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Space Transfer Concepts and Analysis for Exploration Missions

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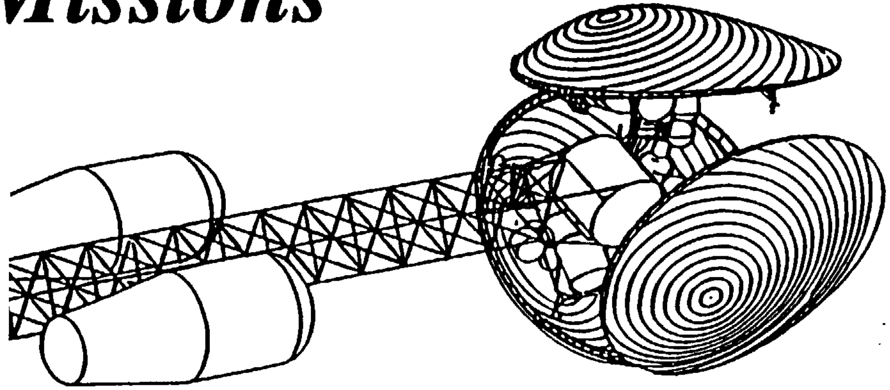

Implementation Plan and Element Description Document
Volume 1: Major Trades Book 1 (draft Final)

February 15, 1991

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ADVANCED CIVIL SPACE SYSTEMS

Boeing Aerospace and Electronics
Huntsville, Alabama
NASA Contract NAS8-37857

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Space Transfer Concepts and Analyses for Exploration Missions

NASA Contract NAS8-37857

Major Trade Studies

Boeing Aerospace and Electronics
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Documentation Set:
D615-10026-1 IP and ED Volume 1: Major Trades, Books 1 and 2
D615-10026-2 IP and ED Volume 2: Cryogenic/ Aerobrake Vehicle
D615-10026-3 IP and ED Volume 3: Nuclear Thermal Rocket Vehicle
D615-10026-4 IP and ED Volume 4: Solar Electric Propulsion Vehicle
D615-10026-5 IP and ED Volume 5: Nuclear Electric Propulsion Vehicle
D615-10026-6 IP and ED Volume 6: Lunar Systems

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Implementation Plan and Element Description Document Major Trades Table of Contents

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Symbols, Abbreviations and Acronyms

ACRV	Advanced crew recovery vehicle
ACS	Attitude control system
AFE	Aerobrake Flight Experiment
A&I	Attachment and integration
Al	Aluminum
ALARA	As low as reasonably achievable
ALS	Advanced Launch System
ALSPE	Anomalously large solar proton event
am	Atomic mass (unit)
AR	Area ratio
ARGPER	Argument of perigee
ARS	Atmospheric revitalization system
art-g	Artificial gravity
asc	Ascent
ASE	Advanced space engine
AU	Astronomical Unit (=149.6 million km)

BIT	Built-in test
BITE	Built-in test equipment
BLAP	Boundary Layer Analysis Program
BFO	Blood-forming organs
BMR	Body mounted radiator

C	Degrees Celsius
CAB	Cryogenic/aerobrake
CAD/CAM	Compter-aided design/computer-aided manufacturing
CAP	Cryogenic all-propulsive
C_d	Drag coefficient
CELSS	Closed Environmental Life Support System
CHC	Crew health care
CG	Center of gravity
C_L	Lift coefficient
cm	Centimeter = 0.01 meter
c/m	Crew module
CM	Center of mass
c/o	Check out
C of F	Cost of facilities
conj	Conjunction
COSPAR	Committee on Space Research of the International Council of Scientific Unions
CO2	Carbon dioxide
Cryo	Cryogenic
C3	Hyperbolic excess velocity squared (in km^2/s^2)
C&T	Communications and Telemetry
CTV	Cargo Transport Vehicle (operates in Earth orbit)

d	days
DDT&E	Design, development, testing, and evaluation
DE	Dose equivalent
deg	Degrees
desc	Descent

DMS	Data management system
dV	Velocity change (ΔV)
EA	Earth arrival
E arr	Earth arrival
Ec	Modulus of elasticity in compression
ECCV	Earth crew capture vehicle
ECWS	Element control work station
ECLSS	Environment control and life support system
EP	Electric propulsion
ESA	European Space Agency
e.s.o.	Engine start opportunity
ET	External Tank
ETO	Earth-to-orbit
EVA	Extra-vehicular activity
F_c	Circulation efficiency factor
FD&D	Fire Detection and Differentiation
F_{ew}	Life support weight factor
FEL	First element launch
F_f	Specific floor count factor
F_{fa}	Specific floor area factor
F_i	Aerobrake integration factor
F_l	Specific length factor
F_n	Normalized spatial unit count factor
F_o	Path options factor
F_p	Useful perimeter factor
F_{pc}	Parts count factor
F_{pr}	Proximity convenience factor
F_{rp}	Plan aspect ratio factor
F_{rs}	Section aspect ratio factor
FSE	Flight support equipment
F_s	Vault factor
F_{ss}	Safe-haven split factor
F_u	Spatial unit number factor
F_v	Volume range factor
FY88	Fiscal Year 1988 (=October 1, 1987 to September 30, 1988. Similarly for other years)
g	Acceleration in Earth gravities (=acceleration/9.80665m/s^2)
GCNR	Gas core nuclear rocket
GCR	Galactic cosmic rays
GEO	Geosynchronous Earth Orbit
GN2	Gaseous nitrogen
GN&C	Guidance, navigation, and control
GPS	Global Positioning System
Gy	Gray (SI unit of absorbed radiation energy = 10^4 erg/gm)
hab	Habitation
HD	High Density
HEI	Human Exploration Initiative (obsolete for SEI)
HLLV	Heavy lift launch vehicle
hrs	Hours

hyg w	Hygeine water
HZE	High atomic number and energy particle
H2	Hydrogen
H_2O	Water
ICRP	International Commission on Radiation Protection
IMLEO	Initial mass in low Earth orbit
in.	Inches
inb	Inbound
IP&ED	Implementation Plan and Element Description
IR&D	Independant research and development
Isp	Specific impulse (=thrust/mass flow rate)
ISRU	In-situ resource utilization
JEM	Japan Experiment Module (of SSF)
JSC	Johnson Space Center
k	klb
keV	Thousand electron volt
kg	Kilograms
klb	Kilopounds (thousands of pounds. Conversion to SI units=4448 N/klb)
klbf	Kilopound force
km	Kilometers
KM	Kilometers
KM/Sec	Kilometers per second
KM/SEC	Kilometers per second
ksi	Kilopounds per square inch
LCC	Life cycle cost
L/D	Lift-to-drag ratio
LD	Low density
LDM	Long duration mission
LEO	Low Earth orbit
LET	Linear energy transfer
LEV	Lunar excursion vehicle
LEVCM	Lunar excursion vehicle crew module
Level II	Space Exploration Initiative project office, Johnson Space Center
LH2	Liquid hydrogen
LiOH	Lithium hydroxide
LLO	Low Lunar orbit
LM	Lunar Module
LOR	Lunar orbit rendezvous
LOX	Liquid oxygen
LS	Lunar surface
LTV	Lunar transfer vehicle
LTVCM	Lunar transfer vehicle crew module
L2	Lagrange point 2. A point behind the Moon as seen from the Earth which has the same orbital period as the moon.
m	Meters
[MarsGram	Western Union interplanetary telegram]
[MARSIN	Martian pornography]
MASE	Mission analysis and systems engineering (same as Level II q.v.)
MAV	Mars ascent vehicle

M/C_DA	Ballistic coefficient (mass / drag coefficient times area)
MCRV	Modified crew recovery vehicle
m_e	Mass of electron
MEOP	Maximum expected operating pressure
MeV	Million electron volt
MEV	Mars excursion vehicle
MLI	Multi-layer insulation
mm	Millimeter (=0.001 meter)
MMH	Monomethylhydrazine
MMV	Manned Mars vehicle
MOC	Mars orbit capture
MOI	Mars orbit insertion
mod	Module
M&P	Materials and processes
MPS	Main propulsion system
MR	Mixture ratio
m/sec	Meters per second
MSFC	Marshall Space Flight Center
Msi	Million pounds per square inch
mt	Metric tons (thousands of kilograms)
mT	Metric tons
MTBF	Mean time between failures
MTV	Mars transfer vehicle
MWe	Megawatts electric
m^3	Cubic Meters

N	Newton. Kilogram-meters per second squared
n/a	Not applicable
NASA	National Aeronautics and Space Administration
NCRP	National Council on Radiation Protection
NEP	Nuclear-electric propulsion
NERVA	Nuclear engine for rocket vehicle application
NTP	Nuclear thermal propulsion (same as NTR)
NSO	Nuclear safe orbit
NTR	Nuclear thermal rocket
N2O4	Nitrogen tetroxide

OSE	Orbital support equipment
OTIS	Optimal Trajectories by Implicit Simulation program
outb	Outbound
O2	Oxygen

PBR	Particle bed reactor
Pc	Chamber pressure
PEEK	Polyether-ether ketone
PEGA	Powered Earth gravity assist
P/L	Payload
POTV	Personnel orbital transfer vehicle
pot w	Potable water
PPU	Power processing unit
prop	Propellant
psi	Pounds per square inch
PV	Photovoltaic

Q	Heat flux (Joules per square centimeter)
Q	Radiation quality factor
RAAN	Right ascension of ascending node
RCS	Reaction control system
Re	Reynolds number
RF	Radio frequency
RMLEO	Resupply mass in low Earth orbit
ROI	Return on investment
RPM	Revolutions per minute
RWA	Relative wind angle
R&D	Research and Development
	Rendezvous and dock
SAA	South Atlantic Anomaly
SAIC	Science Applications International Corporation
SEI	Space Exploration Initiative
SEP	Solar-electric propulsion
SI	International system of units (metric system)
SiC	Silicon carbide
SMA	Semimajor axis
sol	Solar day (24.6 hours for Mars)
SPE	Soalr proton events
SRB	Solid Rocket Booster
SSF	Space Station Freedom
SSME	Space Shuttle Main Engine
STCAEM	Space Transfer Concepts and Analysis for Exploration Missions
stg	Stage
surf	Surface
Sv	Sieviert (SI unit of dose equivalent = Gy x Q)
S1	Distance along aerobrake surface forward of the stagnation point
S2	Distance along aerobrake surface aft of the stagnation point
S3	Distance along aerobrake surface starboard of the stagnation point
t.	Metric tons (1000kg)
TBD	To be determined
Tc	Chamber temperature
TCS	Thermal control system
TEI	Trans-Earth injection
TEIS	Trans-Earth injection stage
t.f.	Tank weight factor
THC	Temperature and humidity control
TMI	Trans-Mars injection
TMIS	Trans-Mars injection stage
TPS	Thermal protection system
TT&C	Tracking, telemetry, and control
T/W	Thrust to weight ratio
UN-W/25Re	Uranium nitride - Tungsten/25% Rhenium reactor fuel
VAB	Vehicle Assembly Building
VCS	Vapor coolled shield
Vinf	Velocity at infinity

WBe_2C/B_4C	Tungsten beryllium cabide/Boron cabide composite
WMS	Waste management system
W/O	Without
WP-01	Work package 1 (of SSF)
w/sq cm	Watts per square centimeter (should be Wcm^{-2})
Z	Atomic number
zero g	An unaccelerated frame of reference, free-fall

[order: numbers followed by greek letters]

100K	$\leq$100,000 particles per cubic meter larger than 0.5 micron in diameter
7*n*7	Where n=(0,2-6): Boeing Company jet transport model numbers
°k	Kelvin (K)
+e	Positive charge equal to charge on electron
-e	Charge on electron
ΔV	Change in velocity
s	Standard deviation
μg	Microgravity (also called zero-gravity)

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Background and Rationale

This document presents trade studies and reference concept designs accomplished during a study of Space Transfer Concepts and Analyses for Exploration Missions (STCAEM) by Boeing for the NASA Marshall Space Flight Center. This volume contains the major top level trades, Level II trades conducted in support of NASA's Lunar/ Mars Exploration Program Office Level II trade studies, and a synopsis of the vehicles for different propulsion systems under trade consideration. The vehicles are presented in more detail in other volumes of this document.

The current study began in August 1989 shortly after President Bush established the Space Exploration Initiative (SEI). The first six months of the study emphasized analysis of cryogenic/ aerobraking Mars transfer systems, supporting the NASA "90-Day Study" of the lunar/Mars initiative conducted in response to the President's directive. The second six months of this study concentrated heavily on trade studies, including alternative propulsion technologies as well as systems, subsystems, configurations, performance, and operations trades. The cryogenic/ aerobraking system provided a reference baseline for comparison of the alternatives. Trade studies (1) improved definition of the baseline, (2) developed concepts embodying alternative propulsion technologies, (3) compared alternatives to the baseline, (4) examined system elements and subsystems such as Mars transfer habitats common to all transfer systems, (5) examined commonality with lunar mission systems, and (6) conducted programmatics, life cycle cost, and return-on-investment trades leading to recommendations for architecture selection and technology advancmeents, reported in the final technical report for this study. This effort supports an overall NASA effort to develop viable alternative SEI architectures for in-depth definition, leading to selection of system architectures for execution of the SEI program.

Assumptions, Requirements and Groundrules

The NASA "90-Day Study" began with a well-developed set of requirements and ground rules derived from the FY88 and FY89 exploration case studies. These were found to be unduly restrictive as a basis for wide-ranging trade studies and were gradually simplified as the trade studies evolved. Requirements used in this study are presented in the final technical report

An important factor in the overall trade effort is the volume and mass of crew modules. Mars transportation system designs are dominated by the size and mass of crew modules; these are the principal payloads for the transportation systems. The trade studies reported herein used parametric estimates based on historical manned space systems and on Space Station Freedom subsystems mass data where

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relevant and available. The resulting volume and mass parametrics, and a summary of the reference crew modules, are included in this section.

The number of crew personnel is also obviously important. A crew of four was taken as the reference for the trades reported herein. A subsequent skill mix analysis indicates the minimum Mars crew to be six or seven people. While this would not appear to influence trade relative results, it will increase the overall size of the systems described herein.

Trade Study Synopsis

This section is an executive summary of the major trade results. Each trade study section of the document begins with a more detailed summary.

Lunar/Mars Commonality Trades

Crew modules: Crew modules are distinguished mainly by their design occupancy duration, which influences internal volume, degree of ECLSS closure, and redundancy level. There are four ranges of capability: 1 to 5 days - open ECLSS and spartan crew accommodations, applicable to lunar and Mars excursion vehicles assuming other accommodations are used for the surface stay; 2 to 3 weeks - partially closed ECLSS and minimum crew accommodations, applicable to the lunar transfer vehicle and a 100-km. class lunar/Mars rover; 1 to 3 months - closed ECLSS and full crew accommodations except private quarters, applicable to a lunar "campsite" module and a surface habitat module for 30 to 90 day Mars excursions; and more than 3 months to indefinite - closed ECLSS, possibly bioregenerative, and full crew accommodations, applicable to the Mars transfer module and permanent surface bases. These ranges lead to some common features. Design for use either in zero g or artificial/natural gravity is not a significant impact if included from the beginning. One key trade result, reflected in the ranges stated, was that living in the excursion vehicle crew module does not pay off for surface stays more than about a week. These commonalities and differences drive vehicle-level commonalities.

Avionics: A common-core avionics system, with unique needs met by optional add-on peripherals, is applicable to all SEI systems. Substantial advances in the avionics and software state of the art are expected to continue over the life of the SEI program. Therefore, the avionics system should be designed from the beginning as a flexible evolutionary architecture with standard protocols and interfaces, able to incorporate plug-compatible improved equipment.

Engines: A common requirement exists for an advanced space cryogenic engine in the 20 to 30 klb thrust range. The exact thrust level has not been determined;

to some extent, vehicle designs can be adapted to engine capability. A reasonable program strategy is to start with a 20k engine, well-suited to early lunar applications, and design for planned product growth to the 30k thrust level. The engine needs to have relatively deep continuous throttling capability for lunar and Mars landing. Benefits of high specific impulse are significant; a target of about 475 seconds seems appropriate. Engine reliability, life, and space serviceability should not be sacrificed for a few seconds of Isp.

The reference cryogenic/aerobrake system also used a space engine in the 150k thrust range for the trans-Mars injection (TMI) stage. Subsequent trades have indicated that (1) a cluster of the 30k engines, combined with multi-burn TMI, has acceptable performance, and (2) except in the case of an early Mars mission or of a small Mars program consisting of 3 or less trips to Mars, advanced space propulsion should be developed for Mars transfer propulsion. At this juncture, the 150k class engine does not seem to have high technical merit.

Stages and Vehicles: The trans-Mars injection stage, whatever its propulsion technology, is much larger than anything needed for the lunar program. While there is excellent opportunity for technology and subsystem commonality, stage commonality does not appear likely. A cluster of lunar transfer stages could be used for TMI, but the design requirements imposed by clustering would probably lead to changes that could be as expensive as a new stage design.

There is a possible commonality between the lunar transfer stage and the trans-Earth injection stage. The propellant loads and thrust level are similar. In the cryogenic/aerobraking reference, the large Mars transfer crew module and integration with the aerobrake led to major configuration differences. In the case of an all-propulsive cryogenic Mars mission, the chance for commonality may increase, but the trend is for the MTV in this case to have a greater propellant load than the LTV. Also, the mission durations are greatly different, leading to a more sophisticated cryogenic insulation design for the Mars vehicle.

Aerobrakes: Differences in vehicle sizes and masses lead to great differences in aerobrake designs. There appears to be a good opportunity here for commonality of both structures and materials.

Lunar and Mars Excursion Vehicles: The high commonality potential for crew modules is described above. When propellant load requirements for lunar and Mars landing and ascent are computed for transporting a common crew module, they are found to be very similar, leading to an interest in a common vehicle design. However, differences in lunar and Mars missions have led to different approaches to LEV and MEV design. The Moon is frequently accessible. LEV operations are typically modular, i.e. cargo and crew trips are separate. For the MEV, cargo and crew have typically been integrated, as in the reference system. They could be separated, and this needs further investigation. However, other significant configuration differences arise because of the need to integrate the MEV with a landing aerobrake. While commonality of the entire vehicle can be forced, the design penalties are such that commonality of subsystems, readily achievable, seems the preferred course.

Lunar and Mars Mission Operations

Alternative Crew Modules

We examined crew modules for 2 to 8 people and 1 to 42 days' duration, for functions of lunar transfer, lunar/Mars excursion, and direct entry into Earth's atmosphere, for an Earth Crew Capture Vehicle (ECCV). One conclusion of this trade study was mentioned above, that excursion vehicle crew modules should be designed for short duration; if occupied surface stays more than a few days are planned, a separate surface habitat that does not have to be returned to lunar or Mars orbit should be provided.

For most of the range stated 4.4-meter diameter modules were adequate . At the upper end, i.e.6 and 8 people for longer durations, the volume requirements cause a single 4.4-meter diameter module to exceed the Space Station Freedom hab module length, and splitting into two joined modules, or a larger diameter, should be considered.

We also found that mass could be minimized by cylindrical modules and ellipsoidal end domes without major penetrations. Airlocks, where required, were separate from the modules. In some instances, this permits the airlock to be disconnected and left behind before ascent from the Moon or Mars, saving propulsion system mass.

ECCVs were defined as Apollo shapes. This permitted the trade to be completed. The Apollo shape is appropriate for lunar return, but for the high end of the Mars return range, a higher lift configuration is needed.

Habitat Trade Summary

A design and integration trade for the Mars transfer habitat led to selection of a 7.6-meter diameter module. This size selection was indicated as valid over at least a crew size range from 4 to 12. This size has significant mass and integration efficiencies over a smaller diameter. A 10-meter design was also investigated but was inefficient for the smaller crew sizes, 4 to 6. The selected design used a single longitudinal floor and included a structural transverse bulkhead that provides two independently pressurizable volumes for redundancy.

Radiation Assessment

The Earth's **van Allen belts** are not a serious problem for Mars missions. In the nominal case, no special shielding is required. For multiple-burn departures, the crew can use the radiation shelter described below for adequate protection. Certain mission profiles involve either protracted low-thrust spirals through the belts, or capture of the returning Mars vehicle in a highly elliptical Earth orbit. In either case, the use of an LTV "taxi" to embark/debark the crew is advisable for other reasons and eliminates the van Allen belt exposure problem.

Solar flares can deliver debilitating or lethal doses to an unprotected human. The shielding afforded by the normal habitat structure and subsystems is inadequate. A heavily-shielded area is required. Present estimates of the shield required are 20 to 30 grams/cm^2. Since solar flares are of short duration (a few days), a "storm shelter" design approach is the preferred one. A small area of the habitat is shielded to the required level and the crew are confined to this area during a solar flare.

Galactic cosmic rays are continuous in nature and extend to very high energies. Mars transfer doses are in the range where pessimistic-side estimates exceed allowable doses but optimistic-side estimates do not. A small storm shelter is not a solution because the crew would need to spend most of their time confined in it. Shielding the entire MTV incurs large mass penalties. Two solutions are evident: (1) design Mars mission profiles for fast (less than six months) transfers and long stay times on Mars, where the atmosphere, and indigenous materials if required, provide shielding; (2) design an enlarged shielded area in the MTV habitat where the crew will normally spend most of their time. Bill Pogue (Skylab 4) stated that the Skylab 4 crew spent 90% of their time in about 10% of the available crew volume.

Certain mission profiles and propulsion systems are more amenable to cosmic ray protection. Fast transfer conjunction missions were mentioned above. Cycler architectures and electric propulsion systems tolerate more massive crew habitats with much less overall penalty than the reference cryogenic/aerobrake system.

Large Crew Size

Most of the SEI transportation studies have assumed crews of 4 to 6. We performed a crew size assessment based on skill mix requirements for Mars missions and concluded that 6 is a bare minimum, and crews of 7 or 8 may be needed. Also, as SEI evolves, larger numbers may be desired, as in the industrialization and settlement scenarios described in the final technical report for this study. Accordingly, we performed a sensitivity study on crew size, examining vehicles for crews of 8, 16, and 32. The overall result was that nuclear propulsion systems adapt to large crews much better than cryogenic systems or solar electric propulsion. Large single nuclear vehicles appear quite practical. Cryogenic vehicles grow in IMLEO much faster than nuclear vehicles. Assembly of large capture aerobrakes becomes a major issue. Cryogenic and solar electric systems appear to adapt to large crew sizes most readily through a convoy approach.

Rescue/Abort

Lunar Missions - The proximity of the Moon enables a number of rescue and abort modes. Powered or unpowered abort flyby of the Moon is possible for any of the mission modes. Direct modes deliver the entire return system to the lunar surface and enable abort directly from the surface to Earth return. Lunar orbit rendezvous modes (LOR) permit abort to lunar orbit on every revolution of the orbiting spacecraft as long as the surface site is at the equator, which was our general assumption. L2 libration point rendezvous enables abort to L2 from the surface at any time from any location.

Site selection considerations discussed at a site selection review in Houston suggest that equatorial sites are not a foregone conclusion. Further, the lunar transportation mode will probably be selected before the lunar outpost sites are selected. For this reason, we concluded that a direct mode or L2 rendezvous mode should be selected for initial implementation. When the lunar base evolves to the point that it has long-duration stays with adequate critical subsystem redundancy, the longer waits (about a week) for LOR abort from non-equatorial sites would be acceptable. At that point the greater efficiency of LOR could be adopted.

Mars Missions - Mars unpowered flyby abort options exist for most conjunction and opposition/swingby profiles. These involve the normal mission duration; rapid returns are not available. For those profiles not having an unpowered flyby abort, a modest amount of propulsion, available from either the MTV or MEV, will place the vehicle on an Earth return path. "Fast" opposition profiles may not have flyby aborts. If they do, the return-to-Earth duration is very long compared to the planned mission duration. Aborts constrained to planned trip time require greater and greater delta V for a powered abort as trip time is reduced. We found that electric propulsion systems can limp home from Mars at half power with very little mission duration extension. We did not perform a general low-thrust abort study.

The MEV is designed so that an ascent abort from a descent can be initiated near the end of the aerobraking descent profile, and during the powered descent. The ascent stage is separate as was the case for the Apollo lunar module. One issue that exists for the cryogenic MEV is the time available for an abort in the event of breaching of the ascent tanks' vacuum jacket on the surface of Mars. Many hours to days would appear to be available unless a very large vacuum leak occurs. The MEV can abort to Mars orbit twice per Mars day, when the surface site passes beneath the parking orbit plane. The ascent initially injects into a low Mars phasing orbit and then continues to the elliptic parking orbit at the proper time to achieve alignment of the apsides.

A further issue is abort return to Earth from the Mars parking orbit on a conjunction profile. During the short stay of an opposition profile, orbit secular precession is small enough that abort is always possible. On a conjunction profile, the interplanetary transfer energy is such that an abort is possible for about the first 100 days at Mars. However, the Mars parking orbit inclination and period are "tuned" to achieve proper orbital plane and apsides alignment at the normal departure time. While we did not investigate early return aborts, it is likely that orbital alignment will be a problem.

Advanced Propulsion System Trades

Lunar missions benefit much less from advanced propulsion performance than Mars missions. Mars missions last months to years; because of their duration, they require massive, costly crew habitats for the transfers from Earth to Mars and back. Further, lunar missions require high thrust propulsion, at least for crew missions, while both crew and cargo missions to Mars can use electric propulsion. A strictly lunar scenario would admit only nuclear thermal rockets as an advanced propulsion option, since the low thrust of electric propulsion systems makes them ill-suited for lunar crew transport and they appear not worth developing merely for lunar cargo transport.

Activity Levels - Selection of transportation architectures is influenced by activity level much more than by purpose or strategy, because transportation basically responds to "how much" and "how often." Three activity levels, "minimum," "full science menu," and "industrialization and settlement," were defined to assess the sensitivity of architecture selection to activity level.

The three levels of activity were as follows: **Minimum**, just enough to meet the President's objectives; in fact "return to the Moon to stay" was interpreted as permanent facilities but not permanent human presence. The minimum program had only three missions to Mars. The **full science** program aimed at satisfying most of the published science objectives for lunar and Mars exploration. The largest activity aimed for **industrialization** of the Moon, for return of practical benefits to Earth, and for the beginnings of **settlement** of Mars. The range of activity levels, as measured by people and materiel delivered to planetary surfaces, is about a factor of 10. The range of Earth-to-orbit launch rates was less, since we adopted results of preliminary trade studies, selecting more advanced in-space transportation technologies as baselines for greater activity levels.

Mars transfer propulsion trades considered **cryogenic all-propulsive** (CAP) systems on conjunction profiles, **cryogenic/aerobraking** (CAB) on opposition and conjunction fast transfer profiles, **nuclear thermal rockets** (NTR), **nuclear electric propulsion** (NEP), and **solar electric propulsion** (SEP) on all profiles, and **gas-core nuclear rockets** (GCNR) on fast-trip profiles. Additional architectures reflecting novel operational strategies included **Mars direct**, **lunar L2 libration-point basing with lunar oxygen**, and various cycler and **semi-cycler** modes. (The term semi-cycler implies combinations of flyby and stopover operations)

A promising variation on a semi-cycler was identified late in the study and labeled **NTR-dash**. In this mode, the MEV separates from the NTR MTV three to four months before Mars arrival. The MTV makes a posigrade burn or the NTR makes a retrograde burn, causing Mars arrival of the two vehicles to be 10 to 30 days apart. The MEV performs a short-duration surface mission; its ascent stage makes a hyperbolic rendezvous with the NTR immediately after Mars flyby. The NTR may be on a free-return flyby or may need to deliver impulse to return to Earth depending on mission profile details. In either case, the NTR delta V is much reduced. This mode reduces initial mass by 10% to 40%. Further profile analysis are needed. The mode appears particularly attractive for crew rotation and resupply at a Mars base.

The cryogenic systems were considered as reference and the others as advanced propulsion. Cryogenic all-propulsive is indicated for the conjunction mission since the Mars capture delta V is so low that the mass penalty for propulsive capture is about the same as the mass penalty for an aerobrake. One variation on the conjunction profile expends additional energy to obtain short transfer times. In this case, cryogenic/aerobraking is a logical choice for the conjunction profile.

Significant performance advantages exist for advanced propulsion, and these advantages increase as trip time is reduced. Therefore, selection of advanced propulsion options and of evolutionary paths for propulsion development is crucial to efficient, economic overall SEI program architectures. If an advanced propulsion option is justified for Mars missions, appropriate use for it is sought in lunar operations. Further, its maturity may need proving on the (relatively) low-risk lunar mission profiles.

Architectures Mass and Reusability Summary

Resupply performance is the key measure of efficiency for repeated missions; for single or expendable architectures, the total initial mass in Earth orbit (IMLEO) applies. Also important is the fraction of propulsion and mission hardware available for reuse at the end of the mission; this is an important mission cost factor since hardware replacement cost can easily exceed ten times the cost to place mass in LEO. Generally, options with lower resupply requirements and higher reusability employ more advanced propulsion and imply more developmental effort. Therefore, one expects low resupply modes to be attractive mainly for larger-scale programs, where the greater technology and development effort is effectively amortized. Less advanced systems are cost effective for lesser programs. The performance of Mars surface rendezvous (Mars direct), as a crew delivery system for a man-tended but long-duration stay base is quite attractive. Since Mars direct is only operable on a conjunction profile, it is not well suited to crew rotation for a permanent base. The NTR-dash profile described below appears well-suited to this requirement.

A mass advantage does not necessarily translate to a cost advantage. For example, lunar oxygen supply to the L2/lunar oxygen architecture appears to have attractive performance. A payback analysis was performed using simple scaling equations. In the case of all-cryogenic propulsion, the IMLEO savings is about 300 t. per mission, compared to cryogenic/aero-braking from LEO. The lunar oxygen production rate needed to fly every Mars opportunity is about 360 t./yr. The quantity of production and power equipment needed for this production rate is not well-defined; a moderately optimistic figure is 1 t. equipment per t./yr production. To save 300 t. per mission, 360 t. of production equipment is delivered to the lunar surface. Since the ratio IMLEO to lunar cargo is about 6, the IMLEO cost is over 2000 t., and breakeven occurs only after 7 to 8 Mars opportunities.

Use of advanced propulsion reduces the resupply requirement and the payback time for lunar oxygen. However, if advanced propulsion is available, why not use it directly for the Mars mission? Use of a mass driver to deliver the lunar oxygen to L2 (in cannisters) also improves the payback, but introduces another new technology. We did not evaluate this option in the present study.

Mission Risk Comparison

Mission risks were compared in a semi-quantitative way. The methodology is rigorous and quantitative, but reliability and safety estimates for SEI hardware and maneuvers are quite rough. We made representative estimates with an attempt to be consistent, i.e. the same type of maneuver was given the same reliability estimate for all cases. Plausible differences were used, e.g. aerocapture was judged higher risk than propulsive capture. Abort modes were included where available. A probability of success value is assigned to each event, and the cumulative probabilities for mission loss, crew loss, and mission success are calculated.

NTR shows the least risk because of its propulsive capture advantage, and because a free return abort was assumed, as it was for the cryo/aerobrake. The NTR/dash mode does not permit free return abort or descent abort at Mars, so some mission loss risk turns into crew loss risk. As Mars transportation matures and a safe refuge on the surface of Mars is available, the NTR/dash mode risk will be comparable to the other NTR mode. The NTR split sprint mode also exhibits higher risk because of lack of abort modes, e.g. no free return. NEP is shown comparable to, but slightly riskier than NTR. The NEP case is sensitive to the lifetime dependability of the propulsion system; this figure is much more uncertain than NTR reliability. Mars direct has a higher mission loss risk because of its complex automated operations, but the crew loss risk is comparable to the others. The perception of crew loss risk for Mars direct is probably higher than the real risk.

Man Rating Requirements

The recommended approach to man-rating includes three elements: (1) Design of systems to manned space flight failure tolerance standards, (2) Qualification of subsystems according to normal man-rating requirements, and (3) Flight demonstration of critical performance capabilities and functions prior to placing crews at risk.

The systems and subsystems for which a man-rating requirement was identified are: aerobrakes, cryogenic rocket engines, nuclear rocket engines, cryogenic propellant systems, attitude control propulsion systems, nuclear and solar electric propulsion systems, environmental control and life support systems (ECLSS), crew modules and hab systems, vehicle electrical power, avionics and communications systems, and surface transportation systems. Integrated man-rating approaches for most of these are presented in this document.

On-Orbit Assembly

The recommended lunar architecture requires no on-orbit assembly until later in the program. The initial tandem-direct lunar transfer vehicle (LTV) can be launched fully assembled and fueled. Two are needed, requiring in-space rendezvous and berthing. Later, a lunar aerobrake may be introduced, but lunar aerobrake assembly can probably be accomplished from a shuttle Orbiter payload bay. The underslung heavy-cargo version of the LTV will require some deployment or assembly as it exceeds the 10-meter diameter we assumed as an HLLV diameter limit.

Operations analyses during the latter part of the study concentrated on alternative assembly concepts for the principal architectural options (cryogenic/aerobraking, nuclear thermal rocket (NTR), nuclear electric (NEP) and solar electric (SEP)). Evaluations of these alternatives have not yet been accomplished. The intent is to simplify the assembly facility by simplifying the assembly process through vehicle design provisions.

The NEP presents the most difficult assembly challenge because of its extensive fluid systems. The SEP is very large, but the assembly process is repetitive and well-suited to robotics. The NTR is easiest, and the cryogenic options somewhat more challenging because of the greater diversity of assembly tasks.

Costs and Schedules

Initial mass in Earth orbit (IMLEO) is often used for comparisons between propulsion options. However, other factors have great cost impact. Reusability and reduction of development cost by reducing the number, complexity, size, or risk of developments are very important.

All cost analyses in this study were performed in constant 1990 dollars, i.e. no attempt was made to forecast inflation. Development and unit costs for architecture elements were estimated using the Boeing Parametric Cost Model. Complete DDT&E costs were developed by estimating the equivalent number of production units consumed in the development program, e.g. for test articles. The first flight article was assumed produced by the DDT&E program; subsequent flight articles were assigned to production programs. We adopted a minimum production rate of 0.3 units per year. Our scenarios avoided shutdown and restart of production lines. Technology advancement and advanced development programs were estimated for each option, using available estimates for these activities together with our judgments as to what technology advancements and advanced developments are necessary. Cryogenic/all-propulsive offers the least development cost, with cryogenic/aerobraking and nuclear thermal rocket slightly more. Electric propulsion systems were significantly higher; the development cost for solar electric is dependent on production cost for large solar arrays. Only with dramatic reduction from today's array costs is solar electric attractive.

In order to get life cycle costs, technology advancement, advanced development, and full-scale development (DDT&E) and production scenarios were established for each program activity level. Launch and flight operations were manifested according to the top-level schedules shown earlier. These were loaded into a spread-sheet life cycle cost model that generates annual funding profiles

To evaluate alternative architectures, the alternatives were manifested (e.g. NTR versus cryogenic aerobraking) for the same program scenario, including changes in technology advancement, advanced development, DDT&E, production, and HLLV launches. Life-cycle costs were generated for the alternatives. This provided comparative life cycle cost profiles for which return on investment (ROI) could be calculated.

Results of Return on Investment Analyses

The representative ROI scenario is that one option has greater front-end investment cost, leading to savings later in the program through more advanced technology, more hardware reuse, fewer HLLV launches, or combinations thereof. We used a criterion that a ROI of 5% or better is acceptable. That is the approximate cost of money to the government in constant dollars. Return on investment is a severe criterion for front-end investment. To obtain a favorable ROI, 10% or better, an investment must generate large savings. The most advantageous situations occur when a technology advancement reduces near-term DDT&E cost. Technology advancement costs are usually small compared to the DDT&E savings and the savings occur soon after the investment. An ROI can be calculated only when funding streams cross.

The case for reusable lunar transportation is negative for a minimum lunar program and weak for a median program; it is strong for an industrialization-class program.

The other main results were that nuclear thermal rocket has a favorable ROI compared to cryogenics except for the minimum Mars activity level; solar electric is attractive if (and only if) array production costs can be brought down to about $100/watt; nuclear electric DDT&E costs are too high to be attractive, leading to a recommendation to reexamine these costs and see what can be done to reduce them; technology advancement and advanced develolpments in cryogenics management, advanced engines, and avionics have large positive returns.

Specific Results for In-Space Transportation Options

Each of the principal options could become a preferred selection under plausible program circumstances.

a. For a minimum lunar activity level, a simple tandem-staged direct expendable mode is attractive. While expendable systems require continuing hardware production, the production lines must be kept open in any case. For minimum activity levels, the cost of having open lines produce hardware is quite small. At two lunar missions per year, the return on investment for developing an efficient reusable lunar orbit rendezvous (LOR) system is only about 5%. Programs with activity levels of four or more lunar missions per year benefit significantly from efficient reusable lunar transportation.

b. For a minimum Mars program, consisting of perhaps a half-dozen landings of a few days' stay time each (that is what Apollo accomplished on the Moon), cryogenic all-propulsive minimum-energy missions with multiple landers, e.g two or three per mission for two or three missions, are very attractive. This offers the opportunity to briefly explore six sites at minimum cost and minimum technology risk (Apollo explored six sites, spending a few days at each site). Carrying multiple landers per trip provides a desirable rescue capability.

One concern with this implementation is that astronauts are committed to almost three years in space each Mars mission. There are significant issues regarding zero-g and cosmic ray exposure. While these concerns can presumably be dealt with, i.e. through zero-g countermeasures, artificial g, or suitable shielding, solutions may be costly in mass and complexity .

c. The performance potential of a nuclear thermal rocket (NTR) leads to less initial mass than cryogenic/aerobraking for most mission profiles. A nuclear rocket can eliminate the need for high-energy aerocapture at Mars; this is an important advantage. On the other hand, the development program for a nuclear rocket requires significant investment in effluent containment test facilities. Return on investment tradeoff of nuclear rocket versus cryogenic/ aerobraking at the median Mars activity level favored the nuclear rocket. If Mars exploration progresses to a permanently-occupied base, aerocapture and NTR are complementary technologies in the NTR-dash mode; this traded favorably versus nuclear electric propulsion in ROI analysis.

The nuclear thermal rocket improves mission flexibility and reduces constraints on mission profiles. A nuclear rocket is the most promising propulsion system for fast Mars trips (a year or less). Fast trips, however, are indicated as expensive in terms of total mass and hardware expended. While fast trips are technically interesting, they are probably not affordable in a space program with constrained funding.

d. We found that electric propulsion systems are suitable for Mars crew transportation if (1) operated from high-altitude nodes such as L2, or (2) boarded by the mission crew at about lunar distance, where the crew fly to the electric propulsion vehicle on a lunar transfer vehicle (LTV). Trip times are competitive with all but fast-trip split-sprint nuclear thermal rocket systems, i.e. about 450 days for nuclear electric propulsion (NEP) and about 550 days for solar electric (SEP). On conjunction fast transfer profiles, NEP delivers 150 to 200 day transfers each way and SEP about 250.

The inherently high reusability and low resupply mass of electric systems offers life-cycle cost advantages at high activity levels. Development cost for NEP and array production cost for SEP are major issues. Resolution of the array production cost issue will require a manufacturing technology program. Cost and return-on-investment results showed that estimated NEP development costs are not effectively amortized even at the settlement activity level when compared with a nuclear rocket operated in the dash mode. SEP, at current array costs (~ $1000 per watt), is estimated as more expensive to develop than NEP. SEP becomes very attractive at $100/watt, showing about 10% return on investment versus NTR at the median activity level. If a low-cost SEP is possible, it is also attractive for lunar cargo.

e. Special architectures offer unique advantages in particular circumstances. For example, lunar libration point staging is attractive for low-thrust systems because spiral operations out from and into Earth's gravity well can be conducted by an electric orbit transfer vehicle in parallel with interplanetary transfers by the interplanetary SEP or NEP.

Lunar libration point operations offer reduced Earth launch mass for cryogenic/aerobraking profiles through use of lunar oxygen (the return on investment in lunar oxygen facilities is not favorable), and to electric propulsion systems because the interplanetary vehicle need not execute low-thrust spirals out of and returning into Earth's gravity well. Neither of these potential advantages applies to nuclear thermal rockets; libration point operations for nuclear rockets were not considered.

Mars direct simplifies flight operations at Mars at the expense and risk of propellant production on Mars; it is more efficient than Mars orbit rendezvous in a crew transport mode after a base is established, but not as efficient as NTR-dash. It appears too risky (lack of abort modes) for an initial mission. Mars direct offers potential advantages where galactic cosmic ray concerns drive us to conjunction fast transfer profiles with long surface stays. It is not suitable for crew rotation and resupply of a permanent base because it is confined to the conjunction profile, and leads to gaps in crew presence at Mars.

Reusable MEVs using Mars oxygen, and methane or hydrogen if available, are interesting as an evolutionary development, mainly because their greater reusability may have significant life cycle cost benefit. In our settlement scenario analysis, the reusable MEV came on line too late to have a net payoff. This concept needs further evaluation.

Cyclers may be advantageous if interplanetary transfer habitats need extensive radiation shielding or if large crews and consequent massive transfer habitats are needed to satisfy mission objectives. Early in a Mars program, full cyclers do not have enough advantage over simple conjunction all-propulsive (cryogenic or nuclear) or aerobraking to merit their need for infra-structure pre-positioning, operational complexity and give-up of abort modes. The NTR-dash semi-cylcer requires no pre-positioning; it holds much promise for reducing system mass and nuclear rocket lifetime requirements

Artificial Gravity Configurations

Artificial gravity configurations were developed for the principal propulsion options. Practical solutions were found in all cases. The artificial gravity configurations ranged from 5% to 20% more massive than zero g configurations. Cryogenic all- propulsive and NTR all-propulsive systems adapt to artificial g with minimum penalty -- the vehicle is simply fitted with a structural truss of adequate length and tumbled at an appropriate spin rate during coast periods. The cryogenic/aerobraking system is equipped with a deployable tether system since the entire Mars transfer vehicle, including artificial g system, must fit within the protected region of the aerobrake during Mars aerocapture. Nuclear and solar electric systems need to thrust while artificial g is operative, and therefore require spin and despin sections with slip rings and mechanical rotating joints between. The solar electric stage suffers from not having a dense, compact element that can be used as an artificial g rotator counterweight. A concept using the entire SEP in an "eccentric rotator" configuration was developed, but the cyclic loads placed on the SEP structure and arrays probably require so much beef-up that an inert ballast counterweight would prove simpler and less costly.

The conclusion of the artificial g trade was that all propulsion options can be adapted to artificial g, and that the complexity costs will be greater than the increased mass costs. Complexity costs for NTR and cryogenic all-propulsive are less than for the other alternatives. The solar electric system adapts least well to artificial g.

MTV/MEV Mission Scenarios

Operational Orbit Selection

Mars missions are presumed to depart Earth from the Space Station Freedom orbit, adopt an orbit of convenience at Mars, and if reusable, return to the Space Station Freedom orbit. Selection of the Mars orbit must observe two factors: minimizing interplanetary transfer delta V and access to desired landing sites.

For high thrust systems, minimum interplanetary delta V occurs when a highly elliptic Mars orbit is chosen, arrival and departure occur in plane, and transfers between approach and departure hyperbolas and the Mars parking orbit occur periapsis- to-periapsis and tangentially, i.e. the parking orbit line of apsides is properly aligned for arrival and departure. On conjunction mission profiles, all of these conditions can be nearly satisfied by selection of the parking orbit inclination and period. On opposition missions, parking orbit period has little effect and there are not enough degrees of freedom in the mission design to meet all the conditions. The usual result is that the parking orbit line of apsides is misaligned for departure, and a delta V penalty is accepted.

Optimal orbits range from 30 to 60 degrees inclination at Mars. Some opposition mission profiles have an optimum delta V parking orbit with retrograde inclination and (usually) periapsis on the dark side of the planet. Since landing occurs near periapsis unless severe deorbit delta V penalties are accepted, this study has rejected optimal orbits with periapsis in the dark and instead selected the best light-side periapsis orbit.

Most scientific sites of interest at Mars are within 30 degrees of the equator and hence accessible from optimal orbits. Availability of Mars water is also an issue; this issue may demand access to higher latitudes. The polar caps are also scientifically interesting; human missions to the polar caps would require parking orbits at higher than 60 degrees inclination or cross-range capability on the part of the lander both on descent and ascent.

Some mission architectures involve establishing an orbital node at Mars and therefore return to the same Mars parking orbit on successive missions. This is labeled the "standard orbit" problem. A preliminary investigation of standard orbits did not find any that had acceptably low delta V penalties. The subject needs more analysis, with tailoring of interplanetary profiles to obtain better matches to the standard orbit. Architecture recommendations of this study did not include architectures that need a standard orbit.

Low-thrust interplanetary propulsion systems, i.e. nuclear and solar electric, can adopt any desired parking orbit with little penalty. This is because low-thrust systems must approach and depart from Mars in a rendezvous fashion, and therefore the direction from which they approach Mars, in terms of relative velocity, has little effect on mission design and system performance.

MEV propellants

While the high performance of hydrogen and oxygen is desirable in almost any mission situation, storage of these cryogens on Mars' surface brings about duration and atmosphere issues. Some Mars mission profiles involve long stay times on the surface, up to 600 days. Under our general ground rule of passive cryogenic storage means, long stays are problematical. Further, the usual design for high-performance cryogenic insulation presumes a vacuum environment. Mars' atmosphere, while tenuous, is hardly a vacuum. Therefore, MEVs depending on cryogen storage on Mars must have vacuum-jacketed propellant tanks.

The propellant trade is between the mass penalties of cryogenic insulation, vacuum jackets, low-density propellants and boiloff but high Isp, versus the opposites for storable propellants: very little insulation, no vacuum jacket, high-density propellants, and no boiloff, but less Isp. Trade analyses indicate moderate to strong advantages of the high Isp cryogens despite their disadvantages. Our conclusion is that cryogenics are indicated for short to intermediate surface stays of 30 to 90 days but that storables are preferred for longer stays despite a moderate mass penalty. Development of advanced storable propellant technology including pump-fed engines and gelled hydrazine fuels with aluminum added is indicated as having high leverage on mission/ system performance.

Aerobrake Issues

Aerocapture lift-to-drag ratio (L/D): Several investigators have converged on the result that aero- capture L/D 0.5 is adequate for human Mars missions. The delta V to trim the aerocapture orbit to a nominal operations orbit is reduced by higher L/D; the amount of reduction needs further study. Our present state of understanding permits us to set the design requirement for Mars aerocapture and landing L/D between 0.5 and 1.0 but does not enable selection of an optimum within this range.

Navigation Aids: Successful aerocapture requires precision navigation at Mars. Position knowledge to 5 km. or less is needed. In the time frame of interest, adequate precision may be attainable by Earth-based tracking and navigation. Mission safety demands an onboard capability. Two general avenues are promising - (1) artificial navigation aids, i.e. satellites, placed in orbit about Mars. Suitable navigation aids can be added to satellites placed in Mars orbit mainly for other reasons; (2) optical navigation by sightings of Mars and its natural satellites. It may also be helpful to use a radar altimeter upon close approach to accurately predict atmosphere entry angle.

Pinpoint surface landing, e.g. return to a base site, needs much higher accuracy, i.e. tens of meters, than aerocapture. When landing from a Mars parking orbit, internal inertial navigation will be adequate during all but the terminal portion of aero-assisted landing. Final propulsive descent will need landing point aids such as a transponder at the landing site or the Mars equivalent of GPS. An alternative may be to use terrain matching to obtain a very accurate state vector during aero descent. If pinpoint landing directly from a transfer path is required, either a Mars GPS or a terrain matching scheme operating during the high-speed portion of the descent will be required.

Structures and Materials: Aerobrake structures are highly loaded and Mars vehicle overall mass is sensitive to aerobrake mass. High-performance structures have important payoff. Our investigations considered plastic and metal-matrix composites and at titanium aluminides as candidates. We have not yet selected among these but all are preferable to aluminum structures.

Aerobraking temperatures for Mars and Earth aerocapture range from shuttle tile temperatures to 500 - 1000 C higher. High L/D shapes confine the severe heating to a small region on the nose and leading edge. Low L/D shapes have very large nose radii, and are predicted to experience severe radiation heating over most of the thermal protection system TPS. Lightweight ablators could be used, but technology advancements are needed. There is some expectation that advances in reradiative TPS materials will be able to handle the 500-1000 C increase over current materials. For the high L/D shapes, it appears reasonable to use carbon/carbon in the small high-temperature region; most of the high L/D brake is within the capabilities of current materials.

Advanced propulsion systems, and the cryogenic/all propulsive option, do not require aerocapture at Mars. In this case, the MEV has a landing-only brake. Thermal analyses predict that most of the brake will be within the temperature capabilities of titanium aluminide without a TPS.

Landing Locations: We performed a brief analysis of Mars landing locations. This was not to pre-empt the eventual site selection problem, but to get an idea of whether sites of interest lie near the equator, which many do. There are reasons to want access, at least on some missions, to higher latitudes.

High-thrust opposition profiles usually lead to a preferred Mars parking orbit inclination of 30 degrees or less. Conjunction profiles sometimes have preferred inclinations as high as 60 degrees, giving much greater site access. Low-thrust transfer systems can provide any desired inclination with little penalty.

High L/D Concepts: We developed a family of high L/D shapes, including swept hyperboloid wing shapes and bent biconics. These have L/D max somewhat greater than 1, and are blunt enough to avoid extreme heating. The high L/D shapes avoid the severe radiation heating predicted for the L/D 0.5 shape, except for a small nose region. The wing family provides high-normal-force shapes for effective use of Mars' thin atmosphere. Center of gravity and wake protection requirements are reasonably met. The biconic has less normal force and is more difficult to package, but could eliminate aerobrake on-orbit assembly. One of the wing shapes was carried through conceptual design as a reusable MEV; the integration of the shape into a vehicle was straightforward. The resulting configuration is similar to the Langley HL-20.

Risk Analyses and Trades

Risk analyses were conducted to develop an initial risk assessment for the various architectures, considersing development risk, man-rating requirements, and several aspects of mission and operations risk.

Development Risk

All of the architectures and technologies investgated in this study incur some degree of development risk; none are comprised entirely of fully developed technology. Development risks are correlated directly with technological uncertainties. We identified the following principal risks:

Cryogenics - High-performance insulation systems, propellant transfer and zero-g gauging need technology advancement. The tandem-direct system recommended elsewhere in this report presents the opportunity to evolve these technologies with operations of initial flight systems, avoiding special flight tests.

Engines - The risk of developing more advanced engines is minimal. An advanced development program in this area serves mainly to reduce development cost by pioneering the critical features prior to full-scale development.

Aerocapture and aerobraking - There are six potential functions, given here in approximate ascending order of development risk: aero descent and landing of crew capsules returning from the Moon, aerocapture to low Earth orbit of returning reusable lunar vehicles, landing of Mars excursion vehicles from Mars orbit, aero descent and landing of crew capsules returning from Mars, aerocapture to low Earth orbit of returning Mars vehicles, and aerocapture to Mars orbit of Mars excursion and Mars transfer vehicles.

Efficient aerocapture aerobrakes require low mass per unit area, demanding efficient structures made from very high performance materials as well as efficient, low mass thermal protection materials.

Launch packaging and on-orbit assembly of large aerobrakes presents a significant development risk not yet solved even in a conceptual design sense. Existing concepts package poorly or are difficult to assemble or both.

Nuclear thermal rockets - The basic technology of nuclear thermal rockets was developed and demonstrated during the 1960s and early 1970s. The development risk to reproduce this technology is minimal, except in testing as described below. Current studies recommend advances in engine performance, both in specific impulse (higher reactor temperature) and in thrust-to-weight ratio (higher reactor power density). The risks in achieving these are modest inasmuch as performance targets can be adjusted to technology performance.

Design and development of full containment test facilities presents a greater development risk than obtaining the needed performance from nuclear reactors and engines.

Electric Propulsion Power Management and Thrusters - Power management and thrusters are common to any electric propulsion power source (nuclear, solar, or beamed power). Unique power management development needs for electric propulsion are (1) minimum mass and long life, (2) high power compared to space experience, i.e. megawatts instead of kilowatts, (3) fast arc suppression for protection of thrusters. The unique requirements of electric propulsion introduce development risks beyond those usually experienced in space power systems.

Only ion thrusters and magnetoplasmadynamic (MPD) arc thrusters can deliver the performance needed for space transfer electric propulsion.The development alternatives all have significant risk: (1) Advance the state of the art of MPD thrusters to achieve high efficiency; (2) Develop propulsion systems with large numbers of thrusters and control systems; or (3) Advance the state of the art of ion thrusters to much larger size per thruster.

Nuclear power for electric propulsion - The development risk in this area arises because these are complex systems; there is no experience base for coupling a space power reactor to a dynamic power conversion cycle; there is no space power experience base at the power levels needed; and these systems require in-space assembly and checkout.

Solar power for space transfer propulsion - Required array areas are very large. Low-cost arrays, e.g. $100/watt, are necessary for affordable system costs, and automated construction of the large area structures, arrays, and power distribution systems appears also necessary. Most of the solar power risk appears as technology advancement risk. If the technology advancements can be demonstrated, development risk appears moderate.

Avionics and software - Avionics and software requirements for space transfer systems are generally within the state of the art. New capability needs are mainly in the area of vehicle and subsystem health monitoring. This is in part an integration problem, but new technoques such as expert and neural systems are likely to play an important role.

Environmental Control and Life Support (ECLS) - The main development risk in ECLS is for the Mars transfer habitat system. Other SEI space transfer systems have short enough operating durations that shuttle and Space Station Freedom ECLS system derivatives will be adequate.

Man-Rating Approach; Mission and Operations Risk

These risk categories include Earth launch, space assembly and orbitasl launch, launch windows, mission risk, and mitigation of ionizing radiation and zero-g risks. Results were summarized earlier in this section.

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Lunar/Mars Vehicle Commonality

Both initial and long-term lunar/Mars vehicle commonality have been addressed in the STCAEM contract using the reference MEV, from the 90-day study, as a starting point. In an effort to create a "mini-MEV", which would allow two simultaneous landings on Mars for the same mass as the reference MEV (~ 80 t), a potential early commonality between the "mini-MEV" and an LEV was realized. The commonality between these two landers is evident in both the configuration and the subsystems. Commonality was also realized between the MTV and LTV, although the commonality does not occur as directly as in the lander case. Commonality between the MTV and LTV occurs mainly at the subsystems (avionics, engines, etc.) and fabrication level. Since the mission durations of the two vehicles are substantially different, the transfer habitation modules are obviously different; however, the fabrication technology and module subsystems have potential commonality.

The addition of a 30 t of cargo to the surface requirement for both lunar and Mars missions made the commonality problem substantially different and more complicated. At root is the different gravity fields of the Moon and Mars, and the need to accommodate aerobraking at Mars. The aerobraking complication can be resolved in a number of ways. Four viable solutions were identified and studied through preliminary configurations and mass analysis.

