

5. Attitude and rate constraints must be defined for this type facility to begin control system design.

6. A serious thermal analysis of the interior hangar environment is needed to simply understand the thermal aspect of the requirements for hangar walls, and particularly the bottom, Earth-facing wall in the hangar.

7. A new RMS translation concept, such as rails or tracks may be needed for the STN. The current Freedom Station concept may prove too slow and awkward for the many uses proposed in the STN.

12.0 Lunar Orbit Transportation Node Space Station

Some scenarios have called for a transportation node space station in lunar obit. The need for such a facility is scenario dependent and is generally felt to be a requirement for a far-term, second generation lunar base which is permanently occupied and involves a lunar surface based and maintained lander. This section investigates the advantages and disadvantages of a lunar orbit space station in more detail.

12.1 Advantages and Disadvantages Summary

Advantages of a lunar orbit transportation node:

1. Launch Flexibility - For an equatorial or L2 node, the OTV and lander schedules and payloads can be somewhat decoupled. The OTV and lander can both carry the optimum (maximum) payloads they are designed for on each mission. The LLO station is a storage location to hold these payloads. The equatorial plane node assumes the base is also on the equator or at a latitude (<10 to 15°) such that the equatorial plane can be reached with a small plane change that does not unreasonably penalize the lander, therefore a lander can launch to or from the base at any time. The L2 node also can be launched to at any time from a lunar surface base at any location but the higher delta V needed to get to it may require a two stage launcher.

2. Safety - A node in any orbit will serve as a safe haven for vehicles that have had failures. It may even allow some decrease in redundancy on the lander or OTV.

3. Lunar Oxygen Mass Payback - For lunar oxygen utilization, the OTV must carry the payload it is designed for to achieve maximum efficiency. By decoupling lander and OTV payloads, the node can increase the LEO mass gains associated with oxygen propellant production. In some scenarios, maximum utilization of the OTV and lander is a requirement to get reasonable returns from a lunar oxygen plant and the LLO node is therefore mandatory.

4. OTV Stay Time - For missions longer than a certain stay time (180 days?) the OTV will either have to carry enough consumables and boil-off propellant to allow it to stay in orbit that long or return to the LEO Space Station and then return to lunar orbit at a later time to pick up the lander and/or lander crew. A lunar orbit node might provide the OTV enough solar power to reliquify boil-off and shields to protect its tanks from meteoroid strikes. This might allow it to stay in lunar orbit rather than returning to the Space Station and then coming back, thus saving one outbound load of propellant. On the other hand, the lunar orbit station must be provided with RCS propellant and other consumables and will have some maintenance requirements. The flights required to put it in place must also be considered. There may or may not be a net gain.

Disadvantages of a Low Lunar Orbit Transportation Node:

1. Launch Windows - For equatorial or L1 stations there is no problem, launch windows for departure from and arrival to a 28.5 deg. LEO station occur roughly every 9 days. This 9 day window is controlled by the interval at which the Moon comes into the LEO STN plane and is therefore independent of lunar orbit arrangements. For higher inclination lunar orbits, the arriving OTV may have to insert into the orbit of the lunar station, which may only be accessible in an optimum mode once a month, if the lunar station orbit and LEO space station orbit are properly synchronized. Figure 12.3-1 plots total delta V from LEO to LLO as a function of the longitude of the ascending node of the lunar orbit for several higher inclination lunar orbits. As the inclination of the LLO increases, the optimum arrival becomes more important, thus a high inclination LLO node may reduce the monthly opportunities for arrival to one instead of three. See section 12.3 for a more detailed discussion of this problem.

2. A similar problem occurs on the return from the Moon. The lander must wait until the LLO station orbit plane contains the base to launch and then the OTV must wait until the LLO station orbit plane is properly oriented with respect to Earth, and the LEO Space Station node orbit is also properly oriented to receive the OTV to launch. This may again be a once per month occurrence for high latitude bases, if the LEO and LLO Stations are properly synchronized. Figure 12.3-2 plots total delta V from LLO to LEO as a function of lunar departure orbit longitude of the ascending node for several lunar orbit inclinations.

3. The requirement to deliver, assembly, maintain, and supply with consumables an LLO node is a significant disadvantage. An LLO node could be anything from a little truss work with an attitude control system and power to a large manned facility similar to the STN described for LEO in this report. Depending on the nature of the LLO node anywhere from 1/3 to four dedicated missions/year may be required for resupply and maintenance. The benefits, such as in the lunar oxygen scenario, must be weighed against this upkeep cost.

12.2 Lunar Orbit Node Location

A variety of locations have been proposed for a lunar transportation node including low and high lunar orbit from equatorial to higher inclinations, and the Earth-Moon libration points, L1, L2, L4, and L5.

12.2.1 Low Versus High Lunar Orbits

As shown in Eagle (March 30, 1988), page 33 and 34, for OTV/lander transportation systems such as discussed in this report, both lander mass and Earth departure stack mass increase as lunar orbit altitude increases. The increase is not great until altitudes of 1,000 km are reached, but the lower lunar orbits show a definite advantage. The lower limit may be on the order of 100 km. At this point the orbit tends toward instability and may impact the Moon within a few months. Early Apollo work found a lower limit of roughly 50 nm (93 km) related to abort concerns for short stay times.

12.2.2 Lunar Orbit Inclination

Figures 12.3-1 and 12.3-2 show total delta V inbound to the Moon and returning from it to Earth for a variety of lunar orbit inclinations. From a launch window viewpoint, the most desirable inclinations are those that result in a fairly constant total delta V over the range of possible ascending node longitudes for the lunar orbit. Based on this, the lower inclinations, roughly 20° and less are optimum, with 0° being the best. The lower inclinations do however limit base latitude to the inclination value. The higher lunar latitudes are then not accessible via a direct landing, which may be serious disadvantage for long term lunar exploration.

12.2.3 L2 Libration Point (Between the Earth and Moon)

The L2 point remains fixed relative to the Earth and Moon and is therefore accessible from the lunar surface at any time. On the other hand it requires an additional approx. 0.7 km/sec to/from LLO and therefore increases the size of the lander significantly without reducing the Earth-L2 delta V much. This may increase the LEO stack mass by as little as 30% to several hundred % (see Eagle, March 30, 1988, p. 34), depending on the exact mission circumstances.

12.2.4 Other Libration Points, L1, L4 and L5

The L1 point, on the far side of the Moon, is expected to show the same characteristics as discussed above for the L2 point, though it requires more study.

The L4 and L5 points are also expected to have the same delta V characteristics as L1 and L3, and in addition require long flight times for transfers from the points to the Moon. This also requires more study however.

12.3 Launch Windows for Low Inclination LLOs

As discussed above, the node locations without significant disadvantages from an orbital mechanics/window standpoint are the low inclination (<20°), low altitude (<500 km) orbits. The following tables and figures quantify the limits of this advantage.

Table 12.3-1 plots LEO departure stack mass as a function of specific delta Vs for a transportation system consisting of a large single-stage, aerobraked OTV and a single-stage reusable lander. This table, used with Figures 12.3-1 and 12.3-2 gives an idea of how launch window for higher inclinations can be paid for by an increase in stack mass.

Figure 12.3-1 and 12.3-2 show how total delta V (inbound and outbound) varies as the longitude of the ascending node of the lunar orbit varies. Table 12.3-2 discusses the assumptions and approximations used in these plots. Figure 12.3-3 illustrates the geometry of the situation.

The longitude of the ascending node of the lunar orbit is defined as the angle between the line of intersection of the local lunar orbit plane and the plane of the Moon's orbit about the Earth and the Earth-Moon line. A longitude of 0° would place the intersection of the two planes at the Earth-Moon line.

A major objection to lunar orbit space stations is that they will seriously reduce available launch windows for high latitude lunar bases. The Moon is only in the plane of the LEO Space Station's orbit roughly three times per month so this is the maximum number of arrival or departure opportunities available. Lunar equatorial orbit is accessible at any of these three times, but as lunar orbit inclination approaches 90°, a given orbit can only be entered inexpensively twice a month and will therefore only match with the Space Station plane once a month, if it is synchronized.

As lunar orbit inclination decreases, as can be seen from looking at Table 12.3-1 and Figures 12.3-1 and 12.3-2, the lunar orbits become accessible at a wider range of ascending nodes. If a penalty of 15% growth in the LEO stack mass can be paid, arrival Total Delta V could be as high as 4.4 km/sec or lunar departure delta Vs could be as high as 4.76 km/sec. For inclinations of 10° and below, this allows arrival in LLO at any of the three opportunities.

For inclinations of roughly 25° and below, departure from LLO at any of the three opportunities is possible for a 15% LEO stack mass penalty.

The 9 day interval is roughly 1/3 of the lunar month, and a lunar orbit changes longitude of the ascending node roughly 120° in this interval. To be able to use two arrival/departure opportunities per month requires the delta Vs be within reason at a 120° interval. Unfortunately, the natural interval for the lunar orbits is 180°. Some slight increase in allowable inclination for a 15% LEO stack mass penalty may be possible, if only two opportunities rather than three are desired, but careful examination of the plots indicates it will not be much.

Figure 12.3-4 is a plot of a variety of trajectories at a single inclination from which a single line on Figure 12.3-1 was generated. The line taken from Figure 12.3-4 is the line resulting in the minimum total delta V.

Table **12.3-1,** LEO Stack Mass Versus Total Delta V

Payload to lunar orbit = 48 m tons (Manned lunar lander)
Payload returned to Earth = 16 m tons (Crew capsule + lander inert)
Percent of entry mass that is aerobrake = 15%, OTV Isp = 455 sec.

Single-Stage Aerobraked OTV does TLI, SOI, LOI, TEI, SOI2 and other midcourse and perigee raise burns. Inbound and outbound trajectories assume three burns; TLI and LOI or TEI and EOI, as well as inbound and outbound burns at the sphere of influence (SOI).

LEO Stack Mass Versus TLI + SOI + LOI Total Delta V
(TEI delta V held at .846 km/sec, TLI held at 3.1 km/sec)

LEO Stack Mass metric tons	% Increase	Total Delta V, km/sec (TLI + SOI + LOI)
189	0	3.95
192	1.5	4.0
196	3.7	4.1
207	10	4.2
212	12	4.3
217	15	4.4
222	17	4.5
232	23	4.6
237	25	4.7
243	29	4.8
255	35	4.9
261	38	5.0

LEO Stack Mass Versus TEI + SOI Delta V
(TLI held at 3.1 km/sec, LOI at .846 km/sec)

LEO Stack Mass metric tons	% Increase	TEI + SOI Delta V km/sec	TEI + SOI + EOI* Delta V km/sec
189	0	.846	4.0
191	1	.9	4.06
193	2	1.0	4.16
196	4	1.1	4.26
199	5	1.2	4.36
211	12	1.4	4.56
217	15	1.6	4.76
225	19	1.8	4.96
236	25	2.0	5.16
244	29	2.2	5.36
260	38	2.4	5.56
269	42	2.6	5.76

* For aerobraked vehicles there is no Earth Orbit Insertion burn, only a perigee raise, but this column includes one (fixed at 3.16 km/sec) to make it easy to use with Figure 12.3-2 which assumes an EOI burn of approx. this magnitude.

Table **12.3-2,** Assumptions Used in Figures 12.3-1, 12.3-2, and 12.3-4

1. Perigee altitude of Earth departure or return orbit = 250 nautical miles (463 km).

2. Circular orbit altitude of lunar departure or arrival orbit = 60 nautical miles (111 km).

3. Angle between the Earth's departure or arrival orbit plane and plane of the Moon's orbit around the Earth = 30°. This angle is a function of the date. The lunar orbit inclination relative to the Earth's equator varies from 18° to 28° over an 18.5 year period. It is also a function of the launch date and regression rate of the Space Station. 30° was chosen as an average number. The effects of varying this number through its range are expected to be minimal, though a few runs are needed to confirm this.

4. The lunar orbit inclination shown in the figure is really the angle between the local lunar orbit plane and the plane of the Moon's orbit around the Earth. The rotation axis of the Moon is only inclined 1.6 degrees relative to a perpendicular to the plane of the Moon's orbit around the Earth, therefore this is a good approximation.

5. Flight time - fixed at 60 hours.

6. The trajectories in the figures assumed three burns inbound and outbound. Free return was not sought and minimum delta Vs and maximum flexibility was felt to be found in the three burn scheme. The burns are:

   Outbound from Earth
   Trans-lunar Injection (TLI)
   Sphere of Influence (SOI)
   Lunar Orbit Insertion (LOI)

   Return to Earth
   Trans-Earth Injection (TEI)
   Sphere of Influence (SOI)
   Earth Orbit Insertion* (EOI)

   The sphere of influence is the surface in space where the Earth and Moon's gravitation are approx. equal. It is the location of the third non-Apollo type burns.

   * The EOI burn is not needed for an aerobraked vehicle. The EOI delta V is relatively constant for all the trajectories plotted in Figure 12.3-2 and can be removed by subtracting 3.16 km/sec from the total delta V.