The first solution is to design for a larger aerobrake. This solution requires an answer to the question, "How large an aerobrake is reasonable?" Some innovative operations concepts (like launching an intact aerobrake on the side of a launch vehicle) are much more tolerant of larger brakes than others (like cutting the brake into sections, packing them into shrouds, and assembling them on orbit). The second solution is to allow payload penetration into the wake heating zone by putting a thermal protection shroud over the payload so that the aerobrake can remain smaller. This solution adds complication to the aerobrake as well as mass, which may not be desirable, and cannot avoid some increase in size anyway. A third solution is to alter the flight geometry (fly < 0.5 L/D), which would allow virtually no cross-range, and is not thought to be possible with current GN&C technology. The fourth solution is to design a flatter vehicle to fit within the protected wake cone. This can be done in three different ways: (1) flying no-engine-out when the

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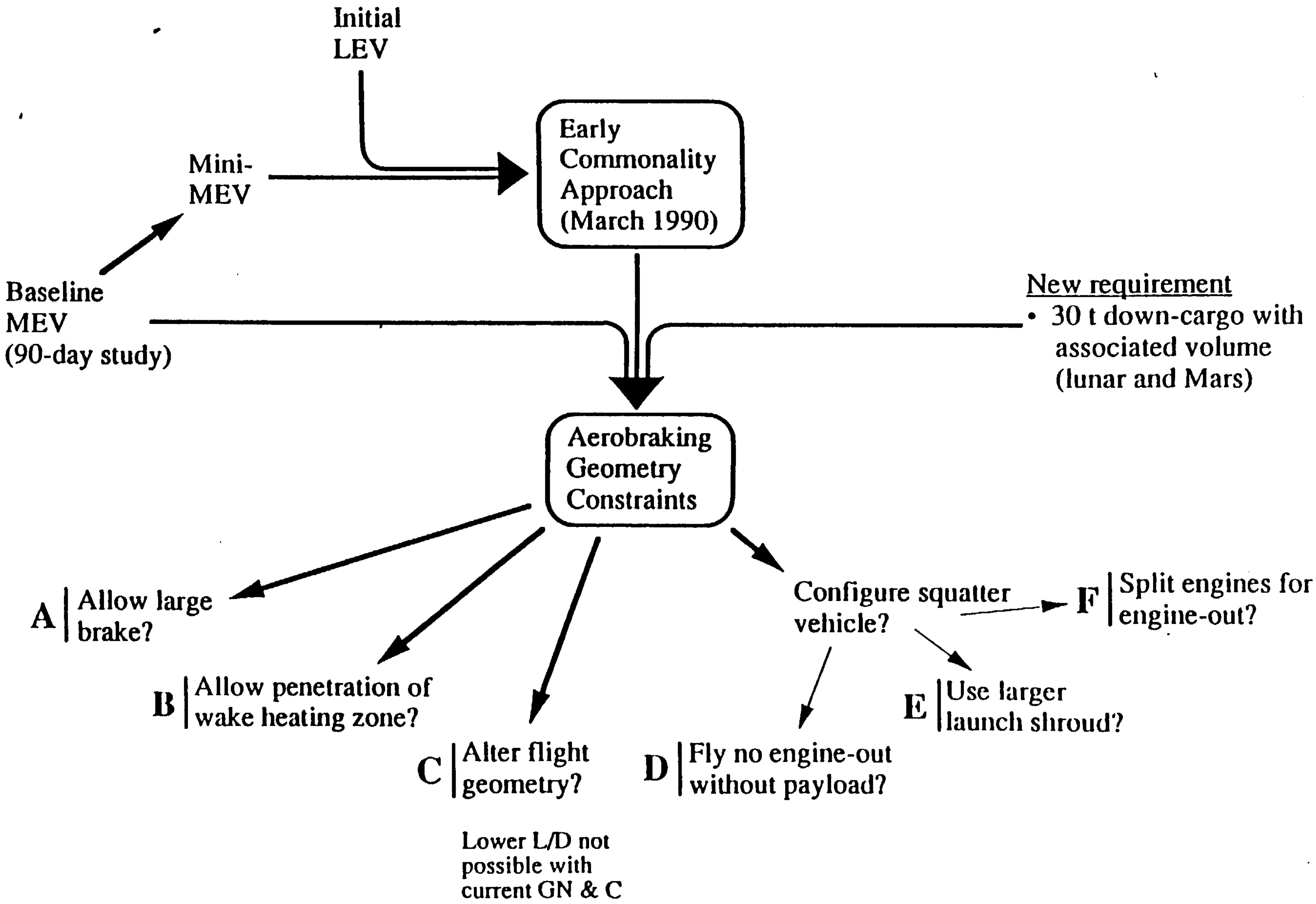
vehicle is empty (the presence of top-loaded payload raises the CM into the gimbal-accessible range), which permits sacrificing the vehicle; (2) using a larger launch shroud (10 m) while still maintaining the ability to launch the propulsion stage intact, which would seem to lead to a squatter vehicle but suffers from the same engine-out problem; (3) or splitting engines geometrically, and increasing their number, to accommodate engine-out through opposing shut-off (this requires 50 % more engines than a clustered configuration).

None of the potential solutions are particularly appealing from a balanced-requirements point of view. Commonality can apply at the level of subsystem, system or entire vehicle. There are many differences between lunar and Mars missions (aerobraking, trip time, environment, etc.) which are difficult, if not impractical, to accommodate simultaneously at the vehicle level. Forcing such extensive vehicle commonality turns commonality from an asset into a driving requirement, resulting in performance liabilities for both lunar and Mars cases. Our conclusion is that commonality is best achieved at the system level (engine, tank, crew system) and subsystem level (avionics modules, ECLSS components, propulsion subassemblies).

Full-sized Lunar/Mars Excursion Vehicle Commonality

The following chart outlines the evolution of our Lunar/Mars commonality assessment. From the baseline MEV (90-day study), common mini-MEV/initial LEV were assessed, and an early commonality approach was studied. The new 30t cargo to surface transport requirement for both lunar and Mars presents many new issues to the forefront. The main issue is the aerobrake geometry, which is outlined with 6 critical questions. The question at this point is which decision path is the most feasible?

Full-size Lunar/Mars Excursion Vehicle Commonality



Initial LEV

Mini-MEV

Baseline MEV (90-day study)

Early Commonality Approach (March 1990)

New requirement

- 30 t down-cargo with associated volume (lunar and Mars)

Aerobraking Geometry Constraints

A | Allow large brake?

B | Allow penetration of wake heating zone?

C | Alter flight geometry?

Lower L/D not possible with current GN & C

D | Fly no engine-out without payload?

Configure squatter vehicle?

E | Use larger launch shroud?

F | Split engines for engine-out?

/STCAEM/sdc/07Sep90

Space Transfer Design for Commonality

This matrix summarizes the required design features, in several subsystem categories, for an evolutionary LTV/LEV system, the mini-MEV, and a small MTV to match what could be applied to achieve the 2010, "easy" opposition opportunity. The right column collects similar requirements into a set of design parameters which would encourage direct commonality between lunar and early Mars missions, based on cryogenic propulsion and aerobraking technology. Aerobrake structures tend not to scale well geometrically, quite apart from the requirement to tailor the structural weight of each to its design payload (so that its mass-reducing benefits can be realized); however, for the purpose of this investigation we have chosen an identical size to work with for both Earth return from the Moon and Mars landing.

Space Transfer Design for Commonality

	LTV	LEV	Mini MEV	2010 MTV	Design Case
Crew cab	- 6 crew, 9 d - rad shelter - partial closure	- 6 crew, 3 d - contam. ctrl. - open ECLSS	3 crew, 10 d - contam. ctrl. - open ECLSS	- 3 crew, 1020 d - rad shelter - closed ECLSS	- 4.4m diameter system - modular lengths - optional rad shelter
Avionics	- Deep space - Orbital - R & D	- Terrain - Orbital - Rend & Dock	- Aerobraking - Terrain - Orbital - Rend & Dock	- Deep space - Aerobraking - Orbital - Rend & Dock	- Modular avionics
Propellant tanks	110t load, cryo	25t load, cryo	- 21t load, cryo - vacuum jackets	140t load, cryo	- 25t tankset, vacuum jacket upgrade - 110t tankset
Engines	3 30klbf 1 engine-out	3 30klbf 1 engine-out	3 30klbf 2 engine-out asc.	3 30klbf 1 engine-out	- 30klbf engine - structure & plumbing
Structure	- strut frame - 7.6m pieces integr. on orbit	- strut frame - 7.6m compat. launched intact	- strut frame - 7.6m compat. launched intact	- strut frame - 7.6m pieces integr. on orbit	- common approach
Landing legs	-------------	22m footprint	16m footprint	-------------	- common approach
Aerobrake	- L/D = 0.2 - 23m length - engine port	-------------	- L/D = 0.5 - 26m length - engine port	L/D = 0.5	- 0.5 L/D shape - 26m length - opt. engine section
Payload accommod.	- mission unique - transferable	- mission unique - transferable	- 1t total - rover & science	- comsats - transit science	- lunar system pallet - Mars unique attach.

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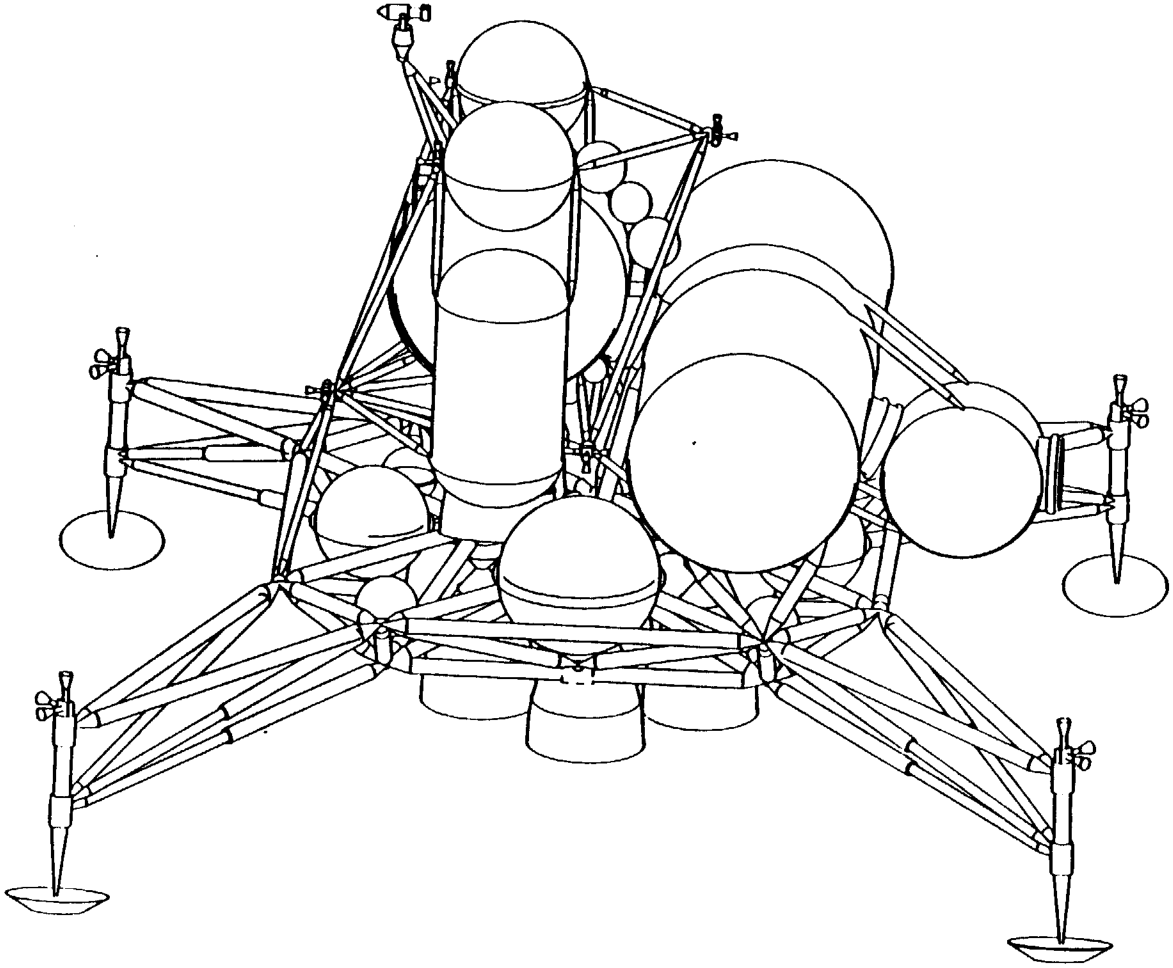
Reference MEV



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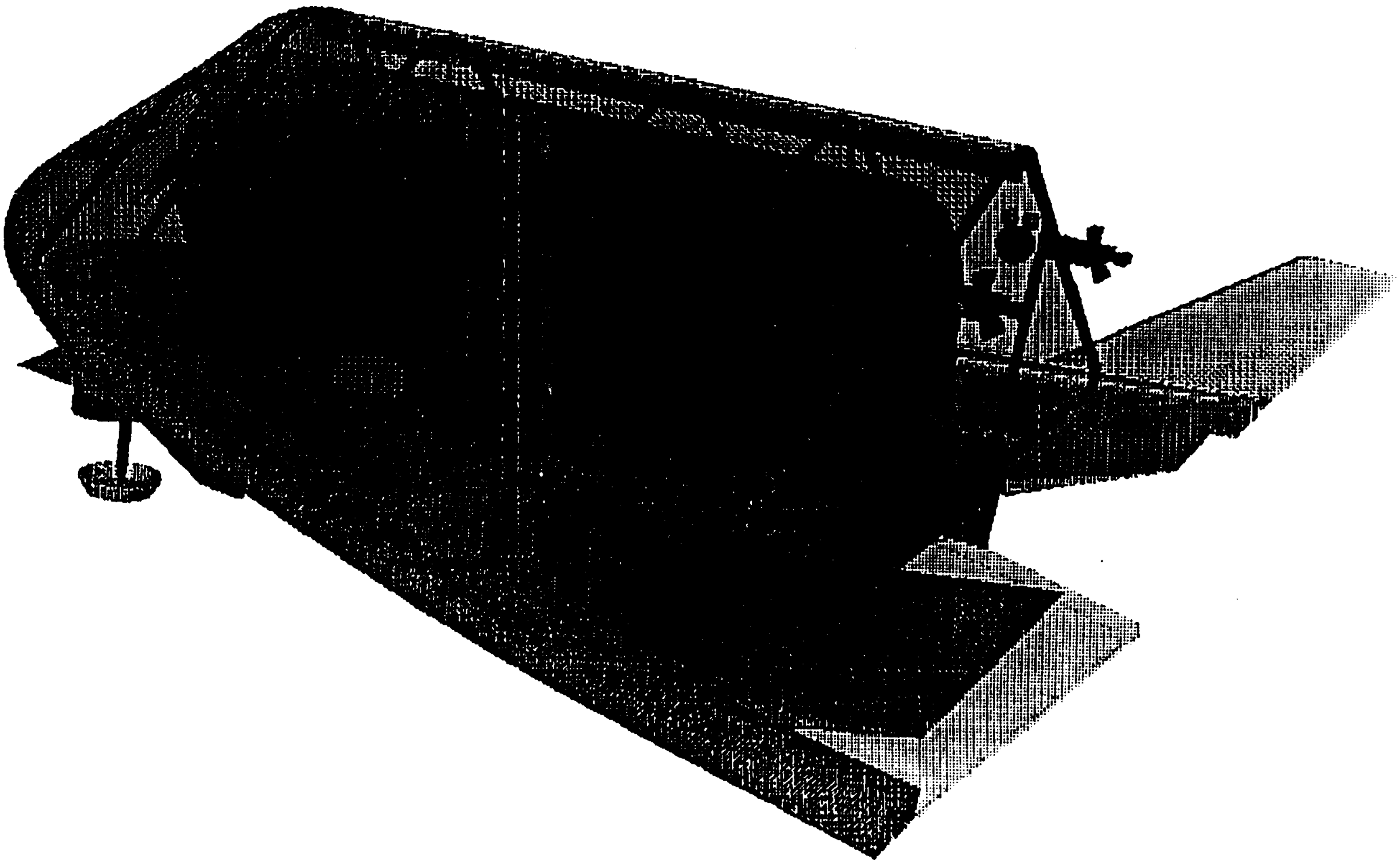
MEV CADD Models

Shown on the following pages are the reference low L/D (0.5) Mars excursion vehicle, and the high L/D (L/D 1.1) reusable MEV. The low L/D MEV acommodates a crew of 4 to 6 for a thirty day surface mission, and returns the crew to mars orbit via a separate ascent stage(MAV).

Evolution of the high L/D MEV would follow for later, more aggressive missions. The high L/D vehicle has been configured in an expendable cargo version, and in a crew/reusable version that refuels with mars surface produced propellant

High L/D MEV Configuration



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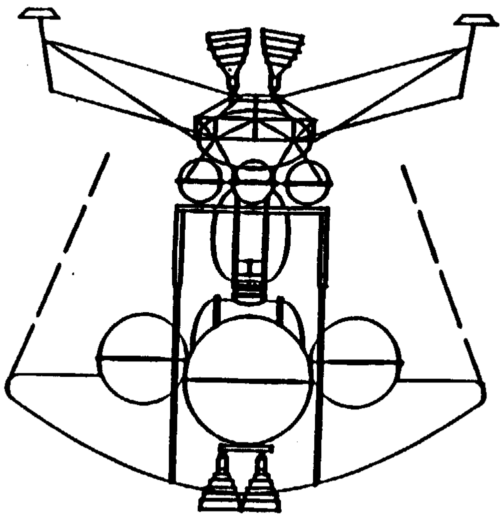
LTV/LEV Configuration

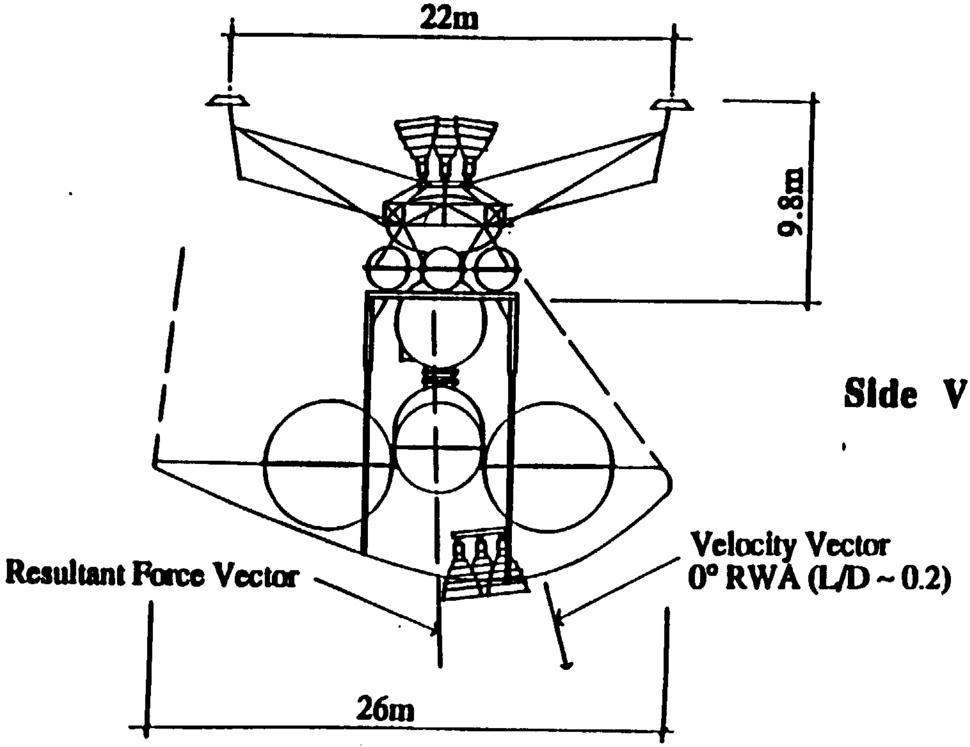
This and the next chart sketch configuration concepts for the cis-lunar system which uses the common parameters just developed. Sized for the Mars case, the aerobrake is somewhat larger than strictly required by the lunar mission, despite retaining all propellant tanks throughout the mission profile. Furthermore, it has the higher-L/D shape required for Mars, albeit flown here in a lower L/D attitude. The LTV engines, while oriented to accommodate the vehicle's changing mass center, are positioned according to Mars landing requirements. Direct transfer of crew from LTV to LEV is accomplished in the same configuration as propellant transfer (whether pumped with rotational settling or transferred using a μg technique).

A single, unconstrained payload pallet is transferred at this time also. The pallet can be integrated at SSF separately from the LTV processing, then mounted for TLI, transferred to the LEV, and unloaded on the surface by a straddling payload transporter. The LEV's height is reduced as much as possible, given the constraint of engine-out on ascent, to maximize unloading efficiency. The landing gear would permit settling the LEV lower to the ground after touchdown to facilitate unloading as well, and are configured in plan to accommodate a triangular straddler. The center section of the pallet is removable, and passes over the LEV-mounted crew module for cargo transfer during crew missions (heavy, bulky, singular payloads like habitat modules or process reactors cannot be accommodated on crew flights due to mass capacity considerations anyway), allowing manifesting of resupply cargo. For unmanned cargo flights, the full payload pallet would be used. The pallet retracts close to the LTV tanks for the aeromaneuver upon return to Earth.

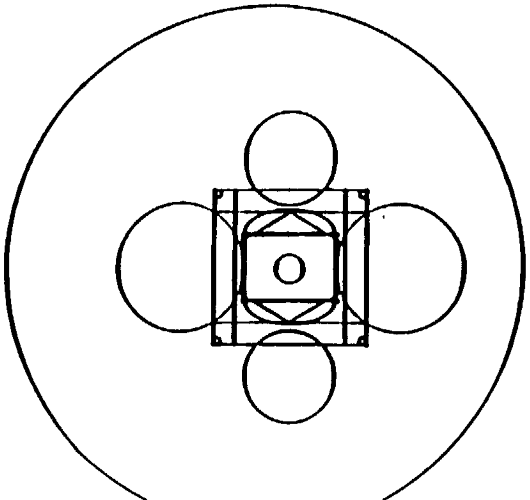
LTV/LEV Configuration



Front View



Side View



Top View (LTV)

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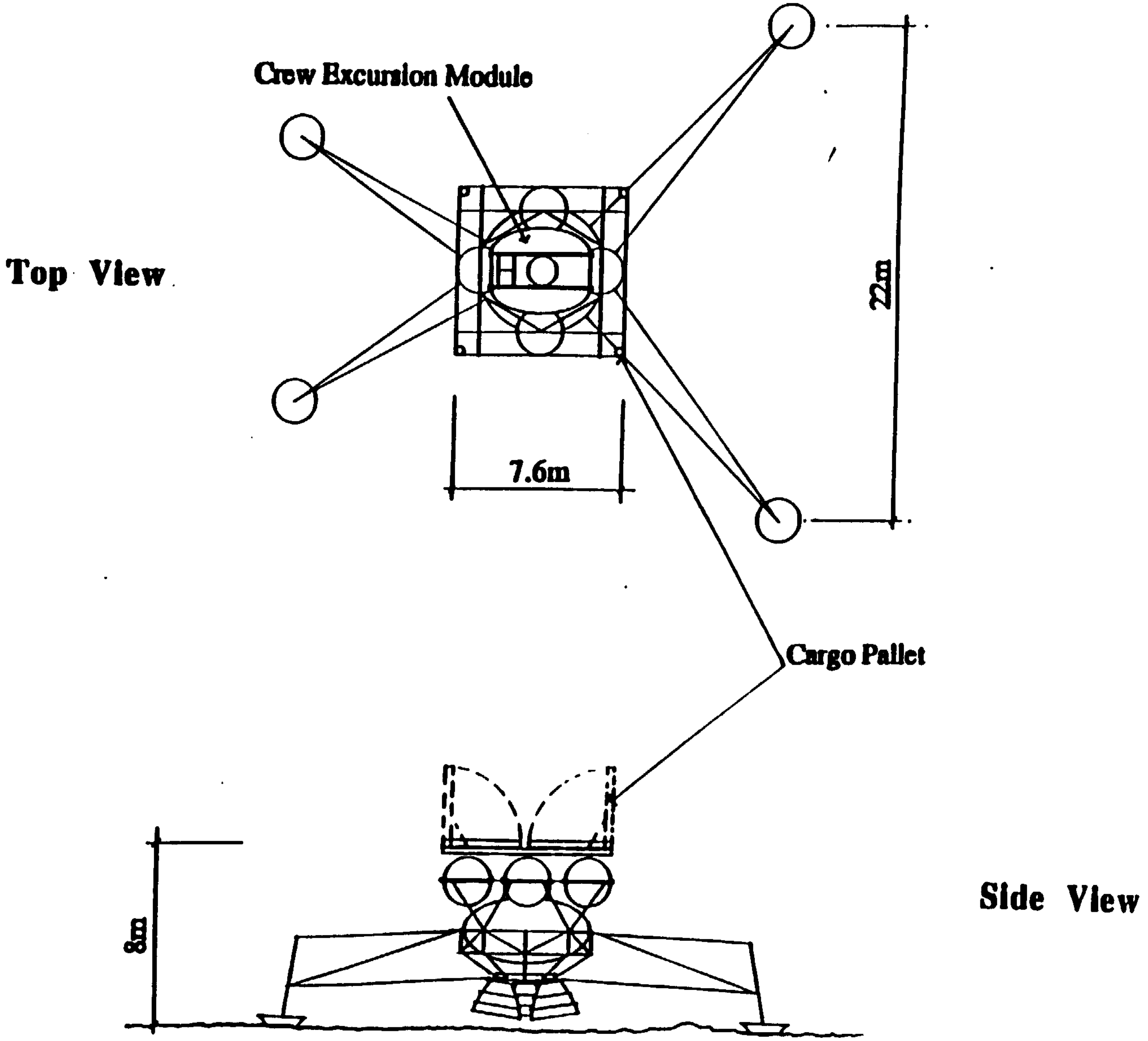
LEV Surface Configuration

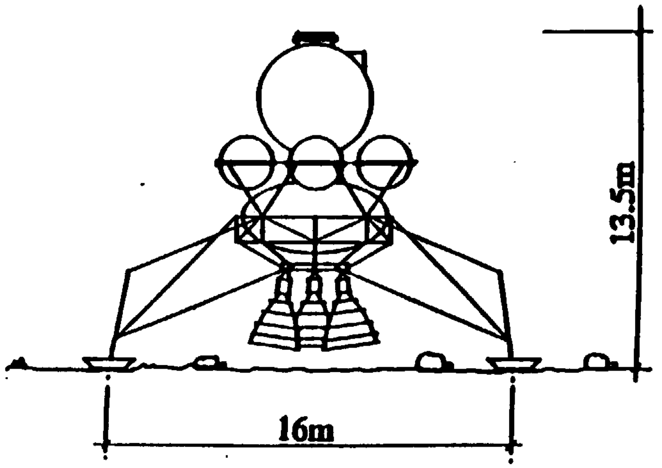


Mini-MEV

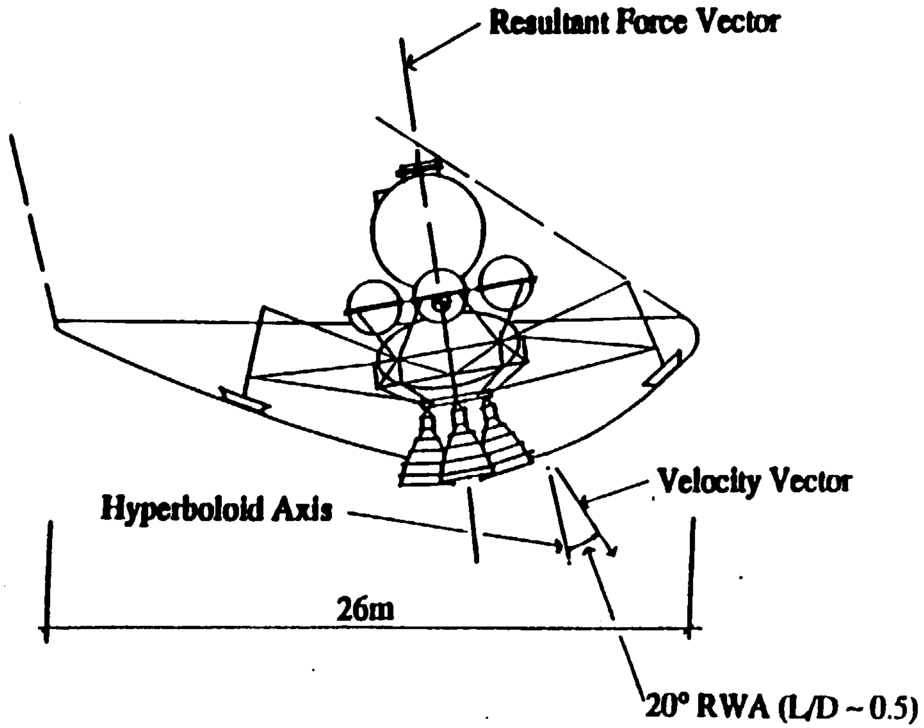
Shown here is a configuration sketch for the mini-MEV, designed according to the same parameters as the evolutionary lunar system just described. The aerobrake is as small as it can be, given the already-minimized MEV height and a requirement for $L/D = 0.5$.

Commonality, exercized through the maturation of a system into meeting the performance requirements for a later mission, has great potential to keep program costs down. It requires pulling a reference concept toward distinct performance goals to develop approaches capable of satisfying both.

Mini-MEV



Surface Configuration

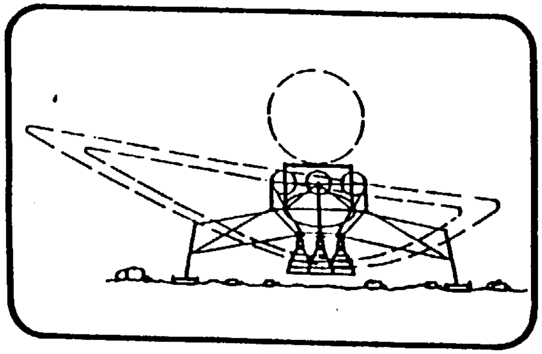


Aerobrake Configuration

Common Lunar/Mars Vehicle Configuration Options

This chart shows configuration options which tie in to the decision tree questions asked previously. The configurations show the impact to the overall system depending on the decision path taken.

Common Lunar/Mars Vehicle Configuration Options

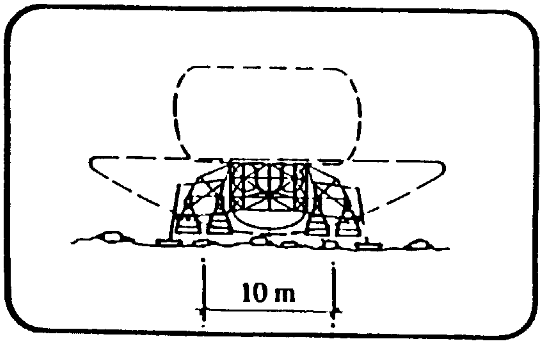


A

- 40 m aerobrake required

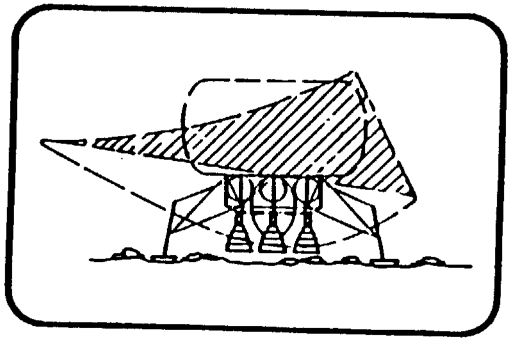
F

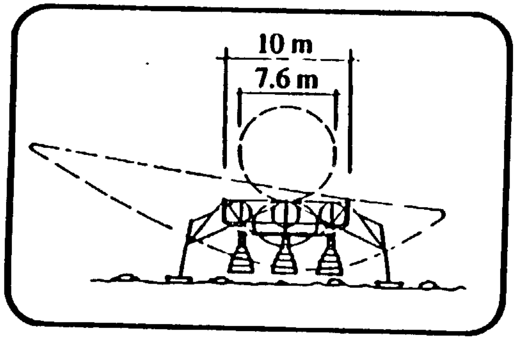
- Relax central engine cluster constraint
- Opposing shut-off for engine-out
- 4 engines required
- 30 m aerobrake required



B

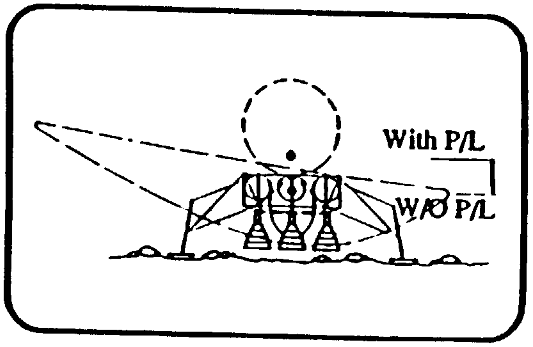
- Use topside thermal protection shroud
- 31 m aerobrake required





E

- Relax 7.6 m launch shroud constraint
- Use 10 m shroud
- Early lunar use questionable
- 35 m aerobrake required



D

- No engine-out without payload
- 35 m aerobrake required

Common Mars/Lunar Lander

The objective here was to design a common Mars/Lunar lander that could operate either as an unmanned cargo carrier, or as a piloted vehicle carrying personnel and cargo to and from the surface. The primary design criterion was to keep the complete propulsion stage identical for all vehicles. The vehicle inert weight is a function of several requirements, foremost among these being the propulsion requirement to provide a minimum vehicle T/W ratio of approximately 1.6 throughout the landing and ascending phases with an engine out margin. Other important parameters include the stage structural frame and landing leg weight, both of which are a function of the load that each must support. Inert weight is also a function of the number of tanks and their size as selected for both the MPS and RCS propellant loads which include boiloff allowances for surface stay time in the case of the piloted sorties. For this preliminary design, the decision was made to size the common inert stage based on the piloted Mars lander mission with a 30 day surface stay time. A total cryogenic propellant load of 31 tons was necessary for this case. To provide a T/W of 1.6, three 30k lbf advanced cryogenic engines (Isp=475) were selected. One large fuel tank, and 4 small oxidizer tanks were selected and sized to hold the 31 ton maximum. Adding tank insulation, meteor shields, VCS, propellant line wts, frame structure, landing legs and mass growth allowance to the engine/tank set produced a total stage inert weight of 7.4 t. The following comments will serve as a brief description for the 6 vehicles given summary weight statements on the following chart.

Column (1): Mars cargo: 7.5 t aeroshell & 30 t cargo to the surface requires only 8.1 tons of prop (1/4 of tank vol) .

Column (2): Mars piloted with 1 year stay: 7.5 t aeroshell, 4.8 t ascent cab (crew of 6), 500 kg of cargo, and 1 year boiloff allowance requires 37 tons total desc & asc prop, which is *above* the common stage tank set load capacity, thus *excluding it from the group sharing the common propulsion stage*.

Column (3): Mars piloted with 30 day stay: requires 31 t prop; selected as the capacity for sizing the tanks for the common stage.

Column (4): Lunar cargo: 30 tons to the surface implies off loaded tanks: only 21.5 tons propellant required (2/3's full)

Column (5): Lunar cargo: 45 tons to the surface are possible if the tanks are at the 31 ton maximum capacity.

Column (6) Piloted Lunar: ascent cab with 23 tons of surface cargo are possible if tanks are filled to their 31 ton capacity.

For the Mars missions a 7.5 t aeroshell decelerates the craft for the majority of the descent dV. Part way through this aerobraking phase the nozzles of the 3 engines are extended through the brake doors. Supplemental braking is provided by these engines until the brake drops off from its own weight, after which the engines alone provide terminal descent to hover and final touchdown. The Lunar cargo case is identical to the Mars cargo case excepting the use of the aeroshell. For the manned sorties the lander functions as a single stage descent/ascent vehicle. The entire vehicle ascends to orbit leaving behind only the surface cargo for the piloted lunar case, while leaving the landing legs as well for the piloted Mars case.

Common Lunar/Mars Lander Mass Statement Cargo and Manned Versions

Mars desc propul ΔV: 773, Asc ΔV: 5319, Lunar asc ΔV: 2100, desc ΔV: 2000, all cryo prop Isp=475
***Piloted vehicles (single stg)** Mars: aeroshell, cargo & landing legs left on surface, Lunar: cargo left on surf*
***Cargo landers (unmanned)** Mars: aerocapture, aerobrake & propulsive descent, Lunar: all propulsive*
RCS ΔV: Mars: desc 100 m/s, asc 35 m/s, Lunar: desc 35 m/s, asc 35 m/s

Element *Stay time*	Mars *Cargo (desc only)* *n/a*	Mars *Manned (single stg)* *365 days*	Mars *Manned (single stg)* *30 days*	Lunar *Cargo (desc only)* *n/a*	Lunar *Cargo (desc only)* *n/a*	Lunar *Manned (single stg)* *6 months*
Ascent cab	0	4847	4847	0	0	4847
Stg inerts	*7396	8656	*7396	*7396	*7396	*7396
Aeroshell	7500	7500	7500	n/a	n/a	n/a
Surf Cargo	30000	500	500	30000	**45000	**23000
Asc prop	n/a	28001	23389	n/a	n/a	7193
Desc prop	8125	9188	7713	21502	30126	24277
RCS prop	1704	2121	1854	705	988	914
Total kg mass	**54725**	**60810**	**53227**	**59602**	**83510**	**67627**

Manned: crew of 6, immediate transfer to surf hab module
Stay time: for Asc stg propellant boiloff calculation only
** *Maxium surface cargo load for these lunar cases when all tanks are full (not off loaded as in column 4)*
* *Commonality across vehicles is realised by using a common propulsion stg: identical engs, structure, MPS & RCS tank set differences consist in: aerobrakes, cargo load, tank prop load (off loaded tanks in columns 1 & 4)*

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Options / Alternatives

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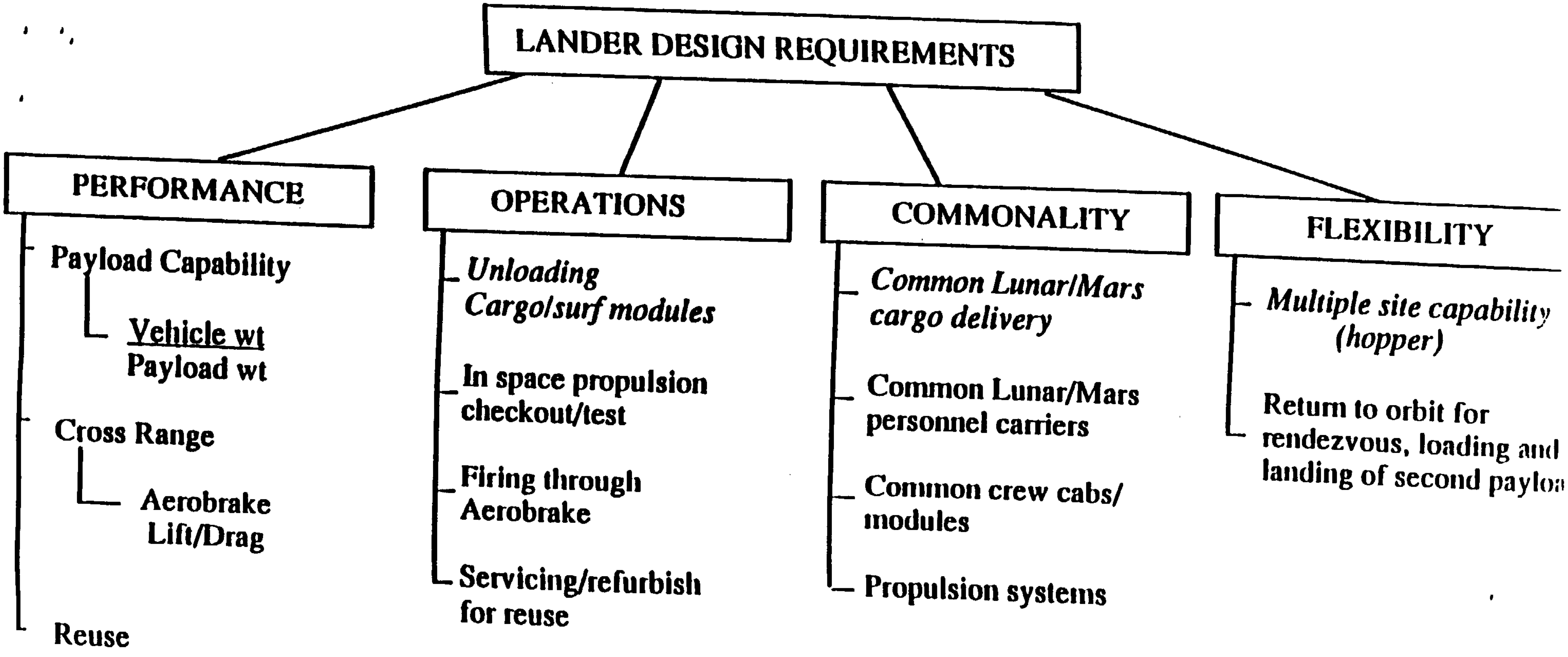
Mars/Lunar Lander Trade Categories

LEV/MEV design requirements are represented under four categories, as shown below. The items in italics are presented in this section.

Mars/Lunar Lander Trade Categories



LANDER DESIGN REQUIREMENTS

PERFORMANCE

- Payload Capability
  - Vehicle wt / Payload wt
- Cross Range
  - Aerobrake Lift/Drag
- Reuse

OPERATIONS

- *Unloading Cargo/surf modules*
- In space propulsion checkout/test
- Firing through Aerobrake
- Servicing/refurbish for reuse

COMMONALITY

- *Common Lunar/Mars cargo delivery*
- Common Lunar/Mars personnel carriers
- Common crew cabs/ modules
- Propulsion systems

FLEXIBILITY

- *Multiple site capability (hopper)*
- Return to orbit for rendezvous, loading and landing of second paylo

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Unloading of Large Cargo payloads

The split payload LEV/MEV designs are inhibited by the fact that the two cargo sections must be relatively equal in weight, otherwise the vehicle CG for the descent phase of the mission may be significantly out of line with the thrust centerline of the engines. Some of the impacts involved with handling and removing large payloads from these split cargo landers are given below with several configuration sketches given as examples.

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Unloading of Large Cargo Payloads

Popular 'side loaded' cargo designs seem to favor the piloted mission application, with these veh's modified to accommodate cargo for the heavy cargo mission applications — **BOEING**

Off loading characteristics of example LEV designs shown:

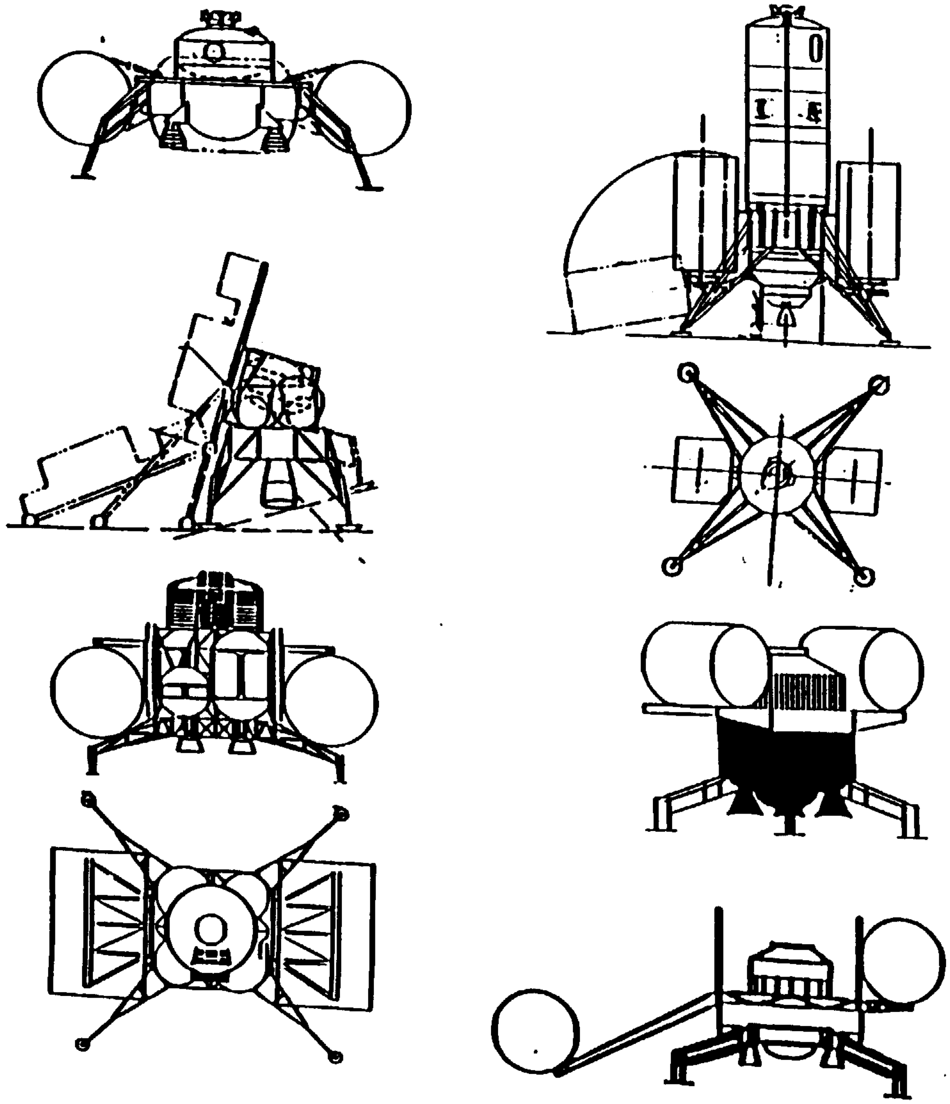
I. Payload must be divided into two equally heavy sections for vehicle CG considerations.

II. Payload off loading requires that movable ramps or large hinged side mounts must be built on to vehicle

III. Additional off loading veh's required Cranes, straddlers, or other lifting off loaders that are necessary for top loaded (cargo) veh's may possibly be necessary for these side loaded vehicles as well

IV. Emergency ascent from surface difficult. In an emergency, these landers can't ascend to orbit w/o necessary offloading steps which may include a series of time consuming manual RMS tasks

V. Unloading failure grounds vehicle. If a cargo release/unload mechanism fails, veh will be incapable of operation as an ascent veh for return to orbit. Simplifing the off loading task would be highly beneficial in this respect



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LEV/MEV Cargo Unloading Issues

All lander designs can not be serviced by the same surface cargo unloader vehicles. The top mounted cargo landers typically require a large crane unloader or a large gantry type lifter that can be driven over the completer lander/cargo combination for removal. The side loaded and undercarriage cargo lander designs require flat bed transporters that can be positioned under the cargo.

Lander Operations Trade - Common Lunar/Mars Landers

Cargo Unloading Characteristics of Candidate Designs

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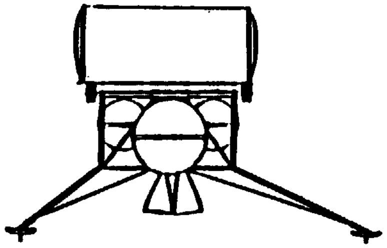
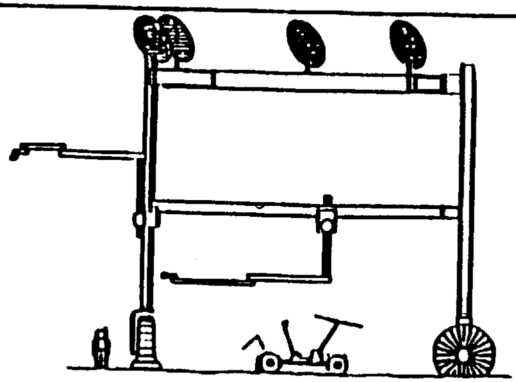
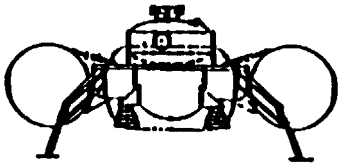
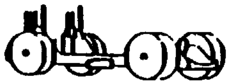
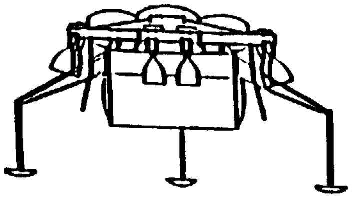
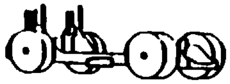
Lander type	description	Vehicle sketch	Unloader sketch	Unloader type, wt., dedicated flights to deliver unloader
Top loader	*Carries large cargo loads or combination cargo and/or crew cab on top*			*'Straddler'; gantry type overtop lifter & transporter self propelled: solar panels or fuel cells. wt: 12 tons, 1 dedicated flight*
Side loader	*Cargo load must be split into relatively equal sections. Crew cab on top. Primarily designed as a piloted vehicle, modified to accommodate cargo*			*Flat-bed transporter; cargo lowered onto carriage, pulled to location by rover, or self propelled. lifting crane may be necessary for these side loaders. wt:3 tons, carried as 1 side load*
Bottom loader	*Large cargo load, or smaller cargo module & crew cab, carried underneath*			*flat-bed transporter; lander design optimized for ease of, cargo unloading, possibly eliminates need for any lifting cranes for base buildup of hab modules. 3 tons, no dedicated flights necessary; carried with cargo underneath*

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Undercarriage Cargo/Crew Cab LEV/MEV

The evaluation of a lander concept that features the undercarriage placement of cargo was begun after apparent operational difficulties surfaced in evaluations of some current LEV/MEV designs. The simplification of unloading operations was the motivation for this analysis. Several key distinguishing features are listed at the top of the following chart, along with a summary weight statement of major elements and a preliminary sketch.

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Cargo Delivery Emphasis-Common Mars/Lunar Lander

Undercarriage cargo/uptop engine position LEV/MEV

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Vehicle Characteristics:

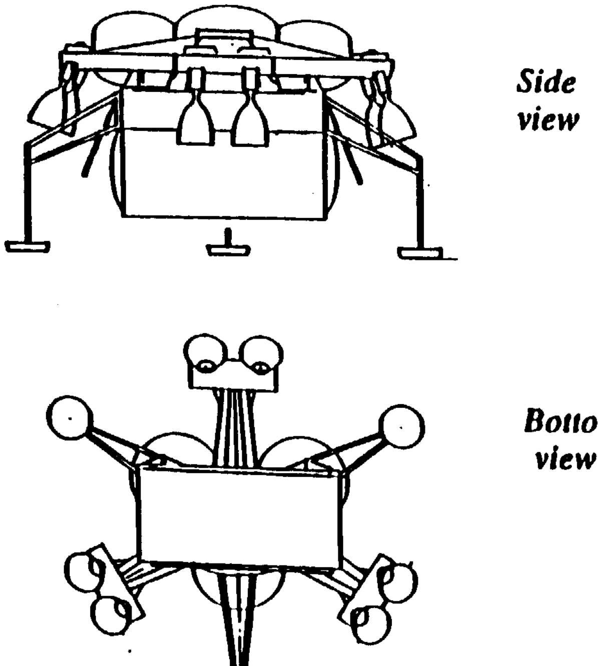
Underneath payload position:
- Eliminates need to split payload into two equally heavy sections
- Ease of payload off-loading; movable ramps, side rails on vehicle, not required
- No overtop lifting/off-loading veh's required; surf based lifting cranes, straddlers, etc.
- Emergency ascent from surface simplified: one step cargo drop minimizes preparation time for unplanned ascent to orbit
- Lower payload (low CG) allows the use of a 3-leg rather than the 4-leg landing gear set.
- Contiguous crew mods for piloted case allows asc crew cab & surf crew hab to be linked together.
- improved ground visibility for piloted missions.

Over top engine position :
- 2 engine out margin; six engines are paired on each of 3 extensions. Slight outward cant at nominal engine position plus impingement shield prevent exhaust plume impingement on cargo.

Weight statement

	Lunar cargo vehicle	Mars cargo vehicle	Lunar piloted	Mars piloted
Return to orbit	yes/reuse	no/expend	yes/reuse	yes
Surf cargo	30000	30000	20000	20000
Asc crew cab	0	0	3500	3500
Asc RCS Prop	128	0	178	277
Asc prop	4510	0	6473	24333
Desc RCS prop	739	1664	660	2575
Desc prop	24655	7293	21996	12962
Fuel tank & inert	1439	580	1433	1722
Ox tank & inert	1113	343	1097	1278
Engines (6)	1140	1140	1140	1140
Total RCS inert	399	540	396	579
Frame structure	2500	2500	2500	2500
Landing legs	1261	1080	1125	1665
Weight growth	795	925	774	633
Aerobrake	0	7500	0	15138
Total	68679	53594	61271	87862

Vehicle sketch



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Disk #7/Cargo emphasis com M/L lander

Unloading Illustrations

The chart shown below illustrates the positioning of surface transporter for unloading of cargo

ADVANCED CIVIL SPACE SYSTEMS

Common Mars/L Lander - Cargo Delivery Emphasis

Operational simplicity & efficiency is a key element in cargo & crew+cargo LEV/MEV design

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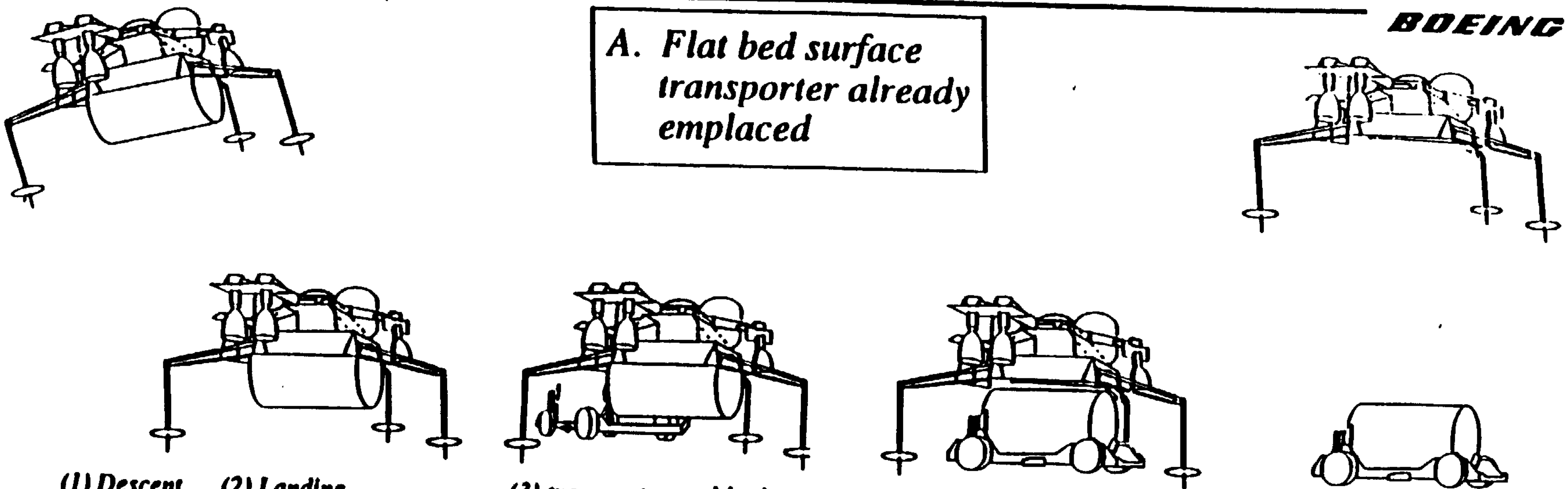
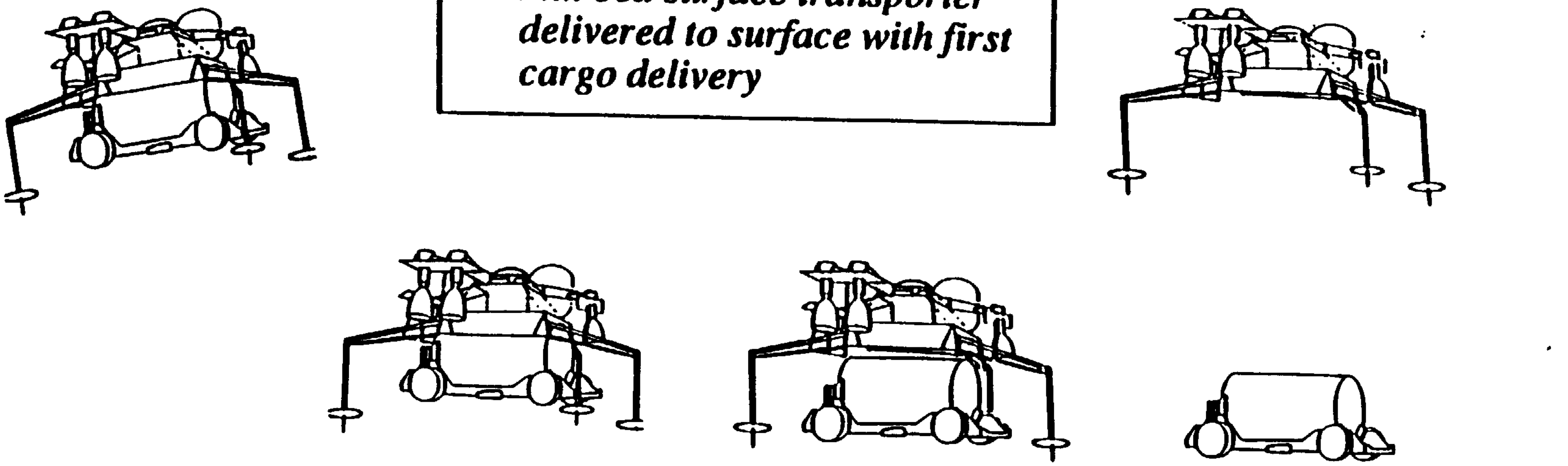


A. *Flat bed surface transporter already emplaced*

(1) Descent (2) Landing (3) transporter positioning (4) Release payload (5) Ascent to orbit



B. *Flat bed surface transporter delivered to surface with first cargo delivery*

(1) Descent with transporter (2) Landing (3) Release payload (4) Ascent to orbit; repeat (lunar)

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Lunar/Mars Surface to Surface Suborbital Transfers

A suborbital transfer is defined as a series of maneuvers targeted to transfer a LEV or MEV from an initial surface location to a final location. Two different approaches are presented on the following two charts.

The first approach, entails executing a series of powered maneuvers to transfer from an initial basepoint on the planet surface, through a specified surface relative transfer angle, to another location on the planet surface. The first burn of this burn-coast-burn scenario achieves the required apoapsis altitude and the final burn achieves the required constraints for landing at the hover altitude. In this case the apoapsis of the transfer orbit is optimized as a function on the desired downrange transfer angle - this is labeled as the Two Burn, or Multiburn Transfer.

The second approach consists of a single-powered maneuver where the LEV/MEV will hover and horizontally transfer through the desired transfer angle - this is labled as the Hover/Horizontial Transfer (HHT).

DV is plotted vs surface transfer range for both maneuver approaches. A chart of the Lunar and Martian surface transfers as used in the second site lander analysis is given in the following charts. This description is a summary of chapter 5 of *Lunar/Mars Common Vehicle Study*, Mission support directorate, Sept 1989 NASA JSC

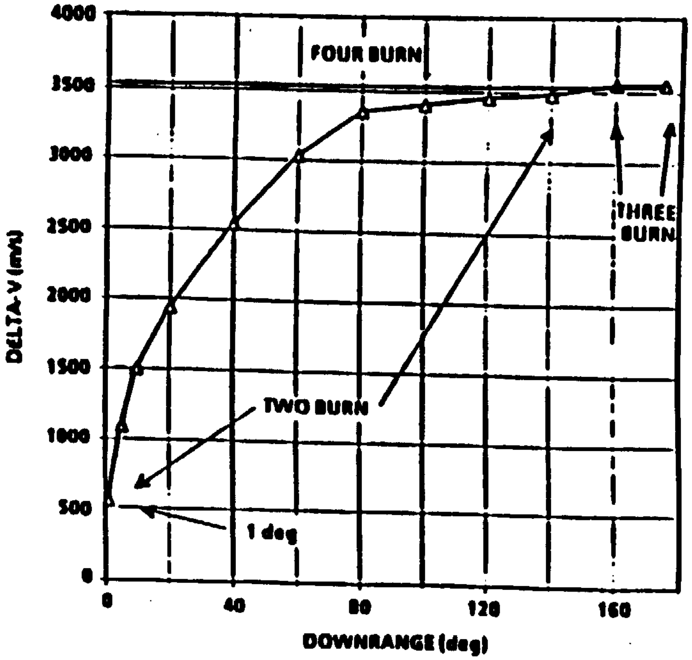
Lunar Surface to Surface Suborbital Transfers

for combination Lander/Hopper Vehicle Second Site Exploration Sorties

Vehicle 'hop' dV (m/s) vs translation range (km)
(T/W = 1.5, Isp=470 sec, W/CdA = 500)

A. Two burn and Multiburn Transfers

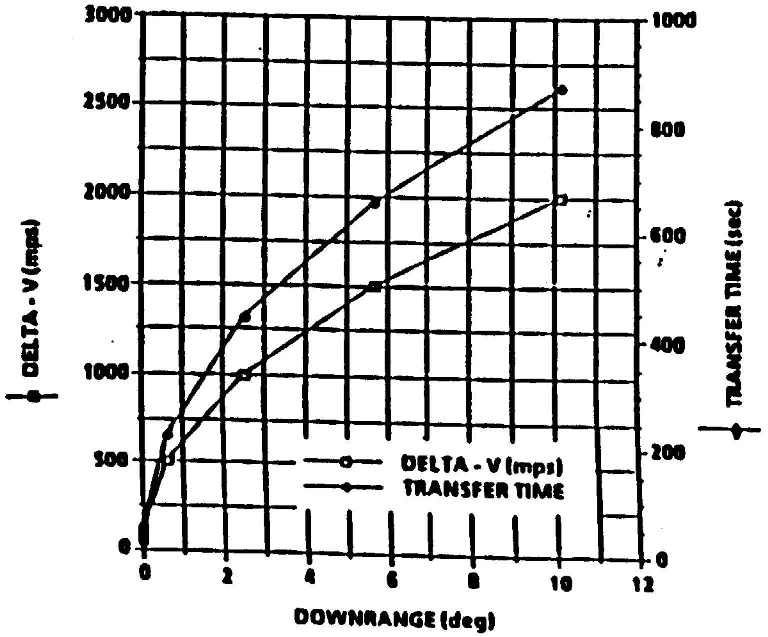
Ascent burn - coast - descent/touchdown burn; Two burn/multiburn more efficient than alternate approach of Hover/horizontal transfers, which require a single constant burn from liftoff to touchdown without a coast period.



Downrange: Lunar: 1 deg = 30 (km)

B. Hover/Horizontal Transfers (HHT)

Ascent to constant hover altitude, with use of a component of thrust acceleration to achieve a desired horizontial acceleration. This acceleration reversed at the appropriate time to slow the vehicle to stationary hover again at the desired terminal location.



Downrange: Lunar: 1 deg = 30 (km)

Data from *Lunar/Mars Common Vehicle Study*, Mission support directorate, Sept 1989 NASA JSC

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ADVANCED CIVIL
SPACE SYSTEMS

Martian Surface to Surface Suborbital Transfers

for combination Lander/Hopper Vehicle Second Site Exploration Sorties

BOEING

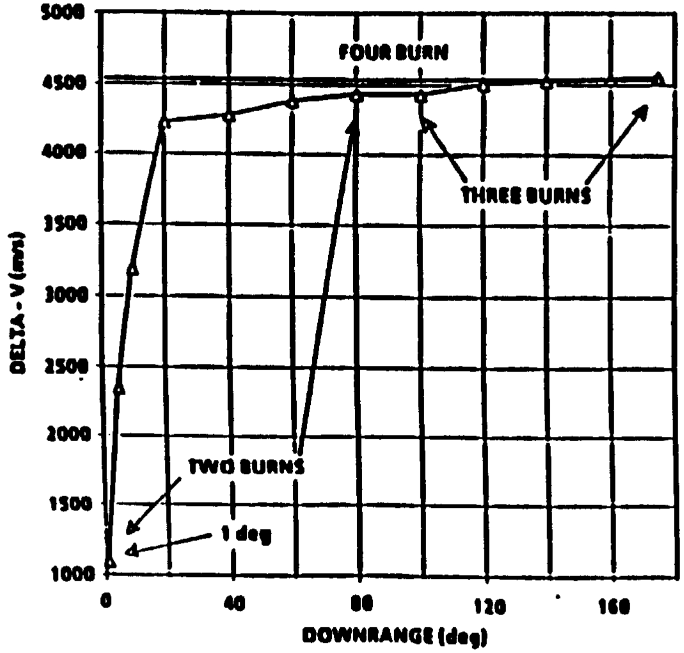
Vehicle 'hop' dV (m/s) vs translation range (km)
(T/W = 1.5, Isp=470 sec, W/CdA = 500)

A. Two burn and Multiburn Transfers

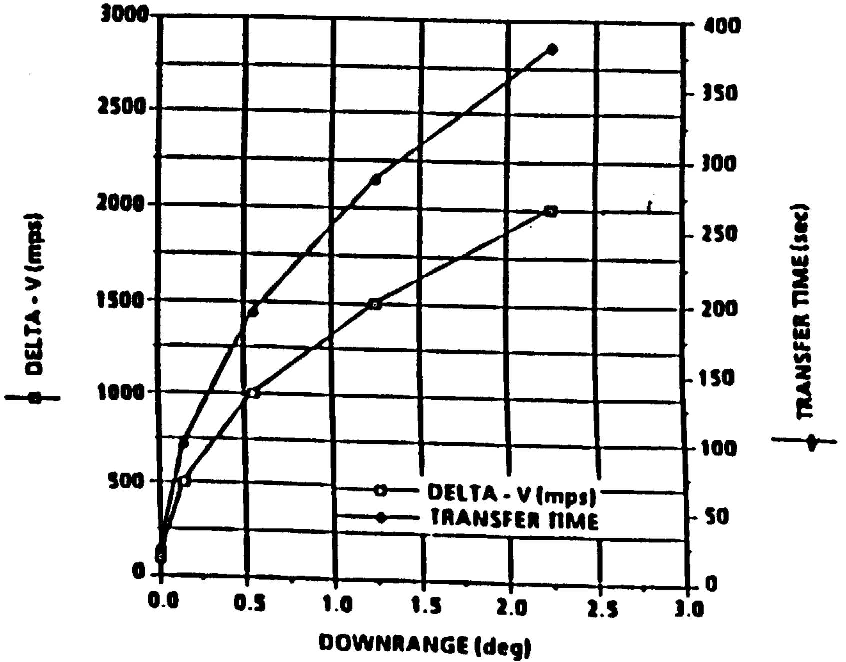
Ascent burn - coast - descent/touchdown burn; Two burn/multiburn more efficient than alternate approach of Hover/horizontal transfers, which require a single constant burn from liftoff to touchdown without a coast period.

B. Hover/Horizontal Transfers (HHT)

Ascent to constant hover altitude, with use of a component of thrust acceleration to achieve a desired horizontial acceleration. This acceleration reversed at the appropriate time to slow the vehicle to stationary hover again at the desired terminal location.



Downrange: *Mars: 1 deg = 60 (km)*



Downrange: *Mars: 1 deg = 60 (km)*

Data from *Lunar/Mars Common Vehicle Study*, Mission support directorate, Sept 1989 NASA JSC

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Lunar/Mars Lander with Second Site Exploration Capability

Once the 90 day study 84 ton reference MEV was presented (see first quarterly briefing), a 'mini' MEV was configured to provide a Mars mission vehicle with the capability to visit 2 sites for the same lander mass by providing 2 of these smaller landers each at one half the mass of the large ref MEV, though obviously with much less science capability and shorter surface stay times. Another approach aimed at increasing exploration potential for a given payload mass delivered to Mars orbit is the development of a lander capable of visiting two sites before ascending back to orbit. Surface 'hopping' (a suborbital flight of a few hundred kilometers) capability would provide both the desirable 15-25 day surface mission (based out of a large surface hab module of ~20 tons) typical of the reference 30 day MEV mission, as well as a short 3-5 day duration exploration of a second site (typical of the 'mini' MEV mission). Because of the heavy crew module requirement for the initial surface stay, the primary design objective was directed towards satisfying the dual requirement for an efficient cargo delivery system that could also serve effectively as a 'hopper' for short duration exploration.

The Undercarriage cargo lander concept, designed to simplify cargo unloading operations at the surface, was utilized in its piloted/cargo configuration for this dual task. A small 3.5 ton asc crew cab is linked to the ~20 ton surf crew module, and both are carried under the vehicles propulsion/frame carriage for the descent and initial surface stay phase. The large surf module is then lowered and released from the propulsion carriage, the asc crew cab remaining in position underneath. Now a factor of 2 lighter, and with a T/W a factor of two higher compared to its earlier initial descent from orbit, the vehicles suborbital flight to site two is undertaken.

vehicle characteristics advantageous for this second site capability are given below:

• Surf module lowered and released w/o the aid of unloaders • Vehicle T/W at 'hop' flight substantially higher than the 1.6 required for initial descent; • Engine out margin of at least 2 for 'hop' flight desirable - provides added confidence for higher risk flights into canons or on to mountain plateaus.

Lunar/Mars Landers Second Site Exploration Capability

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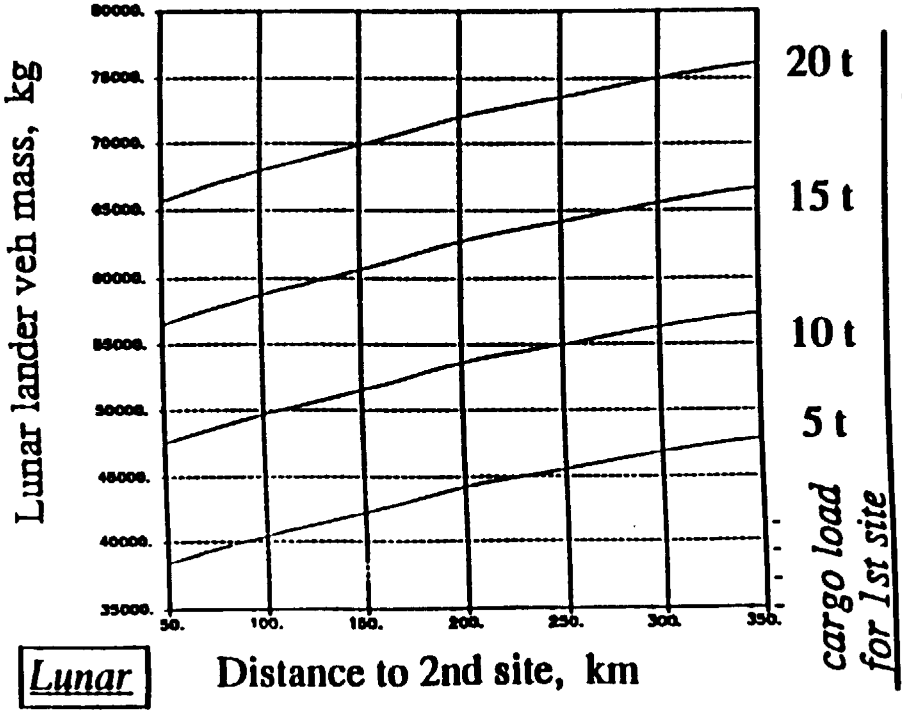
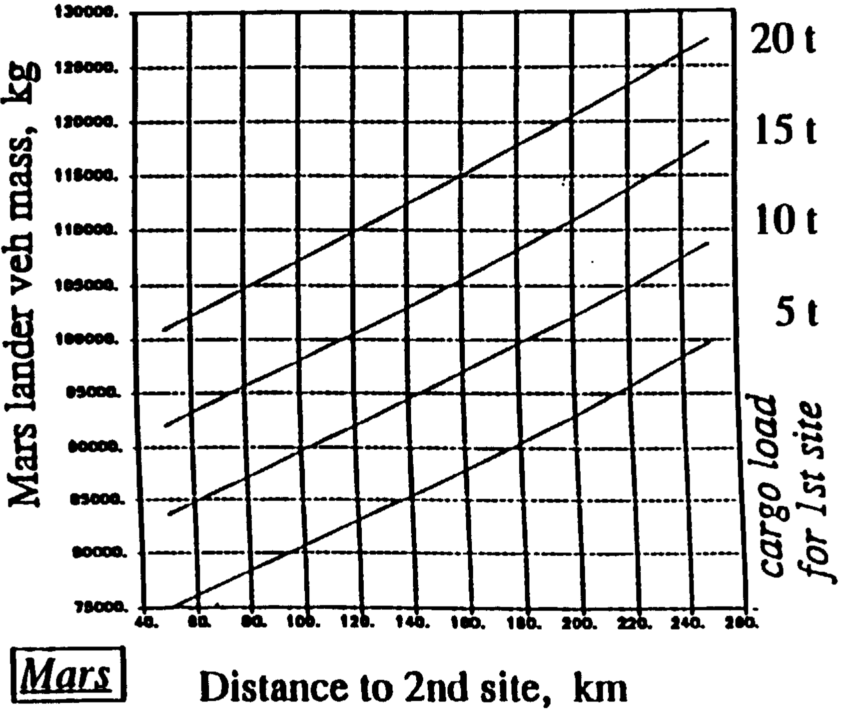
After initial surf stay, veh drops surf hab & flys to 2nd site for short stay

Lunar (left) & Mars (right) lander total veh wt vs 2nd site 'hop' range (distance between sites)

BOEING





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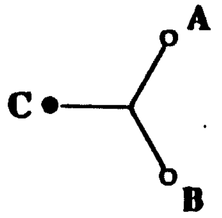
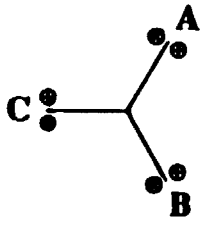
Outrigger Lander
Configuration Options

Engine-Out Recovery Analysis

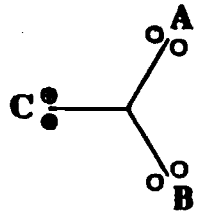
3 Outrigger Configuration



- 3 engines
- Engine-out not possible

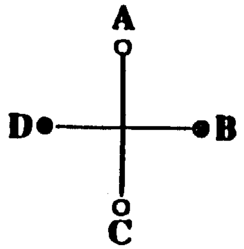


- 6 engines
- 2 engines (1 @ A & B) shut-down
- Throttle remaining engines (@ A,B & C) to 100%
- No gimballing required

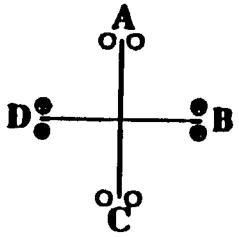


- 6 engines
- Remaining engine (@ C) throttled-up to 100%
- No gimballing required

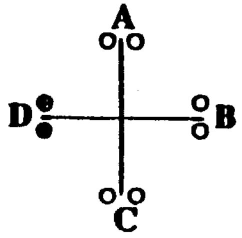
4 Outrigger Configuration



- 4 engines
- 1 engine (@ B) shut-down
- Gimballing *or* RCS required for control



- 8 engines
- 1 engine (@ B) shut-down
- Remaining engines (@B & D) throttled-up to 100%
- No gimballing required



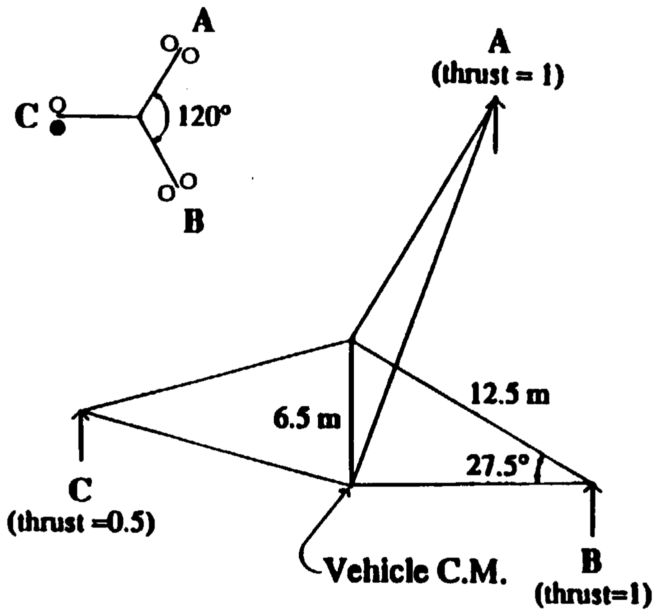
- 8 engines
- Remaining engine (@ D) throttled-up to 100%
- No gimballing required

○ Engine 100%
● Engine-Out
⊛ Engine Shut-Down
● Engine Throttle-Up

/STCAEM/sdc/08Oct90

3 - Outrigger Engine-Out Analysis

6 - Engine Triangular Configuration



- Engine gimballing *through* C.M. is not possible

Rationale:

Engine loss at pod C:

- All engines run nominally at 50%
- Remaining engine at C must throttle-up to 100%
- Gimballing capabilities *do not* ease throttling

Conclusion:

- Engine gimballing has no affect on vehicle stability
- All vehicle steering can be done with throttling, taking *advantage* of "outrigger" moment arm

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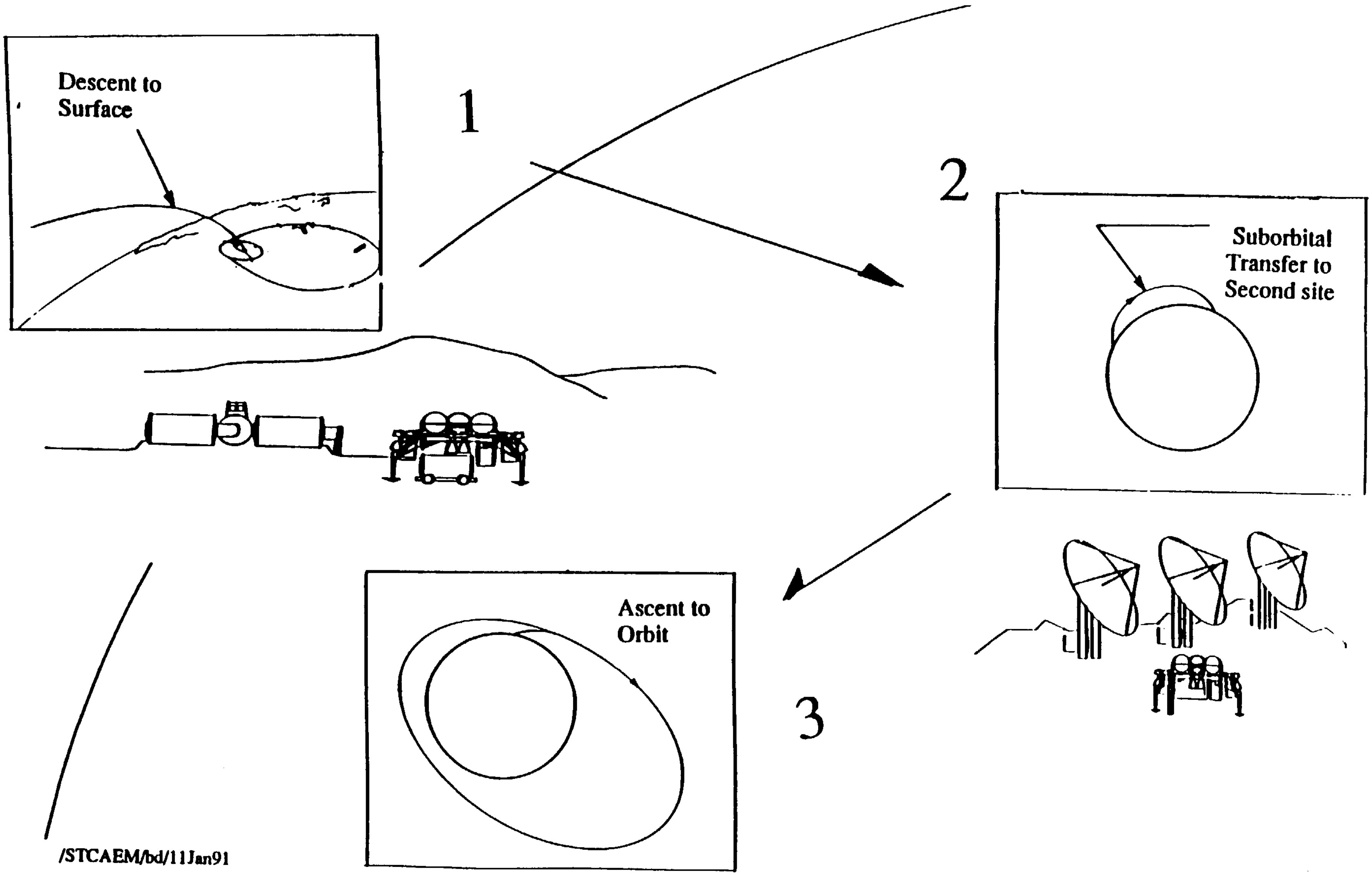
Lander Versatility

In retrospect it was seen that the primary task of the MEV/LEV circumscribed the whole of the descent/ascent task, *delivering cargo and personnel in such a way as to maximize the effectiveness of the surface mission and the crews operations*. The landers usefulness between touchdown and blastoff also being recognized has an essential ingredient to achieving a highly successful surface exploration or site buildup sortie mission. A concept aimed at increasing *exploration potential* for a given payload mass , incorporates suborbital hops to distant sites for additional exploration, i.e. a lander capable of visiting two sites before ascending back to orbit. Surface 'hopping' being the phrase to describe a suborbital flight of a few hundred kilometers.

The *undercarriage cargo* lander concept, designed to simplify cargo unloading operations at the surface, is utilized in a *piloted/cargo* configuration for evaluation of its capability for this dual task of descent/ascent and second site hopping. Both a small ascent crew cab and a larger surf crew habitat module can linked together and carried under the vehicles propulsion/frame carriage for any or all of the mission phases; descent, hop, and ascent. From the propulsion carriage, a large crew module or cargo module can be lowered to, or hoisted up (on-loaded) from the surface with the small ascent crew cab remaining in position underneath. Vehicle characteristics advantageous for this second site capability would include the capability of off-loading the first module without the aid of an unloader, use of a single propulsion stage, and highly responsive flight controllability for higher risk flights into canons or on to mountain plateaus.

Lander Versatility Trades



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Incorporation of Secondary Characteristics into LEV Design:

Initial steps in LEV conceptional design strive for optimization for primary descent/ascent task; one goal of later design iterations is the incorporation of various secondary characteristics that reach beyond primary task

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SPACE SYSTEMS

Impacts on LEV design: Secondary Characteristics

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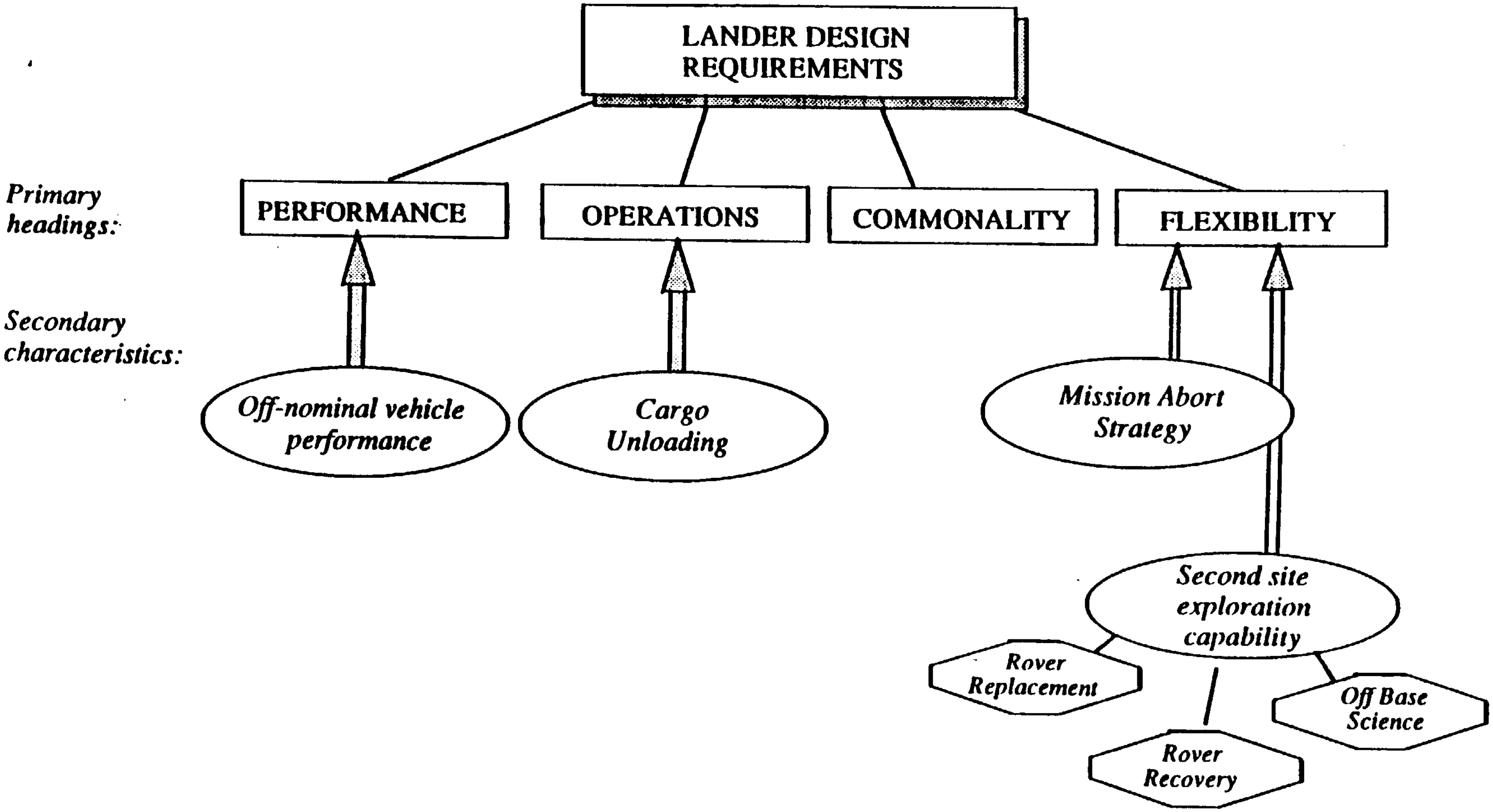

LANDER DESIGN REQUIREMENTS

Primary headings:

PERFORMANCE

OPERATIONS

COMMONALITY

FLEXIBILITY

Secondary characteristics:

Off-nominal vehicle performance

Cargo Unloading

Mission Abort Strategy

Second site exploration capability

Rover Replacement

Rover Recovery

Off Base Science

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Mac Chart/Disk #7/Secondary impacts LEV design/11-29-90

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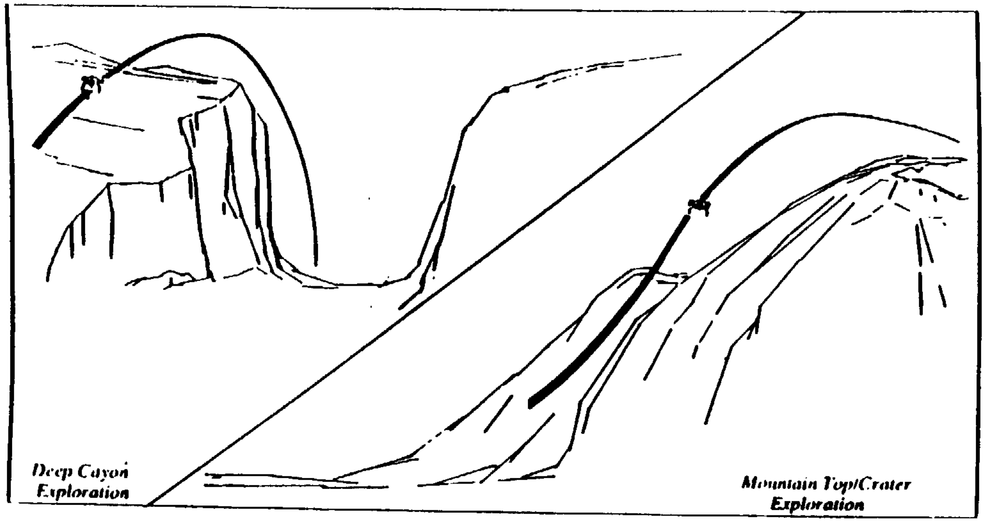
LEV Secondary Requirements Trades:
'Site to Site' Transfer of Surface Cargo

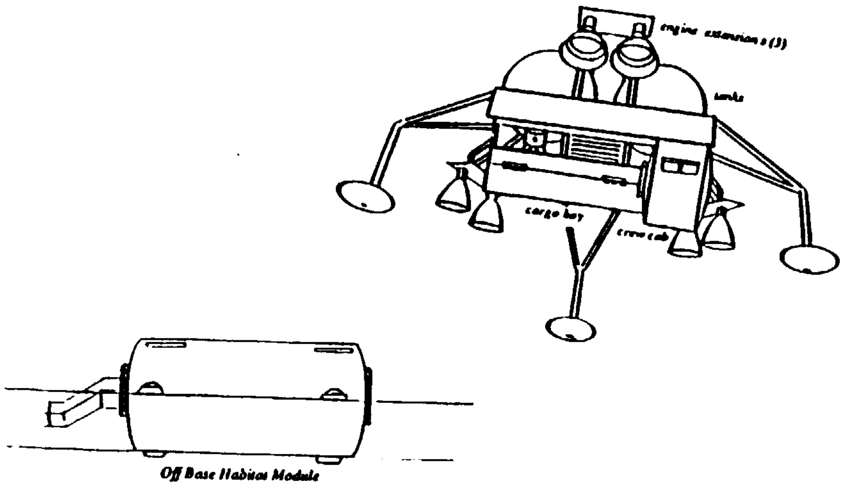


The undercarriage LEV, designed especially for ease of cargo off-loading and surface to surface suborbital transfers, could be utilized at a Lunar base in the following scenarios.

Rover crew recovery
Operating as a rescue vehicle for a pressurized rover sortie mission, the LEV could hop over to the site of the failure, recover the crew, and return.

Rover recovery
Operating as a rescue vehicle for a pressurized rover sortie mission, the LEV could recover a failed rover by flying to the site of failure, 'on-loading' the rover as it would other large cargo, and fly back to base, without requiring the aid of a separate surface based lifting cane or off-loader.

Rover replacement
An LEV designed with second site capability could serve as a replacement for a 15-25 ton class long range pressurized rover via the combination of a suborbital hop and the local exploration capability of the smaller unpressurized rovers carried out to the site by the LEV. Long range travel times on the order of weeks needed by the larger surface rovers could be reduced to a matter of days with this flyer/small rover combination.



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Mac chart/Disk #7/LEV site to site transfer/12 4 90

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Second Site Capability for Surface Rover *Crew* Recovery

The use of a lander capable of suborbital flights to the site of a failed rover for rescue of the stranded crew was evaluated. For this analysis the effect on lander weight was calculated for variations in distance to the site as well as to the rover weight. The rover itself was not 'on-loaded' and returned (for that analysis see Rover Recovery Chart), but the rover mass was important because it was decided that the lander would originally deliver the rover to the surface. Once the rover was delivered and the rover surface mission was underway, the lander would remain on site, or at the base, if and until it was needed to rescue the crew in case of some rover failure. Therefore, on the lander mass vs distance to second site plot given below, rover weights of 5,10,15,20, and 25 tons are listed. The *undercarriage cargo* lander concept, designed to simplify cargo unloading operations at the surface as well as for second site flight capability, is utilized in a *piloted/cargo* configuration for the original delivery of the rover cargo, then in its *piloted only* configuration for the suborbital hop to and from the rover failure site.

Candidate backup systems for a long range rover vehicle are given below:

1. *Second identical rover drives out to return stranded crew:* consideration: long range rover surface transportation is very slow, it takes a long time, on the order of weeks, to carry out such a mission. It may take a matter of weeks for a second rover to reach the site of the failure and that again to return.

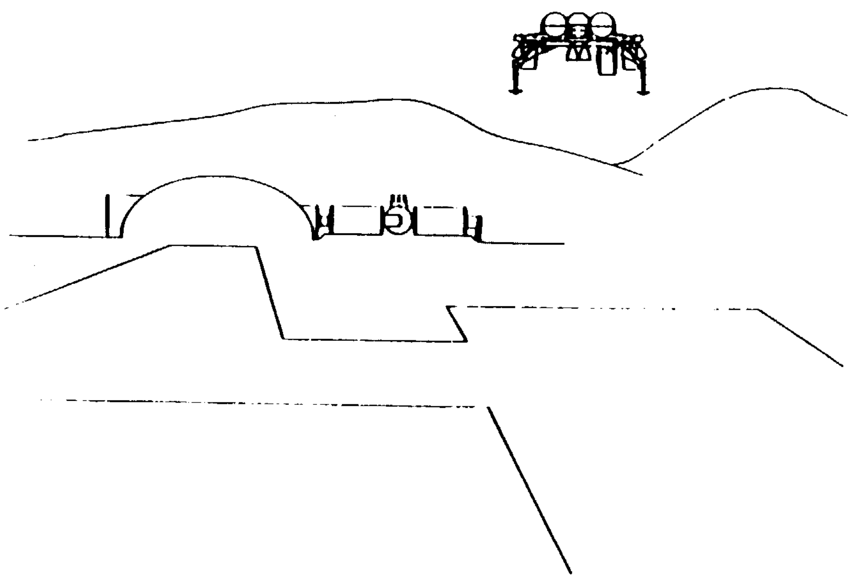
2. *Second identical rover drives out with the first:* consideration: Expensive redundancy. still takes to long a time for a injured rover crew member to be driven back.

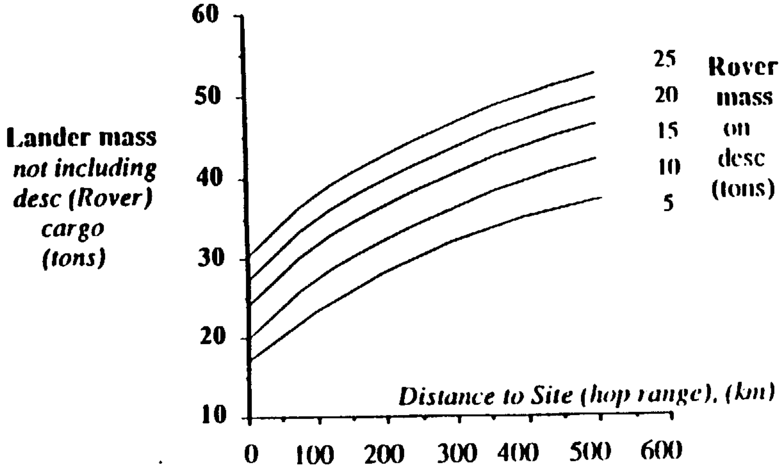
3. *LEV capable of second site flight:* considerations: very fast response time. cost: extra propellant for the flight. all other hardware and flight systems are already necessary for its descent and ascent tasks and are thus already on board. The major modification would be the larger tanks for the additional propellant required. see the propellant vs distance to site plot below.

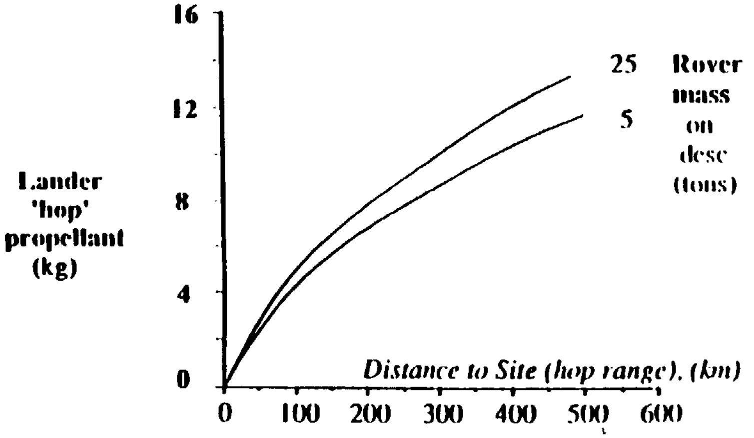
Results: An additional flight would cost any where from 1 to 14 tons of additional propellant for rescue. This is deemed as less of an overall impact then the alternatives. This capability for hopping is beneficial for other reasons as well.

Rover Crew Recovery: Impact on LEV Design

Crew returned to Base via LEV







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Second Site Capability for Surface *Pressurized Rover* Recovery

The use of a lander capable of suborbital flights to the site of a failed rover for rescue of the *Rover itself* and the crew was evaluated. For this analysis the effect on lander weight was calculated for variations in distance to the site as well as to the rover weight. The rover itself *was* 'on-loaded' and returned. The lander originally delivers the rover to the surface. Once the rover was delivered and the rover surface mission was underway, the lander would remain on site, or at the base, if and until it was needed to rescue the crew and return the rover to the Lunar base for repair in case of some rover failure. On the lander mass vs distance to second site plot given below, rover weights of 5,10,15,20, and 25 tons are listed. The *undercarriage cargo* lander concept is again utilized in its *piloted/cargo* configuration for the original delivery of the rover cargo, then in its *piloted only* configuration for the suborbital hop to the rover site, and again in its *piloted/cargo* configuration on return. The LEV lands near the rover. hoist cables are attached to the rover, it is pulled into position under the cargo bay, and it is hoisted up ('on-load') and secured onto the lander. The plot below indicates the cost in delta to LEV mass for this task given range and rover mass. The alternative to LEV recovery would be utilizing a second rover to pull it back or abandonment.

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SPACE SYSTEMS

Rover Recovery : Impact on LEV Design & Ops

Returned to Base via LEV

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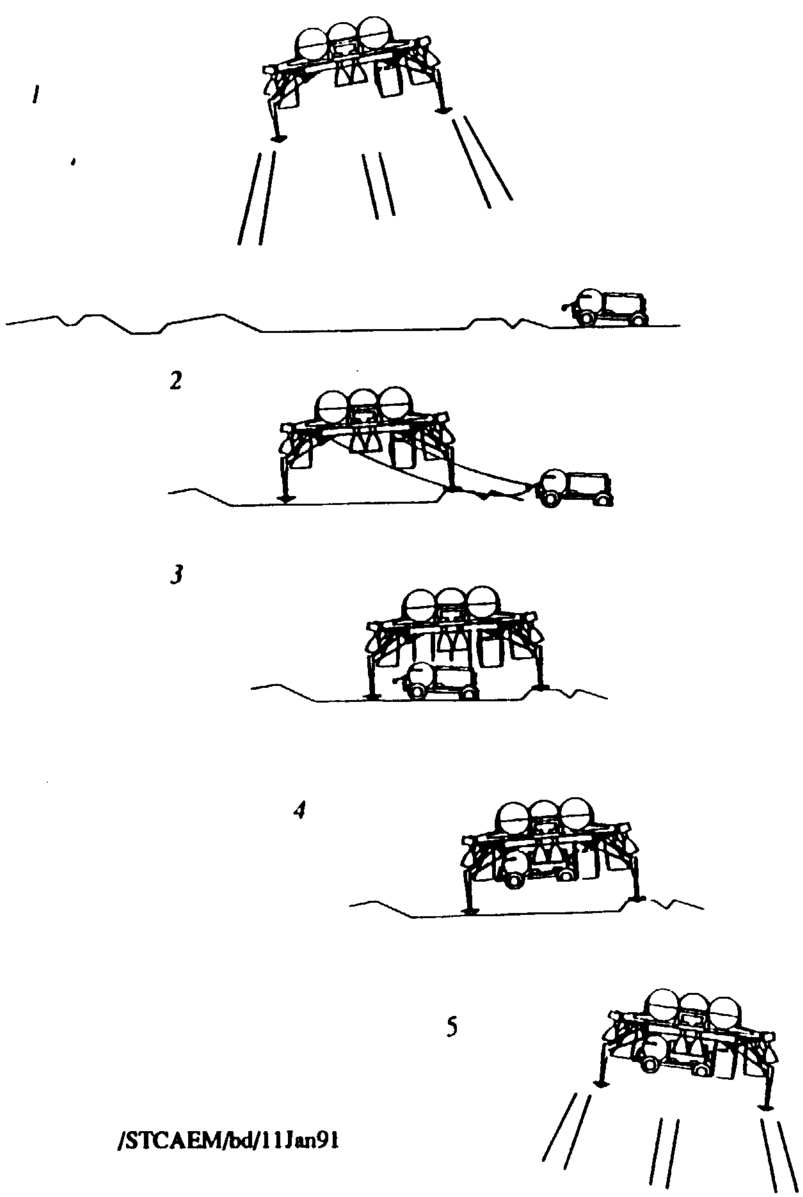

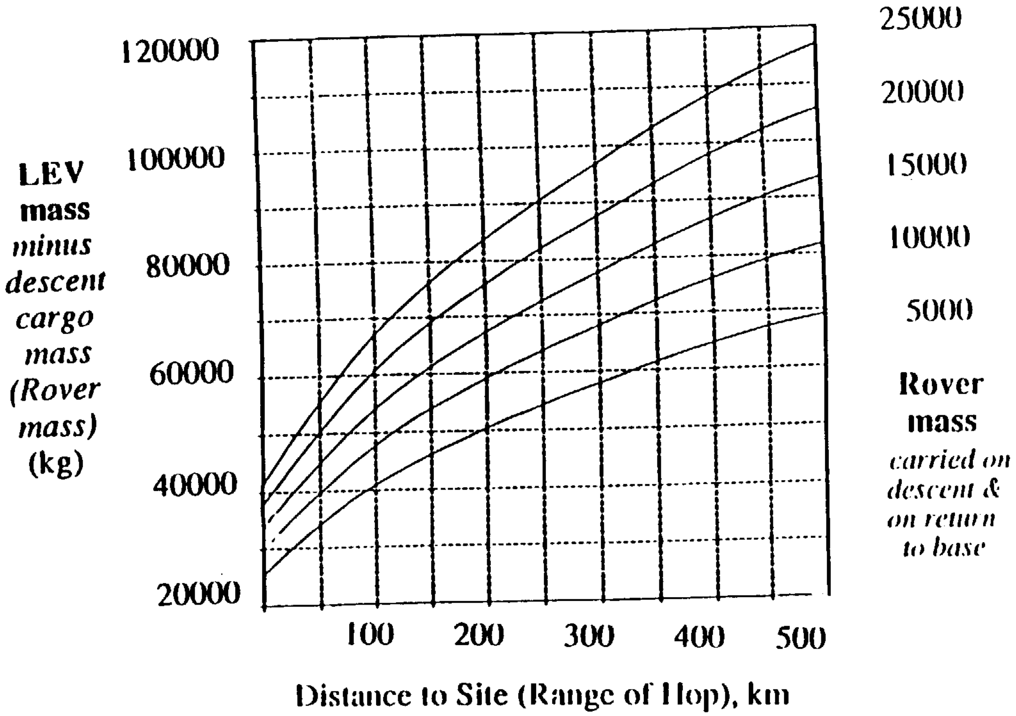

1 Transfer and descent to land near rover

2 Attach hoist cables and pull rover into position

3 'On-loading' hoist

4 Secure rover and preparation for ascent

5 Ascent and transfer to Lunar Base

/STCAEM/bd/11Jan91

Second Site Capability for Surface Pressurized Rover *Replacement*

The use of a lander capable of suborbital flights to explore other sites was evaluated. Consideration was given to replacing a heavy (15 -25 ton) long range pressurized rover with the combination of a light short range unpressurized rover and a LEV capable of hopping. For this analysis the LEV carried 10 tons of cago on its descent task. This cargo was off-loaded for use at the Lunar Base. There at the base was a crew habitat module especially equipped and suited for use as an 'Off-Base Habitat Module'. That is, it provided for exploration surface stays of durations longer than the smaller lander crew cab (aslo called ascent cab) could achieve.(a few days by itself). For this analysis the effect on lander weight was calculated for variations in distance to the site as well as to the 'Off-Base Habitat Module' weight. The extra module was carried both ways; picked up at the base, utilized during exploration, then returned to the base. The 'Off-Base module' served as a base crew module when not in service as a lander science module. Module weights of 1, 2, 3, 4, 5 tons were evaluated. The same undercarriage cargo lander concept is again utilized in its *piloted/cargo* configuration. The plot below indicates the cost to LEV mass, and the hop propellant load, for this task given range and module mass. With the unpressurized rover much actual surface exploration could be done at several sites in a relatively short time. It can be noted that the estimated alternative cost in IMLEO for visiting that second site with a separate LTV/LEV system at a later date would be an order of magnitude higher.

For SEI mission strategies that stress multiple planetary exploration missions that include site visits within as much as several 100 kms of each other, that designing LEV systems for short suborbital excursions is less expensive in terms of mission cumulative IMLEO than utilizing multiple landers.

ADVANCED CIVIL SPACE SYSTEMS

Pressurized Rover Replacement: Impact on LEV Design

Base Surf Hab Module Transfer to Second Site via LEV

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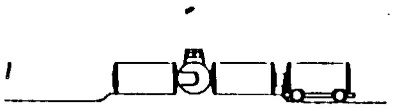
1



Base with off-base capable habitat module attached

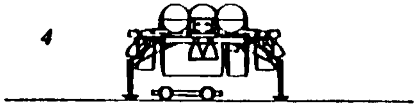
2

Hab module detached and moved to loading pad

3

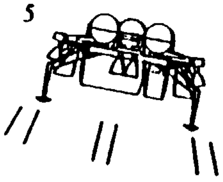
LEV landing and hab module positioning with transporter

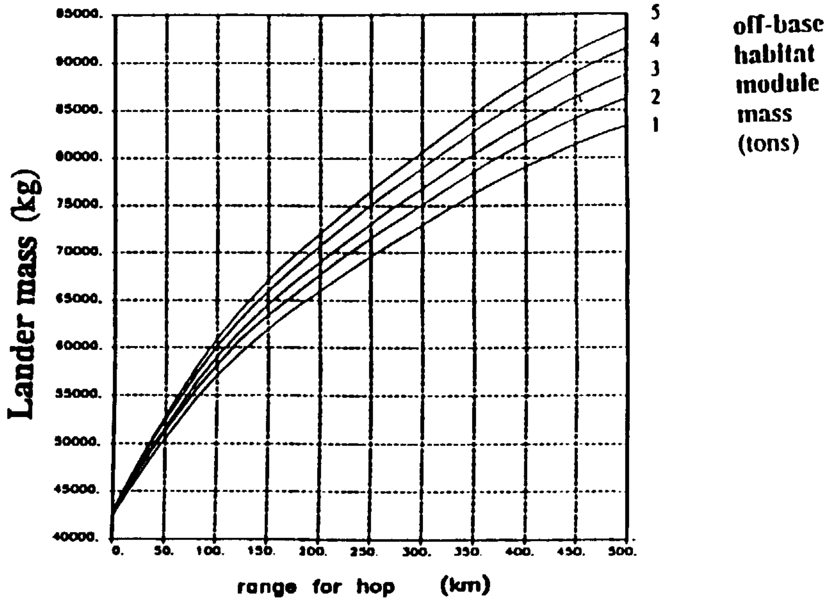
4

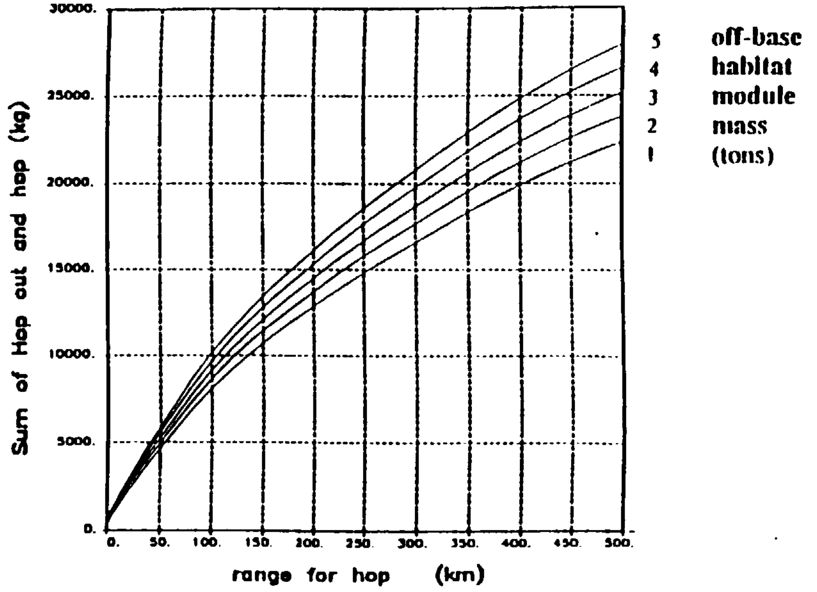


Hab Module hoist-loaded by LEV; transfer flight to distant site

5







LEV/MEV Commonality Findings

The results of the LEV/MEV commonality investigation invite a reassessment of the function of commonality in SEI. Basic astrodynamic conditions are different enough for the Moon and Mars that optimal overall vehicles designs for each place tend in different directions.

Before a decision mandating whole-vehicle commonality is warranted, critical contextual issues must be clarified: the role of the lander in the overall transportation architecture; and the specific nature of the payloads each is likely to carry.

LEV/MEV Commonality Findings

LEV and MEV requirements are inescapably divergent

Differing gravity level (factor of 2)
- The common EV concept flies offloaded for lunar & Mars cargo delivery
- Full tanks (31 t) *could* land 45 t lunar cargo (23 t in crew case)

Aerobraking constraints (Mars only)
- Large payload volumes have non-trivial impacts on the aerobraking cases

Lunar and Mars surface mission requirements may also be divergent

Stay time is a critical parameter
- Campsite or excursion lander only?
- Boiloff or refrigeration?

Realistic lunar and Mars cargo may be substantially different

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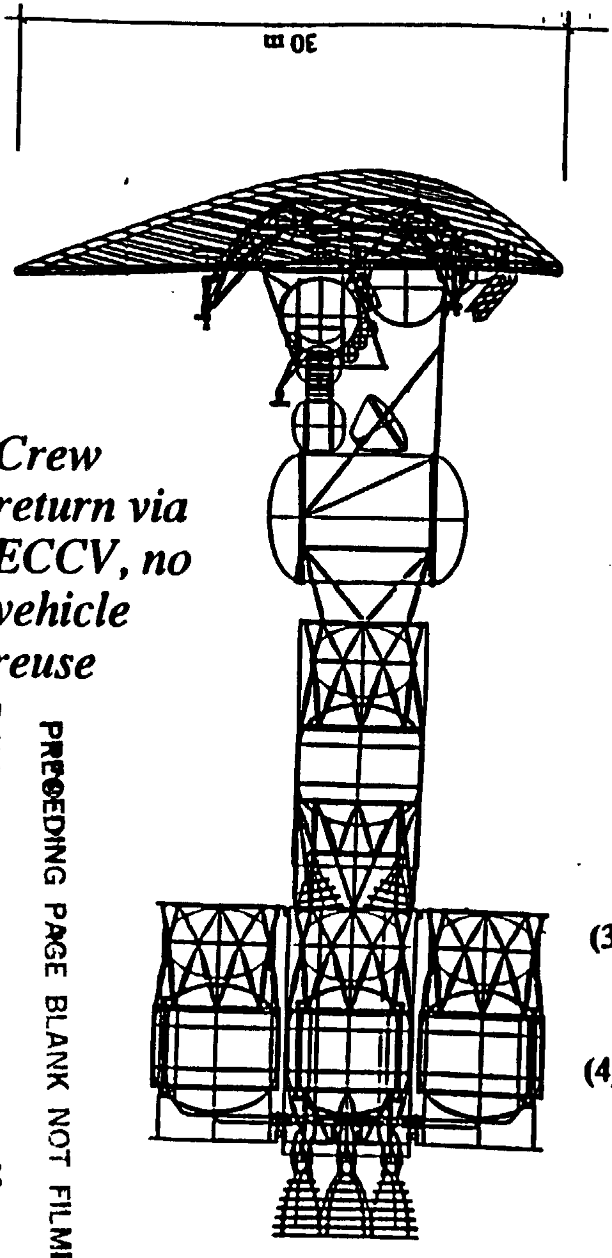
All Propulsive Chemical Vehicle for Conj Mission

Single MEV ;30t surf cargo, crew of 3 for 90 days on surf, 1 crew member left in orbit , Common tank sets for MTV stgs
dV's: TMI dV= 3900 m/s, MOC=1530 m/s, TEI= 860, E arr Vinf=3200,TMI, MOC,TEI eng Isp=475, MEV eng Isp=460



Crew return via ECCV, no vehicle reuse

(1) MOC/TEI Tank
7.6 (m) dia
17.0 (m) length

(3) TMI Tanks
7.6 (m) dia
17.0 (m) length

(4) Engines
at 200k lbf each
(eng out)

Revision 3 8/7/90	*Element*	*Mars*	*Lunar* *offloaded TMI*
[378]	MTV crew hab module 'dry'	39000	39000
[398+371]	MTV hab consumables & resupply	14000	14000
[179]	MTV hab mod science	0	0
sum	***MTV crew hab module***	**53000**	53000
[128]	TEI usable propellant	15713	
[551]	TEI outbound boiloff	1455	
[545+546]	TEI inorbit boiloff	2716	
sum	***Total TEI propellant***	**19884**	27471
[541]	MOC usable propellant	72214	
[538+539]	MOC outbound boiloff	3752	
sum	***Total MOC propellant***	**78910**	48015
sum	***EOC propel (Lunar case: returnto LEO)***	**0**	64056
[118]	RCS propellant	852	917
[121]	Outb midcourse correction prop	2274	0
[122]	Inb midcourse correction prop	663	0
[161]	MOC/TEI propul stg inert	16335	16335
sum	***MTV propulsion stg total***	**115974**	**156729**
[1313]	MEV descent only aerobrake	12000	12000
[63]	***MEV ascent stage***	*24262*	*11716*
	Propellant / Isp	*Cryo/460*	*Cryo/460*
	MEV descent stage	*23190*	*22739*
	Propellant / Isp	*Cryo/460*	*Cryo/460*
[166]	MEV surface cargo (3 crew for 90 days)	30000	30000
[106]	***MEV total***	**89452**	**76455**
[230]	***ECCV for crew return to LEO***	7000	0
[172-173+547]	TMI inert stage wt	45800	45800
[173]	TMI propellant load	407680	315199
[172]	***TMI stage total***	**453480**	**360999**
[171]	***IMLEO (all masses in kg)***	**718910**	**647183**

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Common Mars/Lunar Lander Vehicle - Cargo & Manned Versions

Mars desc propul ΔV: 773, AscΔV:5319, Lunar asc & desc ΔV: 2100, all cryo prop Isp=475

Single stage vehicle - aeroshell , cargo and landing legs left on surface

Element	Mars *Cargo* *(desc only)*	Mars **Manned* *(single stg desc/asc veh)*	Lunar *Cargo* *(desc only)*	Lunar **Manned* *(single stg)*
Ascent cab	0	3500	0	3500
Stg inerts	5374	5374	5374	5374
Aeroshell	7500	7500	n/a	n/a
Surf Cargo	30000	700	30000	**12612
Asc prop	n/a	16082	n/a	5310
Desc prop	7900	5255	20658	16027
RCS prop	893	1341	893	1341
Total kg mass	**51668**	**39752**	**56925**	**44164**

** Manned: crew of 3 or 4 for very short surf stay time (a week or less)*

*** Maxium surface cargo load for manned lunar case when all tanks are full*

Commonality Assessment

Element	potentially common with...	...but with specific exceptions
MEV crew module	LEV / LTV crew modules	Space dormancy time
MAV	LEV / LTV engines, ACS, avionics	Avionics (rendezvous GN&C for elliptic orbits)
MEV descent stage	LEV engines & avionics	Avionics (descent GN&C with aerobraking)
MEV aerobrake	MTV aerobrake	Reduced structural gauges for lighter payload; engine ports; possibly different TPS
MTV crew module	Surface modules, ECLSS	Airlock / dustlock design
MTV / TEIS	LTV	Tank & structure arrangement may be different
MTV aerobrake	LTV aerobrake?	Probably evolution rather than commonality (different size, heating rate, payload mass, L/D)
TMIS	HLLV 3rd stage (Shuttle-Z); LTV engines in low-thrust options	Needs MLI for cryo storage; longer burn time

Lunar/Mars Excursion Vehicle Commonality
Updated Preliminary Conclusions

The conclusions show that commonality is not a "*cut and dry*", simple issue to solve. Many factors affect the outcome, and are outlined below. Commonality also does not have to mean vehicles that are ***exactly*** the same because the Moon and Mars are two very different problems. Forcing lander commonality could force ***uncommonality*** of hab systems (surface and transit), transfer vehicles and aerobrakes. Commonality can, however, be seen at the subsystems level. It may be possible to develop a "kit-of-parts" which are assembled for varying missions. Commonality is an issue that will have to be pursued to a finer level to actually assess the most feasible approach.