Figure 12.3-1, Total Delta V, Earth to Moon versus LLO Longitude of Ascending Node

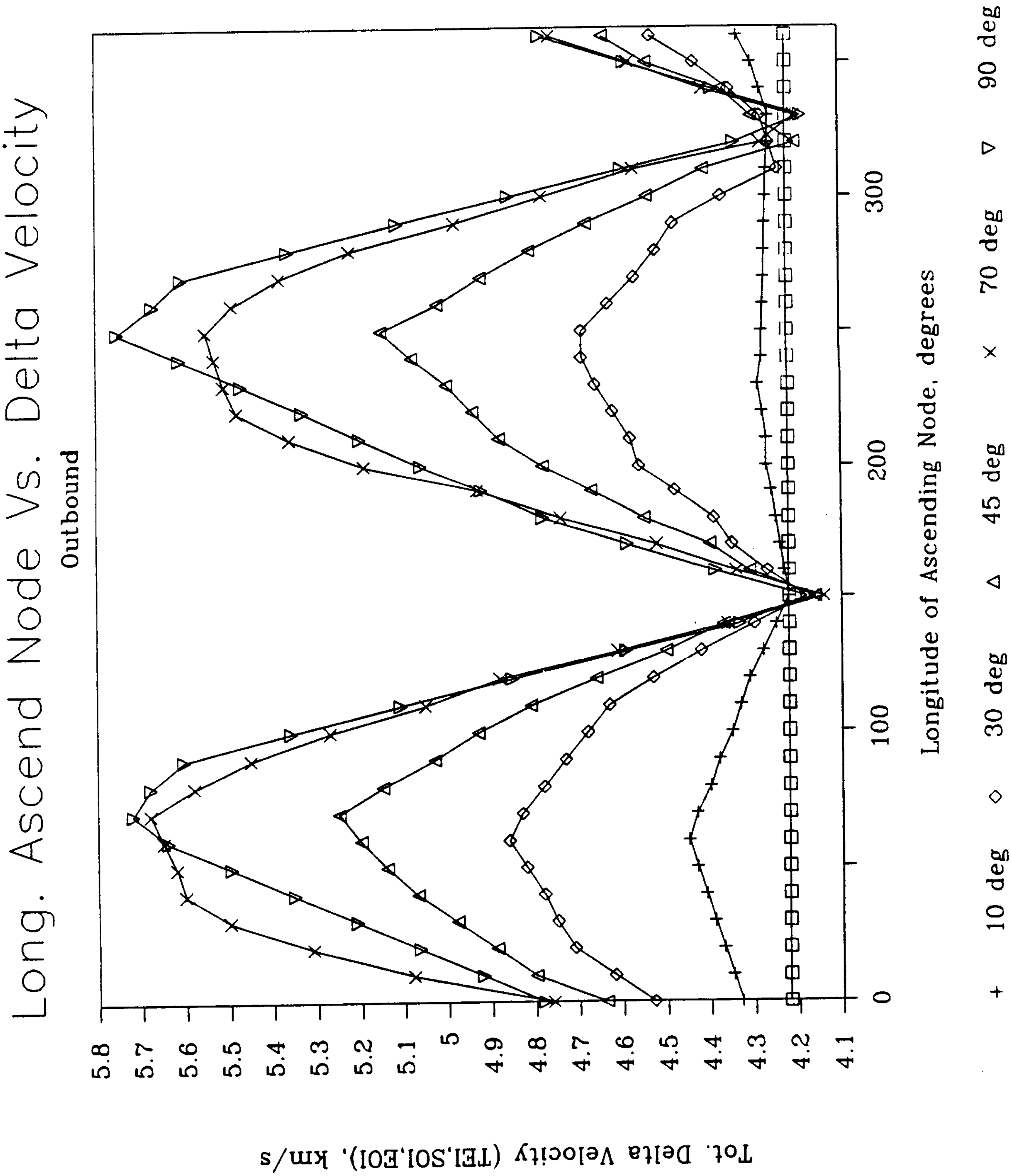


Figure 12.3-2, Total Delta V, Moon to Earth versus LLO Longitude of Ascending Node (for multiple LLO inclinations)