Lunar/Mars Excursion Vehicle Commonality
Updated Preliminary Conclusions

- "Commonality" can apply to *subsystems* or to *entire vehicles*
- There are limited, sensible ranges within which each type of commonality is most feasible
- The lunar and Mars cases *are not* inherently common because of aerobraking, and may begin years apart in any case
- Cryogenic management (boiloff or refrigeration) becomes a dominant assumption for long stay times (≥ 1 year)
- Vehicle commonality at the scale of the "mini-MEV" (our early assessment) *may* indeed be feasible
- Large payload volumes have non-trivial impacts on the aerobraking cases
- Vehicle commonality in the 30 t - payload class forces new priorties, whose costs *may not* be worth it
- The common EV concept flies offloaded for lunar & Mars cargo delivery
- Full propellant tanks (31 t) *could* fly 45 t cargo to the Moon --- 23 t cargo in the lunar crew case
- Subsystem commonality (engines, mechanisms, avionics, ACS) is *probably* more appropriate for the larger class vehicles

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STCAEM Commonality Assessment

Shown is a matrix which encapsulates the current state of knowledge from STCAEM about the type of commonality appropriate for SEI elements "across the architecture". A high degree of potential commonality is apparent, although most often at the system level.

STCAEM Commonality Assessment

②

	SSF	SEI LEO Ops	Lunar Transfer	Lunar Excursion	Lunar Surface	Mars Transfer	Mars Excursion	Mars Surfac
Vehicles								
Lunar Transfer	×	▨	—	●	×	⊘	⊘	×
Lunar Excursion	×	▨	▨	—	×	○	▨	×
Mars Transfer	×	×	×	×	×	—	×	×
Mars Excursion	×	×	×	▨	×	○	—	×
MCRV	—	●	▨	×	×	▨	×	×
Vehicle Systems								
Main Propulsion	×	▨	—	▨	×	TBD	▨	×
ACS Propulsion	×	▨	—	▨	×	▨	▨	×
Aerobrake	×	×	⊘	×	×	▨	—	×
Propulsive Power	×	×	▨	×	▨	—	×	▨
Housekeeping Power	—	▨	▨	⊘	▨	▨	⊘	▨
Communications	○	⊘	⊘	⊘	⊘	—	▨	▨
Avionics	○	▨	—	▨	○	▨	▨	○
Robotics	○	▨	▨	▨	▨	—	▨	▨
Thermal Control	—	▨	▨	▨	⊘	▨	⊘	⊘
Structures	—	⊘	⊘	⊘	⊘	▨	⊘	⊘
Crew Systems								
Small Habitat	—	▨	⊘	⊘	▨	⊘	⊘	▨
Large Habitat	○	⊘	▨	⊘	●	—	⊘	●
Airlock	○	▨	▨	—	▨	▨	▨	▨
Suit	○	▨	▨	—	●	▨	⊘	⊘

/STCAEM/bs/09Oct90 — Exemplar ● Element ▨ Subystem ⊘ Component ○ Technology × None

LEV/MEV Commonality Conclusions

Shown are the basic conclusions regarding LEV/MEV commonality. Commonality at the system and subsystem levels continues to show promise, but whole-vehicle commonality appears to result in really efficient satisfaction of requirements only at the scale of the mini-MEV analyzed earlier. Driving for vehicle commonality in the "full-size" Mars lander range does not result in so good a match.

LEV/MEV Commonality Conclusions

- **Vehicle commonality at the scale of the "mini-MEV" *may* indeed be feasible, as suggested earlier**
- **LEV/MEV commonality in the 30 t payload class forces new configuration priorities, whose costs *may not* be worth it**
- **Lunar and Mars EV flights may begin years apart anyway**
- **System and subsystem commonality (crew cabs, engines, mechanisms, avionics modules, ACS, perhaps tanks) appears more appropriate for the larger class vehicles**

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Lunar/Mars Mission Operations

Lunar Architectures

Seven lunar flight modes have been identified to accomplish lunar missions. The concepts consist of a "kit-of-parts" that could be assembled to accomplish any of the seven mission modes. The "kit-of-parts" includes a 110 t propellant stage, a 25 t propellant stage, an excursion crew module, a transfer crew module, landing legs, a cargo pallet and an aerobrake. These components are configured into LEV-like and LTV-like vehicles, and are shown assembled for each of the seven mission modes.

LTV/LEV Habitat Module Study

This study was conducted to determine the most reasonable crew modules that can be used in the LEV/LTV system. For transfer modules used in conjunction with excursion modules, the modules used with separate aerobrakes are more mass-efficient because of their simplified shape. The single module approach, using one module for both transfer and excursion, is more sensitive to crew size than it is to mission duration, making crew size determination critical. Excursion modules used in conjunction with transfer modules are also more sensitive to crew size than to mission duration. (A separate surface module becomes desirable for mission durations grater than or equal to 5 days.)

Long-Duration Habitation Trade Study

This trade study was conducted to design a reasonable habitation concept for long-duration missions (1000 days) to be integrated into STCAEM Mars transportation system concepts. The study generated a process -- developing metrics and prioritizing them to derive a solution -- which can be applied to any hab system trade study. The study investigated 5 crew sizes, 3 module diameters (most likely launch shroud diameters), and 1480 distinct options. These options were evaluated via topology and geometry comparisons, a preference survey, mass analyses, and integration and fabrication analyses.

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The reference Mars transfer habitat is a single cylindrical module 7.6 m in diameter, divided for safety by an interior pressure bulkhead, which can accommodate crews for up to 1000 d. Major features of the module are as follows:

- 2:1 aspect ratio, unpenetrated end domes
- Cross-section, bisecting bulkhead
- 2 floors parallel to the major axis ("banana-split")
- Diametral tension-tie, deep second floor
- g-field optimized, to provide extensive commonality across architecture

The internal pressurized volume was derived from a plot of historical spacecraft total pressurized specific volume (volume per crew member) versus mission duration. The curve suggests a 112 m^3/person volume for 1000 d durations (worst case round trip time for a flyby abort in conjunction mode).

Functionally, a unitary vessel minimizes leakage and parts count, while the 7.6 m diameter allows a wide variety of internal outfitting designs. The diametral floor maximizes nominal floor area on the upper floor, as well as the potential for a mass-reducing tension tie (analogous to airliner structures). A unitary vessel also provides a compact domain, which is preferable from a crew safety access-time standpoint.

Integration issues addressed were launch, orbital assembly and aerobrake integration. A launch shroud diameter of at least 7.6 m is likely to be available early for SEI. The chosen concept lends itself well to aerobrake integration, and even larger crews (> 12) could be accommodated through simple clustering.

A simple human perception survey showed that technical people, used as a model for early SEI crews, tend to perceive larger diameter concepts as more spacious, independent of actual volume equality. The 7.6 m diameter module also provides a better plan aspect ratio than smaller diameters, when oriented "horizontally", to offset the feeling of living in a tunnel.

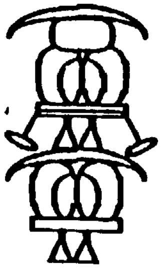
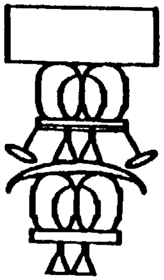
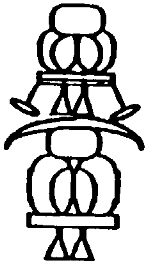
The chosen design is essentially the lightest-mass concept investigated, critical for interplanetary transportation. The concept is also a prime candidate for material and processing improvements, which could lower mass and production costs even further.

Finally, the concept facilitates commonality in growth architectures as well as for surface system applications.

Lunar Modes and Vehicle Options

In examining the architectures, we identified a total of seven lunar vehicle modes shown on the next two pages. The first three are used in all architectures and are shown on the first of the two pages. The next four modes are used in some of the architectures. One is used only in the L2 lunar oxygen architecture, to supply lunar oxygen to the L2 node for use in Mars vehicles. It should be noted that the configuration shown for this particular mode is not a flight configuration but symbolizes the fact that one LTV-sized vehicle can deliver enough hydrogen to L2 to fuel two LTV-sized vehicles to descend to the lunar surface and return to L2 with payloads of lunar oxygen. All of these modes can be implemented with a pair of vehicles: an LTV-like vehicle and an LEV-like vehicle. In examining the modes, we find that both vehicles need to be able to operate with either an LEV-type or LTV-type crew cab. Both need to be able to carry cargo; both need to integrate with an aerobrake; both need to integrate with landing legs; and the larger LTV vehicle needs to integrate with both on the same application. This later case is used only a few times, and can be implemented with expendable landing legs.

Lunar Modes and Vehicle Options

	Mission Mode & Schematic		Application and Rationale	Architectures
1		LTV tandem direct crew, LEO-LS and return	Early crew missions; deferred development of LEV and LEV crew cab.	All
2		LTV tandem direct LEO-LS cargo; booster recovered; lander remains on LS.	Heavy payload capability (45-50 t.); capable of landing campsite intact on single flight	All
3		LTV/LEV crew and cargo LEO-LOR-LS & return; optional lunar oxygen; option to leave LEV on LS in cargo mode.	Most efficient crew mode; efficient with lunar oxygen; lowest lunar oxygen production rate.	All

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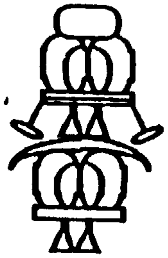
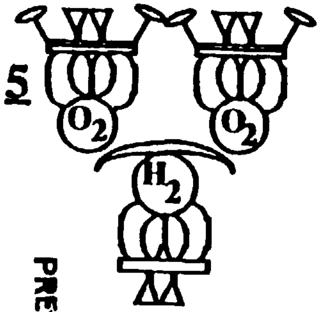
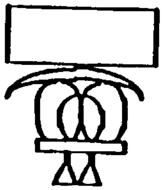
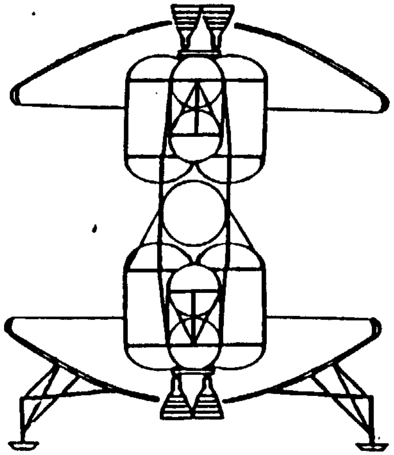
Lunar Modes and Vehicle Options

Mode	Mission Mode & Schematic		Application and Rationale	Architectures
4		LEO-L2-LS crew, rendezvous at L2, with lunar oxygen	As efficient re IMLEO as LOR with lunar oxygen (requires higher lunar oxygen production); simplifies operations for L2 node Mars operations.	L2/Lunar Oxygen; NEP & SEP
5		Lunar oxygen delivery to L2; hydrogen to L2 from LEO	About 1.5 t. lunar oxygen to L2 per t. resupply to LEO; makes for efficient L2 Mars node for cryogenic/aerobraking, or cryogenic all-propulsive conjunction missions.	L2/Lunar Oxygen
6		Crew trips LEO-L2 and return	Mars crews to & from L2 node	L2/Lunar Oxygen; NEP & SEP
7		Cargo trips LEO-L2 with return of LTV	Node and Mars mission cargos to L2	L2/Lunar Oxygen; NEP & SEP

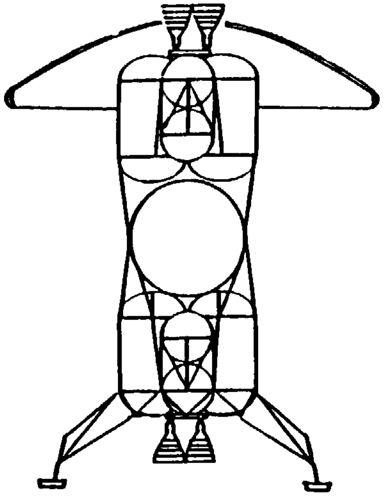
Lunar Vehicle Configurations

Seven lunar vehicle configurations are shown below, to correspond to the mission modes depicted on the previous pages. The vehicles shown are based on a "kit of parts" that include the 26 m aerobrake, a four crew cab, six crew transit hab, 110 t propellant and engine combination, 25 t propellant and engine combination, and a standard cargo pallet.

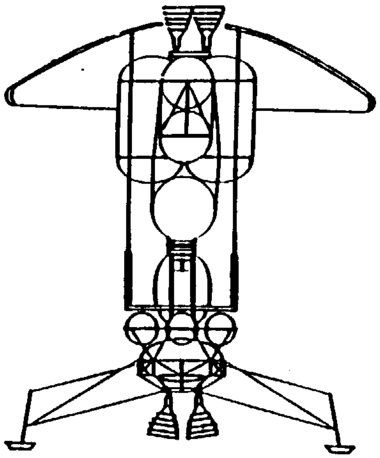
Lunar Vehicle Configurations



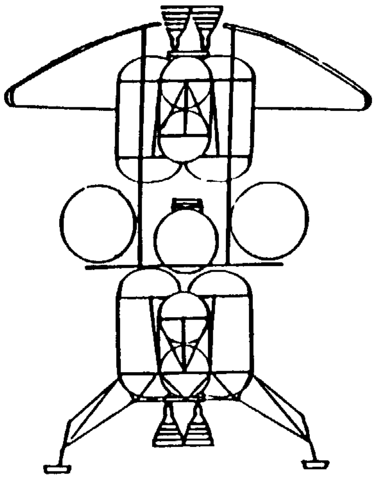
1



2



3



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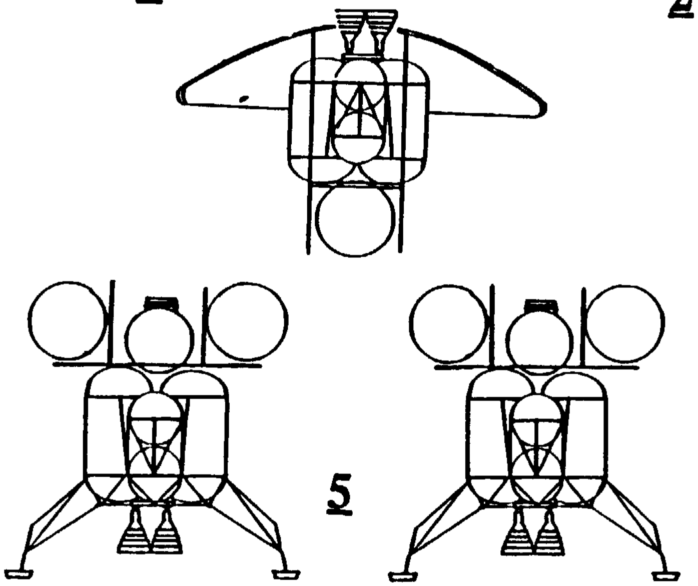
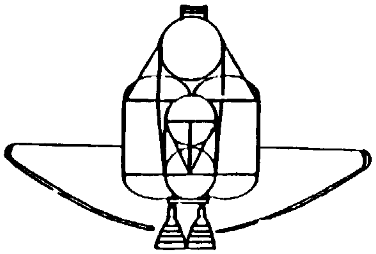
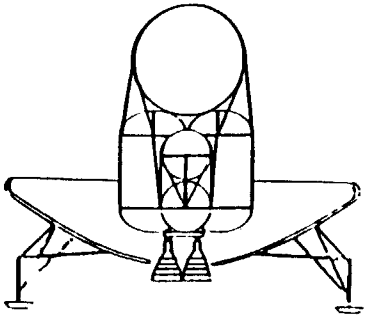


5



6



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LTV/LEV Habitat Module Study

The LTV/LEV habitat module study was conducted to determine the most reasonable crew modules that can be used in the LTV/LEV system. The study encompassed transfer, both aerobraked and direct entry, and excursion modules as well as a combination transfer/excursion direct entry module.

LTV/LEV Habitat Module Study

Goals
- Determine configuration envelopes for LTV/LEV habitat modules
- Develop mass statements for each configuration

Groundrules
- Crew sizes of 2,4,6 and 8
- Surface stays of 1, 14, 28 and 42 days
- Round trip time of 7 days - 24 day free return abort
- Crew volumes extrapolated from historical data
- Mission modes
  - Transfer module in conjunction with excursion module
    - aerobrake at Earth
    - direct entry at Earth
  - Transfer/excursion module
    - single module - direct entry at Earth
  - Excursion module in conjunction with transfer module

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LTV/LEV H[illegible] Modules

Assumptions

- Sized for 24 day free-return abort in the worst-case
- Stored O_2 for the breathing with regenerative molecular-sieve beds for CO_2 removal
- No "hygiene" water allocated for showers, washers, galleys, etc.
  Stored H_2O at 2.0 kg/man day used for drinking, food preparation, and sponge bath
- Food is all shelf-stable - 1.25 kg/man day
- 10 kg/cm2 radiation shielding for shelter in addition to approximately 2-5 g/cm2 of protection provided by skin structure and onboard equipment - 48 hour nominal duration
- No refrigerators, freezers, personal hygiene compartment allocated
- Minimal exercise equipment - "bungee cord" type
- ACS provides cabin air leakage make-up and 3 cabin repress. recharges
- Human waste and urine storage - no urine processing
- Power supply - solar arrays with batteries during lunar night, backup and aeromaneuver periods
- 15% mass growth

Operating Modes

- Transfer module in conjuction with an excursion module
- Transfer/excursion module (1 module) both aerocapture and direct entry

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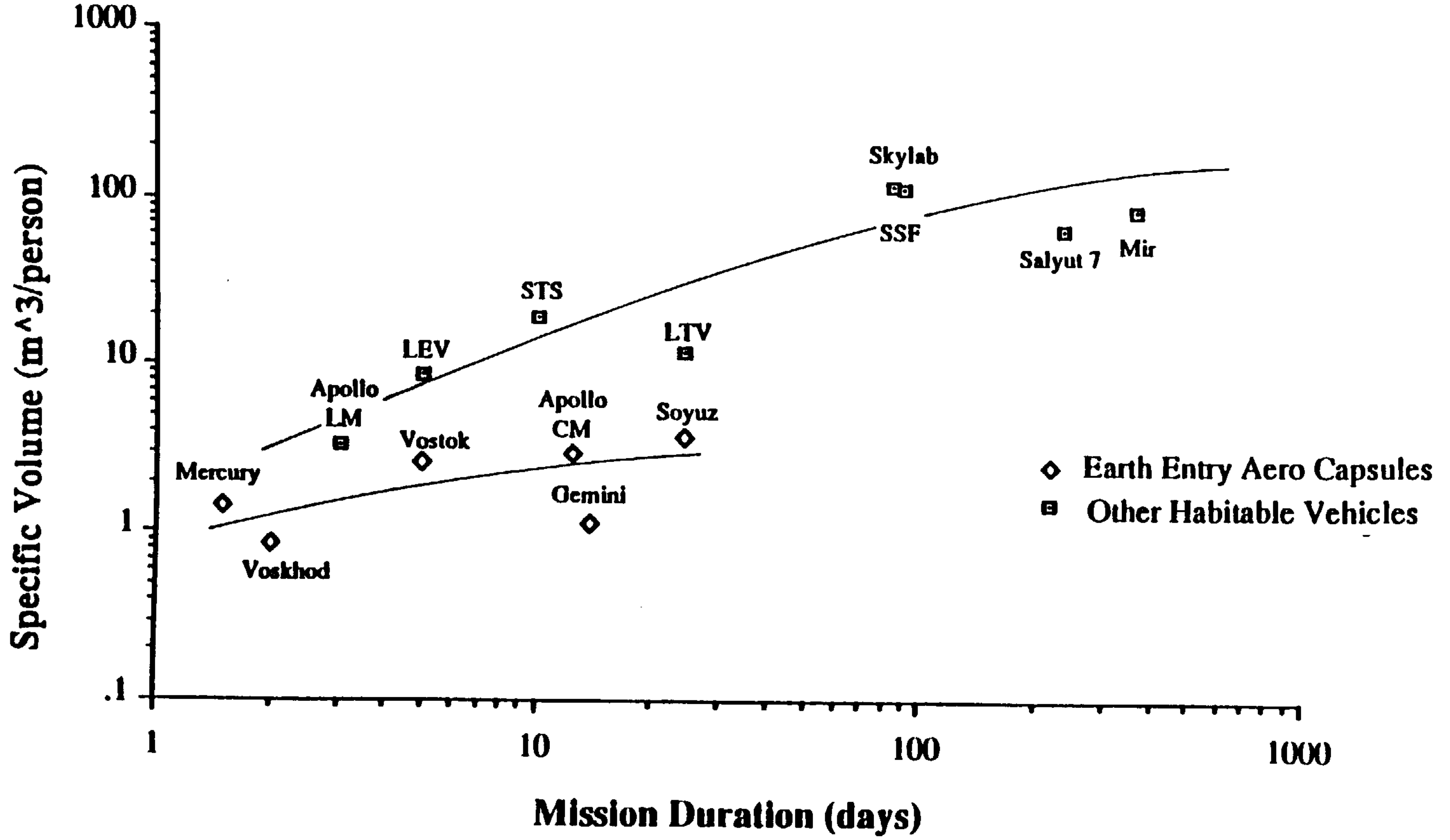
LTV/LEV Crew Volume Guidelines

This graph, which shows historical spacecraft total pressurized volume, was used as a guide for determining optimum crew volumes required for various mission durations. The LTV and LEV modules are plotted on this graph.

LTV/LEV Crew Volume Guidelines

Historical Spacecraft Total Pressurized Volume Data

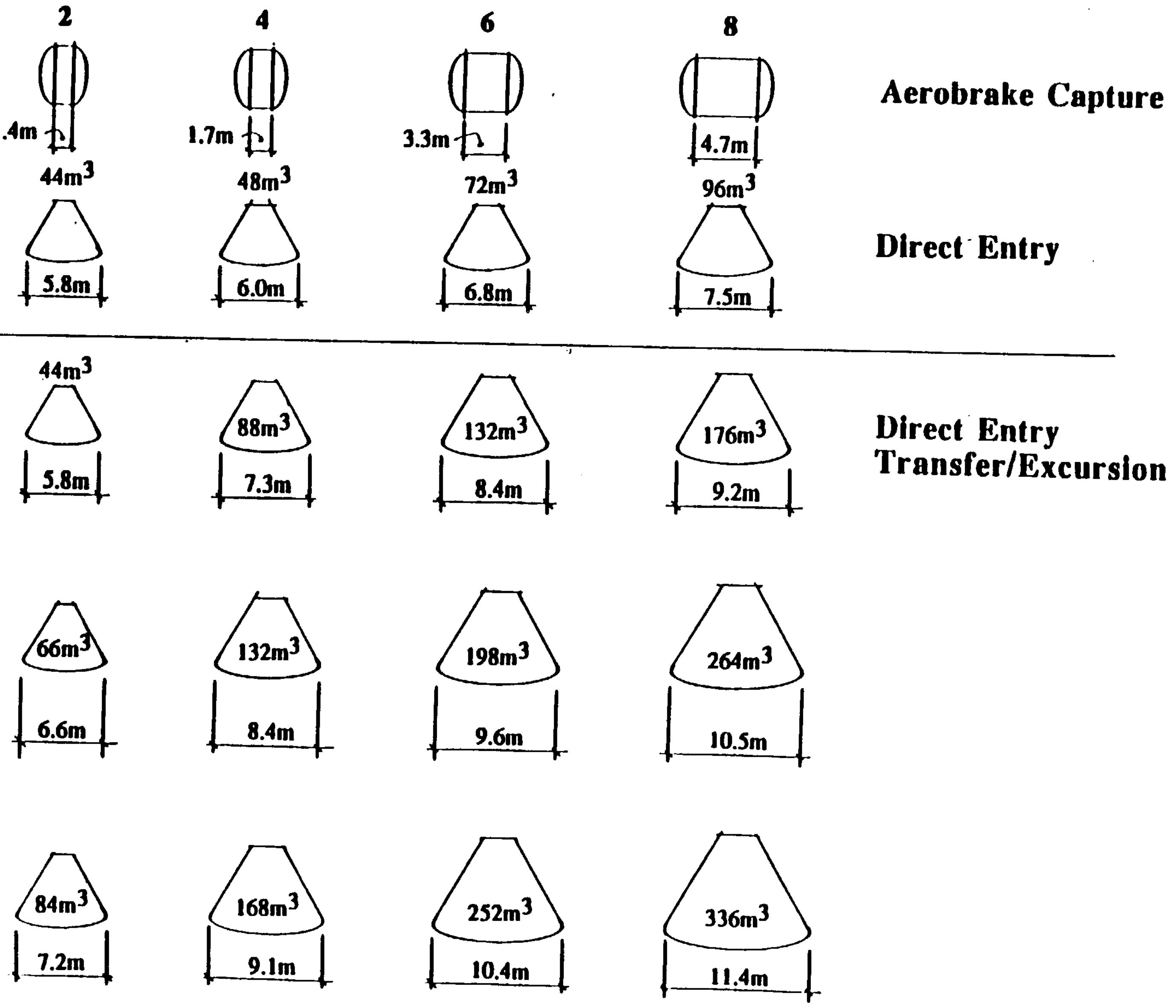


Lunar Transfer Modules Configuration Envelopes

This chart shows the relative sizes of the 3 transfer modes studied: transfer aerobraked, transfer direct entry, and transfer/excursion direct entry. The direct entry shapes are all "Apollo type".

Configuration Envelopes

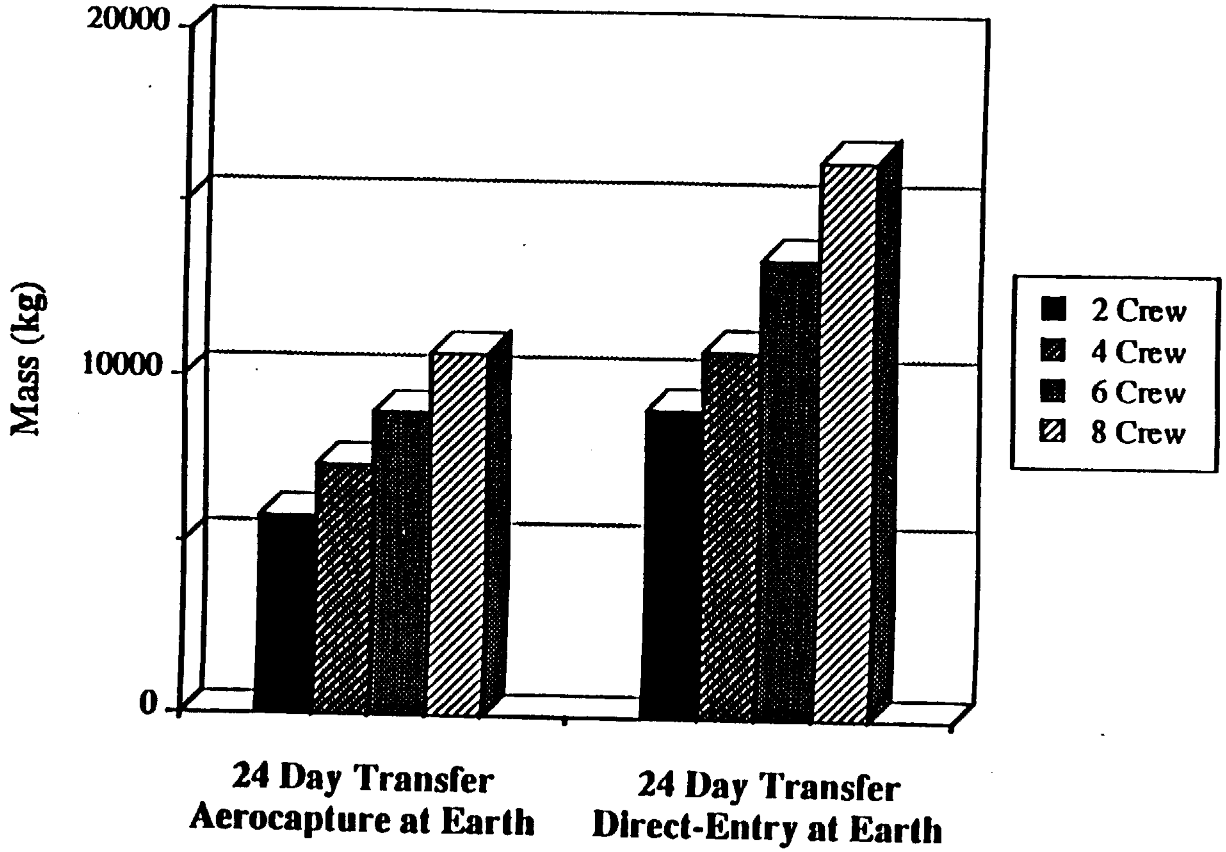
Crew Size/ Mission Duration



Lunar Transfer Module Mass Summary

Shown on this chart are the relative masses of the 2 transfer modes, aerocapture and direct entry. The direct entry is naturally more massive because of the inefficient pressurized shape and the extra equipment required for landing.

Lunar Transfer Module Mass Summary

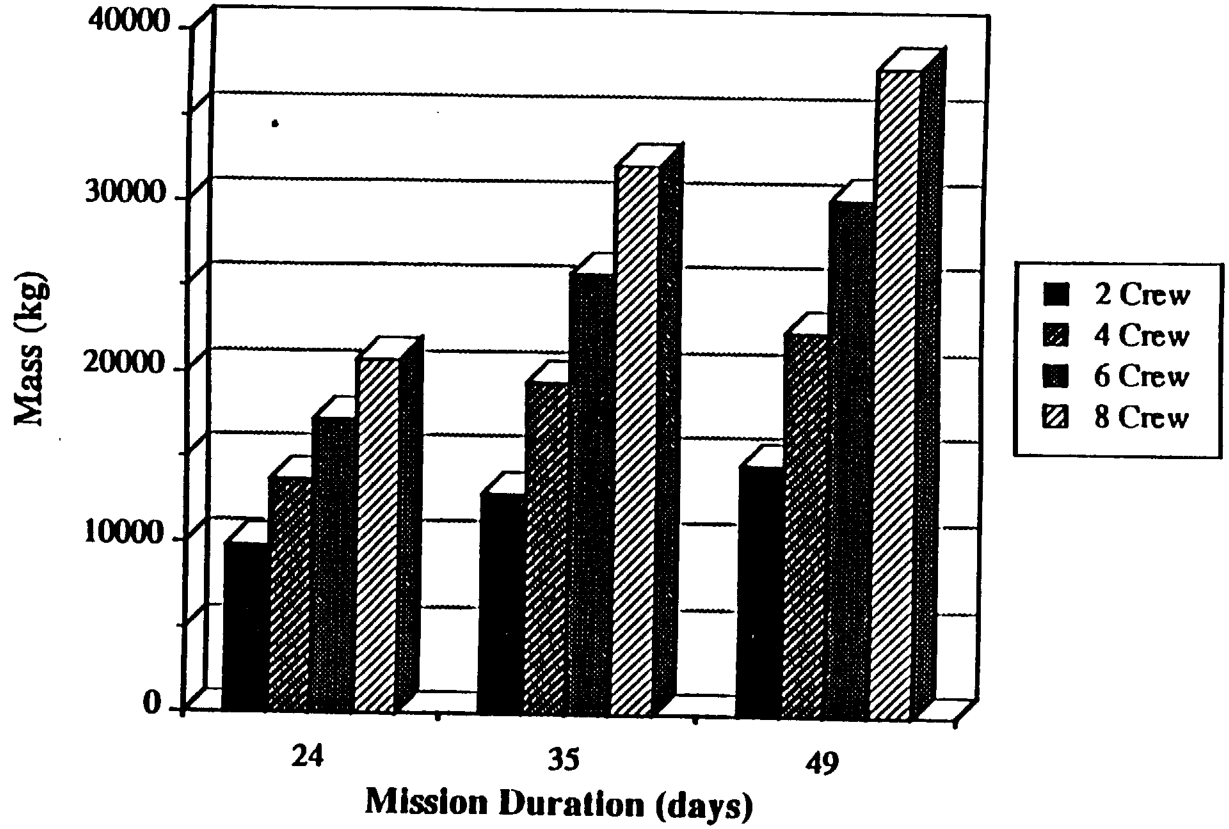


- Operates in conjunction with an excursion module
- Sized for a 24 day free return abort
- "Direct-entry" module is an "Apollo" shape

Lunar Transfer/Excursion Direct Entry Module Mass Summary

Shown on this chart are the relative masses of the direct entry transfer/excursion modules for surface stays of 1, 14, 28 and 42 days with a 7 day transfer time.

Lunar Transfer/Excursion Direct Entry Module Mass Summary



- Single module ("Apollo" shape) serves as transfer and excursion module
- 24 day duration includes 1 & 14 day surface stays and is sized for a 24 day free return abort worst case
- 35 day duration includes a 28 day surface stay
- 49 day duration includes a 42 day surface stay

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LEV Hab Modules

Assumptions

- STCAEM Crew Module Structural System
  - SSF diameter (4.4m) cylinder section
  - All penetrations occur in the cylinder section
  - All structural attachments occur at girth rings
  - Common ellipsoidal end domes (2:1 aspect ratio)
- Volume/crew consistent with historical spacecraft data
- Open ECLS system
- Power supply via fuel cells/solar arrays
  - 1 day = fuel cells
  - 14 + days = solar night capacity
- Minimum mass airlock - 1 airlock cycle/day per 2 crew
- No radiation shielding
- Human waste and urine storage
  - 1 day = bags and storage recepticles
  - 14 + days = toilet and storage bags
- Food - all shelf stable - 1.25 kg/man day
- Minimum medical provisions
- 15% mass growth

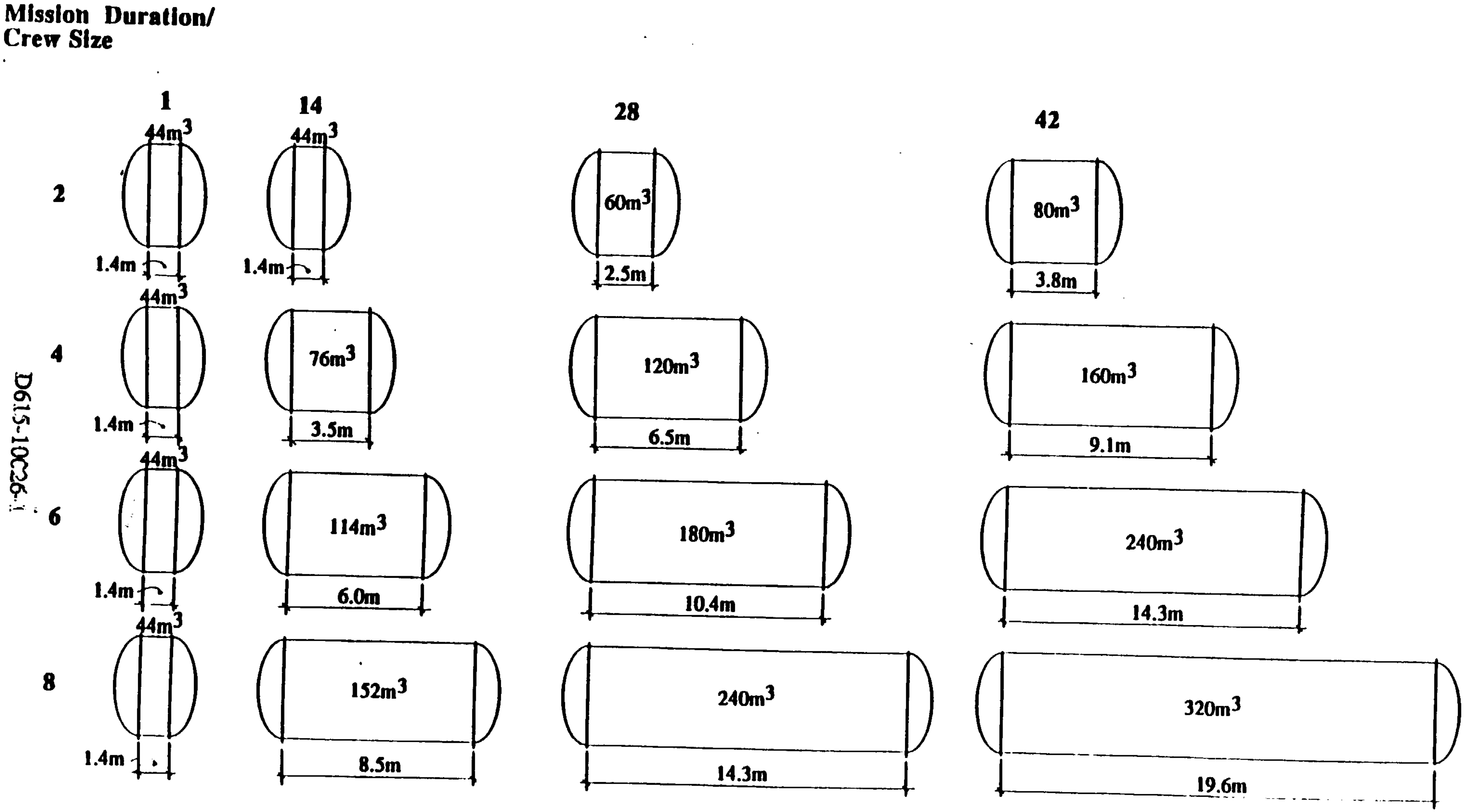
Operating Mode

- In conjunction with a transfer vehicle (LTV/LEV scenario)
- Excursion module only - no direct Earth entry or transfer

Lunar Excursion Modules Configuration Envelopes

This chart shows the relative sizes of excursion modules for crews of 2,4,6, and 8 and surface stays of 1, 14, 28, and 42 days. Volumes presented are derived from historical spacecraft data, as shown on the previous "LTV/LEV Crew Volume Guidelines" chart

Lunar Excursion Modules
Configuration Envelopes

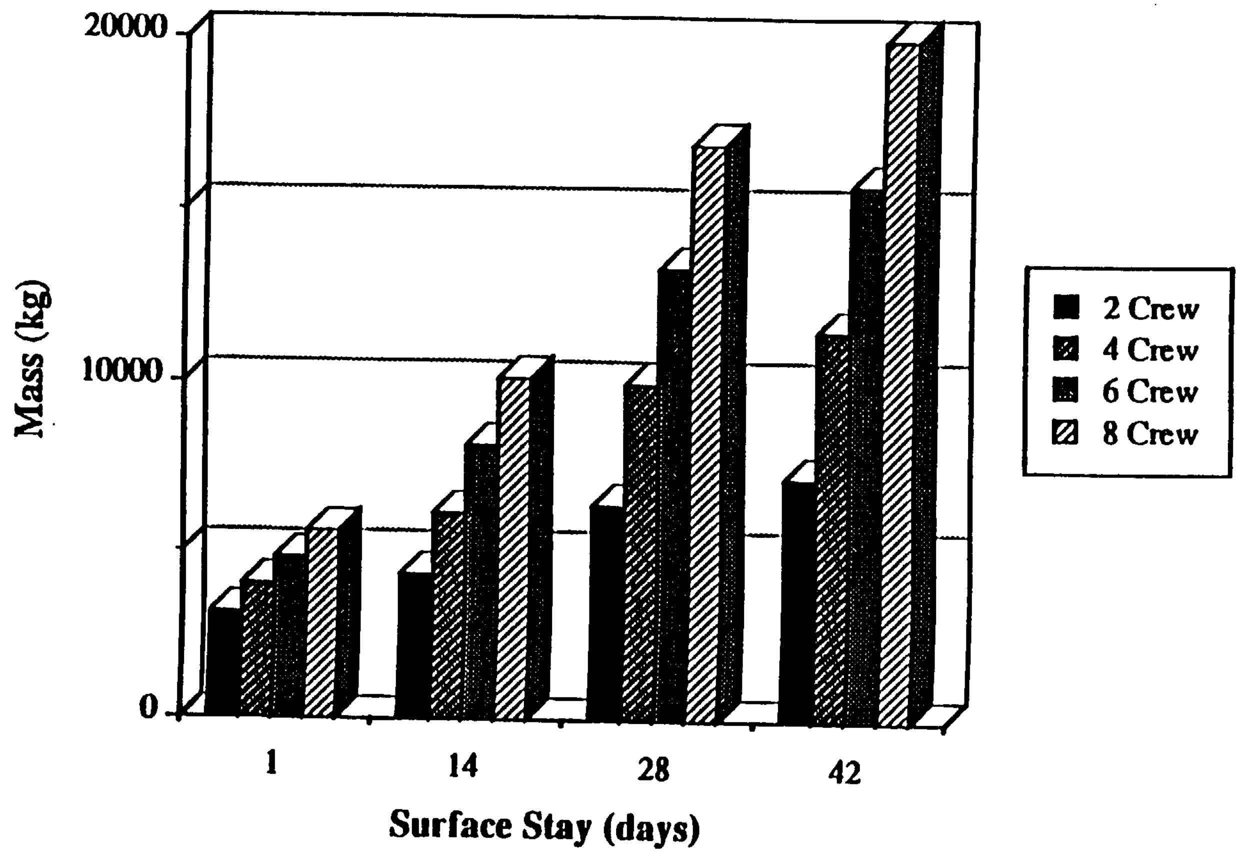


NOTE: All modules 4.4m diameter

Lunar Excursion Module Mass Summary

This chart shows the relative masses of the excursion modules shown on the previous chart. As shown, the modules for 6 and 8 crew for 28 and 42 day surface stays become unusually large and massive.

Lunar Excursion Module Mass Summary

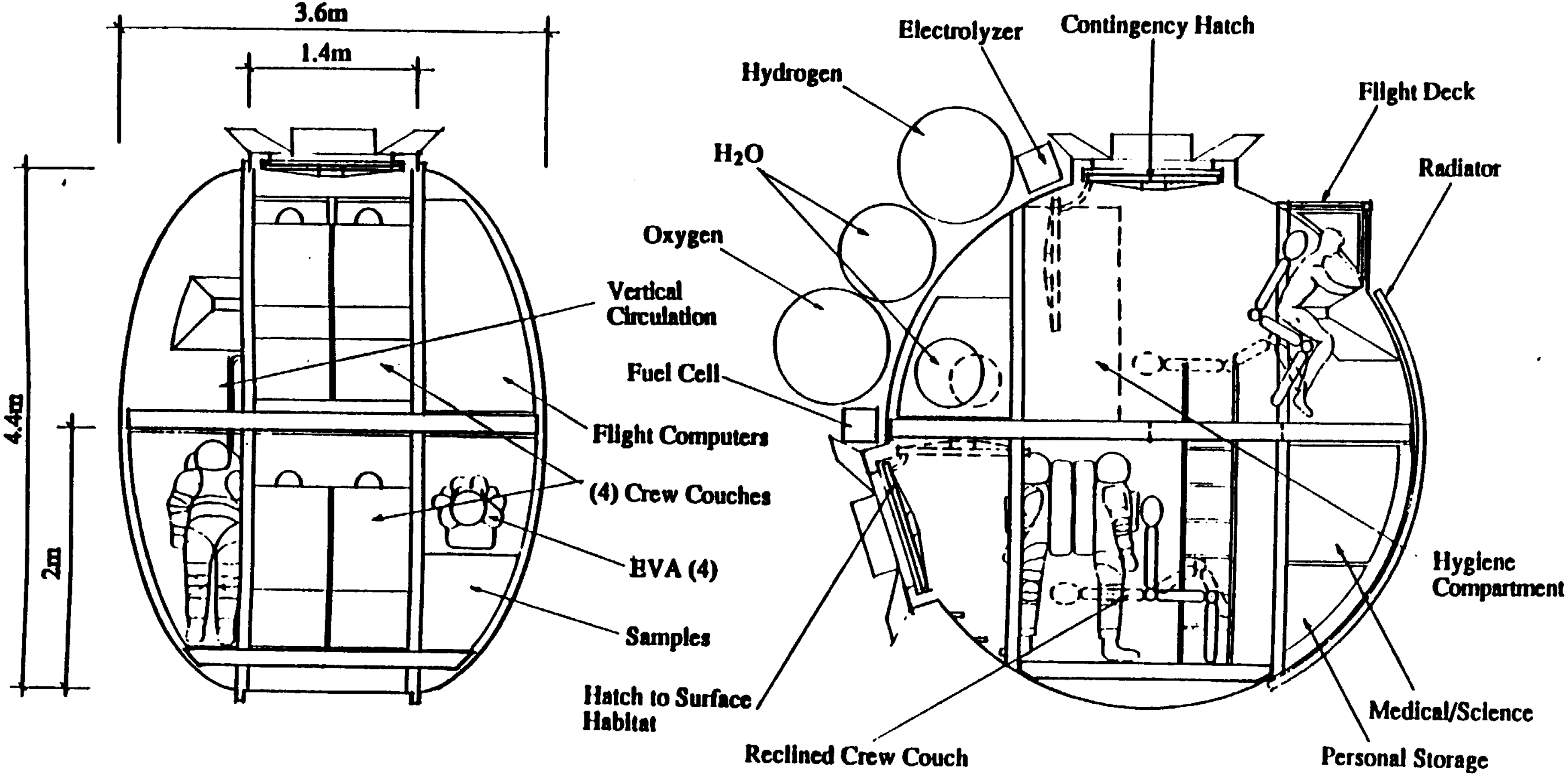


- **Operates in conjunction with a transfer vehicle**
- **1 day duration includes 1 repress**
- **14 and over durations include an airlock and 2 represses**

Common Short-Duration Crew Module

The next 2 charts show a detailed interior configuration of an excursion module that can accommodate 2 crew for a 1 or 14 day surface stay, and 4, 6 and 8 crew for a one day surface stay.

Common Short-Duration Crew Module



Cross Section

Long Section

- SSF diameter cylinder.
- All penetrations occur in cylinder section.
- All structural attachments at girth rings.
- Common ellipsoidal end domes.
- 43.6m3 total volume

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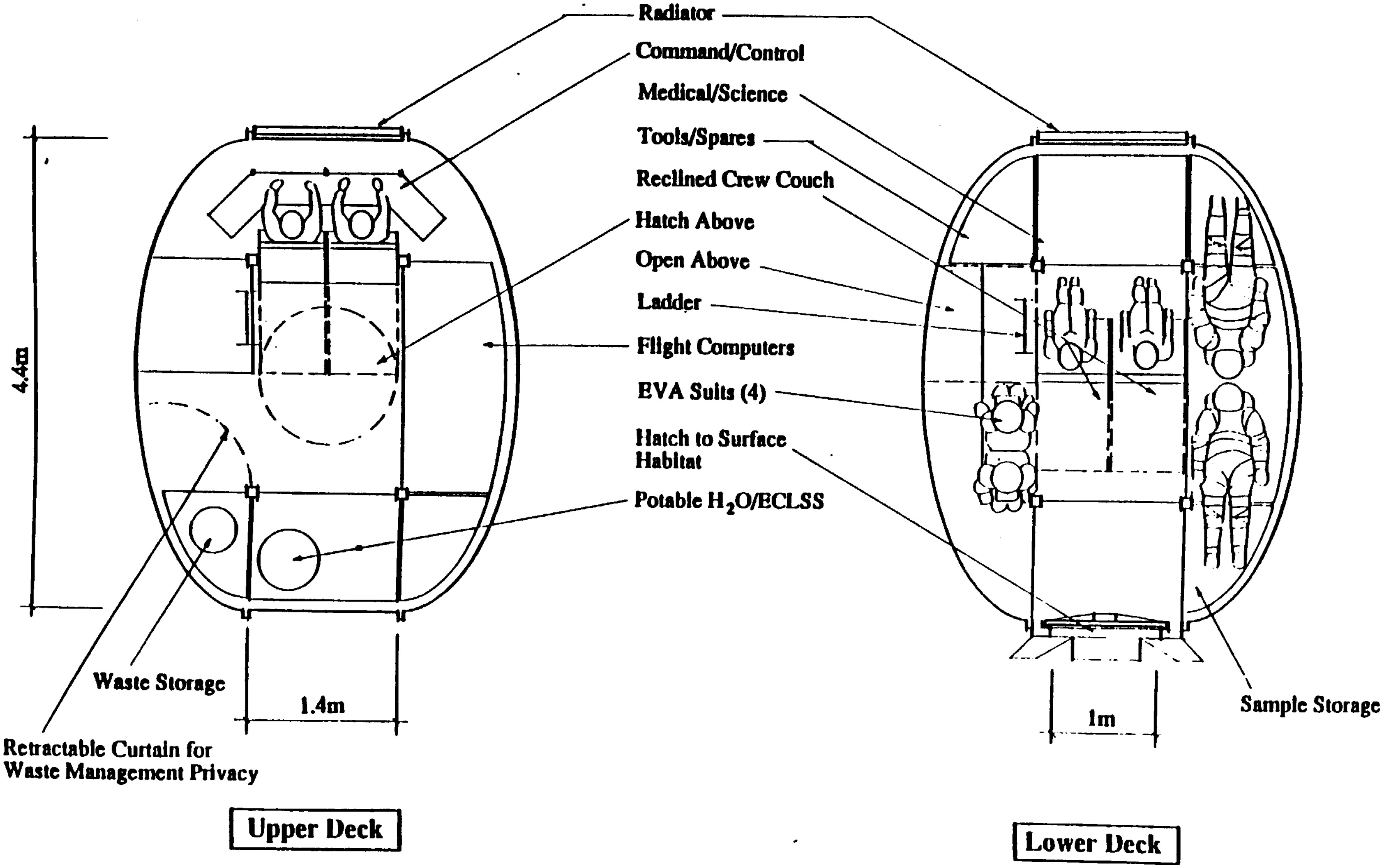
Common Short-Duration Crew Module



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Lunar Transportation Family (LTF)

Threefold Design Strategy

Tough Drivers:
- Current understanding of aerobraking geometry
- Mixed payload manifests, exchange operations, offloading (cargo & crew)
- Cryogen transfer
- Re-usability (retaining tanks also)
- Subsystem/component commonality for later Mars vehicles

Multiple Mission Modes
- Transportation infrastructure decisions will precede site selection, and should not constrain it
- Global lunar access must be preserved

Early Capability: "Campsite" operations (expandable mode, without aerobrake or LEV)

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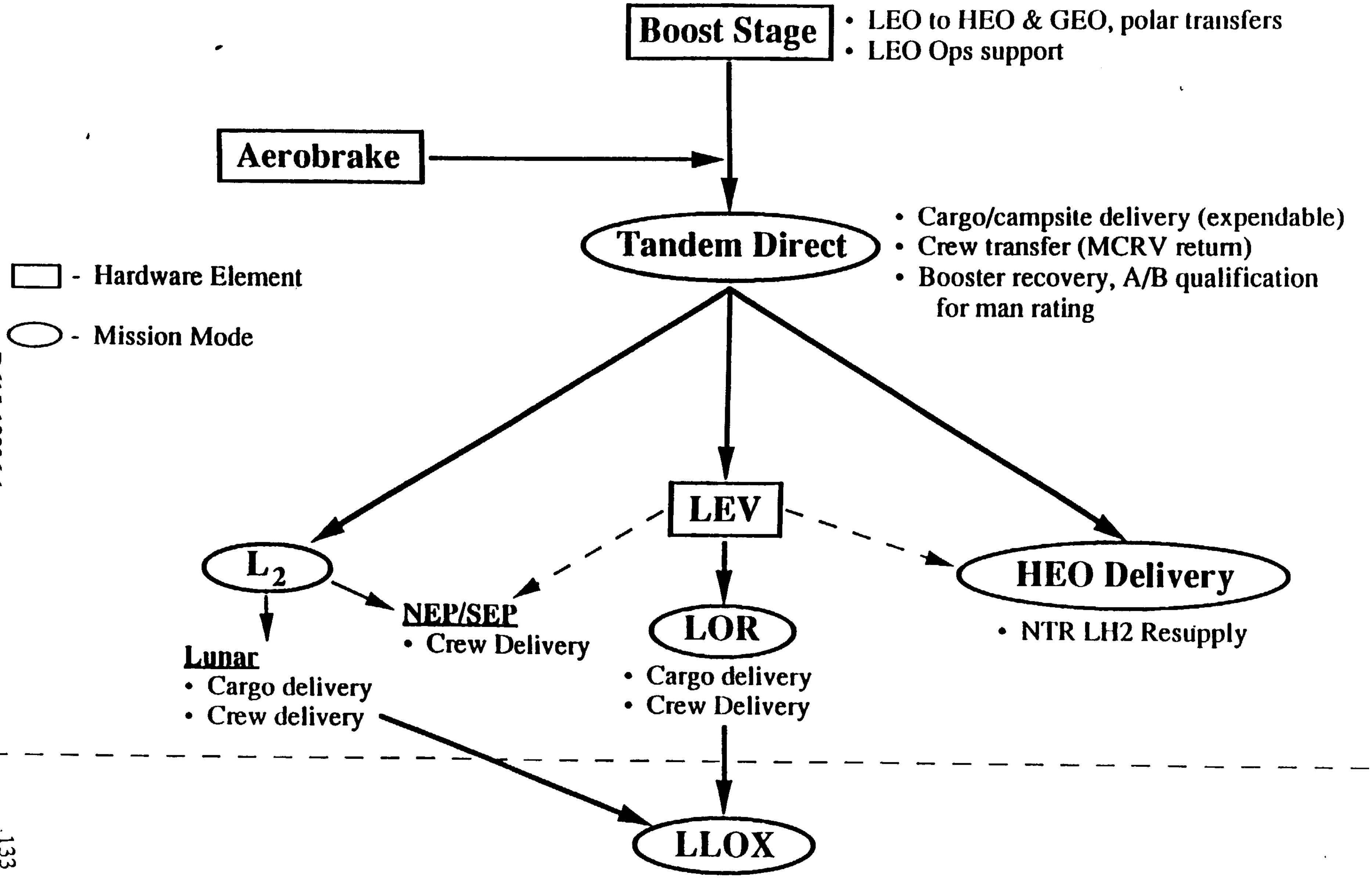
Lunar Transportation Family (LTF)
Preferred Evolution

The LTF concept evolves by beginning with minimal hardware development to phase the costs over the duration of the program. **First**, a boost stage is developed for Earth orbit-to-orbit transfers and operations support in LEO. Completely expendable tandem-direct missions are flown to the lunar surface at this point using an MCRV for crew return. **Secondly**, when the aerobrake comes on-line, tandem-direct missions which recover the boost stage are flown for aerobrake flight qualification for man-rating. Using the same hardware, L_2 missions are flown for crew delivery to NEP/SEP vehicles as well as LOR lunar crew and cargo delivery. NTR LH_2 resupply missions can also be flown at this point to provide propellant to get the NTR to SSF orbit for refurbishment. **Third**, an LEV is brought on line so that classical LOR missions are possible to deliver both crew and cargo to the surface. **Fourth**, LLOX usage becomes available and the missions to the lunar surface begin to take advantage of surface refueling to reduce Earth-to-orbit transfers of propellant.

Lunar Transportation Family

Preferred Evolution



Boost Stage
- LEO to HEO & GEO, polar transfers
- LEO Ops support

Aerobrake

Tandem Direct
- Cargo/campsite delivery (expendable)
- Crew transfer (MCRV return)
- Booster recovery, A/B qualification for man rating

☐ - Hardware Element

○ - Mission Mode

LEV

L_2

NEP/SEP
- Crew Delivery

Lunar
- Cargo delivery
- Crew delivery

LOR
- Cargo delivery
- Crew Delivery

HEO Delivery
- NTR LH2 Resupply

LLOX

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Lunar Transportation Family
Systems Required for Varying Missions

11 different mission modes have been identified and are matrixed with the 5 major lunar transportation family systems. This chart identifies the number of each system required to accommodate the mission modes identified. The last column shows the propellant required from earth for each mission mode, which points out the efficiencies of the LLOX cases once LLOX is available. In the NEP/SEP Crew Delivery mode, the propellant capacity required is not substantially larger than the 25 t LEV propellant capacity. Therefore, this may be a rationale to begin thinking about larger LEV stages that can fulfill NEP/SEP crew delivery as well as extend "hopper" distances on the lunar surface.

Lunar Transportation Family

Systems Required for Varying Missions

Mission Mode / System	110 t Propulsion Stage	25 t Propulsion Stage	Transfer Hab	Crew Cab	Aerobrake	Propellant From Earth (kg)
Tandem Direct Cargo	2	---	---	---	(1)	201,553
Tandem Direct Crew	2	---	1	---	(1)	163,093
Tandem Direct Large Cargo	2	---	---	---	(1)	201,553
LOR Cargo	1	1	---	---	1	130,865
LOR Crew and Cargo	1	1	1	1	1	123,181
LOR Cargo using LLOX	1	1	---	---	1	65,109
LOR Crew using LLOX	1	1	1	1	1	73,274
Cargo Delivery Through L_2	2	---	---	---	1	159,153
Crew Delivery Through L_2	2	---	1	1	1	159,145
NTR Resupply	1	---	---	---	1	42,021
NEP/SEP Crew Delivery	1	---	1	---	1	33,556

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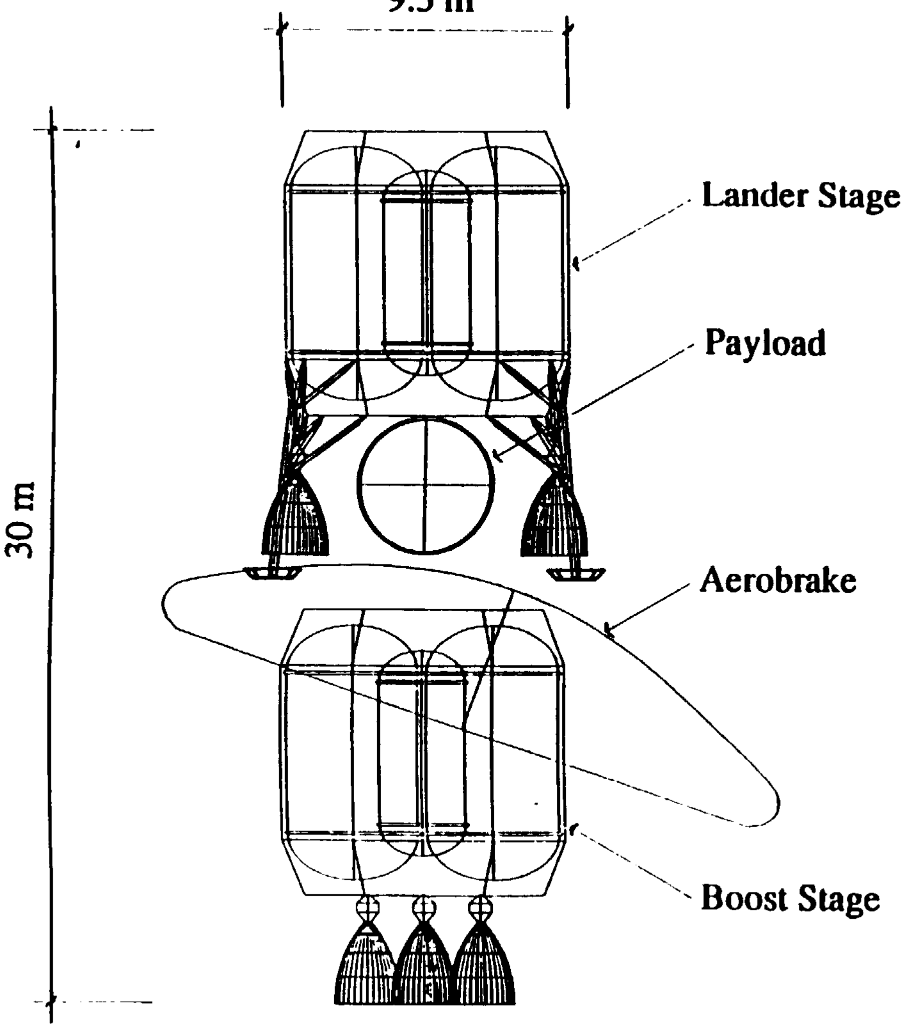
Lunar Transportation Family Configurations

The next 11 charts show configurations for the evolution of the lunar transportation family with accompanying mass statements. 5 major hardware elements were developed and assembled in different configurations to support 11 different mission modes. This kind of system approach minimizes the major elements required, phases in more complex hardware elements, and provides for commonality throughout the system. Although 11 mission modes are identified and represented, their are other possible missions that can be flown using this hardware, such as the cargo and crew delivery missions through L_2 can be flown with LLOX.

Lunar Transportation Family
Tandem-Direct Cargo



Mass Statement

Boost Stage Inerts	10,638
Boost Propellant	108,950
Aerobrake	3,000
Lander Stage Inerts	11,113
Lander Propellant	92,603
Payload	60,000
IMLEO (in kg)	**286,304**

- Boost stage flies without an aerobrake initially in an all expendable mode

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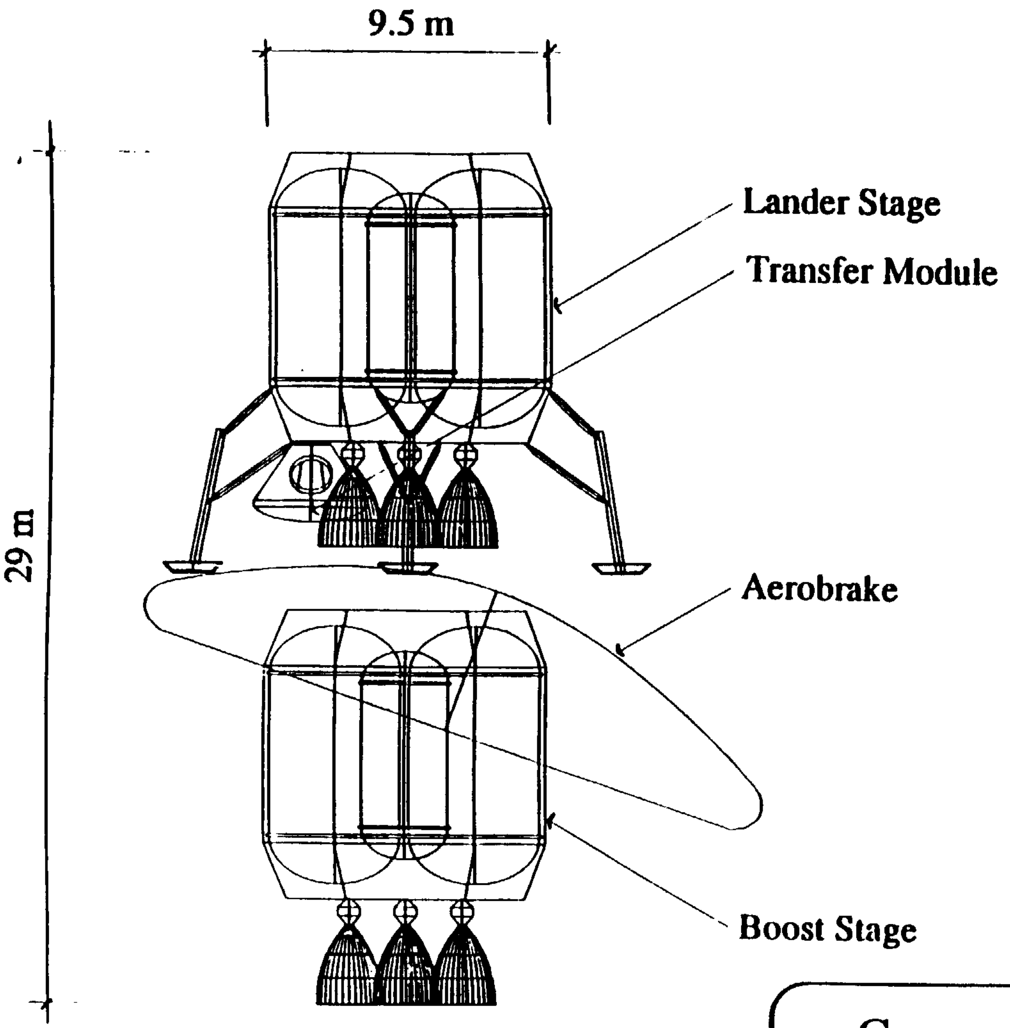
Lunar Transportation Family

Tandem-Direct Crew



Mass Statement

Boost Stage Inerts	10,638
Boost Propellant	78,647
Aerobrake	3,000
Lander Stage Inerts	11,113
Lander Propellant	84,436
Transfer Module	13,633
Payload	5,000
IMLEO (in kg)	206,467

- Can send up to 8 crew in this mode for crew changeout in early base phases

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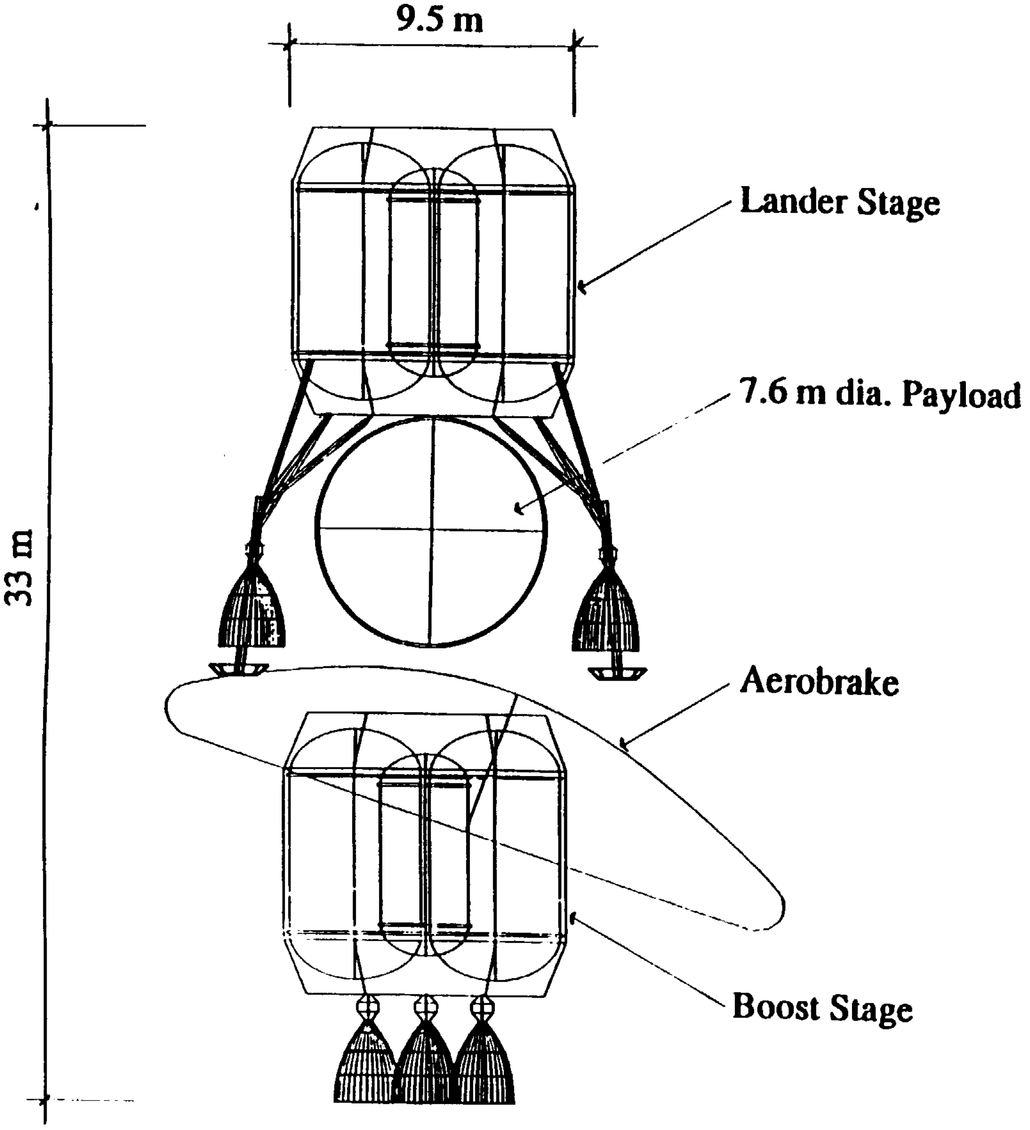
Lunar Transportation Family

Tandem-Direct Large Cargo



Mass Statement

Boost Stage Inerts	10,638
Boost Propellant	108,950
Aerobrake	3,000
Lander Stage Inerts	11,113
Lander Propellant	92,603
Payload	60,000
IMLEO (in kg)	**286,304**

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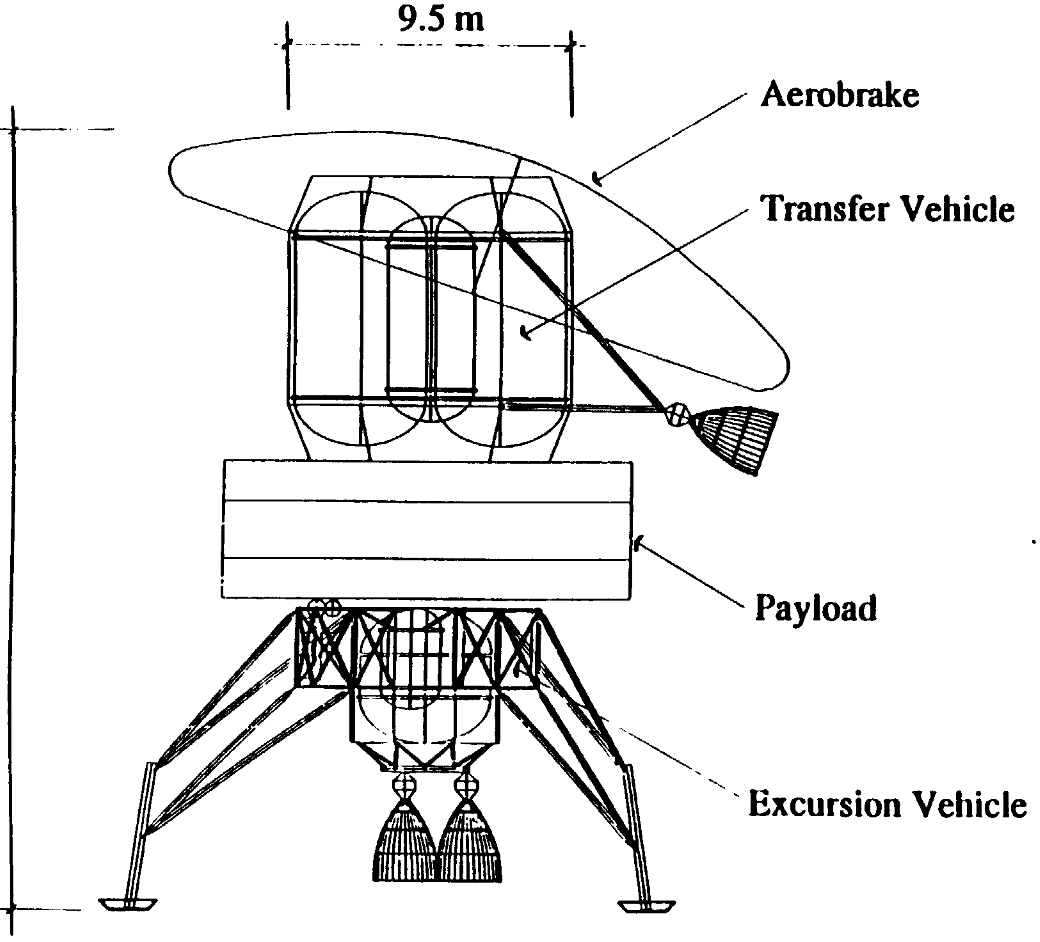
ADVANCED CIVIL
SPACE SYSTEMS

Lunar Transportation Family
LOR Cargo

BOEING



Mass Statement (LEV Delivered)

Transfer Vehicle Inerts	10,638
Transfer Propellant	107,719
Aerobrake	3,000
Excursion Vehicle Inerts	5,592
EV Propellant	23,151
Payload	24,000
IMLEO (in kg)	174,100

Mass Statement (LEV Refueled)

Transfer Vehicle Inerts	10,638
Transfer Propellant	107,100
Aerobrake	3,000
Resupply Tanks	2,377
Resupply Propellant	23,765
Payload	25,000
IMLEO (in kg)	171,880

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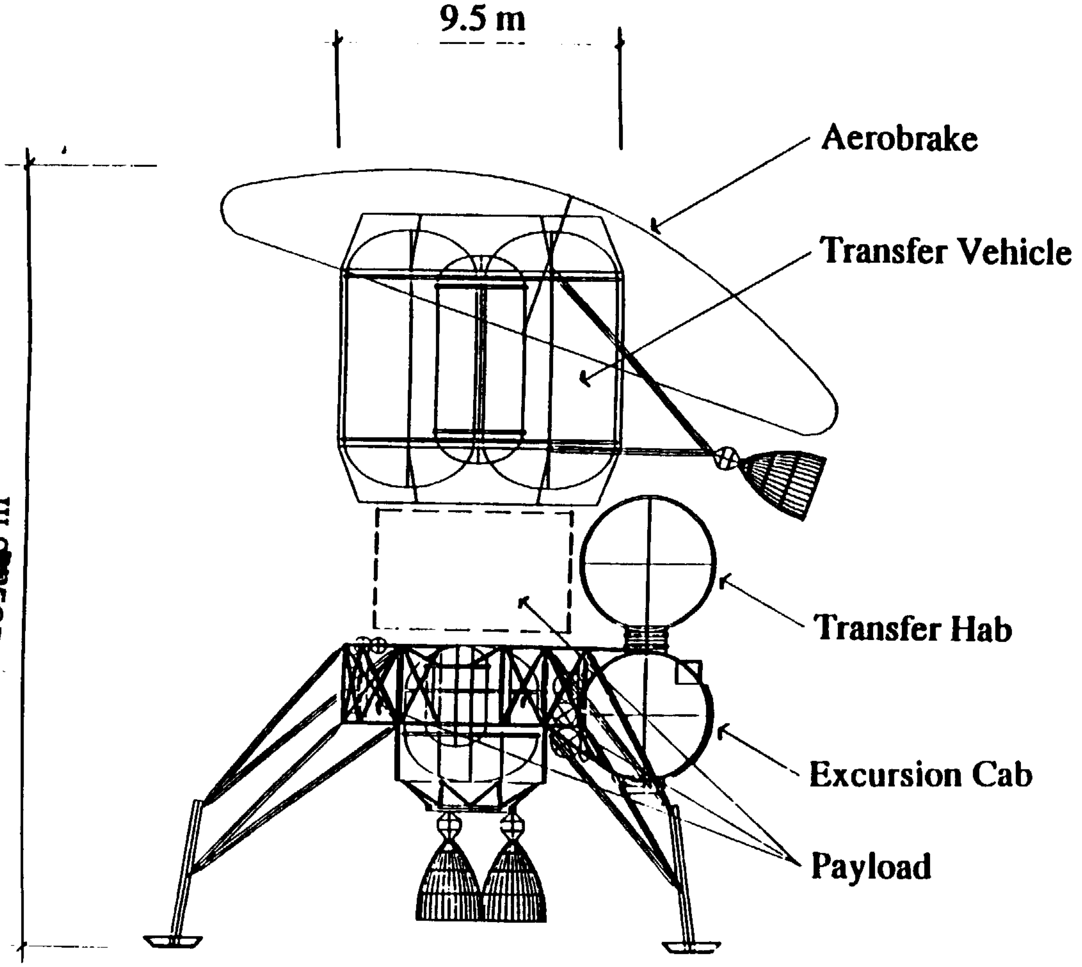
ADVANCED CIVIL
SPACE SYSTEMS

Lunar Transportation Family
LOR Crew and Cargo

BOEING



Mass Statement (LEV Delivered)

Transfer Vehicle Inerts	10,638
Transfer Propellant	106,602
Aerobrake	5,000
Transfer Hab	7,500
Excursion Vehicle Inerts	5,592
EV Propellant	19,768
Excursion Cab	4,000
Payload	8,000
IMLEO (in kg)	**167,100**

Mass Statement (LEV Refueled)

Transfer Vehicle Inerts	10,638
Transfer Propellant	102,184
Aerobrake	5,000
Transfer Hab	7,500
Resupply Tanks	2,100
Resupply Propellant	20,997
Payload	10,000
IMLEO (in kg)	**158,419**

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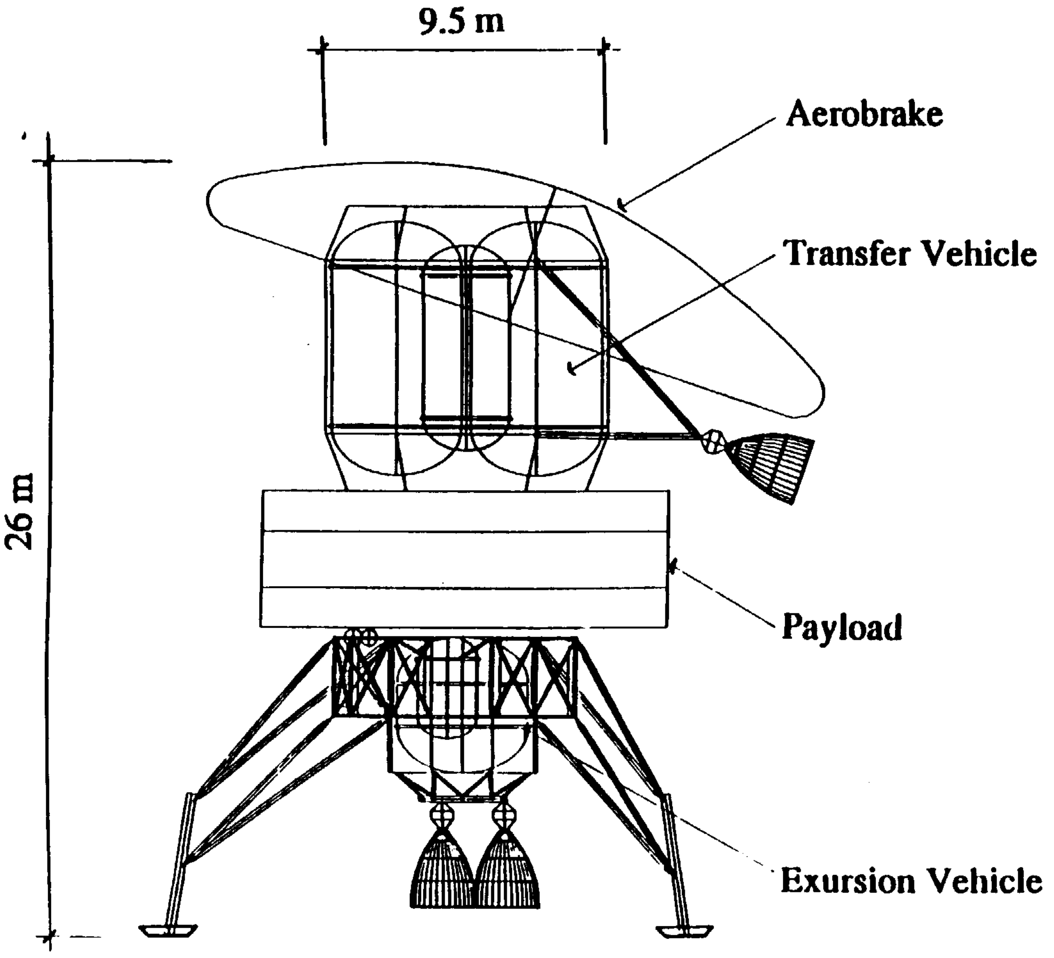
ADVANCED CIVIL
SPACE SYSTEMS

Lunar Transportation Family
LOR Cargo using LLOX

BOEING



Mass Statement (LEV Refueled)

Transfer Vehicle Inerts	10,638
Transfer Propellant	61,645
Aerobrake	3,000
Resupply Tanks	624
Resupply Propellant (LH2)	3,464
Payload	17,000
IMLEO (in kg)	**96,371**

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ADVANCED CIVIL SPACE SYSTEMS

Lunar Transportation Family
LOR Crew using LLOX

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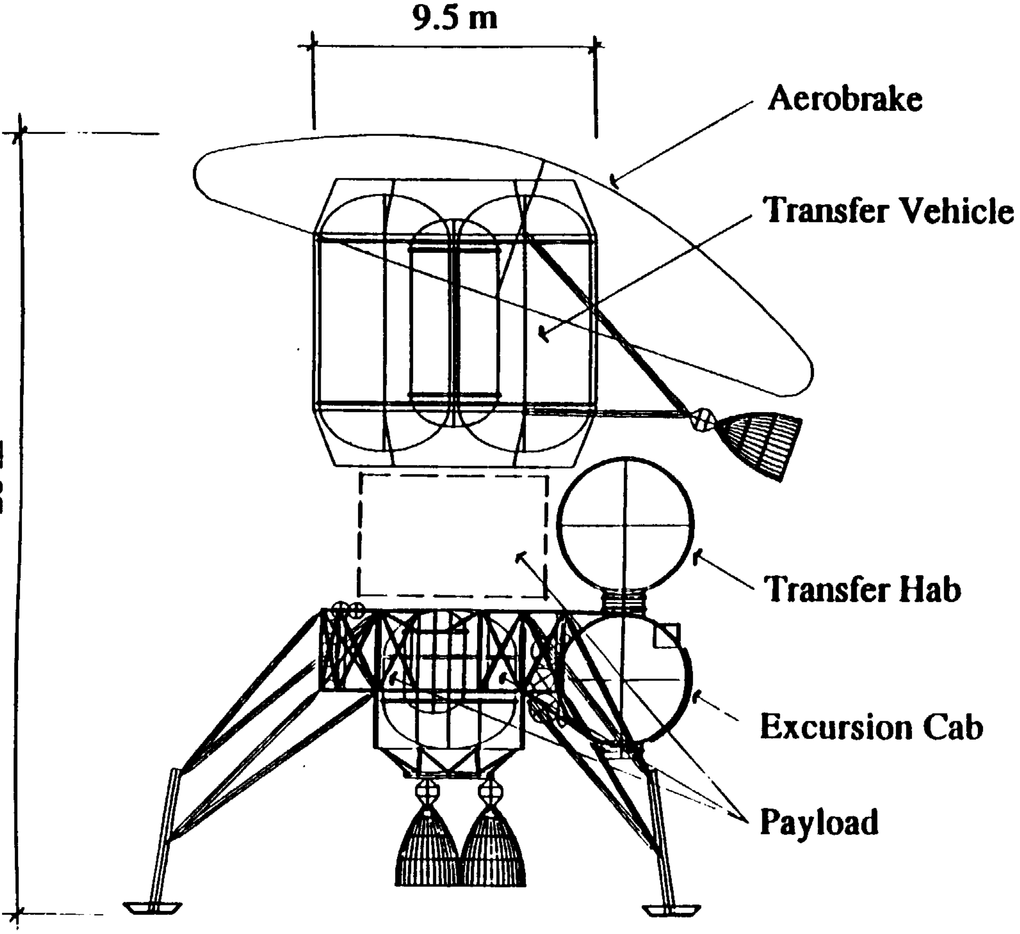

Mass Statement (LEV Refueled)	
Transfer Vehicle Inerts	10,638
Transfer Propellant	70,067
Aerobrake	5,000
Transfer Hab	7,500
Resupply Tanks	577
Resupply Propellant (LH2)	3,207
Payload	8,000
IMLEO (in kg)	**104,989**

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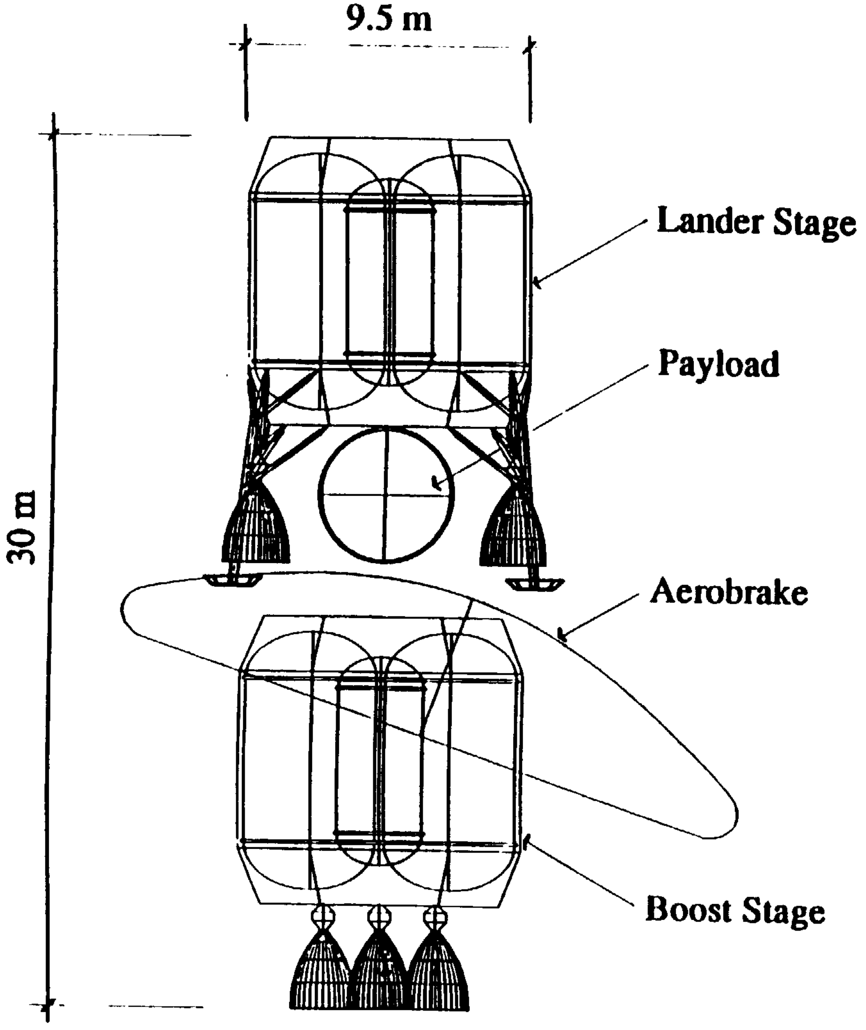
Lunar Transportation Family

Cargo Delivery Through L_2



Mass Statement (Lander Delivered)

Boost Stage Inerts	10,638
Boost Propellant	107,803
Aerobrake	3,000
Lander Stage Inerts	11,113
Lander Propellant	45,637
Payload	18,000
IMLEO (in kg)	**196,191**

Mass Statement (Lander Refueled)

Boost Stage Inerts	10,638
Boost Propellant	107,734
Aerobrake	3,000
Resupply Propellant	51,419
Payload	24,000
IMLEO (in kg)	**196,791**

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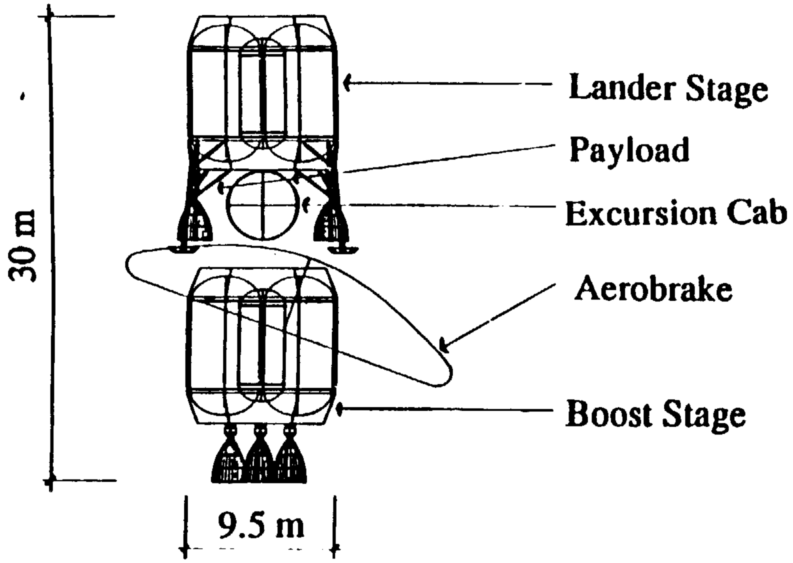
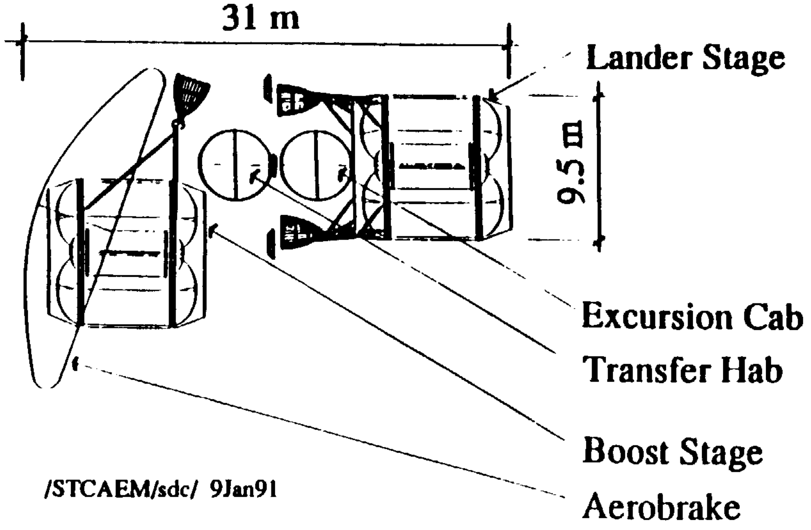
Lunar Transportation Family
Crew Delivery Through L_2

Unmanned Lander Delivery to L_2



Lander Refueled



Mass Statement (Lander Delivered)

Boost Stage Inerts	10,638
Boost Propellant	107,877
Aerobrake	3,000
Lander Stage Inerts	11,113
Lander Propellant	49,415
Excursion Hab	4,000
Payload	10,000
IMLEO (in kg)	**196,043**

Mass Statement (Lander Refueled)

Boost Stage Inerts	10,638
Boost Propellant	107,803
Aerobrake	5,000
Transfer Hab	7,500
Resupply Propellant	51,342
Payload	12,000
IMLEO (in kg)	**194,283**

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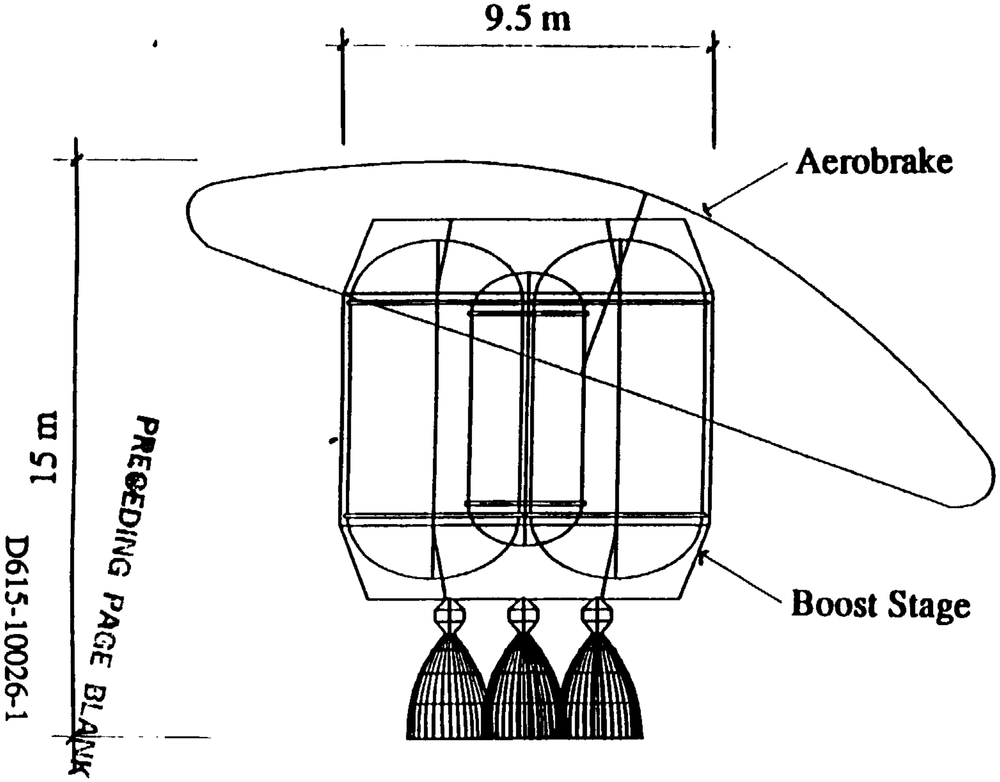
Lunar Transportation Family
NTR Resupply



Mass Statement

Boost Stage Inerts	10,638
Outbound Propellant	41,334
Inbound Propellant	687
Aerobrake	5,000
Resupply LH2	30,000
Resupply Tanks	6,600
IMLEO (in kg)	**94,259**

- Delivered resupply propellant is used to transfer NTR to SSF orbit

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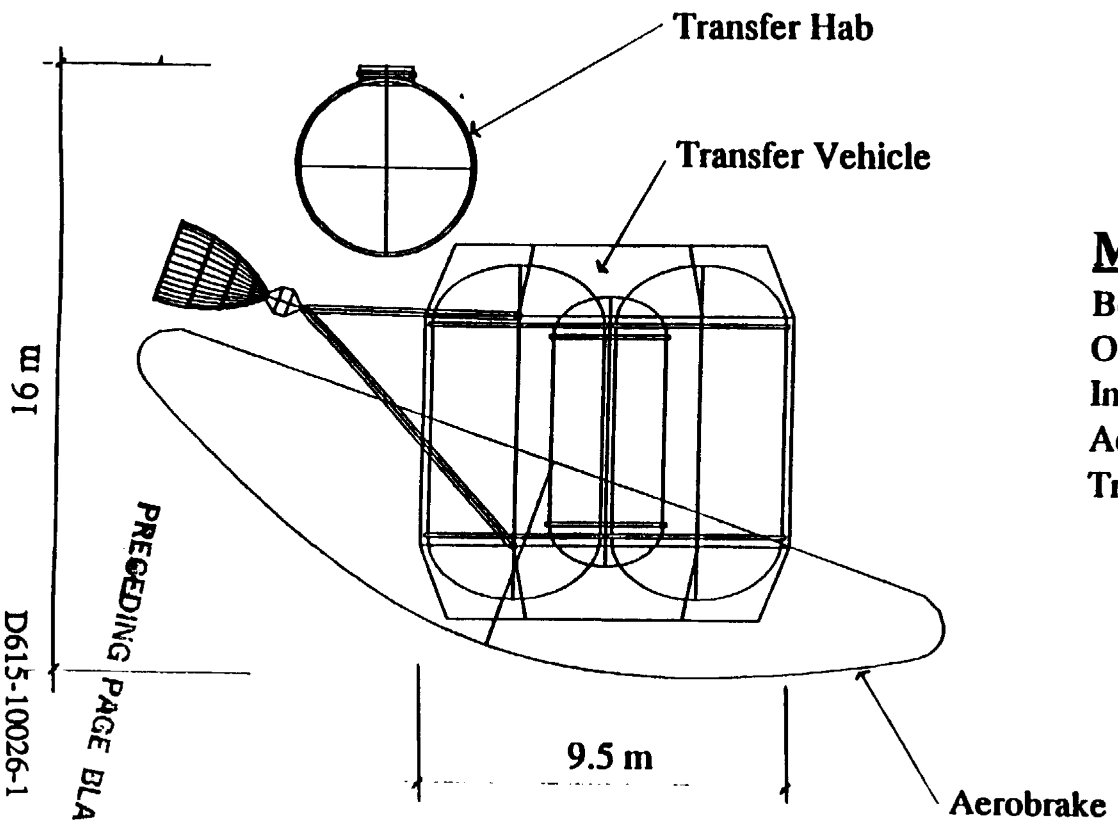
Lunar Transportation Family
NEP/SEP Crew Delivery



Mass Statement

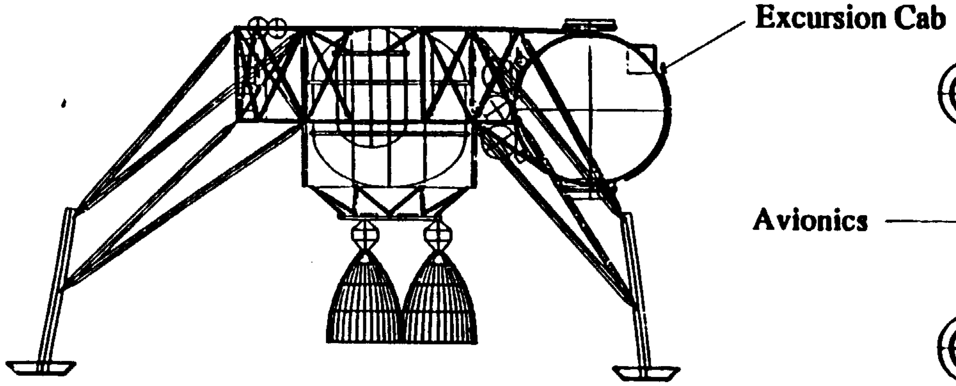
Boost Stage Inerts	10,638
Outbound Propellant	30,624
Inbound Propellant	2,932
Aerobrake	5,000
Transfer Hab	7,500
IMLEO (in kg)	**56,694**

- Assumes LEO to L_2 transfer & aerobrake in LEO return

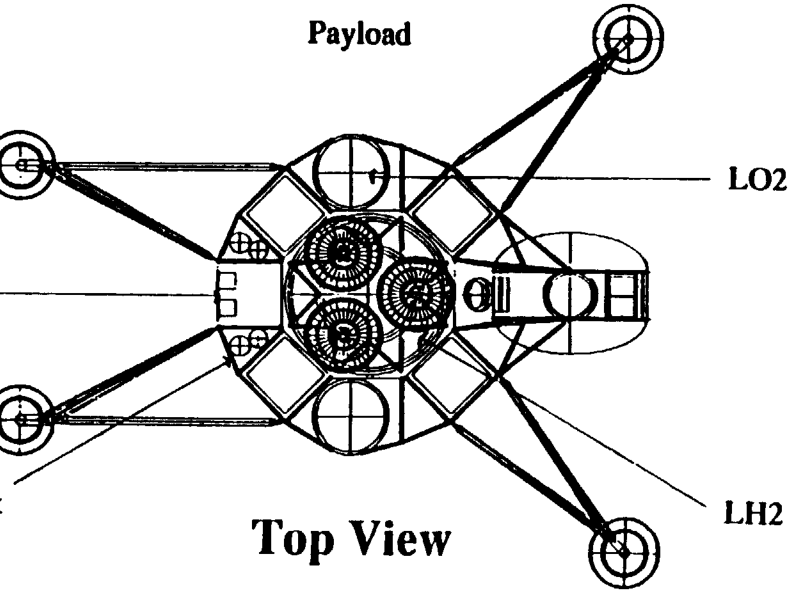
LOR Excursion Vehicle

The LOR Excursion Vehicle shown is a flexible design that can accommodate varying payload sizes, crew + cargo, or cargo only. With the payload imbedded in the lander structure (in standard cargo containers) in the crew mode, this configuration offers the ability to descent abort to LLO by dropping the payload and the landing legs (to have enough propellant to make orbit) and thrusting back to LLO. The placement of the crew cab on the side allows for easy surface access by the crew as well as more direct crew visibility upon landing over top mounted crew cabs. The C.M. shift by placing the crew cab on the side is offset by moving the LO2 off-center and putting the RCS propellant and the avionics on the opposite side of the crew cab. In the cargo-only mode, the crew cab is absent and large cargos can be attached to the top of the lander structure to be unloaded by a straddler. The "triangular" nature of the landing legs is caused by the configuration and size of the straddler, in order for the straddler to easily maneuver over the lander to unload payload.

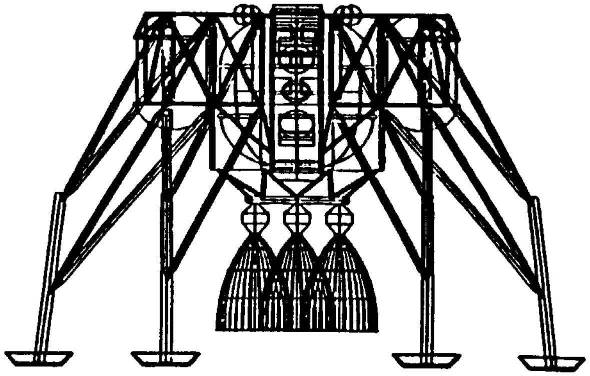
LOR Excursion Vehicle



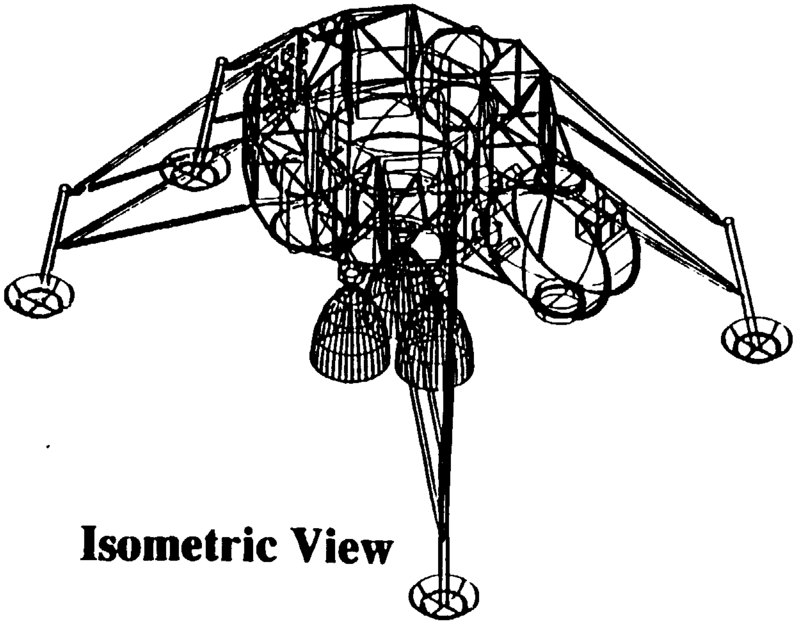
Side View



Top View



Front View



Isometric View

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Long-duration Habitat Trade Study
Contents (1)

ADVANCED
CIVIL
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SYSTEMS

Long-duration Habitat Trade Study
Contents (2)

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Geometry Analysis

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Long-duration Habitat Trade Study
Contents (3)

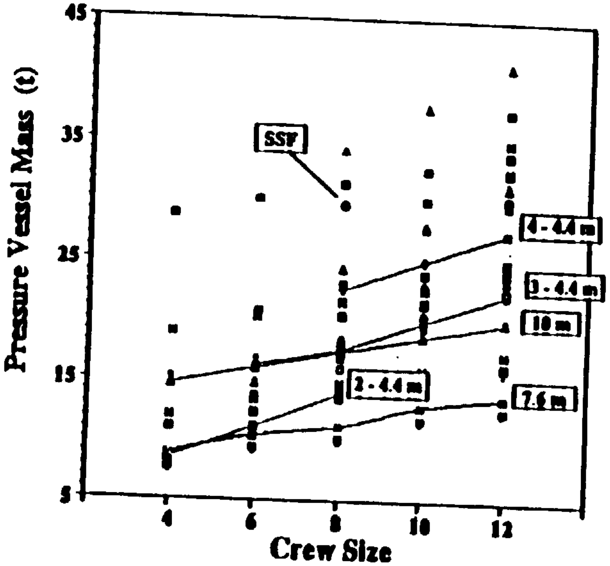
Long-duration Habitat Trade Study Summary

This chart summarizes the process and results of an extensive trade study to compare alternative concepts for long-duration habitats.

Long-duration Habitat Trade Study Summary

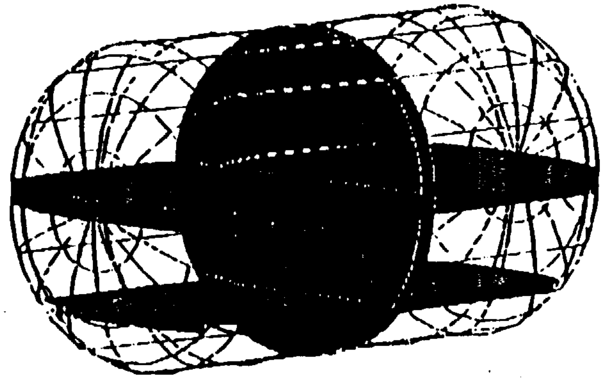
Process

- Trade space matrixed 5 crew sizes and 3 module sizes
- Generated 1480 distinct options, based on gravity, orientation, topology and structure; focused on 150 concepts
- Developed metrics for selecting preferred topologies and geometries; reference configurations for crew response survey
- Weighed pressure vessel structures, estimated equipment outfitted weights; assessed integration impact, commonality, growth potential, manufacturing options



Results

- Generated data allow applying a wide variety of priority sets to determine "optimal" concepts for specific architectures
- First HEI decade can use lightened SSF derivatives for all crew systems: LTV, LEV, surface outposts, safe-havens
- Later, long-duration missions require a larger module, common across architectures and capable of integration with smaller modules
- Trade neckdown led to synthesizing novel module concept, using best features from the studied options
- A 7.6 m diameter vessel, "tunnel-oriented", sized for 6 crew, with a cross-sectional bulkhead, was selected as the reference modular unit



Motivation

This chart explains how the long-duration hab trade study came about, and why its results are critical for further vehicle concept definition in the STCAEM study.

MTV Habitat Trade Study
Motivation

Why trade hab concepts? Mars Transfer Duration & Environment

- 1020 d design duration (SSF is 90 d)
- Deep space (SSF is in LEO)
- No escape, no resupply, no crew rotation

What are the major options? Module Size, Diameter & Number

- Space habitats have traditionally taken advantage of the maximum "throw" diameter of their launch vehicles -- HEI launch vehicles are large
- Volume is at a premium due to mass & packaging

Why is a choice necessary for STCAEM? Vehicle Integration

- Mass more critical for transportation systems than for LEO facilities
- Crew system is the MTV payload; comprises about 1/4 of MTV mass
- Sizes propulsion system, structure, aerobrake (if one)
- Constrains integrated vehicle configuration for some propulsion options

Habitat Module Evolutionary Context

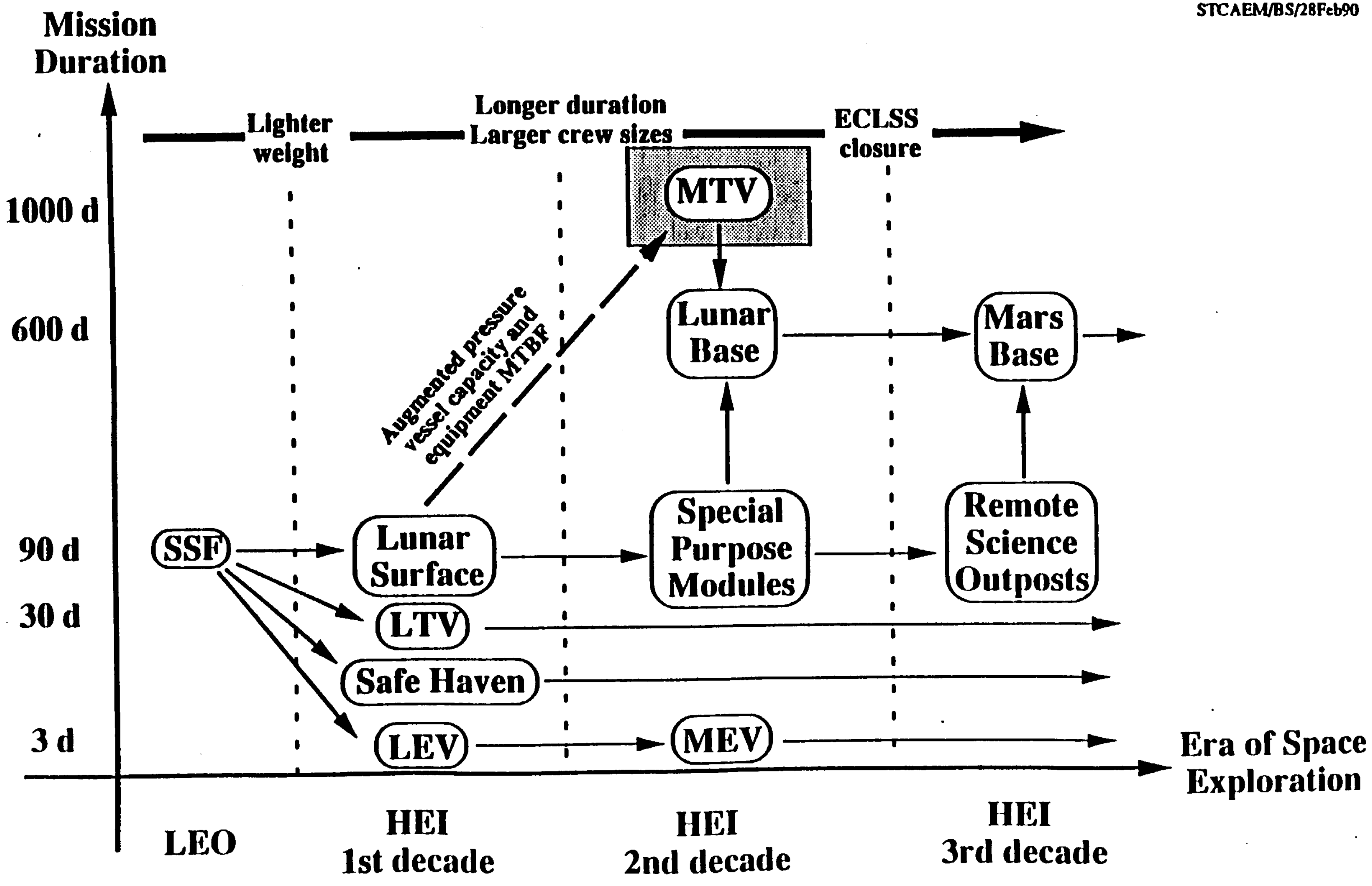
The gross division of HEI into three functional decades subsequent to the 1990s helps organize thinking about habitat system requirements. Here individual crew member mission duration is plotted against program phase, to generate a space populated by various kinds of habitation systems.

SSF-derivatives can serve a great many HEI functions, including crew cabs for LTV, LEV and MEV concepts as well as several kinds of unique applications in space and on planetary surfaces. The key new requirement is decreased structure mass, since uses depending on deep-space transportation are more sensitive to mass than are permanent LEO facilities like SSF.

Some applications, including consolidation-phase surface bases and especially the MTV, must be designed for crew-rotation durations an order of magnitude longer than those best served by the SSF-derivatives. Key new requirements are enhanced equipment reliability and augmented pressure vessel capacity. This trade study therefore concentrates on the MTV application, targeting extremely long durations and the 2nd decade of HEI operations.

Regardless of specific results, we would expect advanced habitation systems (such as planetary bases) to be comprised of both kinds (SSF-derived and advanced) of elements.

Habitat Module Evolutionary Context



Mission Duration

1000 d
600 d
90 d
30 d
3 d

Lighter weight

Longer duration
Larger crew sizes

ECLSS closure

Augmented pressure vessel capacity and equipment MTBF

SSF
Lunar Surface
LTV
Safe Haven
LEV
MTV
Lunar Base
Special Purpose Modules
MEV
Mars Base
Remote Science Outposts

Era of Space Exploration

LEO
HEI 1st decade
HEI 2nd decade
HEI 3rd decade

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MTV Habitat Trade Study
Goals

- Generate and evaluate a reasonably inclusive set of habitat options suitable for evolutionary, deep-space, long-duration missions

- Develop and apply a set of metrics which include criteria of:
  - **Functionality** (Does the concept work?)
  - **Perception** (How will the crew respond?)
  - **Integration** (Does the concept fit into the mission architecture?)
  - **Cost** (Is the concept technologically and programmatically affordable?)

- Develop and present the trade data transparently, so that they can be used for a variety of concept selections under different circumstances

- Determine criteria weighting appropriate for STCAEM goals: light weight, vehicle integration, evolution & growth, commonality

- Select a reference concept for immediate application in current MTV concept definition for the STCAEM Study

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MTV Habitat Trade Space

The fundamental trade space addressed by the study is displayed in matrix form here, plotting five crew sizes against three fundamental sizes of module. Because of the critical constraint of launch vehicle capacity, the candidate module diameters were chosen as: identical with SSF; the 25' diameter commonly discussed for an HEI Shuttle-C or a small ALS shroud, and the 33' diameter which has been suggested for a larger ALS shroud. The study spanned the trade space as shown with combinations of geometrical, weight and configuration analyses.