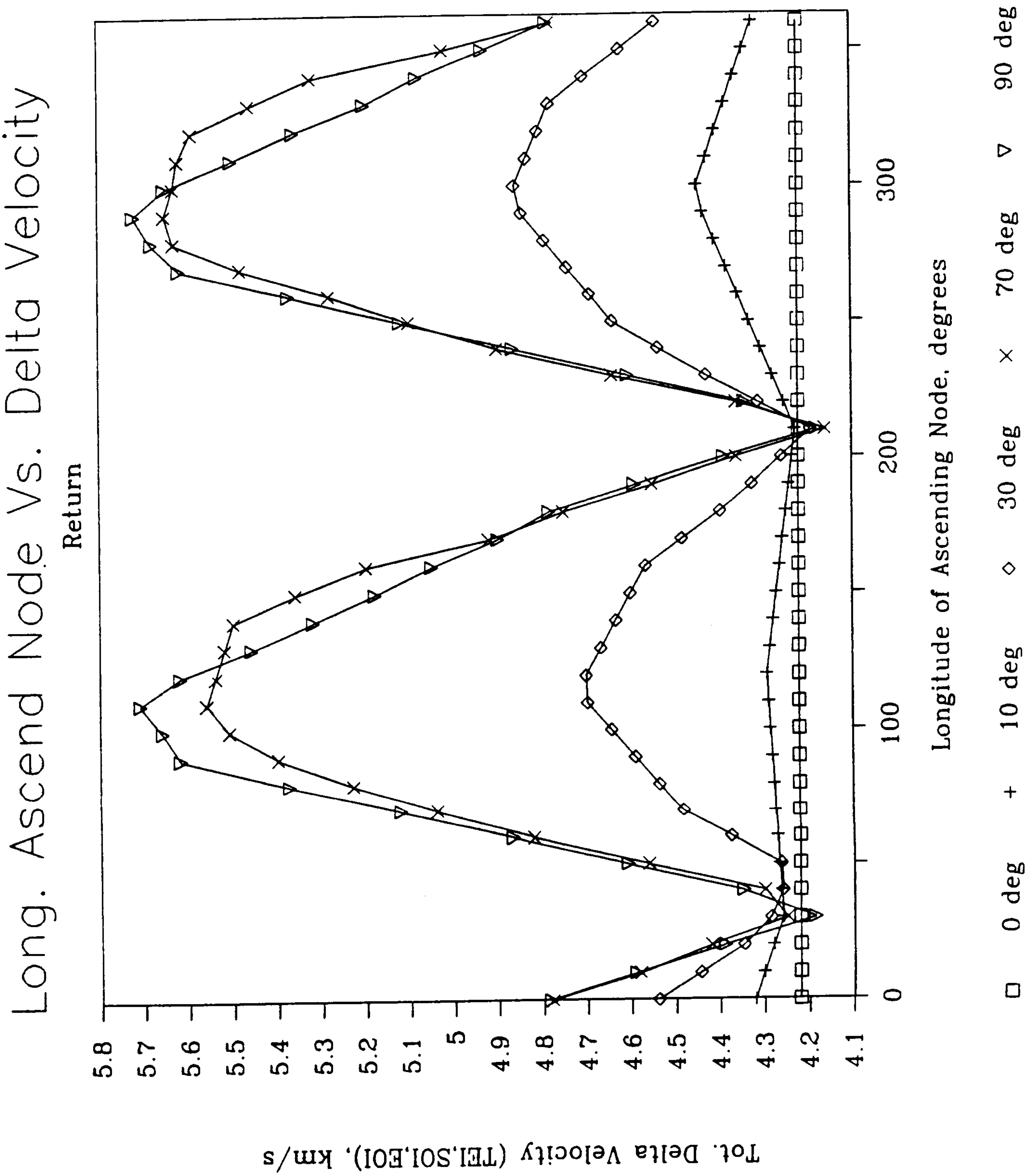


Figure 12.3-3, Earth-Moon Geometry

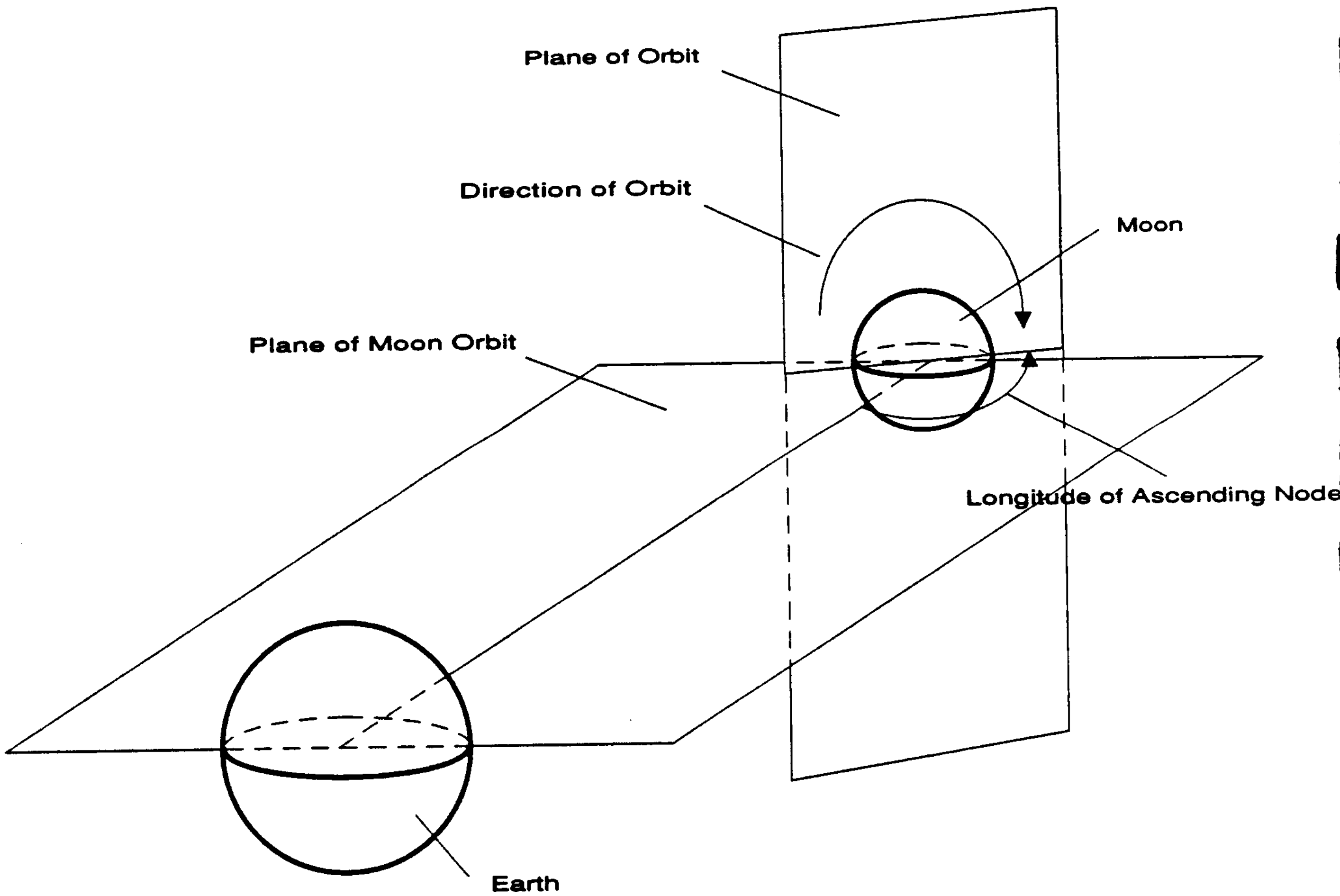
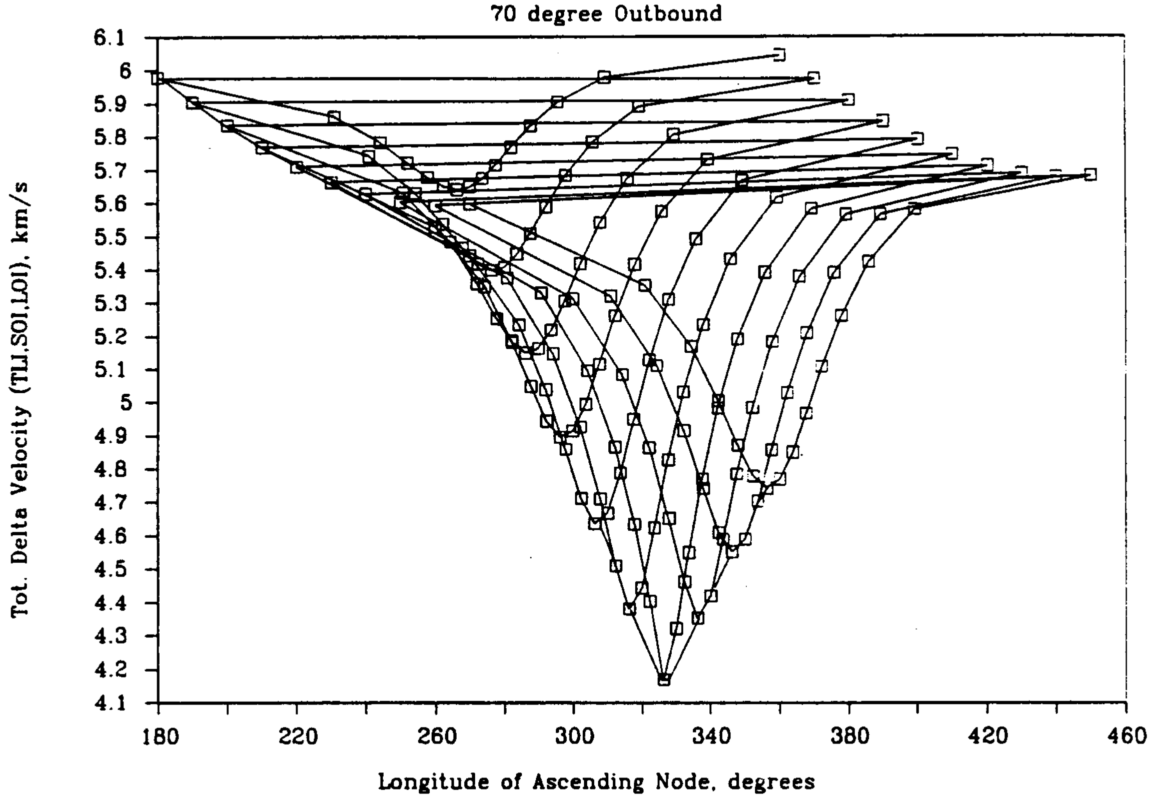
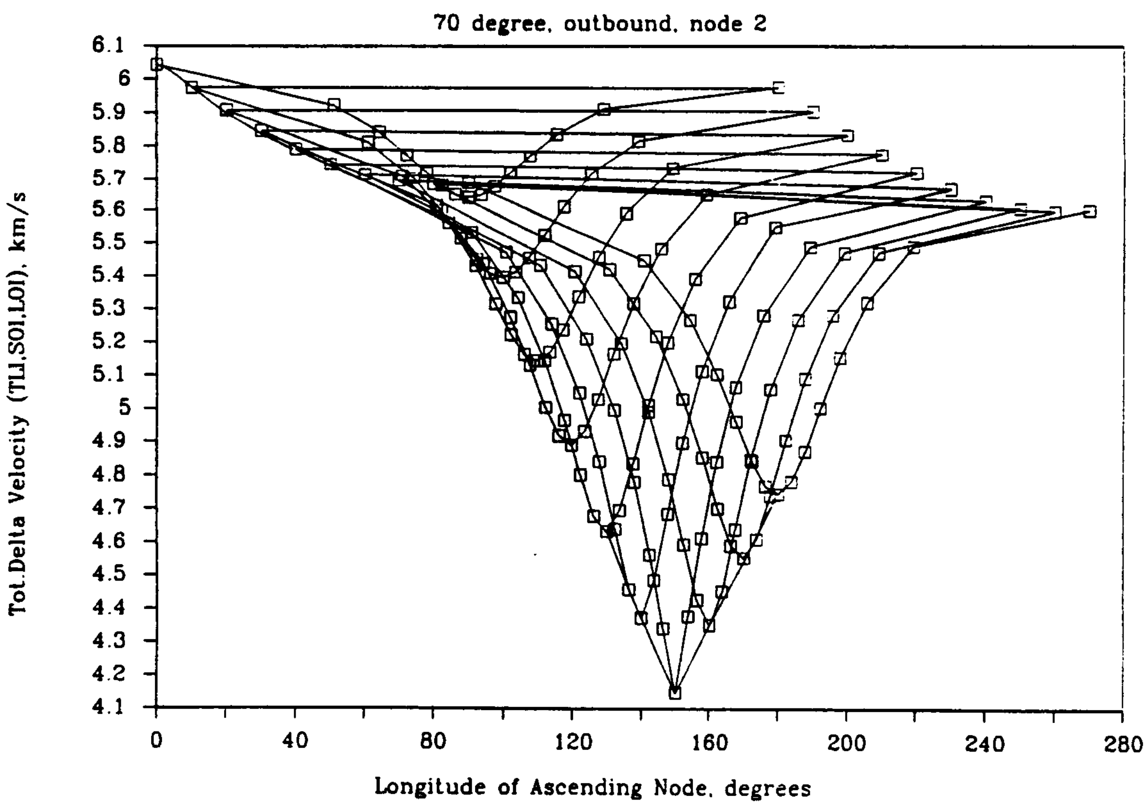