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MTV Habitat Trade Space

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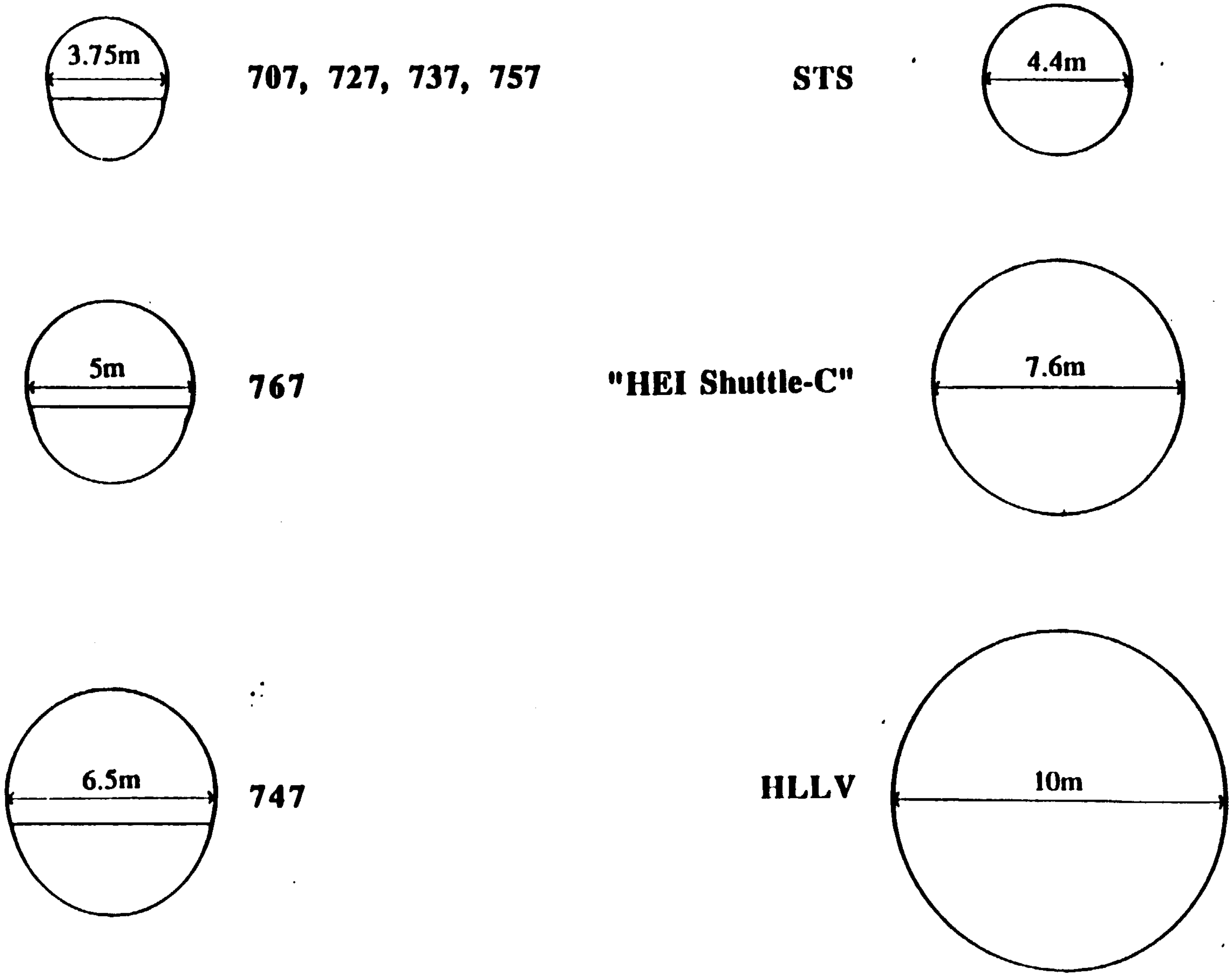
Module Type Crew Size	**Baseline SSF** All equipment identical; "sending SSF to Mars"	**4.4 m Diameter** SSF diameter; saddle hatches; unpenetrated end domes; optimized rack frames;	**7.6 m Diameter** Lunar Shuttle-C diameter; other features unchanged	**10 m Diameter** Mars HLLV diameter; other features unchanged
4		▨	▒ 90-d Study MTV hab concept	▨
6		▒	▒	▒
8	"Off-the-shelf" SSF configurations and weights	▨	▒	▨
10		▒	▒	▒
12		▨	▒	▨

Analysis techniques key: ▒ Structure weight, module geometry, topology ▨ Weight, geometry & configuration

Pressurized Cabin Diameter Comparison

This diagram compares, to scale, the cross-sections of several familiar aircraft, the SSF-diameter module size launchable with the NSTS, and the larger-diameter options considered by this trade study.

Pressurized Cabin Diameter Comparison

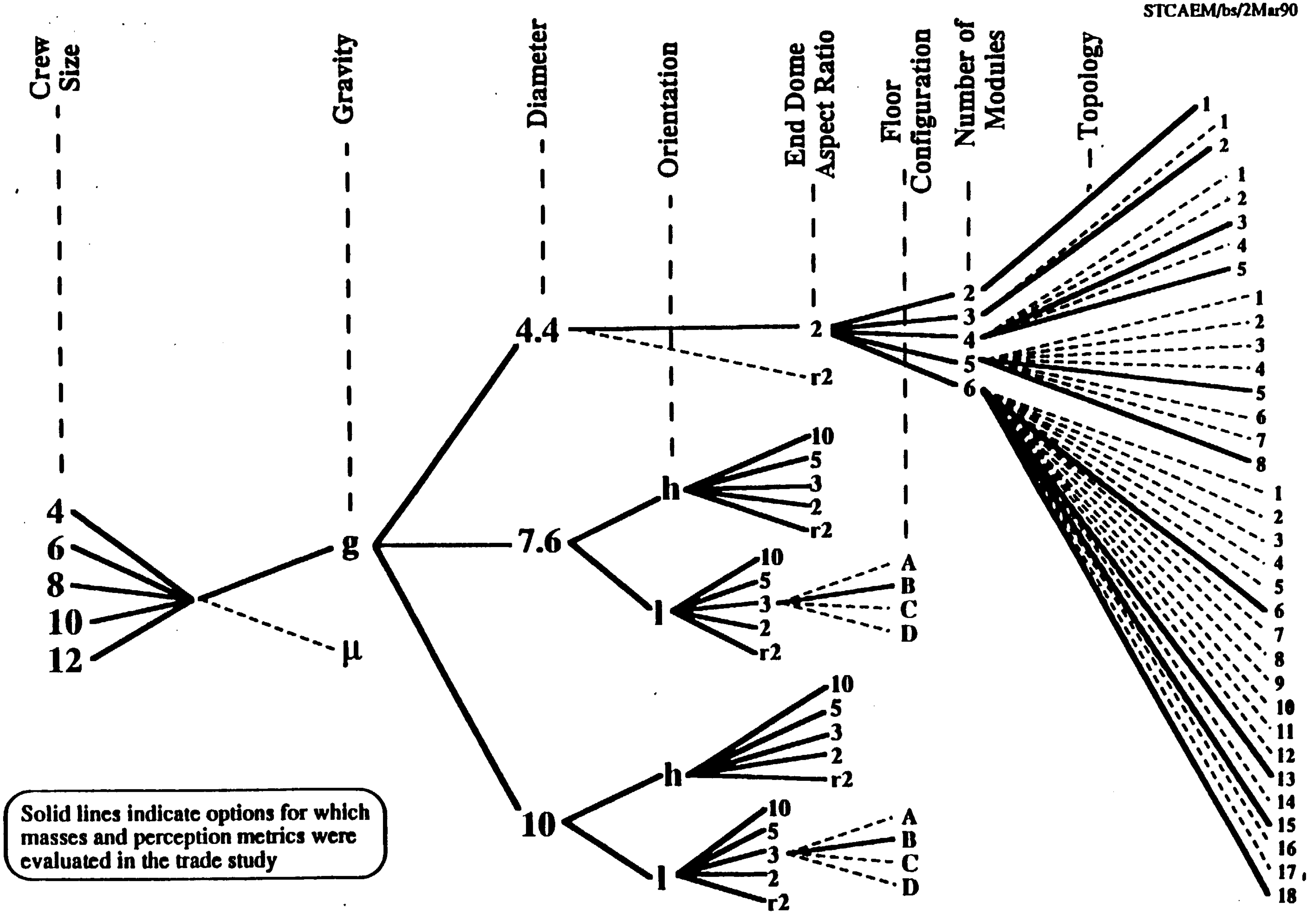


MTV Hab Trade Tree

This diagram shows the parameters varied in the study, to elaborate the trade space:
Crew Size
Gravity Requirement (binary alternative)
Diameter
Orientation of Floors (for the medium and large diameters only --- "h" means high, or stacked like sliced bologna; "l" means long, or arranged like a tunnel on its side)
End Dome Aspect Ratio (five options for the medium and large diameter modules; just two of those for the small module, approximating SSF module end shapes)
Floor Configuration (where in the circular cross section the floors are located for the tunnel-arranged medium and large diameter modules)
Number of Modules (in the clusters of small modules)
Topology (geometrical arrangement, and interconnection, of the cluster options)

The total number of distinct options generated by this trade tree is 1480.

MTV Hab Trade Tree

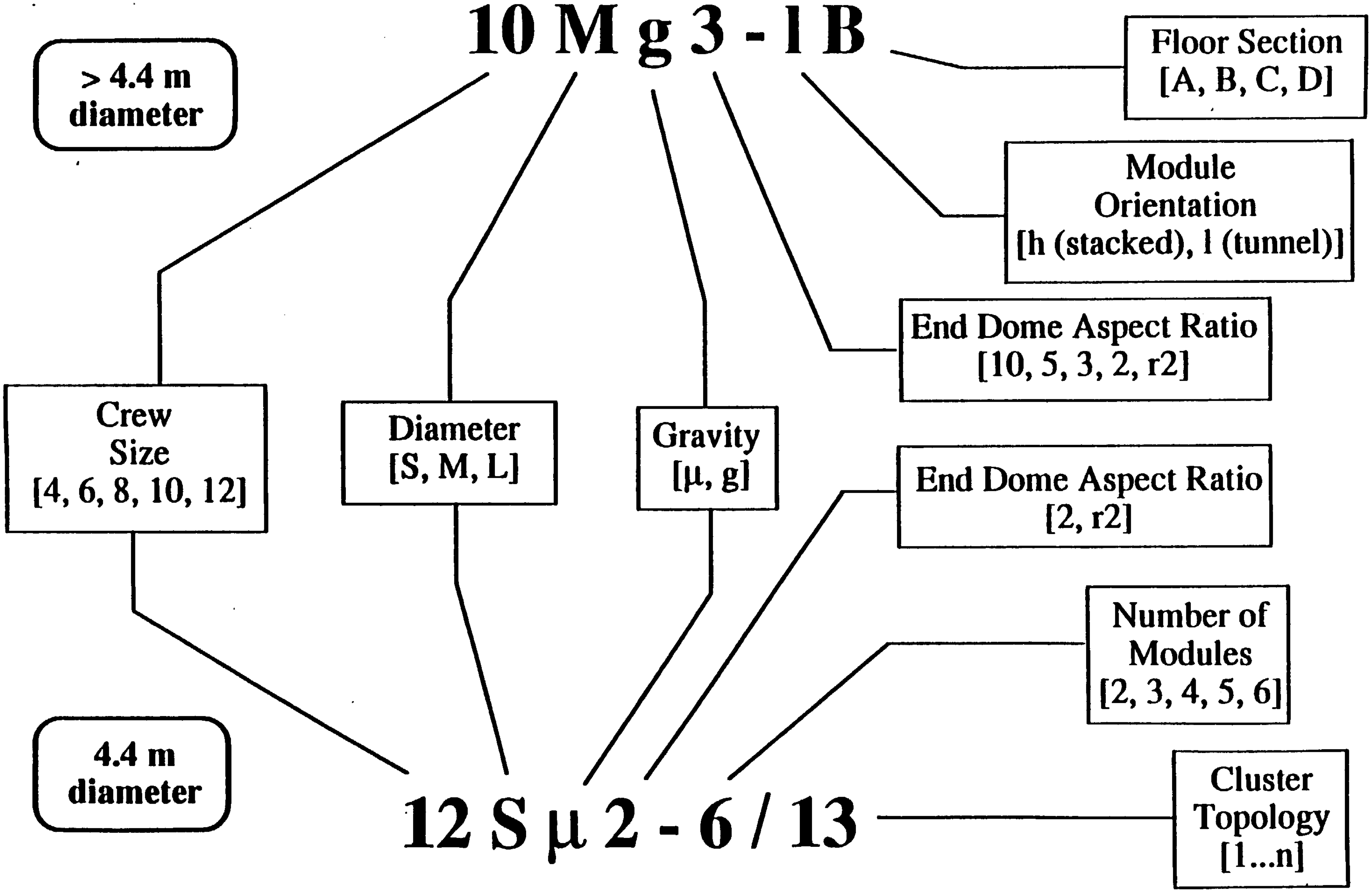


Solid lines indicate options for which masses and perception metrics were evaluated in the trade study

Habitat Concept Nomenclature

This chart explains the nomenclature used throughout the study to designate options.

Habitat Concept Nomenclature



MTV Hab Trade Discriminators

Shown here, with non-exhaustive examples for clarification, are four categories of discriminators identified as dominant in the study.

MTV Hab Trade Discriminators

Functionality discriminators

- **Access** (proximities; maintenance; emergency)
- **Sensory interference**
  - Sound (variety; isolation)
  - Odor (galley; WMS)

Cost discriminators

- **Commonality** (SSF; planetary surface base use)
- **Manufacturability** (M&P; tooling)
- **Processing** (handling; outfitting)
- **Weight** (specific mass; total mass)

Integration discriminators

- **MTV system implications** (aerobrake packaging; docking; assembly)
- **Growth potential** (evolution; larger transfer crews and vehicles)

Perceptual discriminators

- **Proportion**
  - Volume (specific; total)
  - Articulation (shape; modulation; familiarity; versatility)
- **Scale**
  - Views (max sightlines; interior/exterior; choices)
  - Options (pathways; variety)

MTV Hab Trade Non-discriminators

Listed here, with exceptions, are the major characteristics and components identified as non-discriminators for the study. Specifically, effects of varying these "wash out" across the trade alternatives to first order, and so are not accounted for in the study.

MTV Hab Trade Non-discriminators

To first order, the effect of varying these components cancels across the habitat trade study

- **Internal configuration**
  (except as constrained by boundary condition)
- **Science payload equipment**
  (except access)
- **ECLS equipment selection**
  (except configuration)
- **Materials selection & finishes**
  (except M&P technology advances for primary structure)
- **Furnishings**
- **Hatches & windows**
  (used specifically for EVA)

MTV Habitat Trade Study Assumptions

Listed here are the governing assumptions made in the study to facilitate consistency in comparing the various options.

The issue of baselining gravity for long-duration spaceflight is largely sidestepped by this trade study (an artificial gravity impact assessment will be performed by STCAEM in the next quarter). We expect that the complications (mass and configuration) introduced by presuming artificial gravity constitute the superset, since artificial-g vehicles would still have μg flight regimes as well. Furthermore, the effort to exploit commonality between flight and surface habitation systems is best served by module designs which implicitly incorporate the presence of gravity. For these two reasons, quite independently of the possible physiological necessity or operational benefits of artificial gravity spaceflight, we emphasized the gravity options in this trade study. (The principal results which would be different are those which assume that, for instance, only one of the two end domes are available as "overhead" space. In μg, a single structure may serve as the "floor" for both spaces it divides.)

MTV Habitat Trade Study
Assumptions

- **Specific volumes according to extrapolated historical data** (excluding examples of aero-entry vehicles)
- **Space Station *Freedom* habitability standards as the point of departure** (SSF will provide the most sophisticated human environment to have flown in space)
- **"Stacked" module arrangement usable only for module diameters greater than SSF** (previous trades have shown inefficient utilization of space for vertical arrangements of small diameter modules)
- **2.3 m ceiling height used as standard** (for comparative purposes in the trade)
- **0.5 m floor thickness used as standard** (applied to medium and large diameter concepts, accommodates sound insulation & stowage)
- **All major hatch and window penetrations occur in barrel section** (minimizes mass, manufacturing complexity of end domes)
- **Cluster topologies contain no separate connecting nodes** (minimizes mass, vehicle packaging, parts count, additional procurement)
- **Module clusters use all same-length modules** (limits topology options to manageable number)
- **Galley / storm shelter structure integrated with floor structures above and below** (structural advantage of deep-beam configuration to keep weight down)
- **Gravity-condition options emphasized** (higher outfitted weight; must also accommodate µg regimes; result facilitates commonality with surface applications)

Volume Guidelines

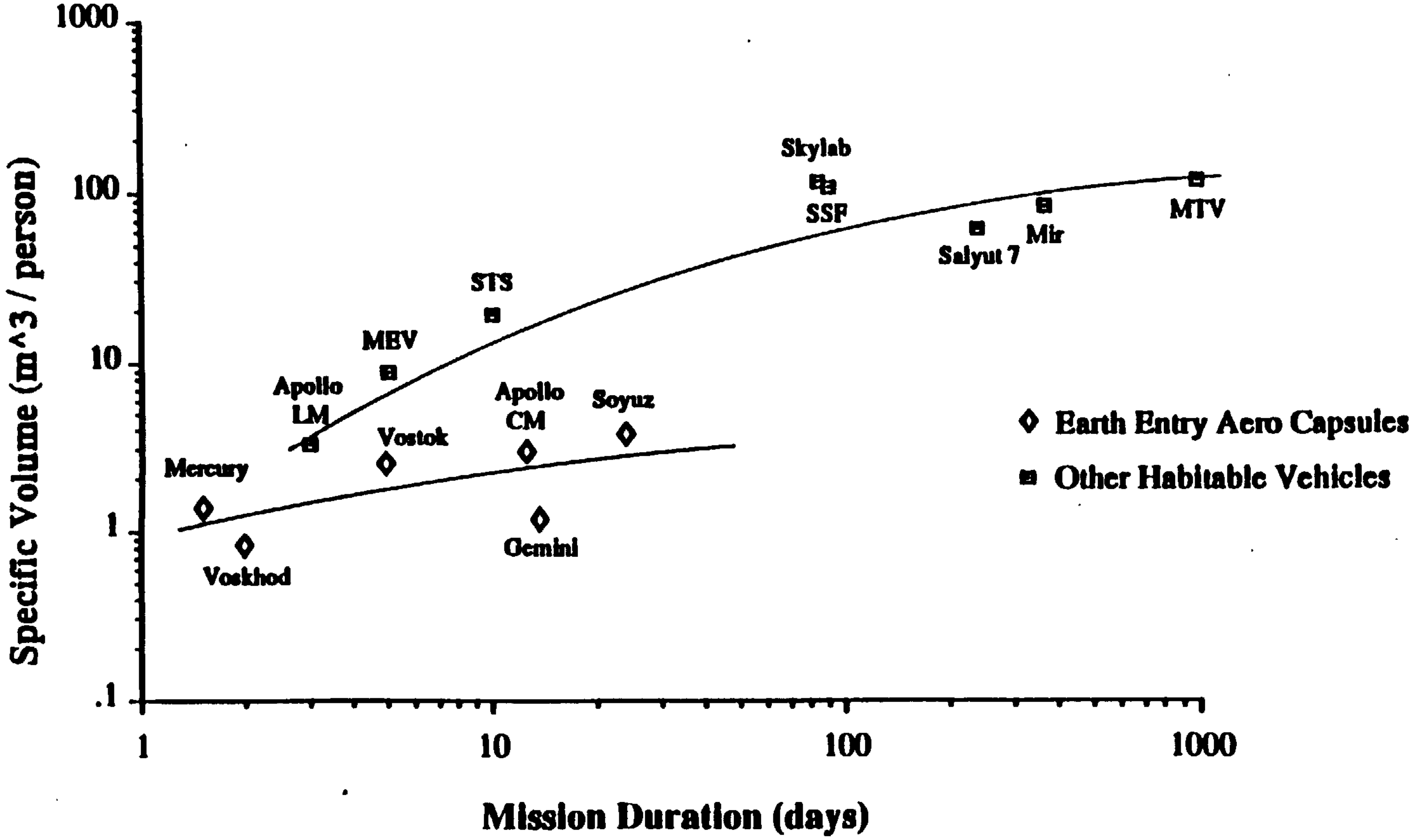
These specific volume curves were assembled from historical sources, and are based on total pressurizable volume (without actual equipment solid volume subtracted). The STCAEM reference specific volumes for the MEV and MTV have been included.

Two features are notable. First, vehicles for which aeroentry was the dominant cabin configuration constraint have typically crowded their crews more than strictly in-space, habitation and non-capsule systems. Second, *Skylab* was not as anomalous as is traditionally claimed; *Freedom* has as much specific volume when hab, lab, all nodes, JEM and ESA modules are included. The key difference is that SSF has much more internal equipment than did *Skylab*, so the free volume is comparatively much smaller.

The upper curve can be used to choose specific volume for new module concepts, based on historical trends.

MTV Hab Trade Volume Guidelines

Historical Spacecraft Total Pressurized Volume Data



Apollo News Reference, Grumman, 1989
Bluth, B.J. Soviet Space Stations as Analogs (NAGW-659), 1986
Boeing SSF WP-01 Data, 1990
Boeing STCAEM Study Data (NAS8-37857), 1990
NASA/JSC Man-Systems Division Data, 1989
Silvestri, G. et al. Quest for Space, 1985

Module Structure Concept Guidelines and Assumptions

Structural approaches are compared here, in several subsystem categories, for SSF and for this habitat trade study. Several subtle advancements have been introduced from the SSF approach, all in the service of reduced mass (which has extremely high leverage for in-space transportation systems). The heavy end cones are replaced by simple, unpenetrated ellipsoidal end domes (aspect ratio to be traded in this study); all module penetrations are in the less geometrically complex barrel section. The barrel sections are of monocoque, rather than waffle-grid, construction. The current precedent for this is in the (thicker-walled) bulkheads surrounding the dock port adapters of the SSF nodes. The advanced concept presumes overpressurization for structural stiffness on ETO launch, assuming ETO launch occurs using unmanned vehicles (e.g. Shuttle-C). This weight-reduction technique is less compatible with manned launchers like the NSTS.

Module Structure Concept Guidelines and Assumptions

Component	Space Station *Freedom*	Reference Trade Study Structure Concept	Comments
Material	2219 - T8 Al (as-welded)	Same	Long experience; ult. strength - 38 ksi
Cylinder	45° waffle grid	Monocoque	SSF uses man-rated, side-mounted launch configuration, not overpressurized for structural rigidity
Cylinder Cap	25° Conical, with flat pressure bulkhead	Ellipsoidal, with no penetrations	Docking loads, assemblage stiffness, axial penetrations drive SSF design
Support Structure	Longitudinal support beams for launch loads; cylinder support rings	cylinder support rings; Intermodule support structure (4.4 m dia.)	Overpressure provides structural integrity for reference unmanned ETO; intermod. support for uneven bending loads on hab system structure
Pressure Bulkhead	Monolithic, integrated into endcones	Al/Al honeycomb (10 & 7.6 m dia.); SSF derived (4.4 m dia.)	Monolithic bulkhead mass prohibitive for large diameter; honeycomb lighter, with acceptable volume penalty
Module Connection	Pressurized nodes	Parallel tunnels with pressure bulkheads between modules (4.4m dia)	Mass critical for reference; no req't for growth flexibility of individual system

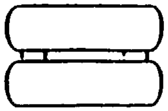
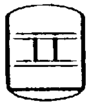
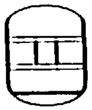
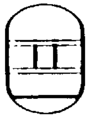
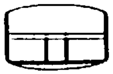
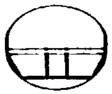
Note: Reference habitat structure design guidelines derived from MSFC-HDBK-505 Rev. A. Structural Strength Program Requirements.

Representative Geometry Options to Scale

The next five charts show, one for each of our crew sizes, comparisons to scale of the S cluster options and the M-1 and L-1 options.

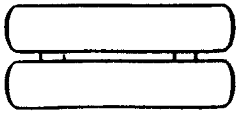
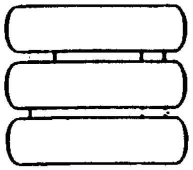
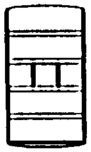
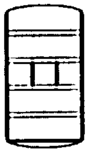
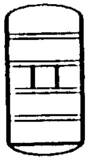
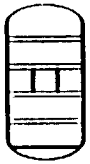
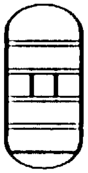
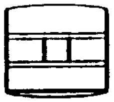
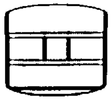
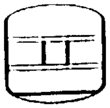
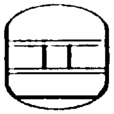
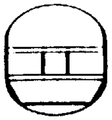
Representative Geometry Options to Scale
4 Crew

BOEING
STCAEM/bs/7Mar90

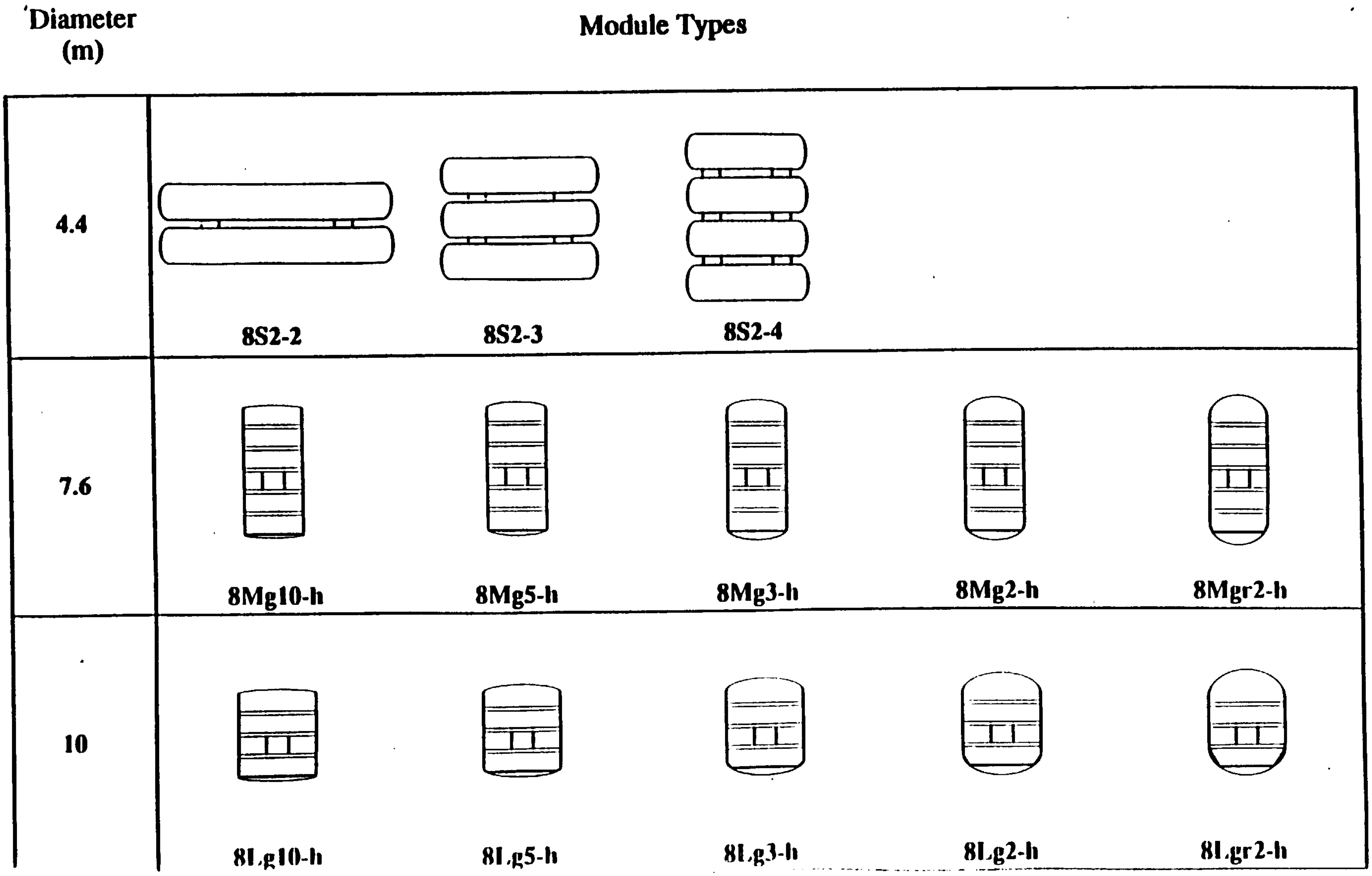
Diameter (m)	Module Types				
4.4	 4S2-2				
7.6	 4Mg10-h	 4Mg5-h	 4Mg3-h	 4Mg2-h	 4Mgr2-h
10	 4Lg10-h	 4Lg5-h	 4Lg3-h	 4Lg2-h	 4Lgr2-h

③

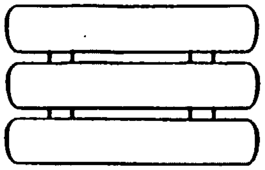
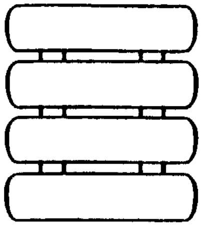
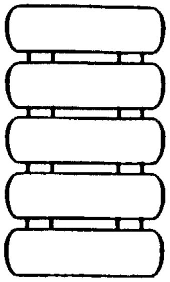
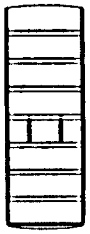
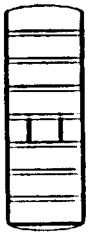
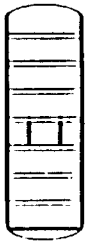
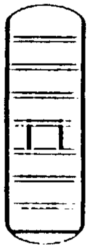
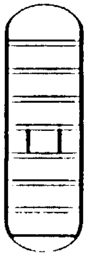
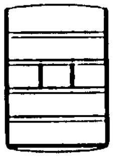
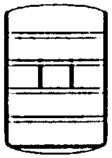
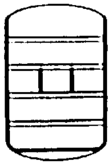
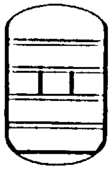
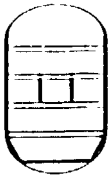
Representative Geometry Options to Scale
6 Crew

Diameter (m)	Module Types				
4.4	 6S2-2	 6S2-3			
7.6	 6Mg10-h	 6Mg5-h	 6Mg3-h	 6Mg2-h	 6Mgr2-h
10	 6Lg10-h	 6Lg5-h	 6Lg3-h	 6Lg2-h	 6Lgr2-h

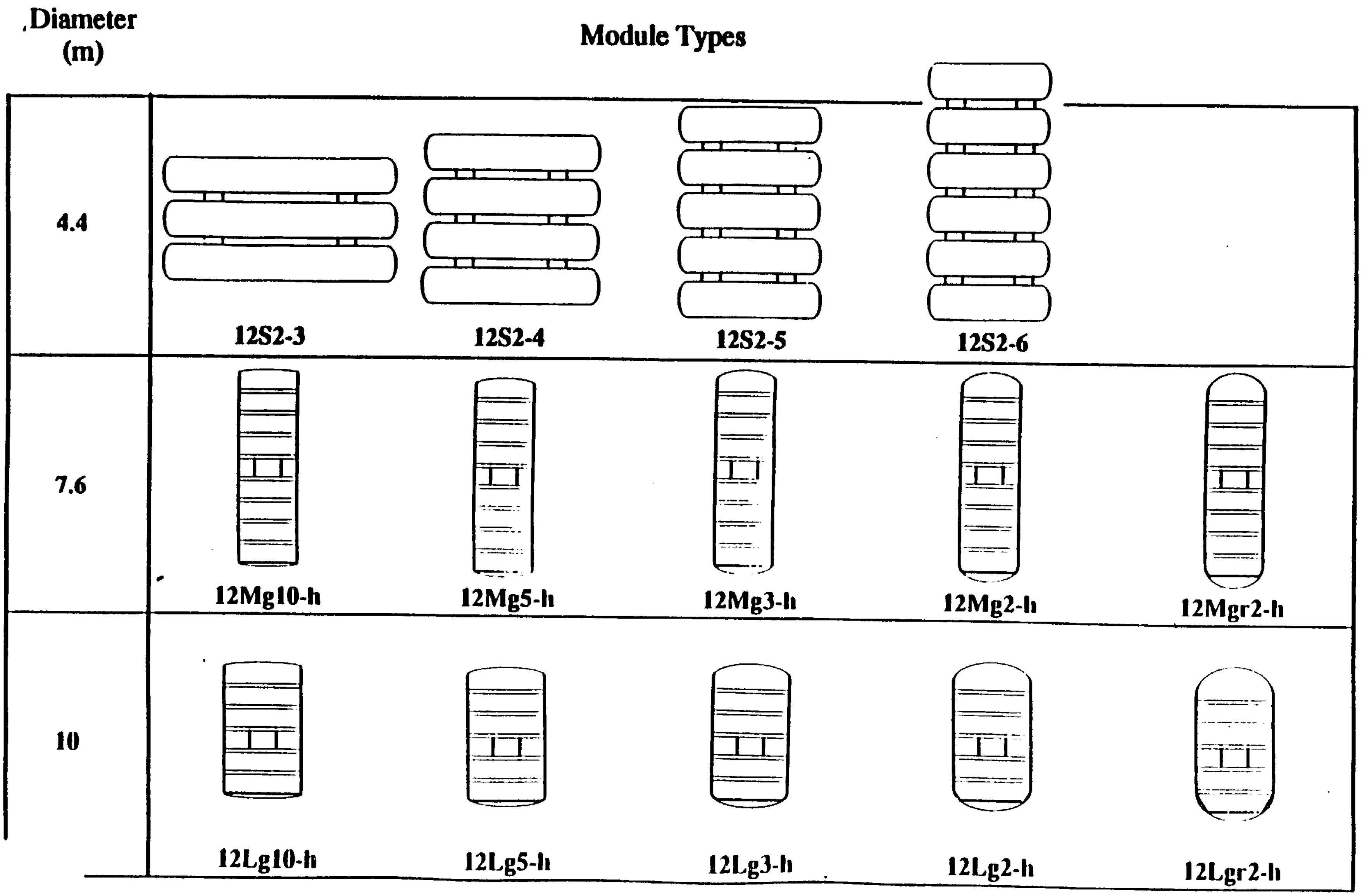
Representative Geometry Options to Scale
8 Crew



Representative Geometry Options to Scale
10 Crew

Diameter (m)	Module Types				
4.4	 10S2-3	 10S2-4	 10S2-5		
7.6	 10Mg10-h	 10Mg5-h	 10Mg3-h	 10Mg2-h	 10Mgr2-h
10	 10Lg10-h	 10Lg5-h	 10Lg3-h	 10Lg2-h	 10Lgr2-h

Representative Geometry Options to Scale
12 Crew



Topology Analysis Metrics

The first of the geometrical analyses is a topology study for cluster configurations of small modules. Seven metrics devised to compare the many cluster topologies are defined on the next two charts. The ranking priorities assigned them are not fixed, but simply record value trends used to neck down the options in this study. The metrics were applied both sequentially and simultaneously to reduce the starting option set to a manageable subset for the purposes of mass analysis later. The most critical metrics for the neckdown were F_i and F_{ss}. The others were used primarily to investigate quantitatively some characteristics of the topologies.

Topology Analysis Metrics

Nomenclature

$n \equiv$ number of modules
n_{ss} = number of modules available under safe-haven conditions
$t \equiv$ number of connecting tunnels
$i \equiv$ each starting module for a circulation pattern, (1...n)
$j \equiv$ each destination module, (1...n-1)

Guidelines

- Each metric ranges between 0 and 1
- For each metric, higher is better

Symbol	Metric name	Definition	Comments	Ranking priority
F_{ew}	ECLSS weight	0 if 4 strings 1 if 3 strings	– 2 fault tolerant requirement favors racetrack topologies – Extra string weight relatively small	1
F_i	Aerobrake integration	Ranked based on configuration experience: 0.1 - 0.9	– Lower numbers are hard to package behind L/D = 0.5 aerobrakes while keeping brake size small – Changes with aerobrake L/D	2

(continued)

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Topology Analysis Metrics (2)

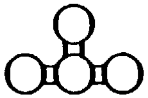
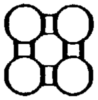
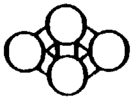
Symbol	Metric name	Definition	Comments	Priority
F_{ss}	Safe-haven split	$\frac{n_{ss}}{n}$	– Worst-case safe-haven condition for best possible ECLSS string distribution among n – Measures how much of the habitable volume is left (for remainder of trip if fix is impossible)	3
F_n	Normalized spatial units	$\frac{n}{6}$	– Normalized to n_{max} –Higher numbers mean greater potential for differently optimized environments	7
F_{pc}	Parts count	$\frac{1}{n+t}$	– higher numbers mean fewer pieces to integrate on-orbit, fewer mechanisms to maintain, less cabin air leakage	6
F_{pr}	Proximity convenience	$\left[\sum_{i=1}^{n}\sum_{j=1}^{n-1} t\right]^{-1}$	– Higher numbers mean fewer tunnels stand between origin and destination, summed over the topology – High numbers mean more convenience – Lower numbers mean potentially greater perception of inhabited domain	5
F_c	Circulation efficiency	$\frac{n}{t}$	– High numbers mean fewer connecting tunnels – Lower numbers may indicate "excessive" tunnels	4

Module Cluster Topology Analysis

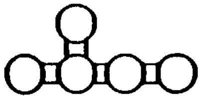
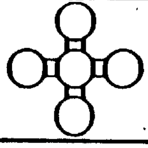
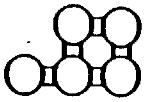
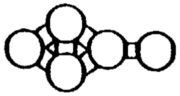
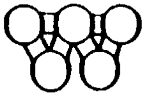
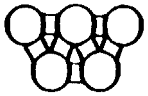
The next five charts diagram the topologies considered for clusters of two, three, four, five and six modules. The topology metrics calculated are tabulated, and the topologies selected for further consideration (as representative of the best candidates from each group) are indicated.

The subsequent six charts graph the six most revealing metrics, to compare all the topology options.

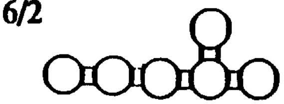
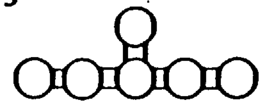
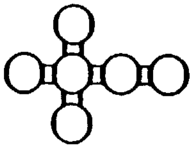
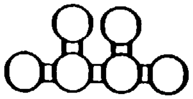
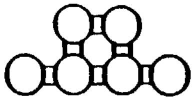
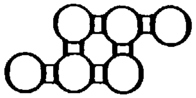
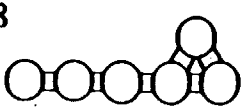
Module Cluster Topology Analysis (1)

Number of Modules	Topology	Number of Tunnels	Number of ECLSS Strings	F_{ew}	F_i	F_{ss}	F_n	F_{pc}	F_{pr}	F_c
2	2/1 	2	4	0	.9	0.5	.33	.25	0.500	1
3	3/1 	4	4	0	.7	0.33	.5	0.143	0.125	.75
	3/2 	6	4 3	0 1	.9	0.67	.5	0.111	0.167	.5
4	4/1 	6	4	0	.5	0.5	.67	0.100	.050	.67
	4/2 	6	4	0	.7	0.25	.67	0.100	.056	.67
	4/3 	8	4 3	0 1	.65	0.75	.67	0.083	.063 0.063	.5
	4/4 	8	4 3	0 1	.6	0.5	.67	.083	.063	.5
	4/5 	10	4 3	0 1	.8	0.75	.67	.071	.071	.4

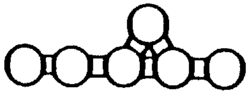
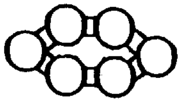
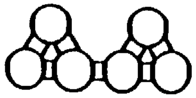
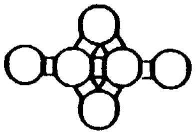
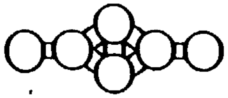
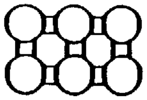
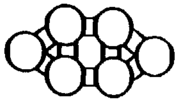
Module Cluster Topology Analysis (2)

Number of Modules	Topology	Number of Tunnels	Number of ECLSS Strings	F_{cw}	F_i	F_{ss}	F_n	F_{pc}	F_{pr}	F_c
5 ,	5/1 	8	4	0	.3	0.4	.83	.077	.025	.63
	5/2 	8	4	0	.5	0.4	.83	.077	.028	.63
	5/3 	8	4	0	.2	0.2	.83	.077	.031	.63
	5/4 	10	4 3	0 1	.5	0.6	.83	.067	.031	.5
	5/5 	10	4 3	0 1	.5	0.6	.83	.067	.031	.5
	5/6 	12	4 3	0 1	.5	0.6	.83	.059	0.33	.42
	5/7 	12	4 3	0 1	.7	0.4	.83	.059	0.36	.42
	5/8 	14	4 3	0 1	.7	0.6	.83	.053	0.38	.36

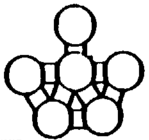
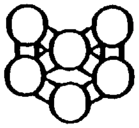
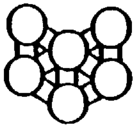
Module Cluster Topology Analysis (3)

Number of Modules	Topology	Number of Tunnels	Number of ECLSS Strings	F_{ew}	F_i	F_{ss}	F_n	F_{pc}	F_{pr}	F_c
6	6/1 	10	3	0	.1	0.5	1	.063	0.014	.6
	6/2 	10	4	0	.2	0.5	1	.063	0.016	.6
	6/3 	10	4	0	.3	0.33	1	.063	0.017	.6
	6/4 	10	4	0	.2	0.33	1	.063	0.018	.6
	6/5 	10	4	0	.5	0.5	1	.063	0.017	.6
	6/6 	12	4 3	0 1	.5	0.67	1	.056	0.019	.5
	6/7 	12	4 3	0 1	.3	0.67	1	.056	0.017	.5
	6/8 	12	4 3	0 1	.25	0.33	1	.056	0.016	.5

Module Cluster Topology Analysis (4)

Number of Modules	Topology	Number of Tunnels	Number of ECLSS Strings	F_{ew}	F_i	F_{ss}	F_n	F_{pc}	F_{pr}	F_c
6	6/9 	12	4 3	0 1	.3	0.5	1	.056	0.017	.5
	6/10 	12	4	0	.7	0.83	1	.056	0.019	.5
	6/11 	14	4 3	0 1	.5	0.5 0.33	1	.05	0.019	.43
	6/12 	14	4 3	0 1	.45	0.83 0.67	1	.05	0.021	.43
	6/13 	14	4 3	0 1	.45	0.83 0.67	1	.05	0.019	.43
	6/14 	14	4 3	0 1	.5	0.83	1	.05	.020	.43
	6/15 	16	4 3	0 1	.55	0.83	1	.045	.022	.38

Module Cluster Topology Analysis (5)

Number of Modules	Topology	Number of Tunnels	Number of ECLSS Strings	F_{ew}	F_i	F_{ss}	F_n	F_{pc}	F_{pr}	F_c
6	6/16 	16	4 3	0 1	.7	0.83	1	.045	.023	.38
	6/17 	16	4 3	0 1	.5	0.83	1	.045	.023	.38
	6/18 	18	4 3	0 1	.5	0.83	1	.042	.024	.33 ●

- designates topologies included for further analysis

Topology Metric Analysis
Aerobrake Integration Factor

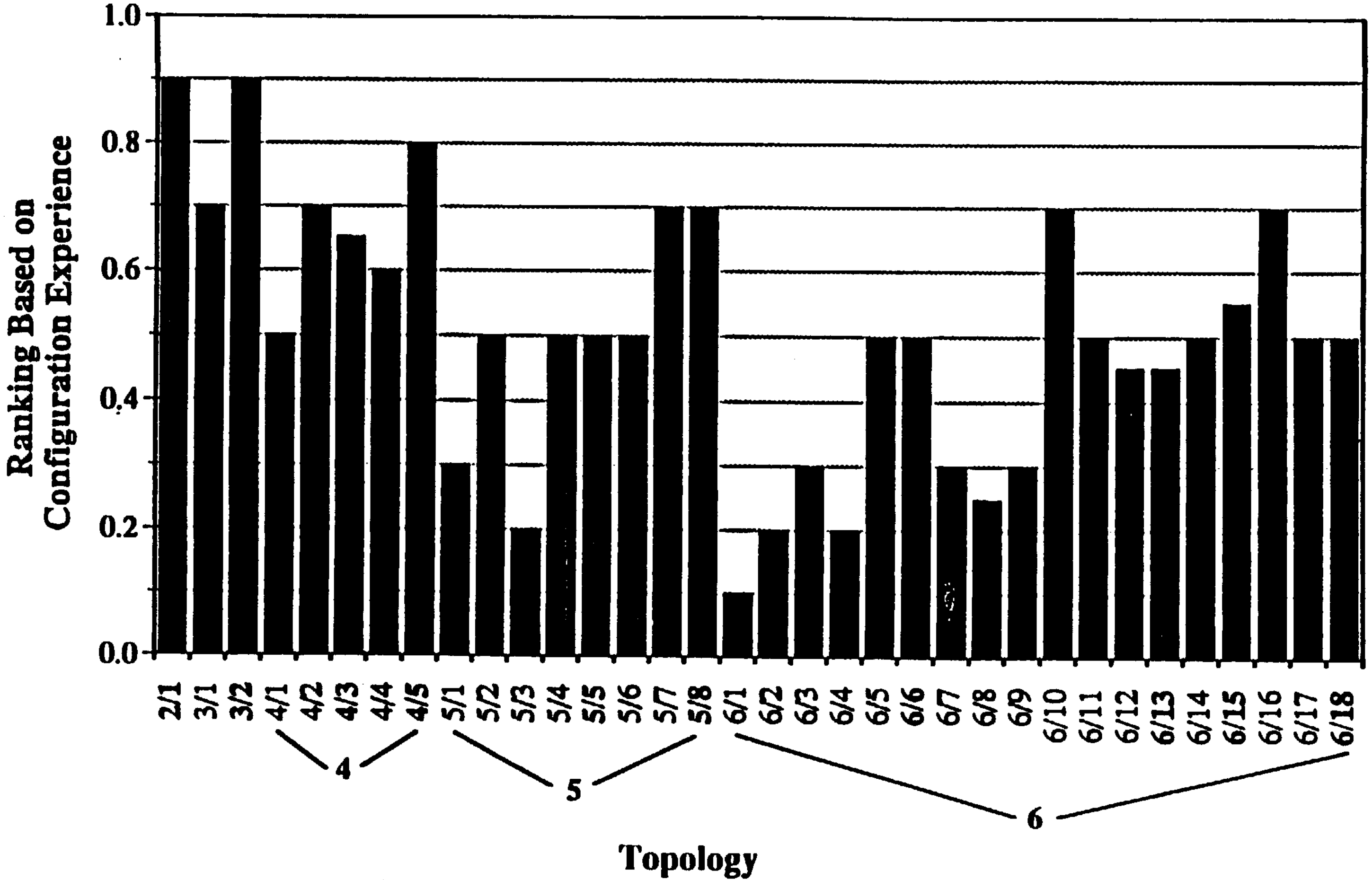
This is the most "subjective" of the metrics; however, its assessment was performed by configuration engineers who have developed contemporary aerobraked vehicle concepts over the last several years and are experienced with the configuration complications introduced by packaging behind aerobrakes. The goal here was to elucidate those topologies which, considered in cross section only (independent of module length), would facilitate configuring the smallest (and therefore lightest) aerobrake possible within each group of module-number.

Star and string configurations are poor; dense clusters, and particularly those which tend to accommodate the curvature of an aerobrake shape and/or the conical aftbody wake-protection zone, trade much better. Selecting a cutoff (0.45 for example) allows rejecting the least favorable topologies.

Topology Metric Analysis

Aerobrake Integration Factor

$F_I = 0.1 - 0.9$



Topologies vary widely in their ability to be integrated easily behind aerobrakes.
The clusters are assessed regardless of module length.
Comparisons are most useful among clusters with the same number of modules.

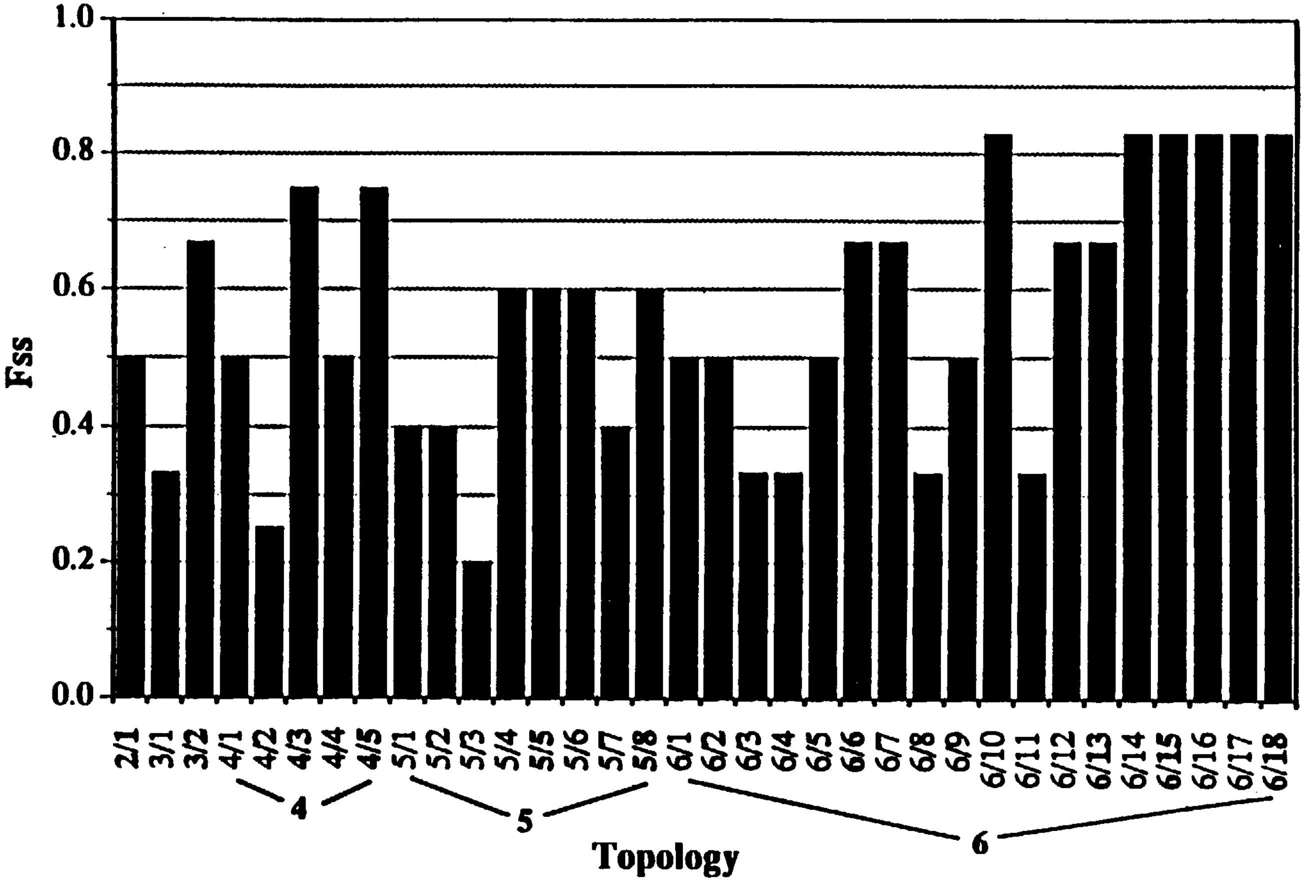
Topology Metric Analysis
Safe-Haven Split Factor

Assuming irrecoverable damage to a single module in flight, assuming individual modules cannot be moved within the topology after departure, and assuming the most favorable distribution of redundant ECLS equipment among the modules for each topology, this metric assesses how much of the original volume would be IVA-available to the crew for the remainder of the trip. Losing half of the total appears a severe scenario; such a criterion allows rejecting several topologies.

Topology Metric Analysis

Safe-Haven Split Factor

$$F_{ss} = \frac{n_{ss}}{n}$$



Requiring the worst-case safe-haven scenario to leave at least half the original habitable volume ($F_{ss} > 0.5$) eliminates many possible topologies

Topology Metric Analysis
Spatial Units Factor

This metric merely compares the total number of available modules to the maximum studied, six. It serves as a quantitative reminder that more separate modules provides more intrinsic opportunity for optimizing spatial units according to distinct functions (sleep, recreation, laboratory, etc.).

Topology Metric Analysis

Spatial Units Factor

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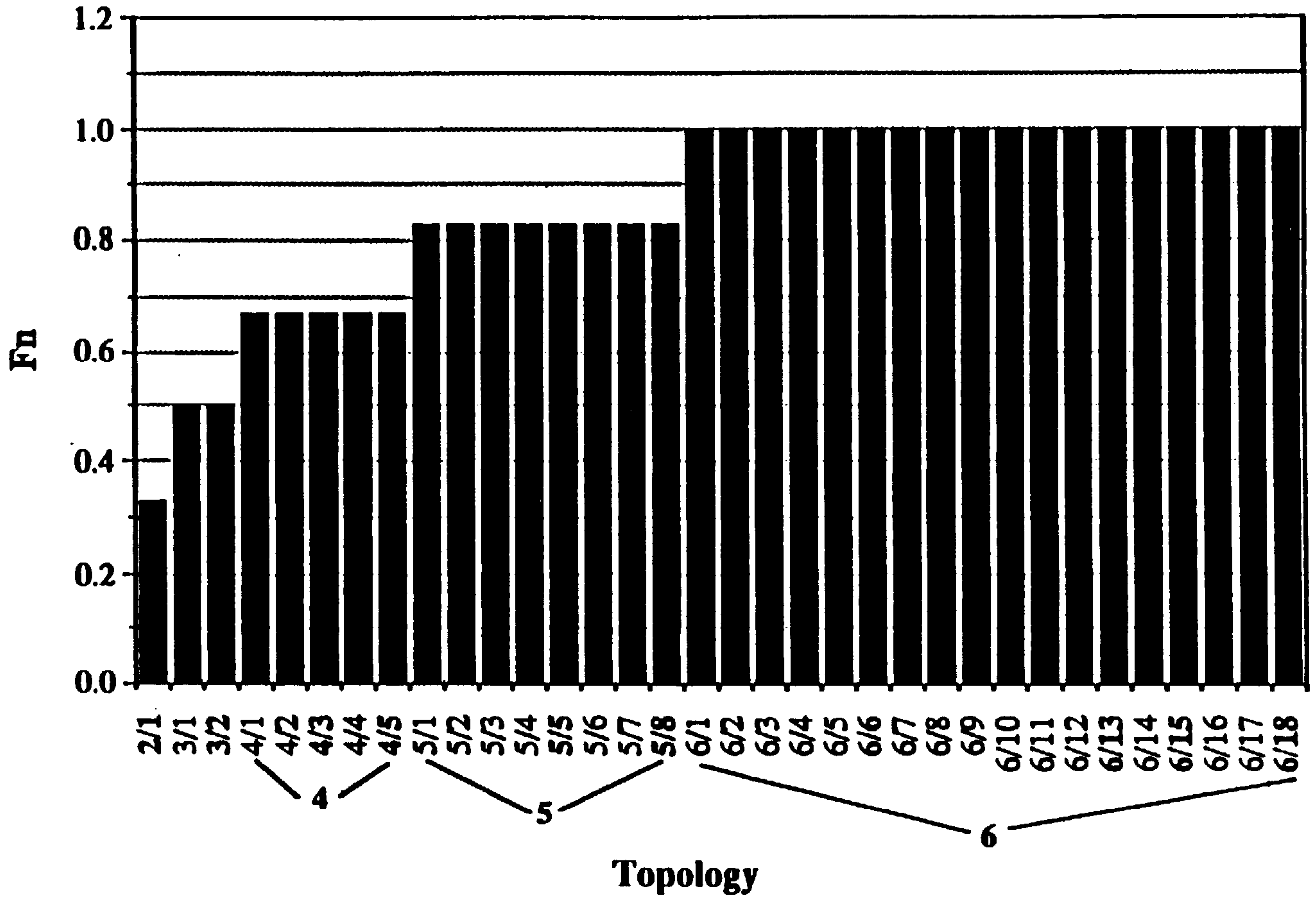
$$F_n = \frac{n}{6}$$



Opportunities for spatial variety may be enhanced in clusters consisting of more modules

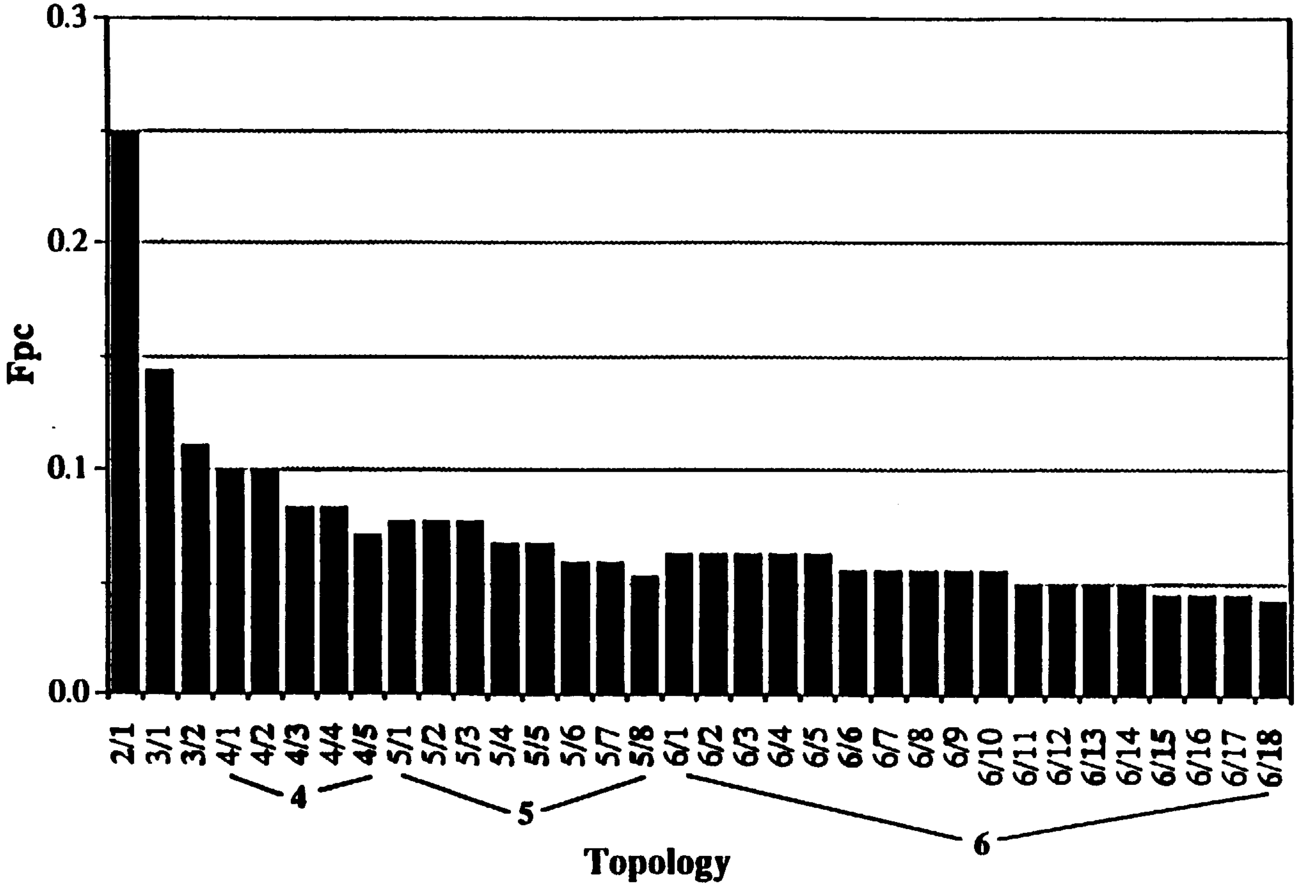
Topology Metric Analysis
Parts Count Factor

The greater the number of modules, however, the greater the parts count (tunnels, hatches, modules, interconnection structure, etc.), and the greater the opportunity for failures and leakage. The parts count metric drops dramatically once module-number exceeds 2 or 3. Subtle differences exist among topologies within each module-number group.