Figure 12.3-4, Total Delta V, Earth to Moon Versus LLO Longitude of Ascending Node for 70° Inclination, Node 1 and Node 2 Plots





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APPENDIX A

Transfer of Storable and Cryogenic Propellants in Micro-G

Appendix A Transfer of Cryogenic and Storable Propellants in Micro-G

1.0 Task Definition

This subtask was performed in support of the Conceptual Design of a LEO Node activity. Its objectives were as follows:

1. Acquire copies of all recent reports and papers that deal with the problem of the transfer of cryogenic and storable propellants in micro-g.

2. Describe all past and proposed propellant transfer experiments. Describe the techniques used.

3. List names and telephone numbers of all project managers/technical monitors working on propellant transfer experiments.

4. Describe the leading proposed techniques of transfer of cryogenic and storable propellants in micro-g.

2.0 Reports and Papers

A review of the material in the Eagle library and a Recon search of the literature data base was performed to identify appropriate papers and reports on the subject. In addition, contacts were made with appropriate individuals in NASA to determine current and past activities on the issues of propellant transfer and storage. Copies of reports were obtained from these individuals where available. A listing of the documents obtained is found in section 6.0.

3.0 Propellant Transfer Experiments and Studies

The number of definitive propellant transfer experiments in micro-g is low. Specific experiments that have been conducted are the Storable Fluid Management Demonstration and the Orbital Refueling System (ORS) experiment packages. Planned for the future are the Superfluid Helium On Orbit Transfer (SHOOT) which is a combined Goddard-Ames-JSC experiment to be flown on the Shuttle and the Cryogenic On-orbit Liquid Depot-Supply, Acquisition, & Transfer (COLD-SAT) flight experiment planned to be launched in 1996 on an expendable launch vehicle.

Studies in the areas of propellant storage, acquisition, and transfer are being pursued by many of the NASA centers. Investigations are also being performed by industry but the resources available did not allow their complete investigation. Studies sponsored by JSC included the Tethered Orbital Refueling Study and the Orbital Spacecraft Consumables Resupply System (OSCRS) Study. JPL has been conducting a series of investigation in the area of Advanced Thermal Control Technology for Cryogenic Propellant Storage. Similar investigations have been performed by the Air Force Rocket Propulsion Laboratory in a Long Term Cryo Storage Study. MSFC has performed several studies in conjunction with its OTV and OMV activities to address the issue of fluid management in orbit. Analytical modeling activities are being pursued at NASA Ames, JPL, LaRC, Goddard, MSFC, and JSC to develop thermal and fluid dynamic models of large quantities of propellants in a micro-g environment.

3.1 Storable Fluid Management Demonstration

The Storable Fluid Management Demonstration was a Shuttle mid-deck experiment which was flown on STS 51-C to investigate micro-g transfer of fluids using different acquisition methods and transfer techniques. The fluid used in the experiment was water, and the primary intent of the first experiment was to demonstrate the capability to transfer fluids by using liquid acquisition devices to collect and direct the flow of fluids from the supply tank to the receiver tank. This experiment was directed towards earth storable fluids and did not address the thermal issues normally associated with cryogenic fluids. Further experiments are planned and scheduled to be flown on future Shuttle missions. The primary objectives of the experiment were as follows:

- Investigate the behavior of fluids in a micro-g environment
- Demonstrate the capability of the Liquid Acquisition Device (LAD) to control the acquisition and flow of fluids into and out of tanks
- Evaluate the effectiveness of performing an evacuated receiver tank fill
- Evaluate the effectiveness of perforated plate baffles for liquid slosh control
- Evaluate the effectiveness of gas vent separation for vented tank fill

The experiment consisted of two tanks using water as the fluid to represent fluid transfer in orbit. The original supply tank contained a diaphragm with positive expulsion provided by pressurized nitrogen. The receiver tank contained a LAD with channels and perforated screen baffles for slosh control and vent gas separation. The fluids were transferred back and forth between the two tanks to evaluate different methods of fill and drain. On the experiment with a vented receiver tank, fluid expulsion was very effective leaving approximately 2% residuals in the supply tank. When the receiver tank was vented as the liquids were induced, mixed gas and liquids quickly reached the vent and the transfer was terminated.

Sufficient tanks were built to perform four flight experiments but presently only one additional experiment is scheduled to be flown. This experiment is sponsored by the Air Force and is investigating the transfer of fluids into and out of a torridial tank.

3.2 Orbital Refueling System

The Orbital Refueling System flight experiment was flown on the STS 41-G mission to demonstrate a monopropellant transfer system which would lead to the development of a space tanker to replenish propellants and other liquids on earth-orbiting vehicles and satellites. The primary objectives of the experiment were as follows:

- Develop and demonstrate the equipment and procedures for a hydrazine fuel transfer system

- Develop and demonstrate the tools needed to interface with existing satellites to accomplish fuel transfer. (Current satellites are not designed for on-orbit refueling.)

- Develop and evaluate specific procedures to refuel present satellites of the Landsat type

The method used to transfer fuel on this experiment was positive expulsion with inert gas pressurization using tanks with elastic diaphragms. Transfer was also accomplished in a blowdown mode to compare its effectiveness to that of pressurized transfer. The material used for the diaphragm is compatible with the fuel but not with oxidizers.

3.3 Superfluid Helium On-Orbit Transfer

The SHOOT experiment is a joint flight experiment involving Goddard, Ames, and JSC to evaluate technologies and demonstrate techniques to enable transfer of superfluid helium on orbit. The experiment is scheduled to be flown on the Shuttle in the 1991 time period. The major objectives of the experiment are as follows:

- Demonstrate the technology required to transfer superfluid helium in a micro-g environment due to induced drag in low earth orbit

- Evaluate the limits of the liquid acquisition device by operation under incremental increases of induced linear acceleration

- Demonstrate the technology of transfer of superfluid helium by a thermo-mechanical pump

- Prove the feasibility of cooling and filling a "warm" receiver tank (i.e., above 20K) in space

- Demonstrate the capability of coupling technology and equipment in conjunction with EVA operations to accomplish helium transfer in earth orbit

- Demonstrate techniques for remote and/or autonomous operation of the transfer operations

This experiment plans to transfer superfluid helium using the unique properties of the fluid to aid the transfer mechanism. The thermomechanical pump consists of a porous ceramic cup and a heat source which will induce the superfluid helium to flow through the porous cup by the induced action of a heat source. Acquisition of the superfluid helium in the supply tank will be accomplished by channel devices or capillary screens. It is assumed that the micro-g environment due to induced drag effects are sufficient to provide the necessary settling action inside the supply tank. The limit of the capability of the liquid acquisition device will be evaluated in a test where induced accelerations of increasing levels will be applied by the use of RCS impulse burns.

3.4 Cryogenic On-orbit Liquid Depot-Supply, Acquisition, & Transfer

The COLD-SAT flight experiment is designed to address the technology issues related to cryogenic fluid supply, acquisition, and transfer. This experiment was originally designed to be flown on the Shuttle as the Cryogenic Fluid Management Flight Experiment (CFMFE) which was managed by Lewis Research Center. The experiment was canceled from the Shuttle for reasons of flight safety, consequently the COLD-SAT experiment was developed to address these technology issues on a flight experiment currently scheduled to be flown on an expendable vehicle in the 1995 time frame. This experiment will be augmented with a 1-G ground experiment and a sub-scale propellant depot program which will also operated in a 1-G environment. The objectives of the COLD-SAT experiment are as follows:

- Address the technology of passive thermal control system performance by evaluating the effects of launch environment on thick multilayer insulation
- Evaluate thermodynamic vent system performance and fluid mixing for stratification control
- Evaluate pressurization system performance using autogenous pressurization and helium pressurization
- Address technology issues related to fluid acquisition and conditioning by fine mesh screen liquid acquisition devices, fluid settling and outflow by impulsive acceleration, fluid settling and outflow under low gravity conditions, impact of heat addition to LAD performance, and thermal conditioning of liquid outflow
- Demonstrate liquid transfer technology related to transfer line chilldown, tank chilldown, no-vent fill, LAD fill, and low-gravity fill
- Investigate fluid handling issues of liquid dynamics/slosh control and fluid dumping/tank inerting
- Address and demonstrate technologies related to advanced instrumentation on quantity gaging, mass flow/quality metering, and leak detection
- Demonstrate technologies related to passive orbital disconnect strut (PODS) performance, composite (light weight) vacuum jacket performance, and long term space environmental effects

The primary method of fluid acquisition is by fine mesh screen LAD. Capillary channels may assist the flow of fluids to the LAD. Propellant settling is planned to be dependent on the micro-g environment due to induced aerodynamic drag in low earth orbit. Transfer of fluids will be accomplished by pressurization of the supply tank and evacuation of the receiver tank. A method of thermodynamic fill will be used to minimize the need to vent the receiver tank. With this method, the initial fluid inflow to the receiver tank is expected to prechill the tank and cause a pressure rise in the receiver tank. When the receiver tank is cooled below the condensation point of the fluid, it is anticipated that the fill process can continue until the transfer is complete. This method of transfer does require that the fluid be prechilled to maintain the temperature of the fluid in the receiver tank below the condensation point as fluid is introduced into the tank.

3.5 Orbital Spacecraft Consumables Resupply System

The OSCRS studies were intended to develop a concept and preliminary design for an earth-storable monopropellant tanker that could be flown in the Shuttle to service spacecraft in low earth orbit. This study was performed by Rockwell International, Martin Marietta, and Fairchild. Other objectives of these studies were to identify the ground support requirements to support the operational scenarios; identify design concepts for a bipropellant system design; and address the operational issues of performing a Gamma Ray Observatory (GRO) resupply mission.

Several methods of liquid acquisition in the supply tanks were investigated. These ranged from capillary sponge reservoir devices to simple surface tension devices with positive expulsion by an elastic diaphragm and pressure feed. The choices investigated for propellant transfer included ullage recompression, ullage exchange, and ullage vent methods. Propellant transfer in these cases investigated pressure fed and pump fed approaches. The pump fed approach seemed to be preferred method in two of the three studies due to versatility and saving in tank masses. The choice for handling the ullage gasses seemed to be dependent upon the spacecraft design for monopropellant resupply while ullage vent or exchange methods were preferred for larger bipropellant systems.