Topology Metric Analysis

Parts Count Factor

$$F_{pc} = \frac{1}{n + t}$$



The parts count increases quickly as topologies get more complex. However, the differential increase becomes less significant with larger numbers of modules

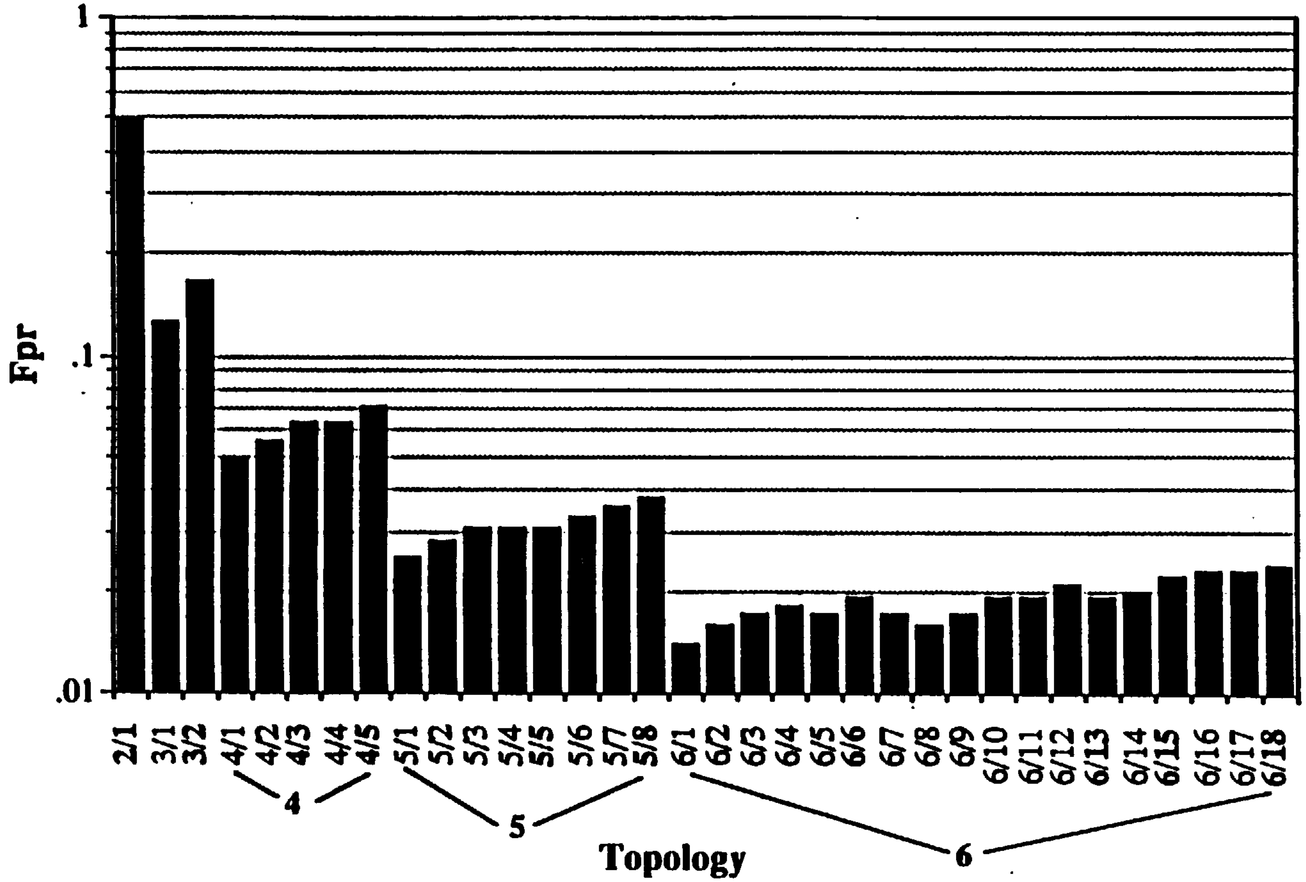
Topology Metric Analysis
Proximity Convenience Factor

This metric assesses how many non-destination modules one must go through to get to the destination module, summed in the best case over all possible combinations of origin and destination modules for all topologies. High numbers mean more convenient circulation, but low numbers may contribute to the perception of a greater habitable domain.

Topology Metric Analysis

Proximity Convenience Factor

$$F_{pr} = \sum_{i=1}^{n} \sum_{j=1}^{n-1} t$$



For high-n clusters, circulation patterns are characterized by having to thread several modules. This interferes with convenience, but contributes to perceptions of a large domain.

Topology Metric Analysis
Circulation Efficiency Factor

This metric merely ratios the number of modules to the number of tunnels used to connect them, for each topology. It is a measure of how much hardware is devoted to interconnection in the cluster approach to habitat design.

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Topology Metric Analysis

Circulation Efficiency Factor

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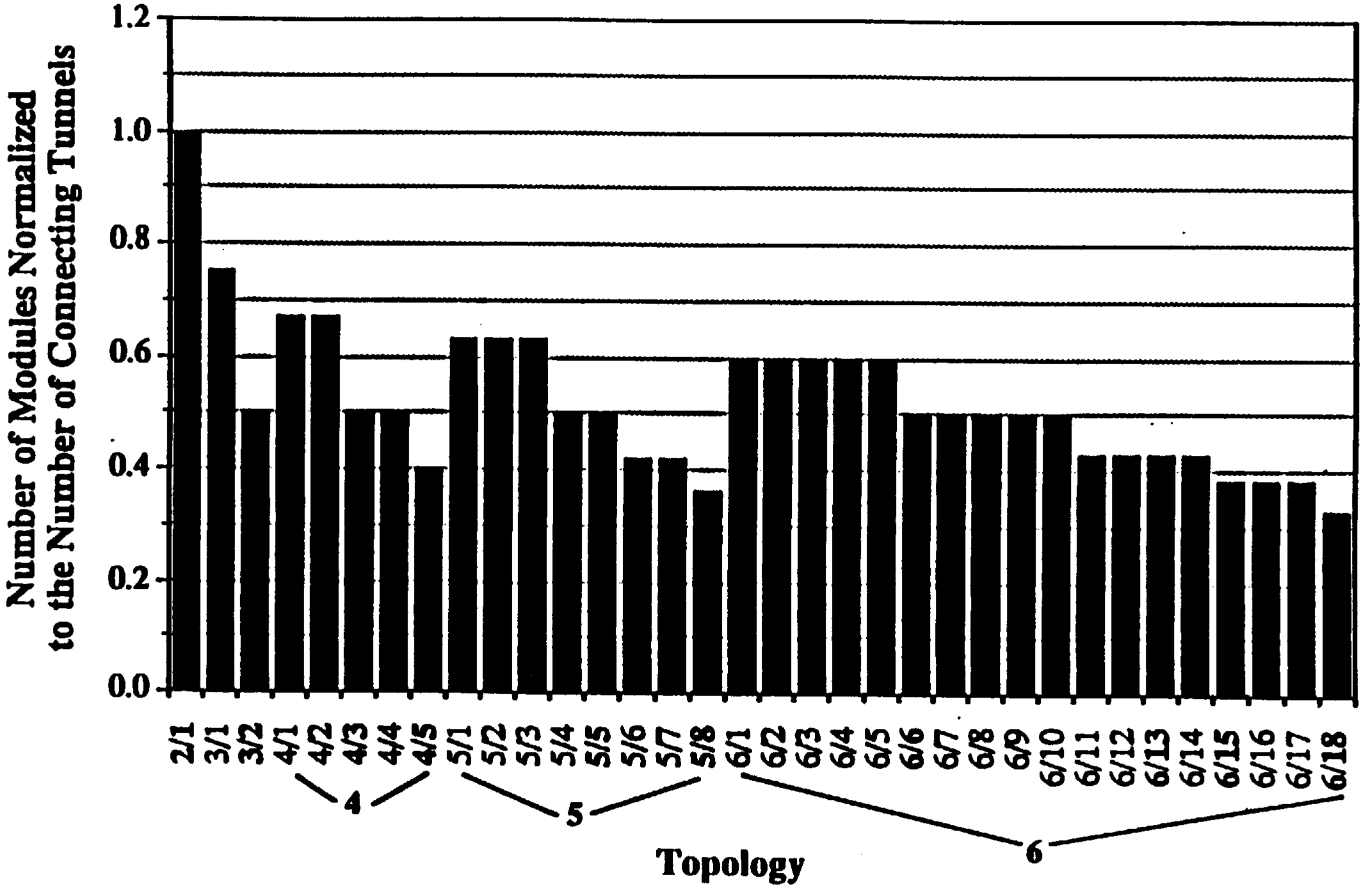
$$F_c = \frac{n}{t}$$



Options preferable for other reasons rarely tend to have the fewest connecting tunnels; however, suitable candidates can be selected from all cluster groups

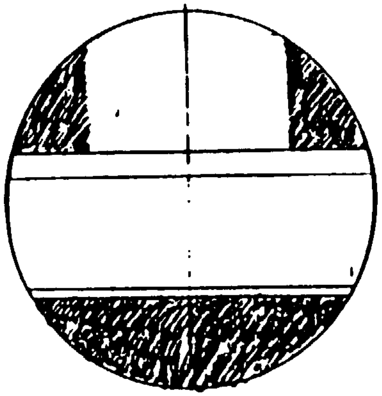
MTV Hab "Tunnel" Arrangements
Cross Section Properties

The next two charts show representative options for floor arrangements in the tunnel-oriented medium and large diameter modules. Off-nominal volume is defined as "uninhabitable", or that which has a ceiling height lower than the 2.3 m nominal assumed by this study; it is a prime location for equipment location. Lettered from left to right on the charts, options "B" and "C" provide the most nominal floor area, the most accessible underfloor volume (useful for ECLSS and stowage), and the advantages of the vaulted ceiling (spaciousness perception) without excessive wasted space. For the quantitative analysis purposes of this trade study, floor option B is selected.

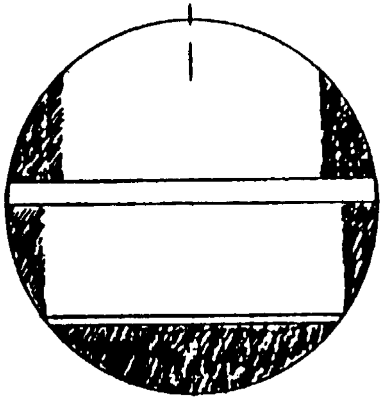
The third chart shows an analogous analysis for the small diameter module, for comparative purposes. For gravity conditions in which spaciousness is important for psychological reasons (long-duration flights), option "C" is most reasonable and is used for quantitative analyses throughout this trade study.

MTV Hab "Tunnel" Arrangements

7.6 m-diameter Cross Section Properties

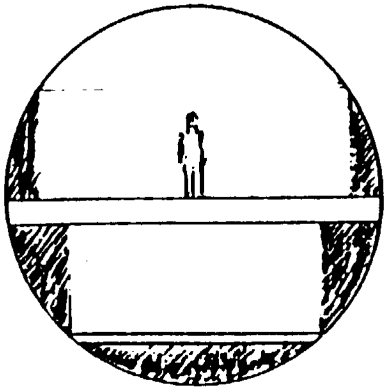


loor area: 10.8 m^2/m
ccessible off-nominal volume: 25 %
laximum ceiling height: 2.75 m
ut-of-reach overhead volume: 2 %

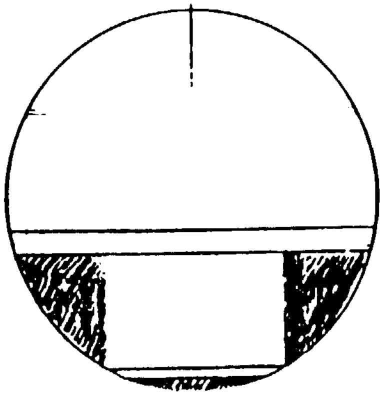


Floor area: 11.4 m^2/m
Accessible off-nominal volume: 22 %
Maximum ceiling height: 3.2 m

Floor area: 11.4 m^2/m
Accessible off-nominal volume: 17 %
Maximum ceiling height: 3.8 m
Out-of-reach overhead volume: 12 %



Floor area: 11.3 m^2/m
Accessible off-nominal volume: 12
Maximum ceiling height: 4.4 m
Out-of-reach overhead volume: 17



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MTV Hab "Tunnel" Arrangements

10 m-diameter Cross Section Properties

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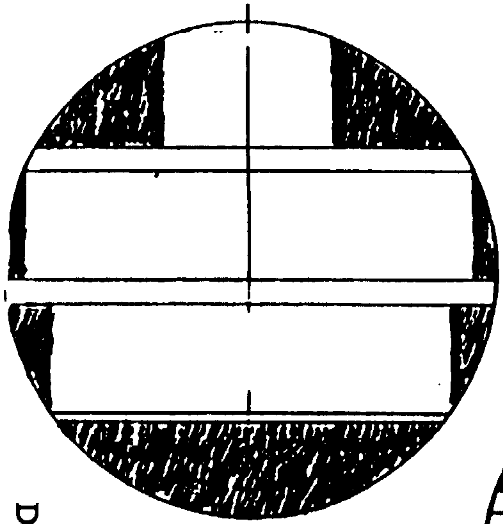
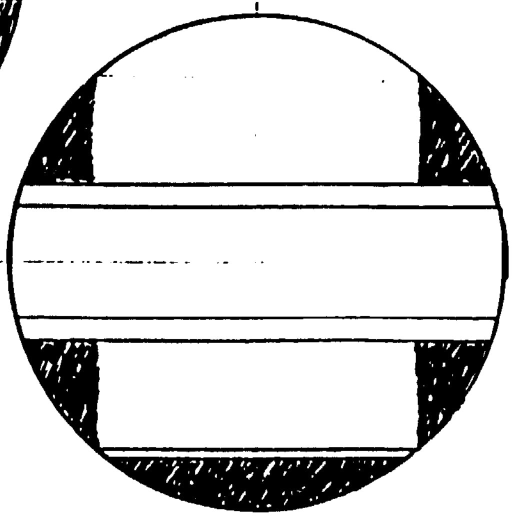
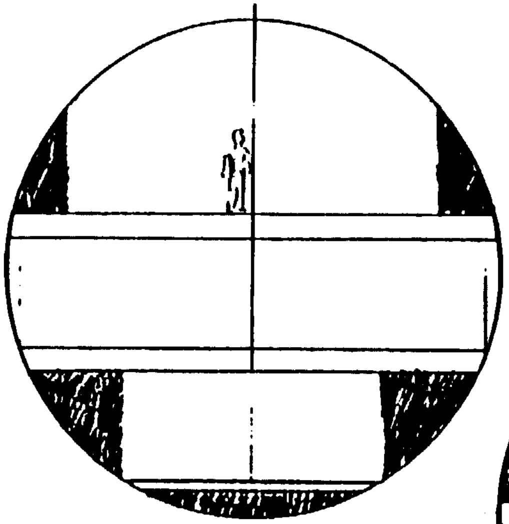


loor area: 21.7 m^2/m
ccessible off-nominal volume: 25 %
aximum ceiling height: 2.5 m
ut-of-reach overhead volume: 0 %

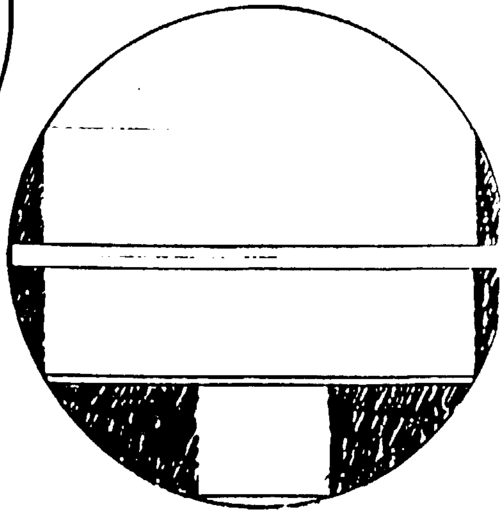


Floor area: 22.8 m^2/m
Accessible off-nominal volume: 15 %
Maximum ceiling height: 3.4 m
Out-of-reach overhead volume: 5 %

Floor area: 22.5 m^2/m
Accessible off-nominal volume: 12 %
Maximum ceiling height: 3.9 m
Out-of-reach overhead volume 9 %

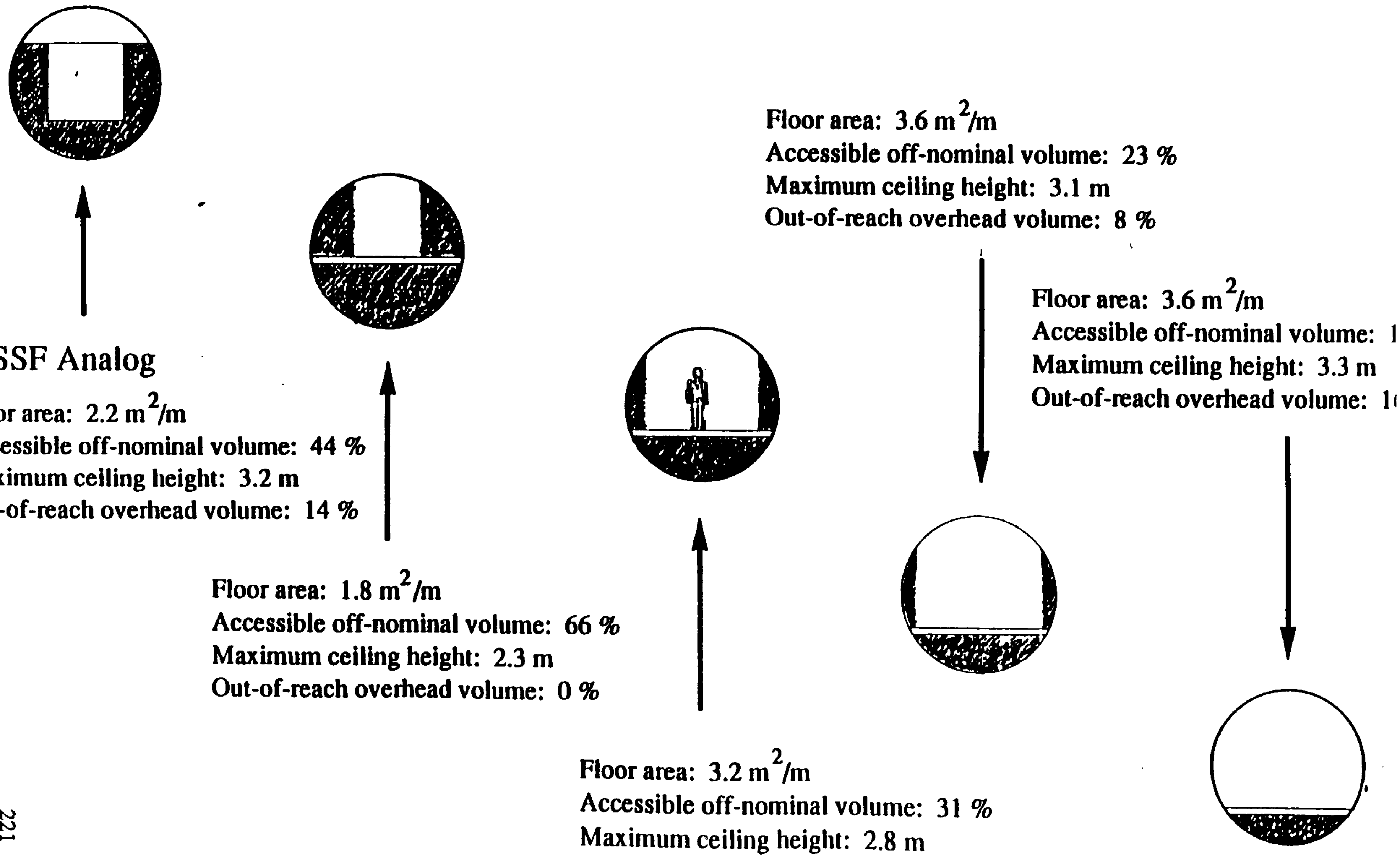


Floor area: 21.3 m^2/m
Accessible off-nominal volume: 11
Maximum ceiling height: 4.75 m
Out-of-reach overhead volume: 19



MTV Hab "Tunnel" Arrangements

4.4 m-diameter Cross Section Properties



SSF Analog

loor area: 2.2 m^2/m
ccessible off-nominal volume: 44 %
laximum ceiling height: 3.2 m
ut-of-reach overhead volume: 14 %

Floor area: 1.8 m^2/m
Accessible off-nominal volume: 66 %
Maximum ceiling height: 2.3 m
Out-of-reach overhead volume: 0 %

Floor area: 3.2 m^2/m
Accessible off-nominal volume: 31 %
Maximum ceiling height: 2.8 m

Floor area: 3.6 m^2/m
Accessible off-nominal volume: 23 %
Maximum ceiling height: 3.1 m
Out-of-reach overhead volume: 8 %

Floor area: 3.6 m^2/m
Accessible off-nominal volume: 1
Maximum ceiling height: 3.3 m
Out-of-reach overhead volume: 1

C-3

Geometry Analysis Metrics

The next three charts define the metrics devised to compare the small module cluster options which survived their own neckdown, with the unitary medium and large diameter options. These have been developed to be configuration-independent; that is, they compare module geometries regardless of internal outfitting and detailing considerations. Each has been assigned a shorthand appellation, shown in quotation marks, which captures the primary way in which the metric has been taken for the purposes of this study.

Geometry Analysis Metrics

Nomenclature

A_n ≡ nominal floor area (having 2.3m ceiling height)
V ≡ volume
A_s ≡ sectional area of largest spatial unit
A_{soe} ≡ off-ergonomic sectional area (e.g. above 2.3m ceiling height)
l_{max} ≡ maximum simple path length within habitat
x ≡ plan dimension within one spatial unit
h ≡ maximum ceiling height
U ≡ number of spatial units
F ≡ number of floors
P_u ≡ spatial unit perimeter consumed by doorways to other spaces, and not available as wall space
O_d ≡ distinct pathways available between origin and destination spatial units
Spatial Unit ≡ 1 floor in multi-floor module, or
1 module in multi-module cluster

Guidelines

- All perception metrics are assessed independently of internal configuration details
- All metrics depend only on the geometry and orientation of the modules, and the arrangement of floors and spatial units within them

Symbol	Metric	Formula	Comments
F_{fa}	specific nominal floor area, "Inhabitability Factor"	$\frac{A_n}{\Sigma V}$	– Higher numbers mean more habitable floor area for gravity conditions. – Equipment can be located in off-nominal spaces or can take up nominal floor area (configuration-dependent)

(continued)

Geometry Analysis Metrics (2)

Symbol	Metric	Formula	Comments
F_s	Specific off-ergonomic section, "Vault Factor"	$\frac{A_{soe}}{\Sigma A_s}$	– Higher numbers indicate habitable spaces with more sectional area beyond the 2.3 m-high ergonomic envelope – Measures spaciousness in section
F_l	Specific end-to-end travel distance, "Domain Factor"	$\frac{l_{max}}{\Sigma V}$	– Higher numbers indicate long worst-case intra-habitat travel times – Lower numbers indicate habitats perceived as having limited territory
F_{rp}	Plan aspect ratio, "Hallway Factor"	$\frac{X_{max}}{X_{min}}$	– Taken in longest spatial unit – Higher numbers indicate more hallway-like spatial units
F_{rs}	Sectional aspect ratio, "Spaciousness Factor"	$\frac{X_{max} X_{min}}{h}$	– Taken in longest spatial unit – High numbers indicate perceptions of low ceiling height – Low numbers may indicate perceptions of being in a "pit"
F_u	Specific number of spatial units, "Variety Factor"	$\frac{U}{\Sigma V}$	– Higher numbers indicate more optimistic opportunitites to optimize different spaces

(continued)

Geometry Analysis Metrics (3)

Symbol	Metric	Formula	Comments
F_f	Specific number of floors, "Elevator Factor"	$\frac{F}{\sum A_n}$	- For small crew sizes, higher numbers indicate more "upstairs-downstairs" variety. - For large crew sizes in gravity configurations, higher numbers indicate functional inconvenience
F_p	Specific useful perimeter, "Perimeter Factor"	$\frac{\sum P_u}{\sum A_n}$	- Higher numbers indicate more intrinsically available wall space for equipment positioning
F_o	Options to destination "Pathway Factor"	$\sum_{U_i} \sum_{U_j} O_d$ $i,j = 1,...n$ $i \neq j$	- Measures distinct pathways available within habitat system - Higher numbers indicate many options (can get extremely high) - Low numbers indicate monotony of movement patterns within the environment, deficiency of variety
F_v	Volume variety range, "Scale Factor"	$\frac{V_{max}}{V_{min}}$	– Measures the range of perceptual scales available to crew - Higher numbers indicate a wider range

Habitation Module Geometry Metrics

The following two charts tabulate the 10 geometry metrics for the options designated, as calculated according to the formulas just defined on the previous charts. An end-dome ellipsoid ratio of 3 was used for the medium and large diameter option calculations; 2 was used for the small-diameter options.

Habitation Module Geometry Metrics

Module Type \ Perception Metrics	F_{fa}	F_s	F_l	F_{rp}	F_{rs}	F_u	F_f	F_p	F_o	F_v
4S-2/1	0.21	0.16	0.05	3.58	14.51	0.005	0.011	0.78	4	27.25
4M3-h	0.35	0.14	0.02	1.0	20.63	0.008	0.022	0.53	6	18.68
4M3-IB	0.20	0.25	0.02	1.24	11.69	0.005	0.026	0.74	4	13.73
4L3-h	0.34	0.23	0.02	1.0	31.25	0.004	0.013	0.40	2	37.57
4L3-IB	0.18	0.27	0.02	1.83	6.99	0.007	0.037	0.87	16	9.00
6S-2/1	0.22	0.16	0.04	5.56	22.50	0.003	0.007	0.70	4	40.88
6S-3/2	0.25	0.16	0.04	4.22	17.10	0.005	0.012	0.75	24	27.25
6M3-h	0.36	0.14	0.02	1.0	20.63	0.008	0.022	0.53	6	18.68
6M3-IB	0.22	0.25	0.02	2.22	20.97	0.003	0.014	0.57	4	24.63
6L3-h	0.35	0.23	0.02	1.0	31.25	0.005	0.013	0.40	6	37.57
6L3-IB	0.21	0.27	0.02	1.08	12.23	0.005	0.021	0.61	16	15.75
8S-2/1	0.22	0.16	0.04	7.56	30.60	0.002	0.005	0.66	4	54.50
8S-3/2	0.22	0.16	0.03	4.92	19.91	0.003	0.011	0.72	24	36.33
8S-4/3	0.21	0.16	0.03	3.58	14.51	0.005	0.011	0.78	112	27.25
8S-4/5	0.21	0.16	0.02	3.58	14.51	0.005	0.014	0.78	148	27.25
8M3-h	0.36	0.14	0.02	1.0	20.63	0.008	0.022	0.53	30	18.68
8M3-IB	0.23	0.25	0.02	2.71	25.61	0.003	0.012	0.53	4	30.08
8L3-h	0.36	0.23	0.01	1.0	31.25	0.005	0.013	0.40	12	37.57
8L3-IB	0.23	0.27	0.02	1.36	17.47	0.003	0.015	0.51	16	22.50
10S-3/2	0.22	0.16	0.03	6.22	25.20	0.003	0.008	0.68	24	45.42

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Habitation Module Geometry Metrics

Module Type \ Perception Metrics	F_{fa}	F_s	F_l	F_{rp}	F_{rs}	F_u	F_f	F_p	F_o	F_v
10S-4/3	0.22	0.16	0.03	4.58	18.56	0.004	0.008	0.73	112	34.06
10S-4/5	0.22	0.16	0.02	4.58	18.56	0.004	0.012	0.73	148	34.06
10S-5/5	0.21	0.16	0.03	3.58	14.51	0.005	0.009	0.78	224	27.25
10S-5/8	0.21	0.16	0.03	3.58	14.51	0.005	0.009	0.78	660	27.25
10M3-h	0.37	0.14	0.02	1.0	20.63	0.008	0.022	0.53	56	18.68
10M3-IB	0.23	0.25	0.02	3.69	34.89	0.002	0.009	0.48	4	40.98
10L3-h	0.36	0.23	0.01	1.0	31.25	0.005	0.013	0.40	116	37.57
10L3-IB	0.24	0.27	0.02	1.77	24.13	0.003	0.011	0.45	16	29.25
12S-3/2	0.22	0.16	0.03	7.56	30.60	0.002	0.007	0.66	24	54.50
12S-4/3	0.22	0.16	0.03	5.56	22.50	0.003	0.007	0.70	112	40.88
12S-4/5	0.22	0.16	0.02	5.56	22.50	0.003	0.009	0.70	148	40.88
12S-5/5	0.22	0.16	0.03	4.36	17.66	0.004	0.007	0.74	224	32.70
12S-5/8	0.22	0.16	0.02	4.36	17.66	0.004	0.007	0.74	660	32.70
12S-6/6	0.21	0.16	0.03	3.58	14.51	0.005	0.007	0.78	392	27.25
12S-6/13	0.21	0.16	0.03	3.58	14.51	0.005	0.009	0.78	800	27.25
12S-6/15	0.21	0.16	0.03	3.58	14.51	0.005	0.009	0.78	1,392	27.25
12S-6/18	0.21	0.16	0.02	3.58	14.51	0.005	0.013	0.78	1,988	27.25
12M3-h	0.37	0.14	0.02	1.0	20.63	0.008	0.022	0.53	72	18.68
12M3-IB	0.23	0.25	0.02	4.18	39.53	0.002	0.008	0.47	4	46.43
12L3-h	0.36	0.23	0.01	1.0	31.25	0.005	0.013	0.40	208	37.57
12L3-IB	0.25	0.27	0.02	2.18	27.95	0.002	0.009	0.42	16	36.00

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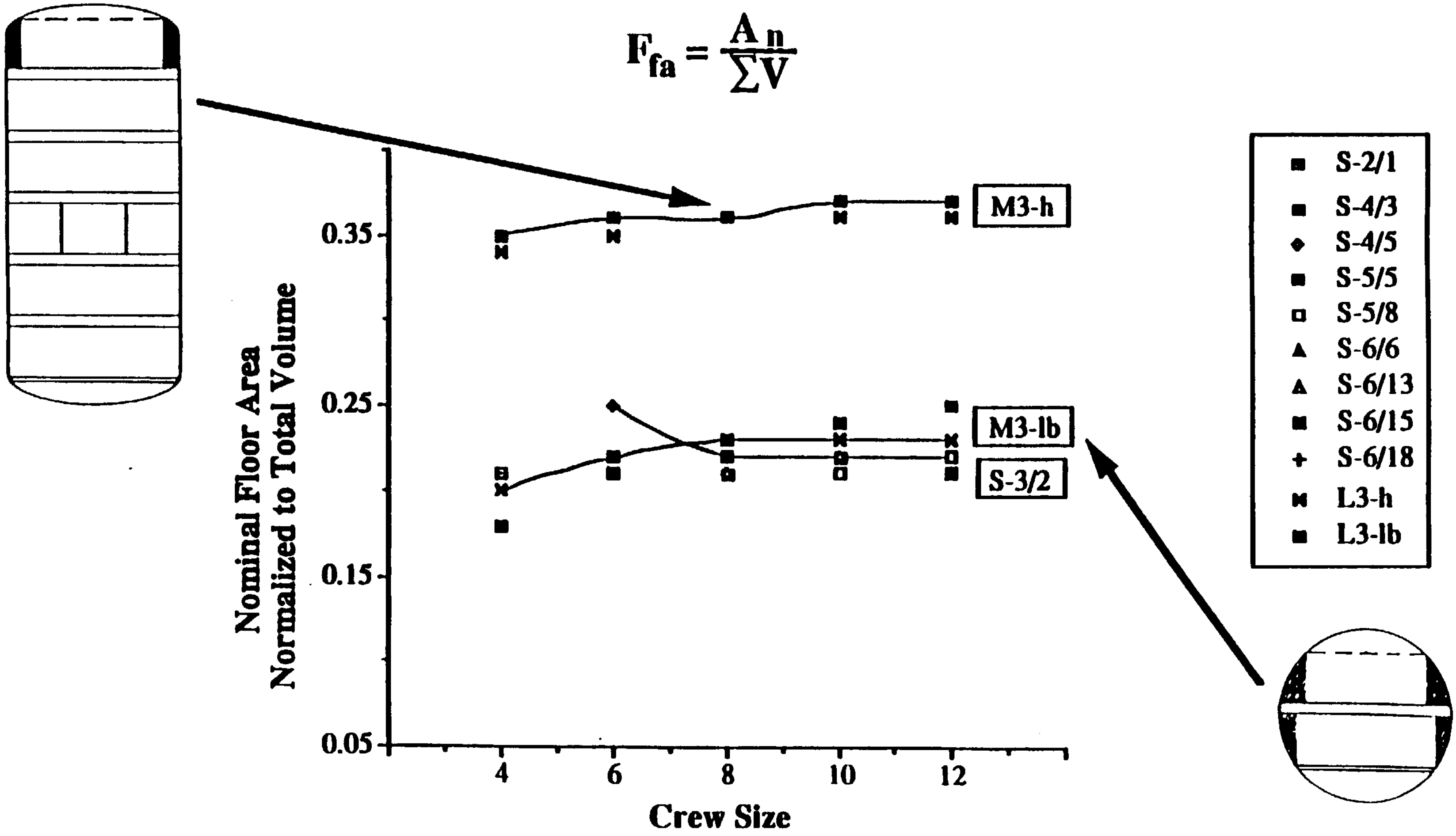
Geometry Metric Analysis
"Inhabitability Factor"

This metric assesses how much of the floor area has a ceiling height of at least 2.3 m (and is therefore nominal by our definition), relative to the total habitat volume. It quantifies the familiar result that walls which curve vertically introduce greater habitability penalties than walls which are normal to the floor.

Geometry Metric Analysis

"Inhabitability Factor"

$$F_{fa} = \frac{A_n}{\sum V}$$



There is more specific nominal floor area in stacked configurations than in tunnel configurations, regardless of module diameter.

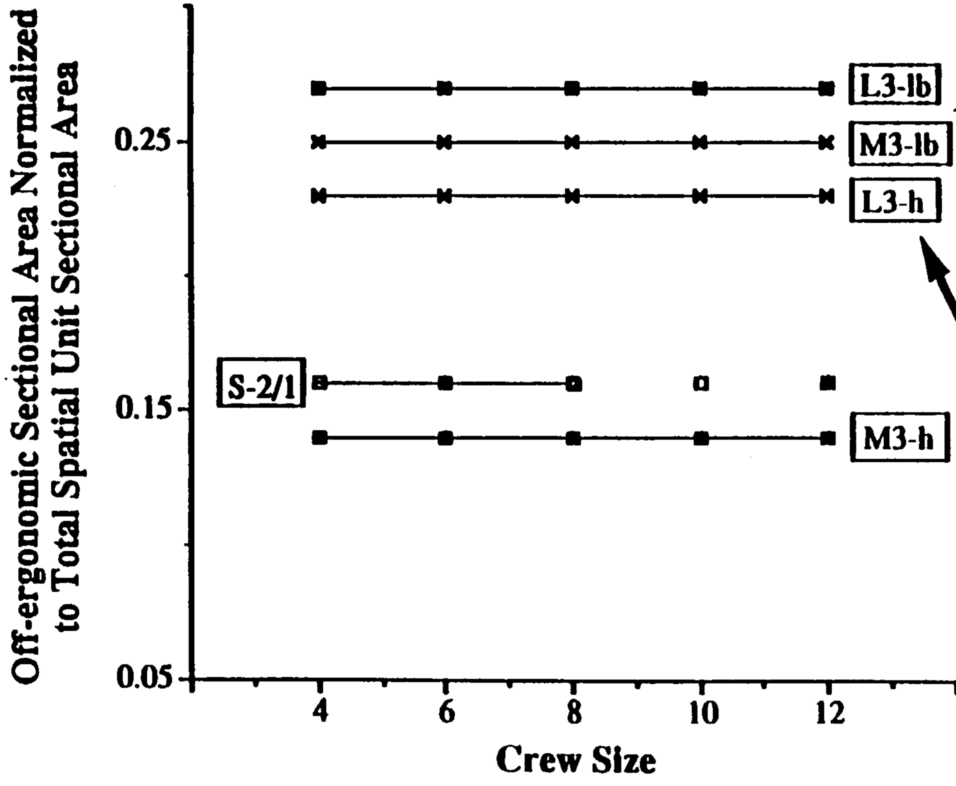
Geometry Metric Analysis
"Vault Factor"

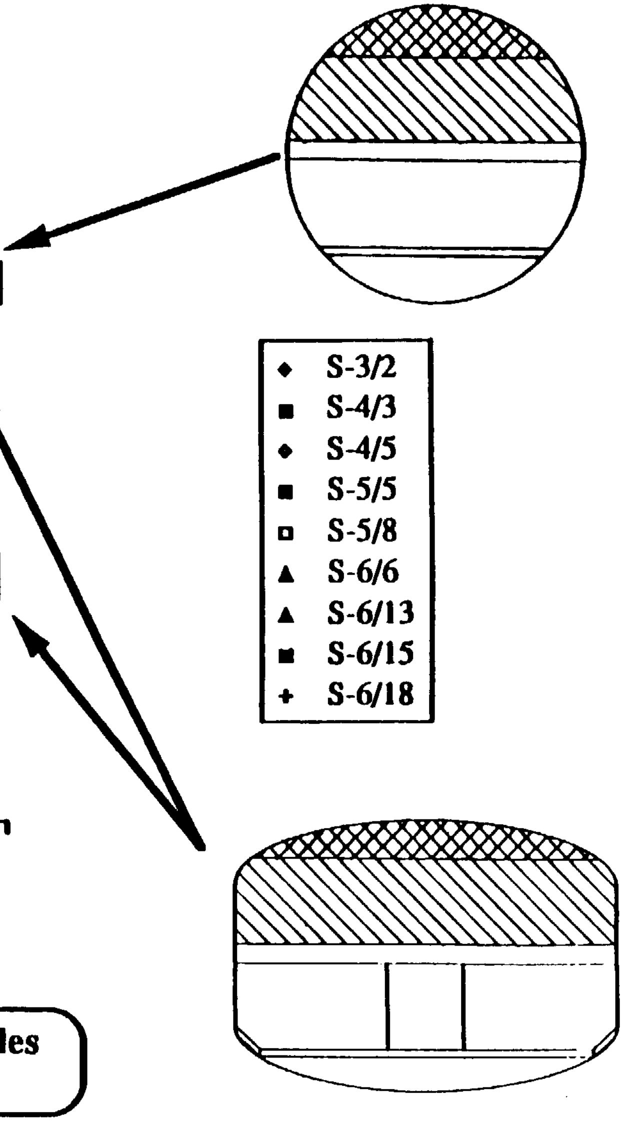
This metric assesses, in the most spatially generous place within each option, the ratio of out-of-reach cross section (in a gravity field) to total cross section. It was developed to pertain to overhead vaults, but the measure remains similar even for µg conditions, since the human reach envelope travels with the body. An indication of how much "height" is out of reach at any time, this metric implies spaciousness in section. Large diameters trade best in both orientations, as does the tunnel orientation of the medium diameter (this result is sensitive to floor configuration assumptions).

Geometry Metric Analysis

"Vault Factor"

$$F_s = \frac{A_{soe}}{\sum A_s}$$





Small-diameter and medium-diameter stacked modules have less generous overhead vaulted spaces.

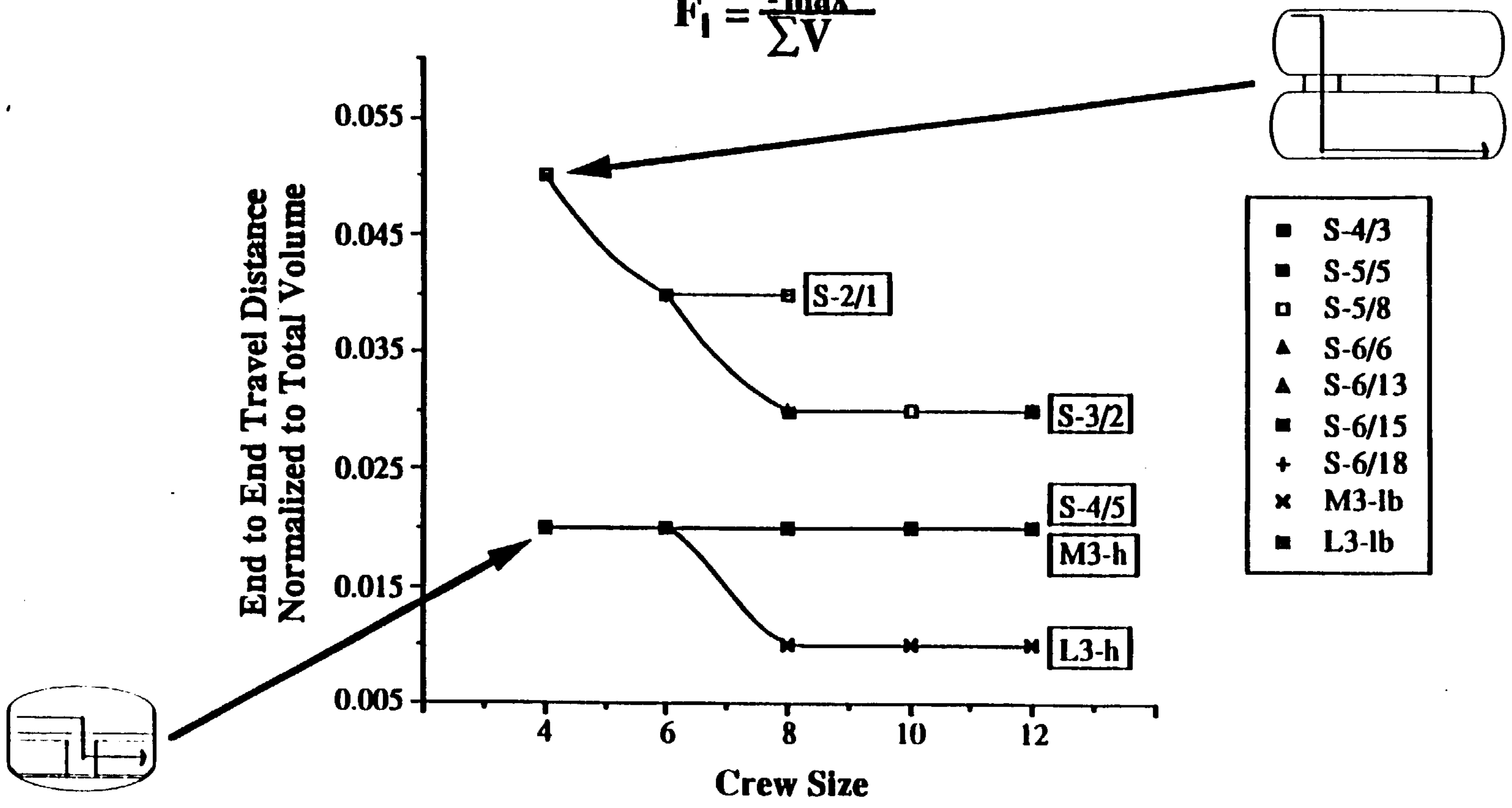
Geometry Metric Analysis
"Domain Factor"

This metric assesses the travel distance from one "end" of the habitat system to the most distal point, normalized to total volume. It is taken as a measure of the domain available in the confined habitat, since long travel times may imply more inhabited territory. However, longer travel times also introduce greater locomotion delays in an emergency. The medium and large diameter options have more compact domains.

Geometry Metric Analysis

"Domain Factor"

$$F_l = \frac{l_{max}}{\sum V}$$



Small-diameter modules would seem to cover more territory, but larger diameter modules allow shorter emergency access times.

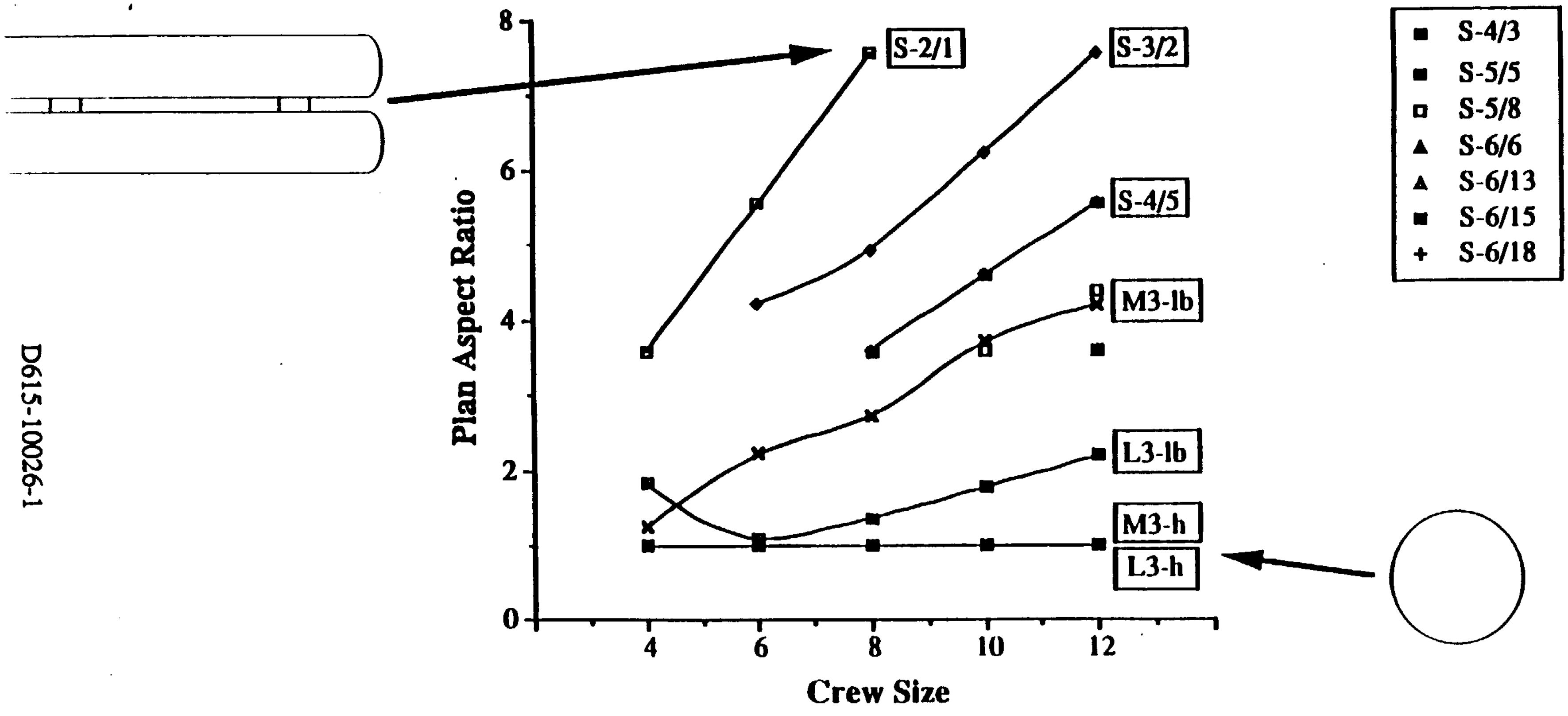
Geometry Metric Analysis
"Hallway Factor"

This metric assesses aspect ratio in plan of the largest single perceivable spatial unit within the habitat (one module in a cluster, or one floor in unitary options). The stacked options remain constant with increasing crew size, because the dimensions per floor remain constant. The tunnel options all become more like hallways with increasing crew size, since they get longer in plan. For the small diameter options, the fewer modules in the cluster, the longer each must become with larger crew sizes, and therefore the steeper the slope of the curve.

Geometry Metric Analysis

"Hallway Factor"

$$F_{rp} = \frac{x_{max}}{x_{min}}$$



Medium and large diameter options provide more evenly-proportioned spatial units.

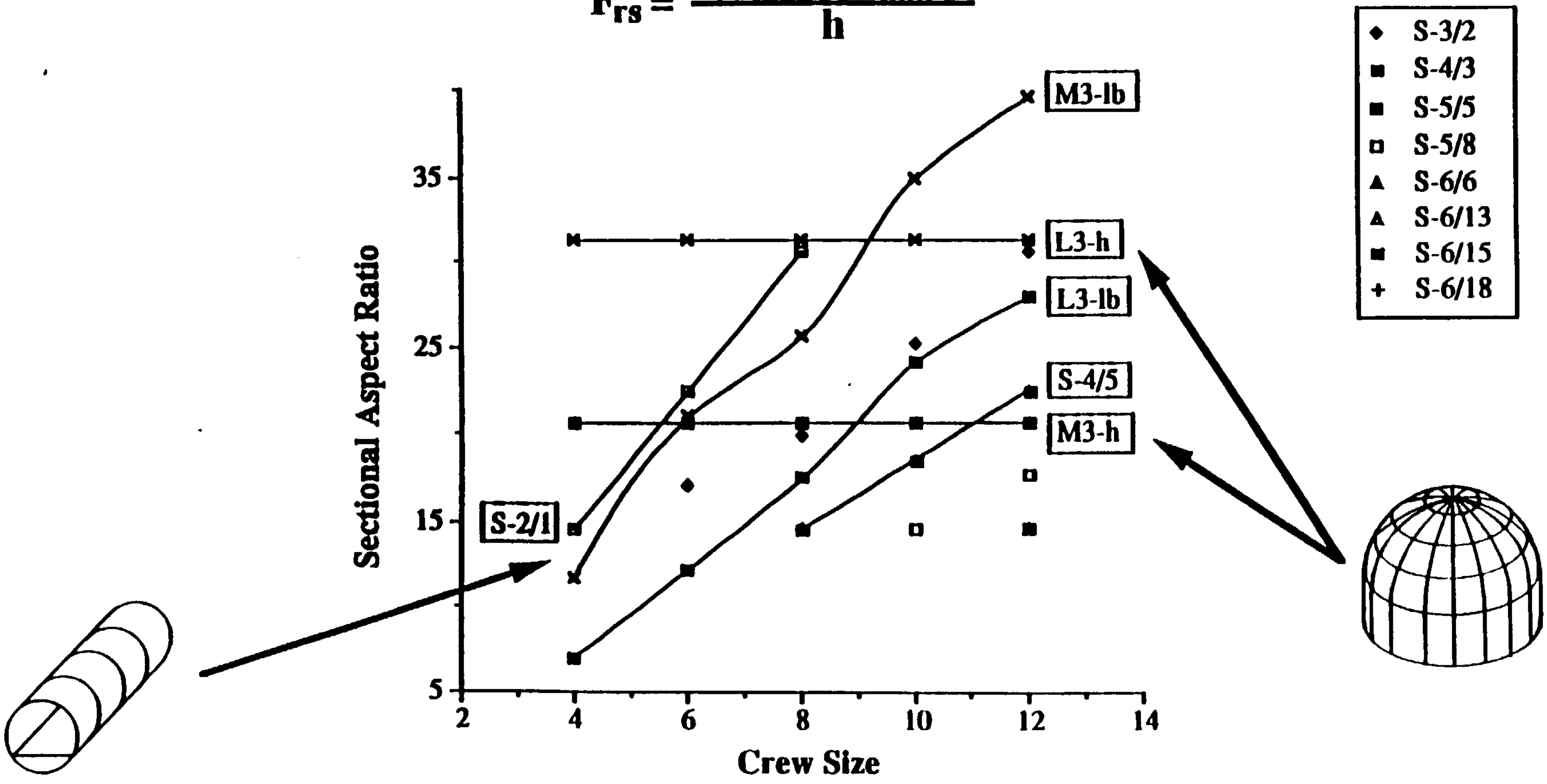
Geometry Metric Analysis
"Spaciousness Factor"

This metric assesses sectional aspect ratio of the largest perceivable volume within each option. This is a more apt measure of overall spaciousness than the "vault factor", because it includes three dimensions. Stacked options remain constant with increasing crew size, because the top floor does not change dimension. The dome they provide trades very well for the smallest crew sizes, but is passed by the tunnel options for the larger crew sizes because their barrel vaults grow in length commensurately.

Geometry Metric Analysis

"Spaciousness Factor"

$$F_{rs} = \frac{(x_{max})(x_{min})}{h}$$



Domed spaces appear more spacious for small crew sizes. With fixed diameter, barrel-vaulted spaces trade better for larger crew sizes.

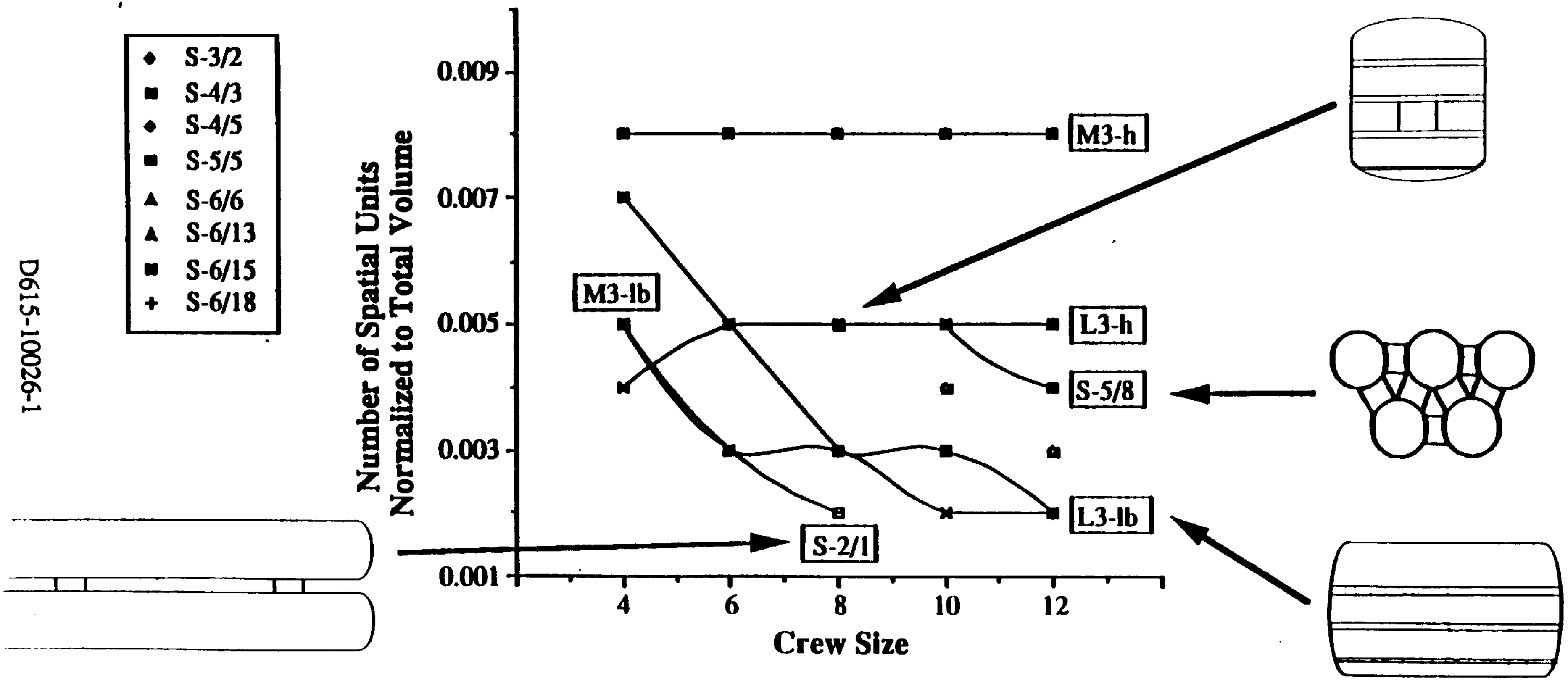
Geometry Metric Analysis
"Variety Factor"

This metric assesses how many perceptual pieces the available volume is broken up into, which measures (among other things) how much opportunity exists for optimizing spatial units for distinct functions. It is similar to the Spatial Units Factor in the Topology Metric Analysis, but includes the larger diameter options as well, and is normalized to total volume. The tunnel options trade poorly for large crew sizes (although interior designs could generate more spatial units with the cavernous volume available).