3.6 Tethered Orbital Refueling Study

This study was done by Martin Marietta for NASA - JSC to evaluate the feasibility of fluid acquisition and transfer under an acceleration induced in a tethered orbital refueling facility. A conceptual design for such a facility was also provided as a product of this study. In this study, large masses of propellants (100,000 lbm of cryogenics and 10,000 lbm of storables) were investigated. Transfer methods investigated included pressure, pump and gravity feed. The study concluded that it was feasible to settle the propellants in the micro-g environment induced but the operational implementation of the may have some difficulties. The transfer of cryogenic propellants selected the method of autogenous pressurized transfer while the earth-storable propellant transfer selected pumped transfer as the best option.

3.7 Long Term Cryo Storage Study

This study was done for the Air Force Rocket Propulsion Laboratory by Martin Marietta. Its primary objective was to identify and plan the technology improvements necessary to enable large quantities of cryogenic fluids to be stored in space for periods up to seven years. Even though most of the effort in this study was directed to the thermal management issues, the needs for fluid management technologies were addressed.

4.0 Propellant Transfer Technology Investigators and Organizations

This section included a list of individuals directly involved with activities and technologies related to micro-g fluid acquisition and transfer.

Organization	Name	Telephone No.	Involvement
JSC	John Griffin/EP4	713-483-9003	Section Head/Vehicle Propulsion and Fluids
	Bill Boyd/EP43	713-483-9020	Technical & Study/ORS/OSCRS/ Superfluid Helium Tanker
	Kenneth Kroll/EP42	713-483-9011	Technical & Study/OSCRS/ Tethered Orbital Refueling Study
	Nancy Munoz/EP43	713-483-9015	Technical & Study/OSCRS
	Gordon Rysavy/EX2	713-483-3269	Flight Project Mgr./ORS
Ames	Bob Lavond	415-694-6521	Technical Mgr./SHOOT
	Peter Kittel	415-694-4297	Systems Engr./SHOOT
	Sanford Davis	415-694-2601	Microgravity Fluid Model
	Walt Brook	415-694-6547	Principal Investigator/SHOOT
AFRPL Storage	Roy Silver	805-277-5651	Study Mgr./Long Term Cryo
Goddard	Orlando Figueroa	301-286-7327	Experiment Mgr./SHOOT
	Michael DiPillo	301-286-8568	Principal Investigator/SHOOT
JPL	David Elliott	818-354-3486	Technical Mgr./Advanced Thermal Control Technology for Cryo Propellant Storage
LaRC	Eugene Symons	216-433-2853	Project Mgr./Cryo Fluid Management Project Office
	John Aydelott	216-433-2472	Fluid Modeling/Cold SAT
	Erich Kroeger	216-433-2843	System Engr./Cold-SAT
MSFC	Norman Brown	205-544-0505	Advanced Projects/Prelim. Design
	John Cramer	205-544-7090	OTV Studies/Cryo Storage
	Bob Durette	205-544-0628	Technical/Propellant Storage and Transfer

5.0 Fluid Acquisition and Transfer

The problem of fluid acquisition and transfer in a micro-g environment is dependent on several factors. These factors include the nature of the fluid, the quantity of the fluid, and the micro-gravity environment. Different transfer techniques are possible with different acquisition techniques, so these issues will be addressed separately. Tables 5.0-1 and 5.0-2 list the advantages and disadvantages of each acquisition and transfer method.

5.1 Fluid Acquisition

The technology of fluid acquisition in a micro-g environment is affected by the nature of the fluid and the size of the container. Generally speaking the methods of fluid acquisition can be divided into those associated with positive expulsion devices and those associated with a free liquid surface. For the most part, small quantities of earth-storable propellants have been handled with positive expulsion devices in the past. This was done to avoid the problem of fluid acquisition problems in a micro-g environment. Most of these techniques are not effective with cryogenic propellants. The methods of positive expulsion are as follows:

- Bladders
- Pistons
- Bellows
- Diaphragms

Currently the preferred technique is to use elastic diaphragms with expulsion with a pressurized gas. This method seems to provide the best performance in terms of expulsion of residuals in a spherical tank. The current problems with this method are those of finding a material compatible with the oxidizer for bipropellant systems, limitations of tankage shapes, and limitations with tankage size.

With a free liquid surface, for earth storable and cryogenic propellants, the liquid acquisition devices used are as follows:

- Fine Mesh Screen LAD
- Capillary Channels
- Screened Channels
- Screened Vanes or Baffles

All of the above designs seem to be effective but the choice of device is more dependent on the nature of the fluid and the amount of acceleration effects available to induce fluid settling to keep the propellants in the vicinity of the LAD. As the acceleration levels increase, the amount of channels and screens required inside the tank are reduced. Superfluid helium is a special case which exhibits a unique property of non-viscous flow. For superfluid helium, the use of capillary or screened channels seems to be very effective in directing fluids towards the inlet.

Studies have indicated that the micro-g environment due to a tethered gravity gradient propellant depot or that caused by induced aerodynamic drag on large space structures in low earth orbit may be sufficient to allow settling and supply of the propellants to the LAD located at the supply tank outlet. These issues will be addressed in the flight experiments of SHOOT and COLD-SAT.

5.2 Fluid Transfer

Fluid transfer of propellants in space should be discussed for several different cases. These cases are the transfer of Earth storable propellants, transfer of cryogenic propellants, and the transfer of superfluid helium. The transfer options for Earth storable propellants fall into the following classes:

- Pressure Fed Transfer
- Pumped Transfer
- Gravity or Induced Acceleration Transfer

For small simple monopropellant systems, the choice of a pressurized transfer seems to be the best choice. For larger systems and bipropellants, the preference is pumped for transfer systems. Gravity or induced acceleration transfer methods presently have operational considerations which limit their application.

For cryogenic propellants, similar methods of transfer as cited above apply. The main difference is the thermal and pressure constraints that apply to cryogenic fluids. Most cryogenics have very low vaporization temperatures and experience sharp pressure rises if that temperature is exceeded. Because of these properties special consideration must be made for the ullage gasses during propellant transfer. With cryogenic propellants, several variations to the supply transfer methods mentioned above are suggested. These methods include the following:

- Helium Pressurization and Recovery
- Autogenous Pressurization
- Thermal Subcooler
- Jet Pump

These methods are illustrated in Figure 5.2-1. The technology of autogenous pressurization has been applied successfully to cryogenic fueled launch vehicles and represents a low mass penalty method of propellant transfer. For long term storage of cryogenics, the addition of thermal energy to the cryogenics may not be desired, so the thermal subcooler or jet pump becomes more desirable.

The problem of venting the receiver tank poses a special problem in space as illustrated in the Storable Fluid Management Demonstration flight experiment. Effective separation of gasses and liquids in the receiver tank during fill becomes a difficult problem in a micro-g environment. For all cases of tank fill of cryogenic propellants, the preferred method is thermodynamic fill where the receiver tank is initially evacuated and the propellant is fed into the receiver tank as a liquid which evaporates to cool the receiver tank. As soon as the temperature drops below the condensation temperature of the liquid, the fill process can be continued until the receiver tank is filled.

The preferred method of transfer of superfluid helium is by a thermomechanical pump described earlier and fill of the receiver tank is accomplished by the thermodynamic fill method.

The determination of the level of micro-g required to settle free surface liquids in large tanks for both supply and receiving will probably be answered after the previously mentioned flight experiments are flown to address the issue.

Table 5.0-1, Fluid Acquisition Methods Advantages and Disadvantages

Positive expulsion devices have primarily been applied to Earth storable fluids of relatively small volumes. The nature of the devices do not incline them to cryogenic applications.

Method	Advantages	Disadvantages
Bladders	Fuel and oxidizer compatible	Moderate residuals, corner fold problems, limited cycle life
Pistons	Low residuals, fuel and oxidizer compatible	Seal design, leakage, sensitivity to dimensional tolerances
Bellows	Fuel and oxidizer compatible, high reuse life	Large total tank volume, cylindrical configurations
Diaphragms	Low residuals	Elastic diaphragm materials are not oxidizer compatible, tank shape sensitive

With a free liquid surface, large volumes of liquids can be accommodated without any active or flexible displacement materials. This is especially attractive for cryogenic fluids.

Method	Advantages	Disadvantages
Capillary Channels	High head capability	Multiple channels needed to cover tank interior, not effective for high flow rates
Screened Channels	High flow in micro-g	Gas entrapment between screen sections
Screened Vanes or Baffles	Very high flow rate capability	Sensitive to acceleration disturbances, low pressure head capability
Local Screen LAD	Liquid retention in area of tank outlet	Requires acceleration to induce propellant settling

Table 5.0-2, Fluid Transfer Methods Advantages and Disadvantages

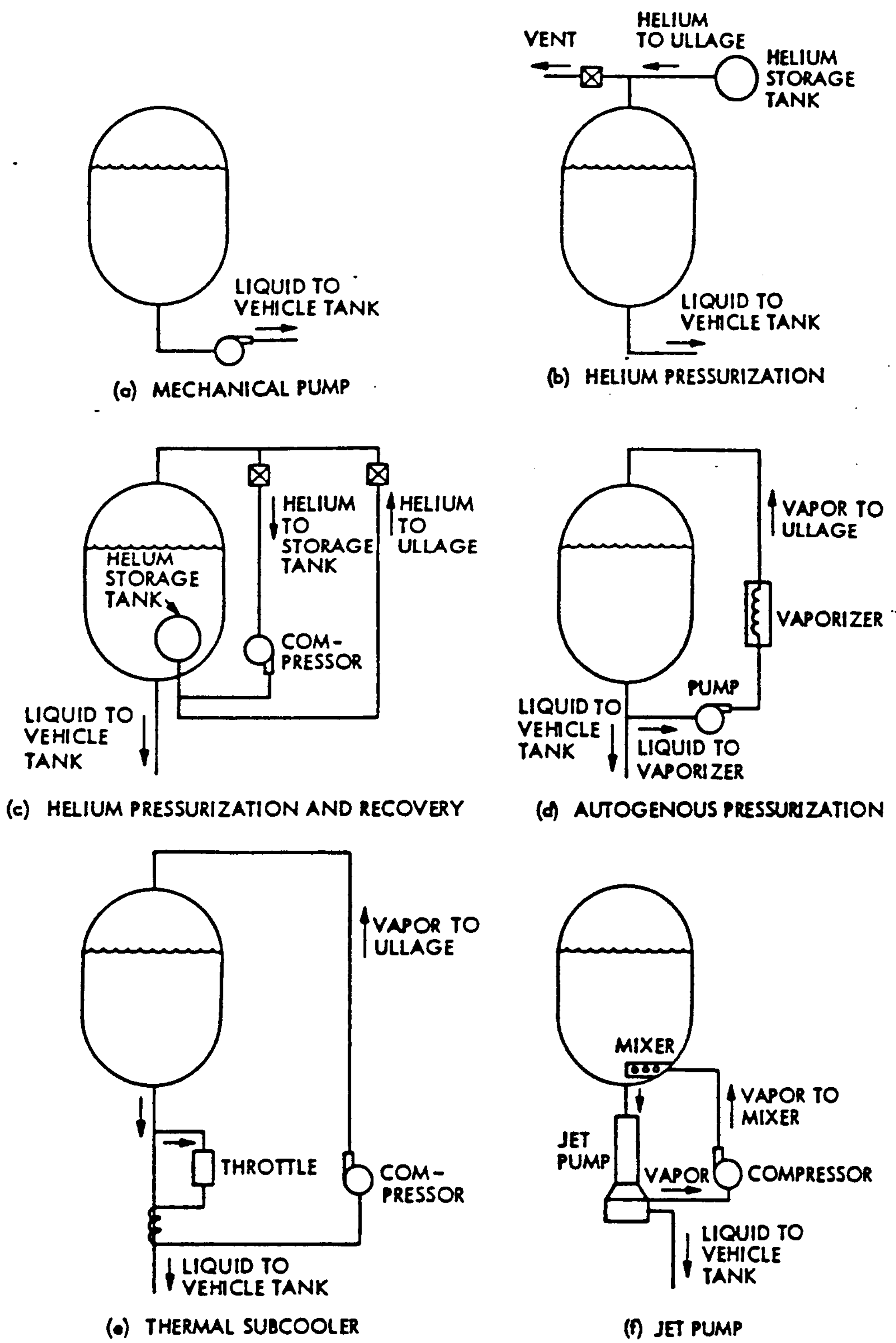
The transfer of propellants in space usually falls into one of three categories, pressure fed, pump fed, or acceleration induced transfer. These methods apply to both earth storables and cryogenic propellants.

Method	Advantages	Disadvantages
Pressure Fed	Simple system, low equipment mass	Requires pressurization gas, tank walls must withstand pressure levels required to induce transfer
Pump Fed	Lighter supply tank, downstream pressures can be high, high flow rates available	Complexity and mass of pump, energy source required for pump
Gravity or Induced Acceleration	Simple tankage and plumbing, low tank mass	Operationally complex, low flow rates available for drag induced or gravity gradient acceleration levels

For cryogenic propellant transfer, minor variations to the methods mentioned above apply. These methods take into consideration the properties of cryogenic fluids and their effect on long term propellant storage.

Method	Advantages	Disadvantages
Helium Pressurization and Recovery	Minimum loss of helium during refill cycle	System complexity, external energy source for compressor
Autogenous Pressurization	Low system mass, utilizes liquid vapors for pressurization	Heat input to cryogenics, requires liquid pump and heat input, may require initial helium pressurization
Thermal Subcooler	Subcools liquid to tanks, vapor used to pressurize supply tank	Compressor and energy source required for pressurization
Jet Pump	Low heat gain, high pressure head available	External power required

Figure 5.2-1, Method of Transferring Propellant From a Depot Tank



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