Geometry Metric Analysis

"Variety Factor"

$$F_u = \frac{U}{\sum V}$$



Stacked, and most cluster, configurations provide more opportunities for functionally unique spatial units than do tunnel configurations.

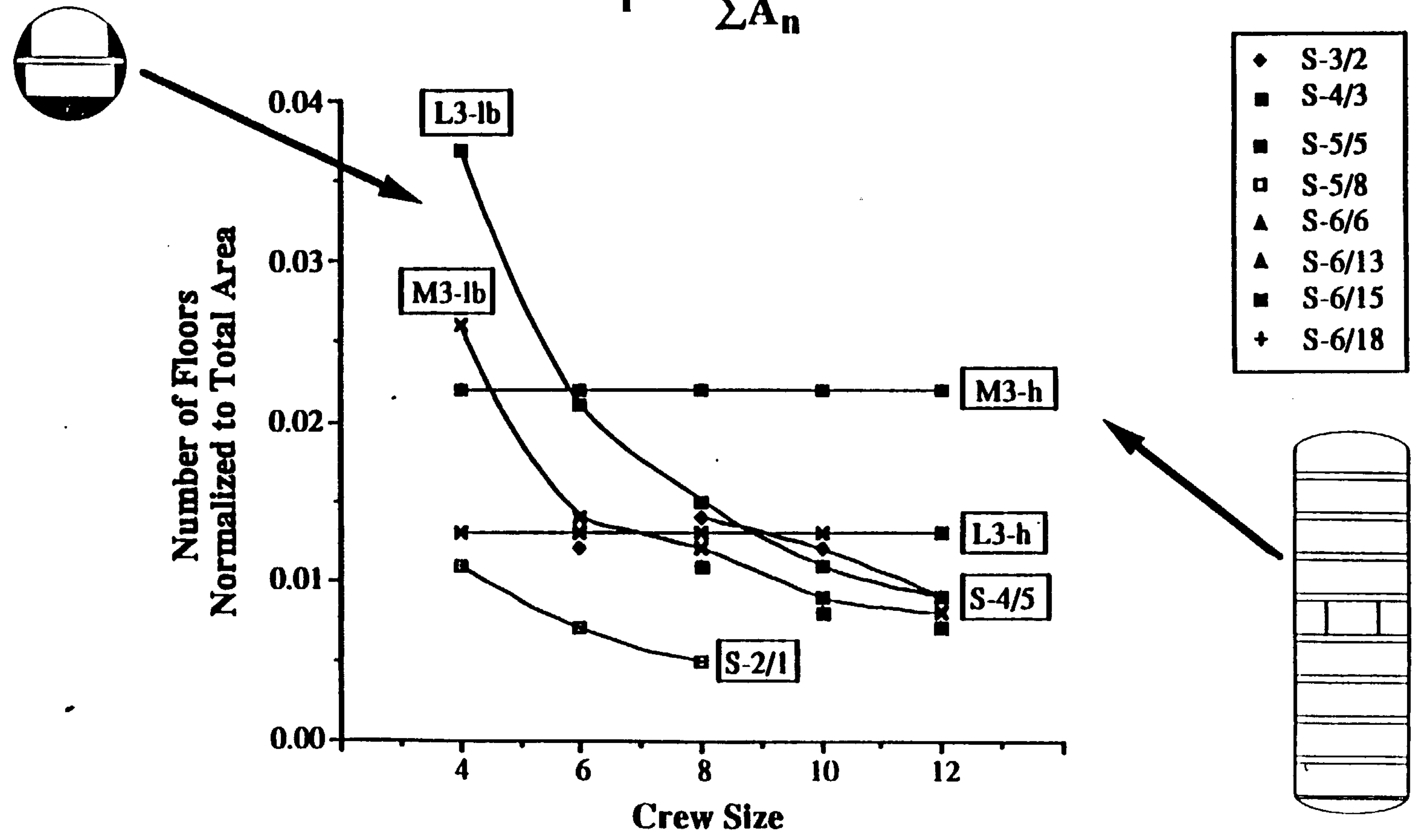
Geometry Metric Analysis
"Elevator Factor"

This metric reveals how many separate floors the available floor area is broken up into. "Split-level" cluster options were assigned fractional numbers-of-floors for the calculations. Whereas the metric may be largely irrelevant for µg conditions, in a gravity field it provides a strong discriminator against particularly the stacked medium diameter option for large crew sizes (analogous to living in a 9-story house with one room on each floor). For the smallest crew sizes, the medium and large modules get so short that the tunnel orientation becomes a less efficient was to organize the internal space.

Geometry Metric Analysis

"Elevator Factor"

$$F_f = \frac{F}{\sum A_n}$$



Stacked options for large crew sizes have an excessive number of floors in gravity orientations. Medium and large diameter tunnel options work inefficiently for small crew sizes.

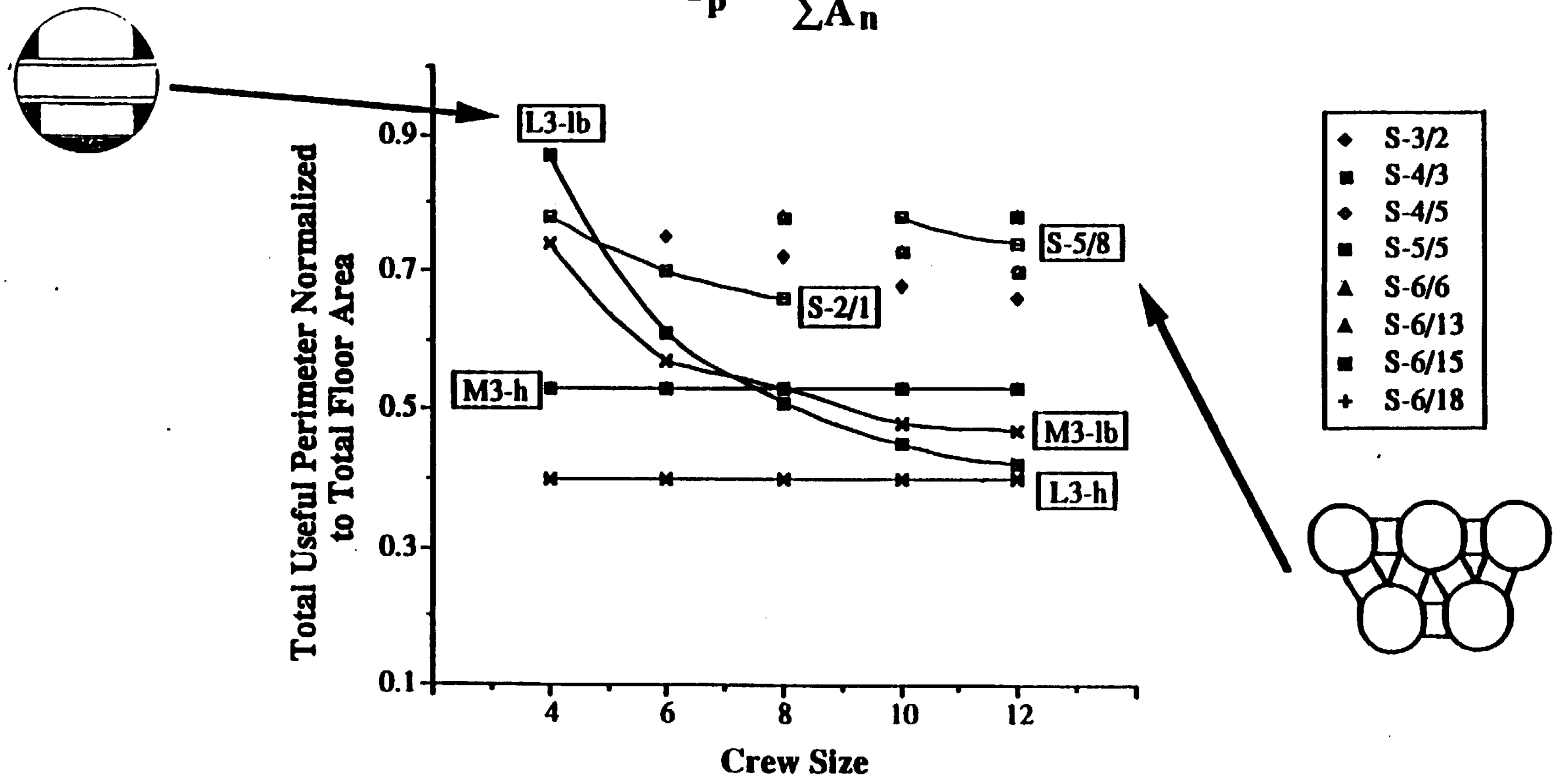
Geometry Metric Analysis
"Perimeter Factor"

This metric was devised to investigate the penalty in usable "room" perimeter suffered by module cluster options sporting a lot of interconnection tunnels. What it reveals, however, is that the small diameter options have so much more specific surface area that the tunnel effect washes out; the larger diameter options have much less intrinsic wall area available. This means that equipment mounting cannot as readily take advantage of wall locations for these latter options; however, their reduced pressure vessel wall area will be seen to confer a mass advantage.

Geometry Metric Analysis

"Perimeter Factor"

$$F_p = \frac{\sum P_u}{\sum A_n}$$



Although there are modest differences among cluster options due to tunnel openings, these modules provide more inherent wall space than larger diameter options because of their greater surface area.

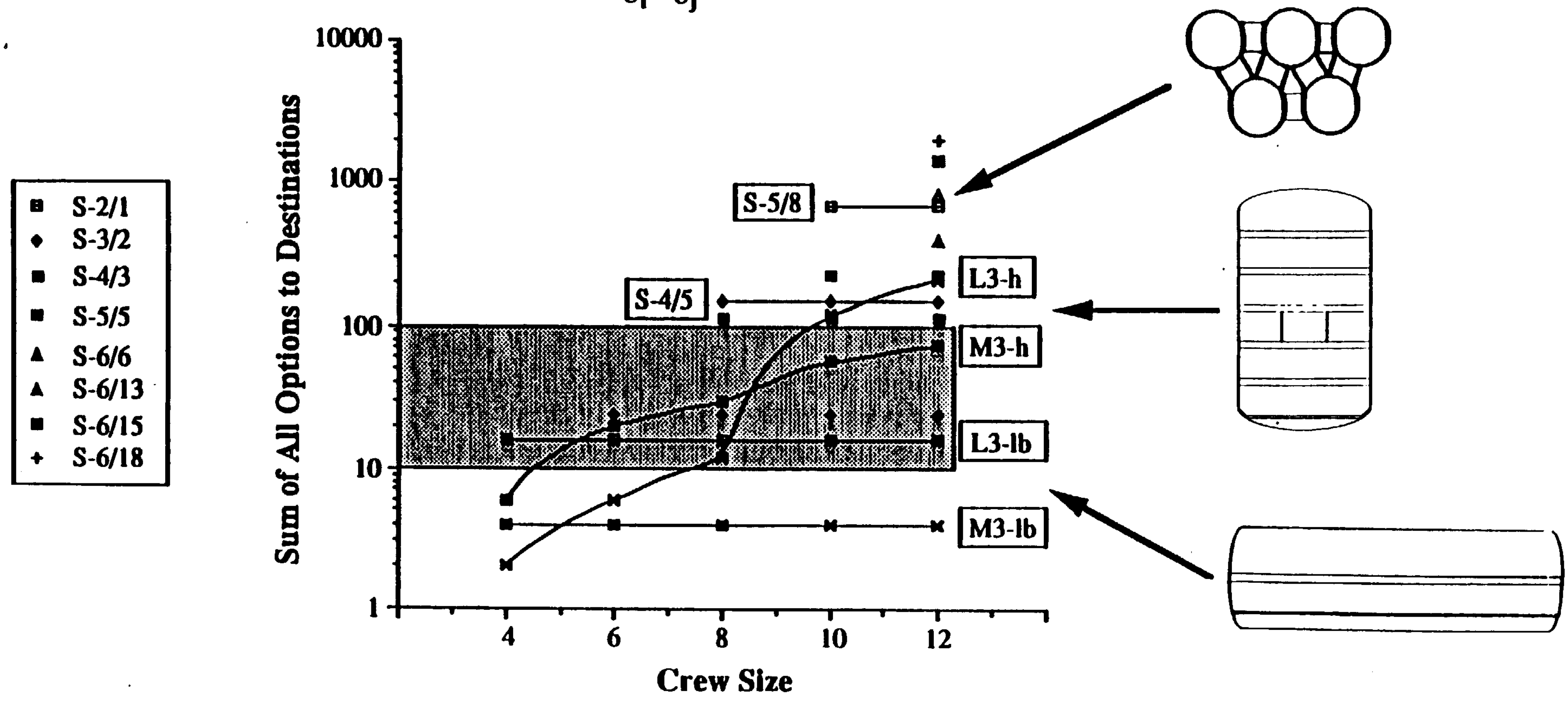
Geometry Metric Analysis
"Pathway Factor"

This metric calculates the number of different ways to get from one spatial unit to another in the habitat options, summed over all combinations of starting and destination units. Note the logarithmic ordinate, which ranges from 2 to over 2000. The discontinuity in the curves for the larger diameter stacked options reveals the assumption than two separate vertical circulation paths are required for crew sizes larger than 8, just to avoid circulation congestion. Interconnected clusters can provide many, many pathway options. This may be quite advantageous in mitigating "domain boredom" over long durations, and in alleviating social concentration when undesirable. Perceptions of inherent habitat privacy accommodation may be enhanced with many pathway options. A suggested range of pathway numbers is indicated on the graph; because it has only two floors, the medium diameter tunnel option trades quite poorly.

Geometry Metric Analysis

"Pathway Factor"

$$F_o = \sum_{U_i} \sum_{U_j} O_d \left({i,j = 1,\ldots.6 \atop i \neq j} \right)$$



The intermediate range numbers (~10 to 100) provide sufficient circulation options to mitigate boredom, without incurring an excessive hardware penalty.

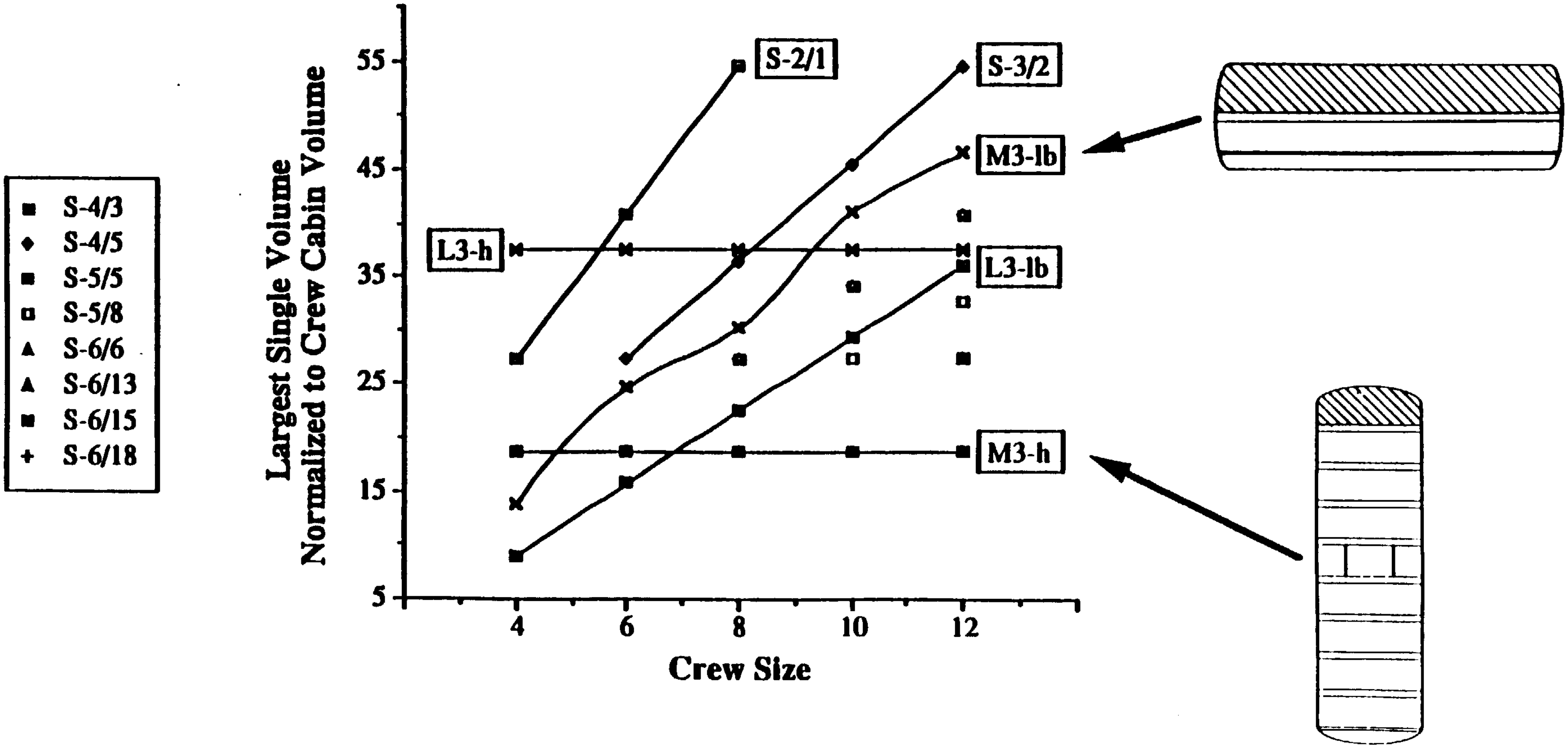
Geometry Metric Analysis
"Scale Factor"

This metric compares the largest single spatial unit available to an individual crew cabin, measured in volume, as an indication of the range of spatial scales available within each habitat option. A greater range may imply greater potential spatial variety when the interior is configured, important to mitigate perceptual boredom over long durations. For small crew sizes, the top floor dome of the large diameter stacked option is dramatic; for larger crew sizes, however, the tunnel options are more favorable. The seemingly good performance of some cluster options for large crew sizes shows that some of those modules get very long; this must be weighed against their large "Hallway Factor".

Geometry Metric Analysis

"Scale Factor"

$$F_v = \frac{V_{max}}{V_{min}}$$



With increasing crew size, tunnel options provide potential for greater variety of spaces.

Activity & Proximity Analysis

In addition to the topology and geometry metric analyses, it is important to determine if any unique complications arise from the interior configuration standpoint for the primary habitat module options.

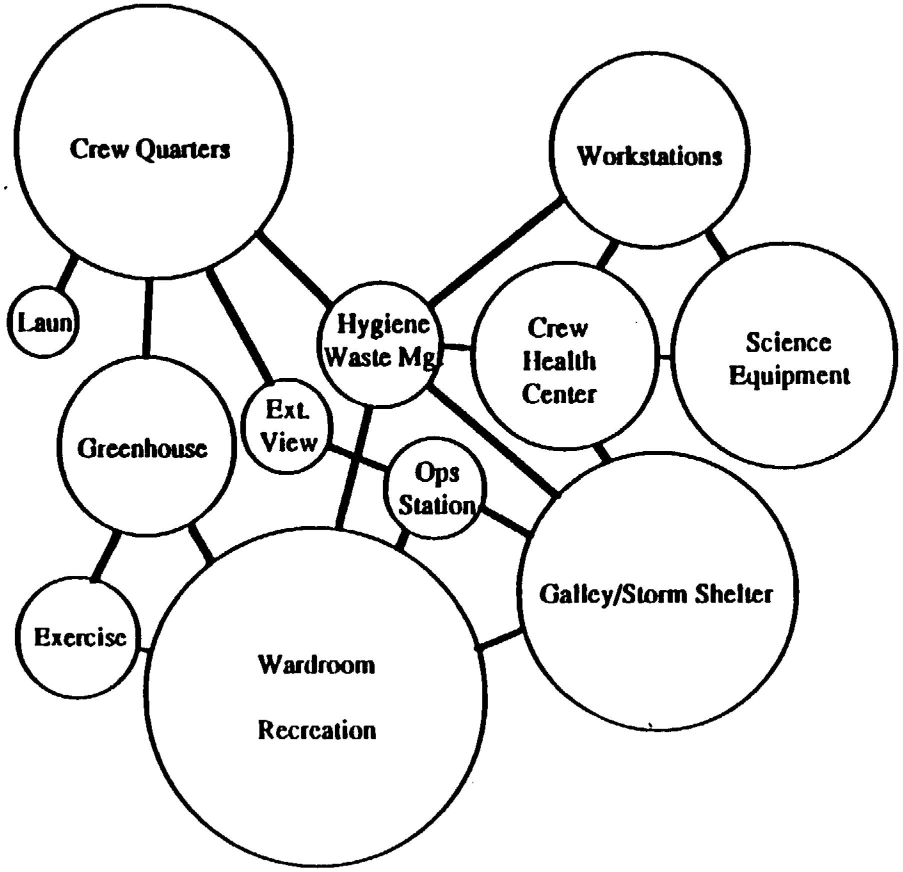
Using SSF and terrestrial design as starting points, we developed representative functional area allocations for habitat activities (the area bias emphasizes the gravity condition --- a μg bias would emphasize packing volume as well as surface area). The allocations listed are totals per crew of four. Excepting those values noted as "per crew", which remain constant to first order despite crew size changes, the relative areas scale with increased crew size. The activities taking place in those allocated areas are related by proximity constraints of varying strengths, to be close together or far apart. For example, recreation activities should be far from sleep areas to avoid disturbing resting crew members. However, most habitation areas and the recreation area should have viewing access to greenhouse facilities. The proximity diagram then serves as a guide for developing interior configurations which satisfy functional and perceptual requirements.

Activity & Proximity Analysis

Representative Allocations
(referenced as floor area for gravity configuration)

Activity Location	Area (m^2)
Crew Quarters	16
PH/WMF	3
Galley/Storm Shelter	16
Wardroom/Recreation	12
Exercise	2
Greenhouse	6
Operations Station	2 (per crew)
Workstations	8
Science Equipment	10
CHC	7
ECLSS	12
EVA Stowage	2
Laundry	1 (per crew)
Spares Stowage	7 (15% active equip)
Circulation	9 (15% crew space)

Scaled Proximity Diagram



The proximity analysis provides the basis for developing interior configurations which work and which address the unique requirements of long-duration spaceflight

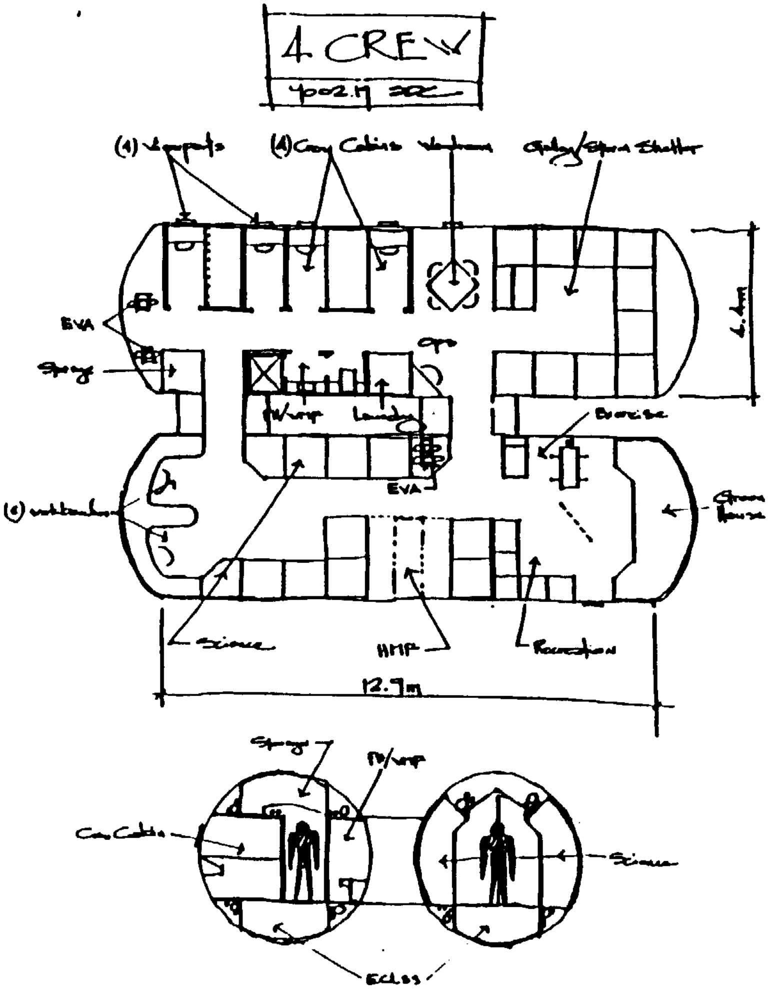
Reference Configurations

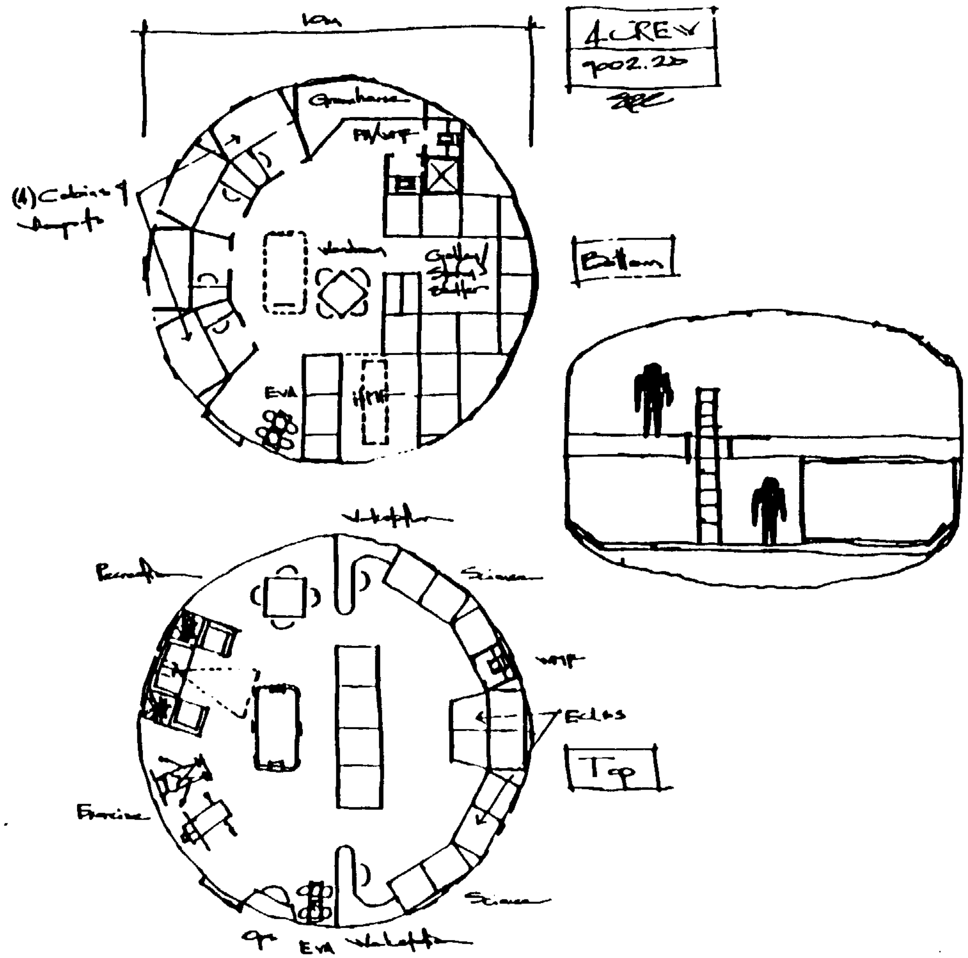
The next five charts show preliminary layout sketches of interior configurations developed for crew sizes of 4, 8 and 12, using either simple clusters of small diameter modules or unitary large diameter modules.

Each habitat type has unique advantages and disadvantages from the interior configuration standpoint; however, no "roadblock" considerations were uncovered with these initial studies.

Reference Configuration

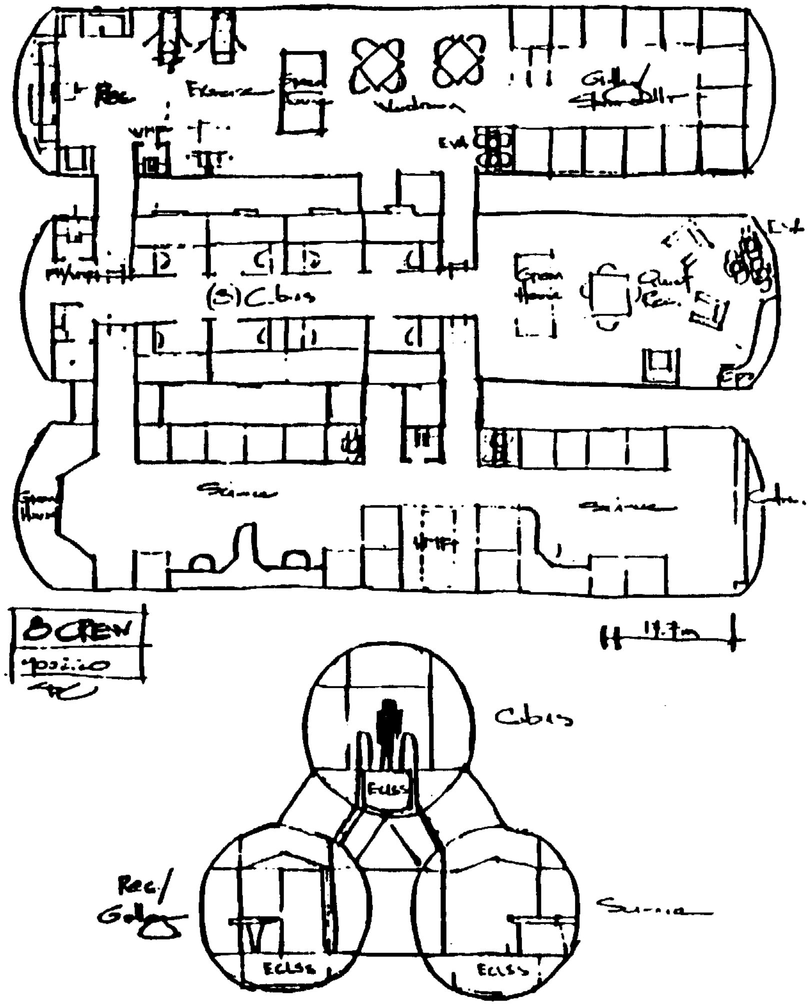
4Sg2-2/1 & 4Lg3-h





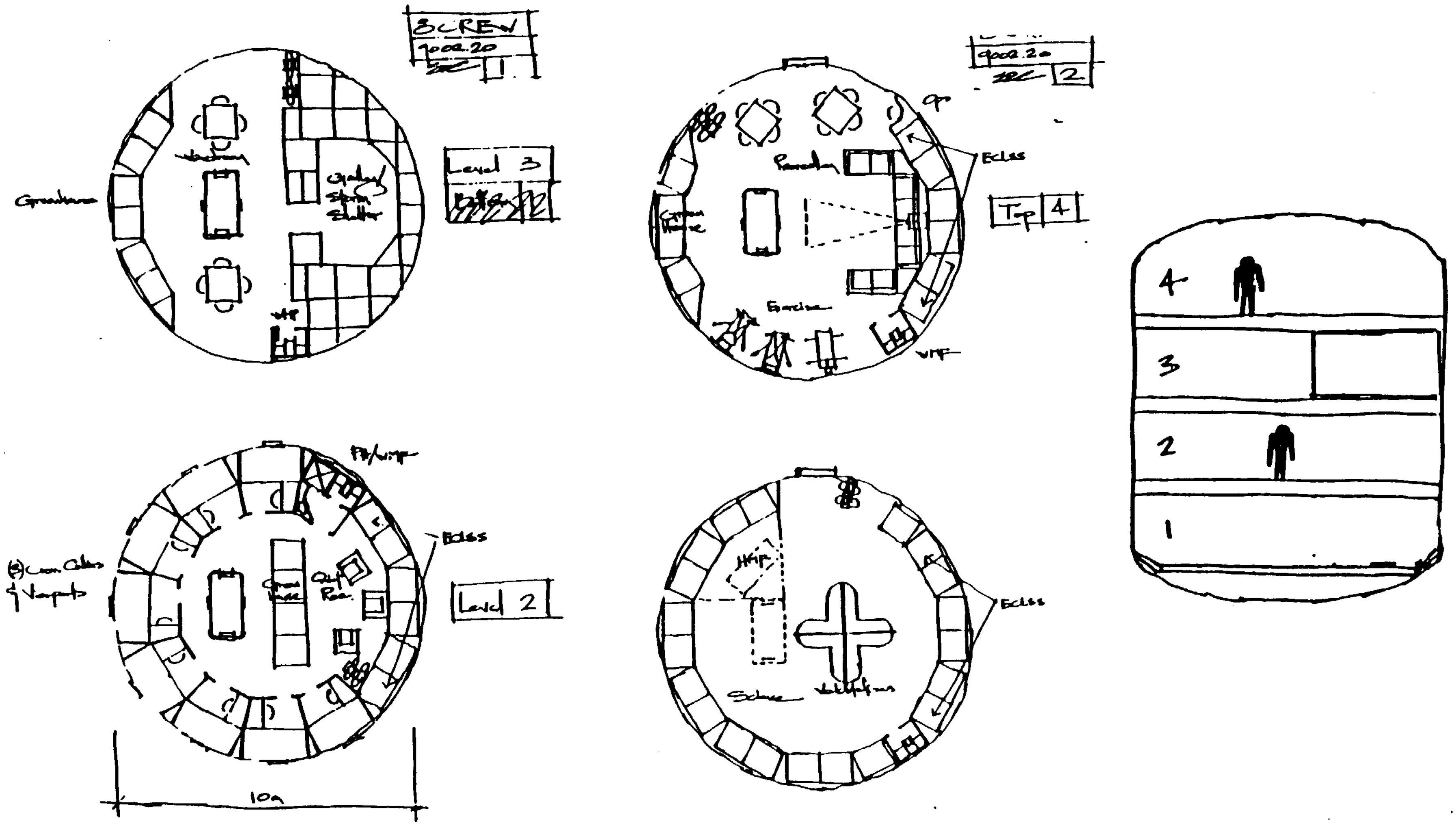
Reference Configuration

8Sg2-3/2



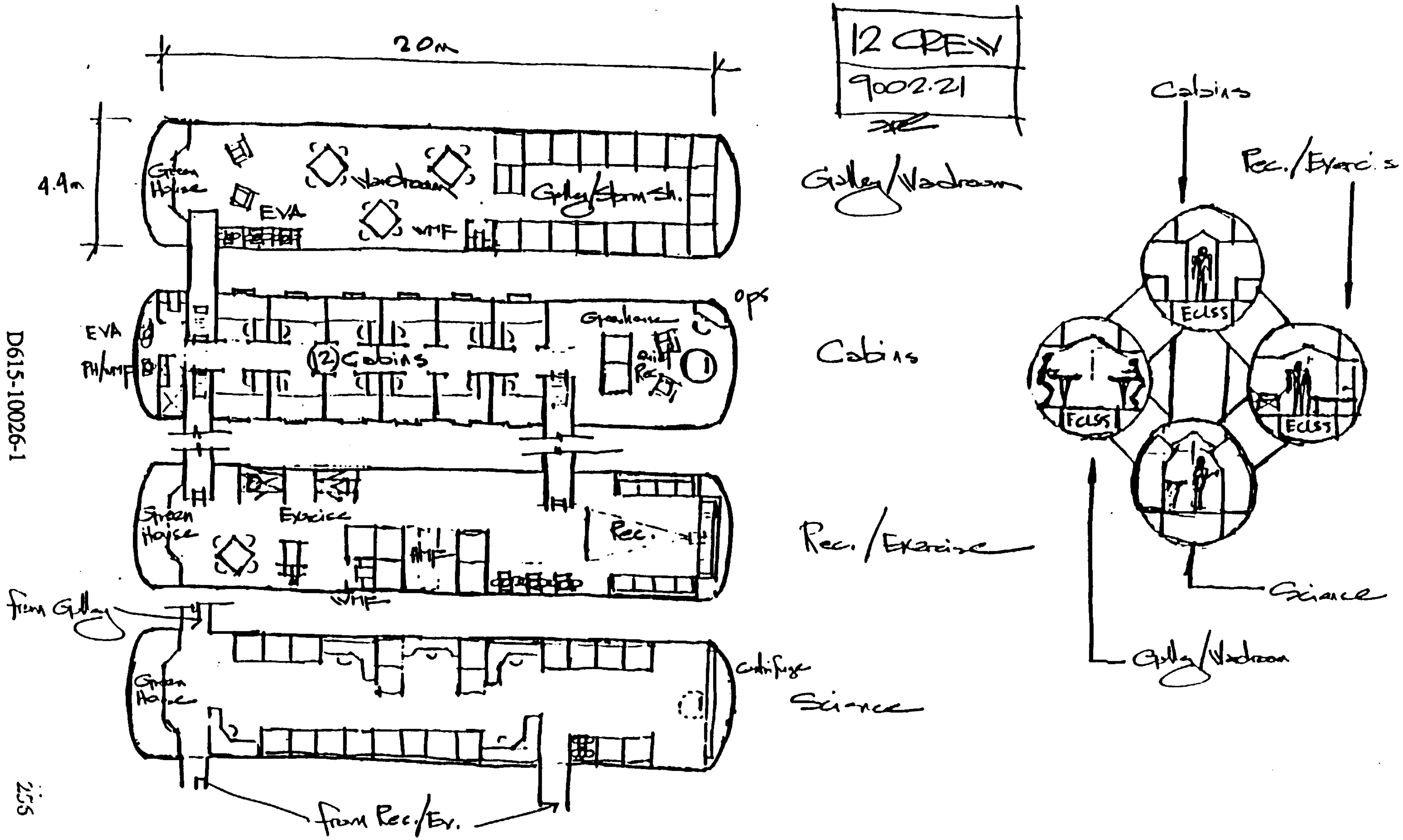
Reference Configuration

8Lg3-h



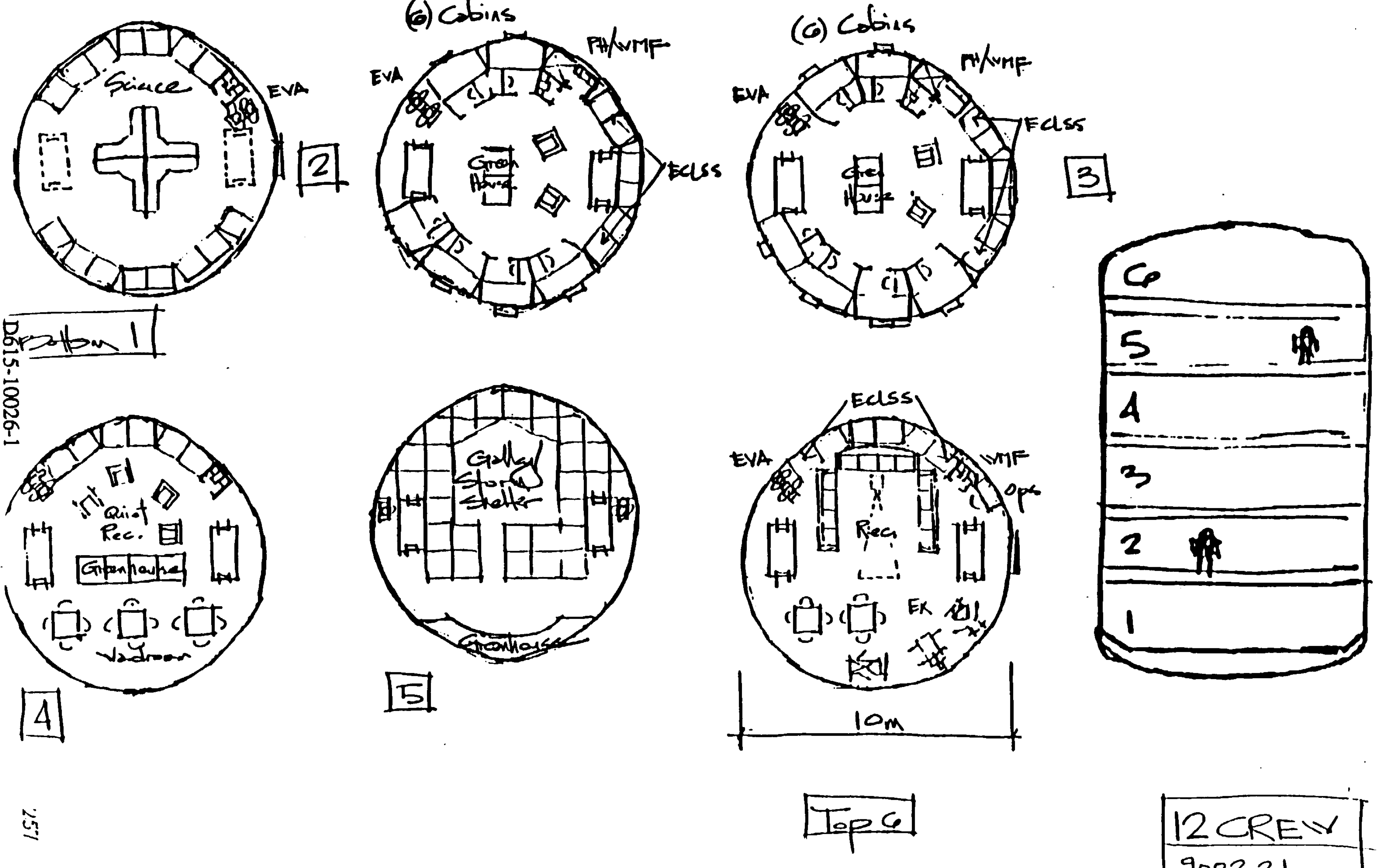
Reference Configuration

12Sg2-4/5



Reference Configuration

12Lg3-h



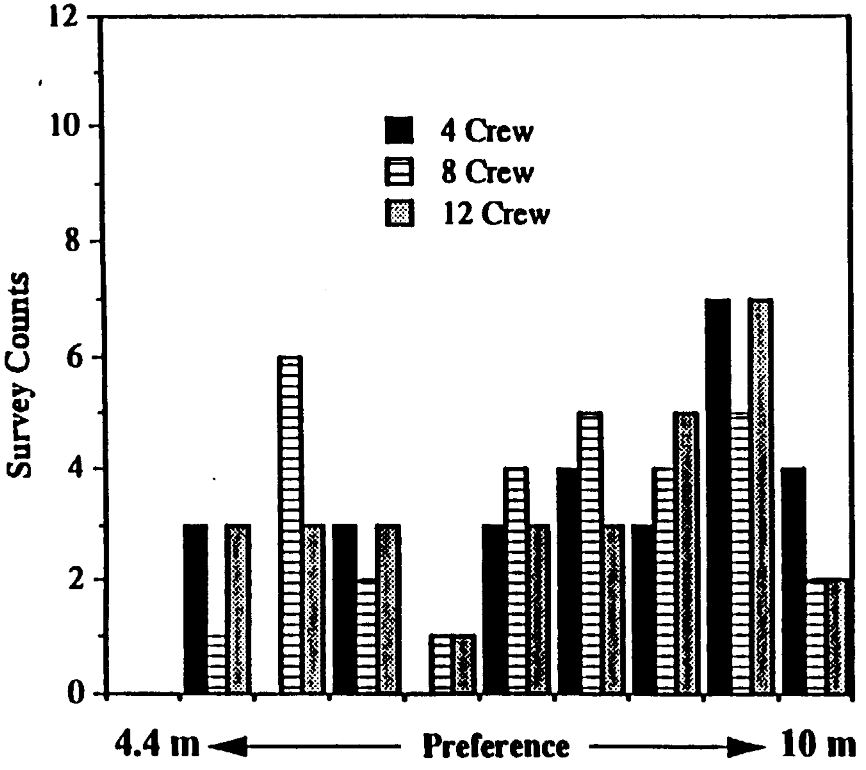
Opinion Survey Results

Using the configuration sketches as points of reference, we solicited the opinions of 56 Boeing employees, asking them to indicate their module type preference on a scale of 1-10 between the small diameter cluster option and the large diameter option, for each of the three crew sizes 4, 8 and 12. We explained to the respondents the type of mission, its maximum possible duration, and the conditions of confinement that characterize it. Each person also had the opportunity to record a simple explanation of the preference indicated. The quantitative results are collected here.

Both classes of respondents showed a statistically bimodal preference. It is not clear whether this is an artifact of the survey technique, or whether people tend actually to develop strong preferences. Engineer-respondents were not as extreme in their preference bifurcation, but tended to prefer the large-diameter option. The breakdown was performed according to engineers and non-engineers because engineers represented the best paradigm available for the type of crew members likely to fly early Mars missions. Many comments were made that the large diameter option seemed more spacious, or "was" larger (even though both options' volumes were strictly the same in all cases). The strongest preference peak for non-engineers was in precisely the place the engineers categorically avoided: complete preference of the small diameter options. A possible explanation, indicated by some of the comments written by non-engineers, is that those people concentrated more on the floor plan than the section cut. In plan, the small diameter option appears more familiar and rectilinear; however, the section (a more accurate experiential estimator of spatial character than the plan, which is a behavioral document) reveals the large diameter option to generate in fact more familiar spaces.

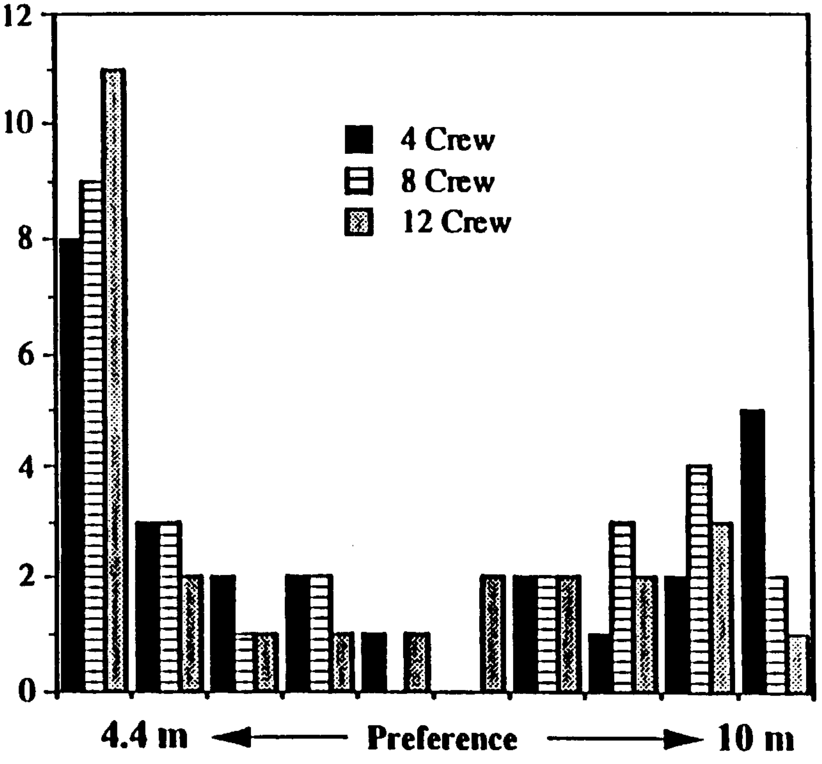
Opinion Survey Results

Responses of 30 Engineers



- Distribution weakly bimodal
- Preference weighted toward large-diameter option
- Sample is probably more appropriate paradigm for technical mission crew

Responses of 26 Non-engineers



- Distribution strongly bimodal
- Strong preference peak for small-diameter cluster option
- Participants tended to focus on floor plan, less attention to vertical section (a more apt indicator of spatial character than plan)

MTV Hab Trade Weight Groundrules

Listed here is the allocation of structural subsystems into discriminators and non-discriminators for the structural mass analysis of the long-duration habitat trade study. Those items which occur in any habitat system regardless of type (EVA-specific and viewing equipment) are not considered. Equipment mounting standoffs are not included because to first order their mass is not expected to be configuration-dependent. Floors and walls, albeit strongly configuration-dependent, were not included because their variations were suspected of being second-order. That turned out in fact to be a valid assumption.

MTV Hab Trade Weight Groundrules

Primary Structure (trade discriminators)

- Pressure vessel
- Structure rings and ribs
- Pressure bulkheads (if any)

Secondary Structure (trade discriminators)

- Inter-module tunnels (if any)
- Inter-module integrating structure (if any)
- Pressure hatches separating redundant volumes
- Meteoroid, debris and thermal protection (surface-area-based)

Secondary Structure (not included; non-discriminators to first order)

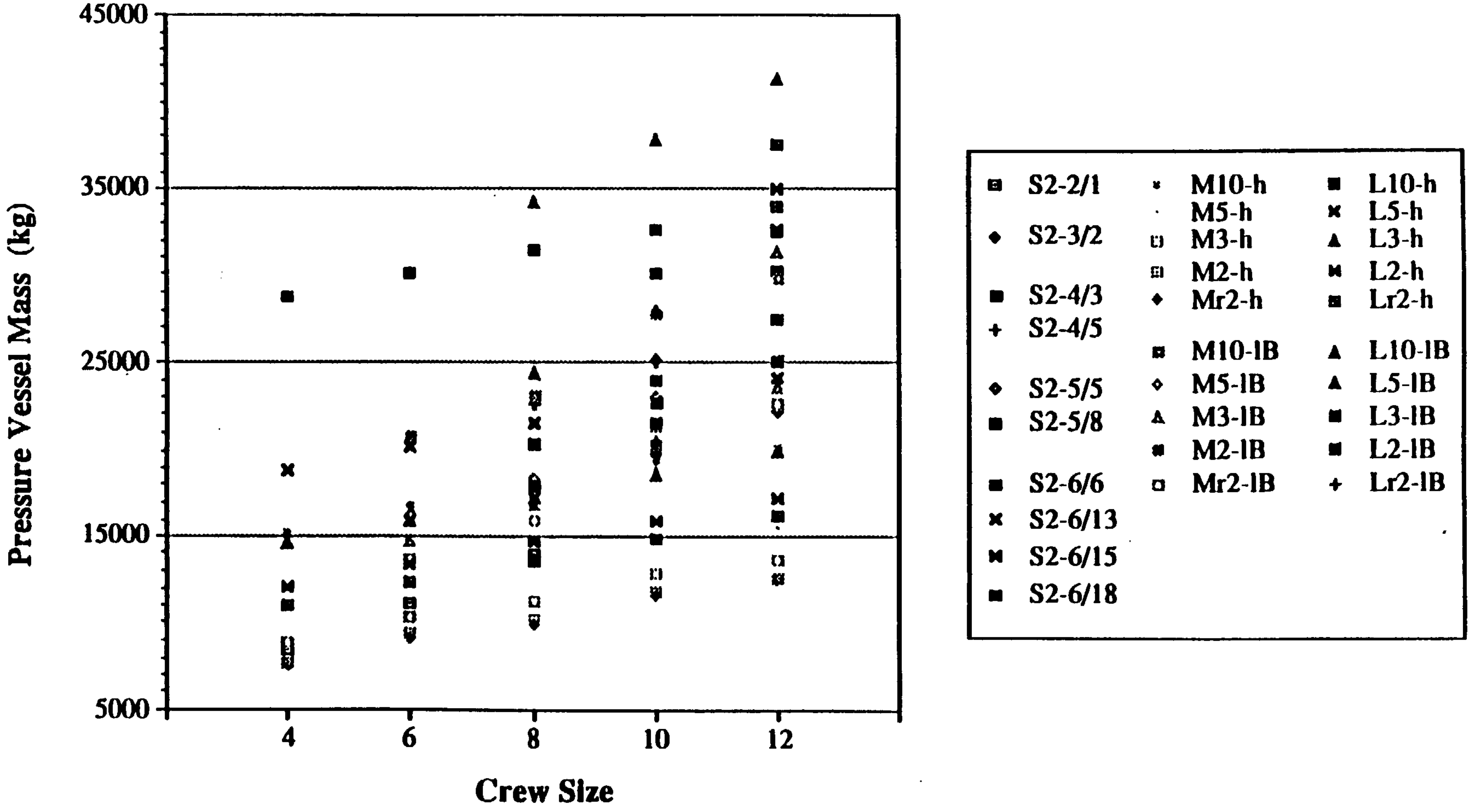
- Airlocks
- Hatches associated with airlocks / EVA
- Windows
- Floors
- Walls
- Subsystem mounting standoffs

Pressure Vessel Mass Analysis

This chart shows the total pressure vessel system masses calculated, including all subsystems just enumerated, for 30 concepts which survived the topology and geometry metric analyses.

Pressure Vessel Mass Analysis

Mass Sensitivity



30 pressure vessel concepts were weighed, covering a wide range of sizes, types and configurations

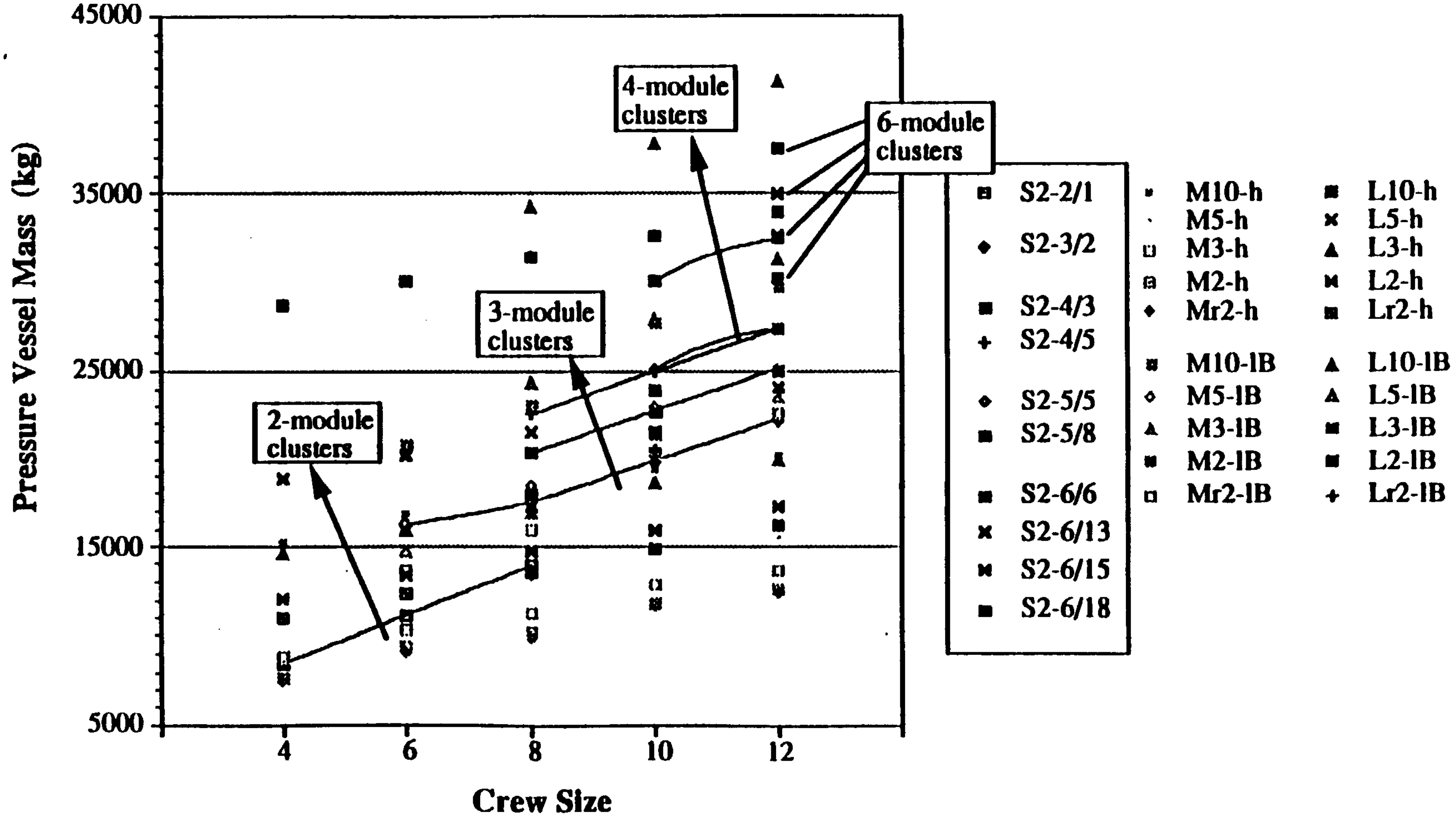
4.4m-diameter Module-cluster Mass Analysis

Plotted as a subset of the 30 concepts are the small-diameter options. Total mass rises rapidly with increasing numbers of modules in the clusters, which penalizes these options for large crew sizes since reasonable single-module length-limits (27 m, commensurate with an HEI Shuttle-C) require clustering for large crew sizes.

4.4m-diameter Module-cluster Mass Analysis

Mass Sensitivity (Small-diameter)

Parameters: Cluster Size
Cluster topology



Clustering modules together weighs much more than extending the modules' lengths

7.6m-diameter Module Mass Analysis

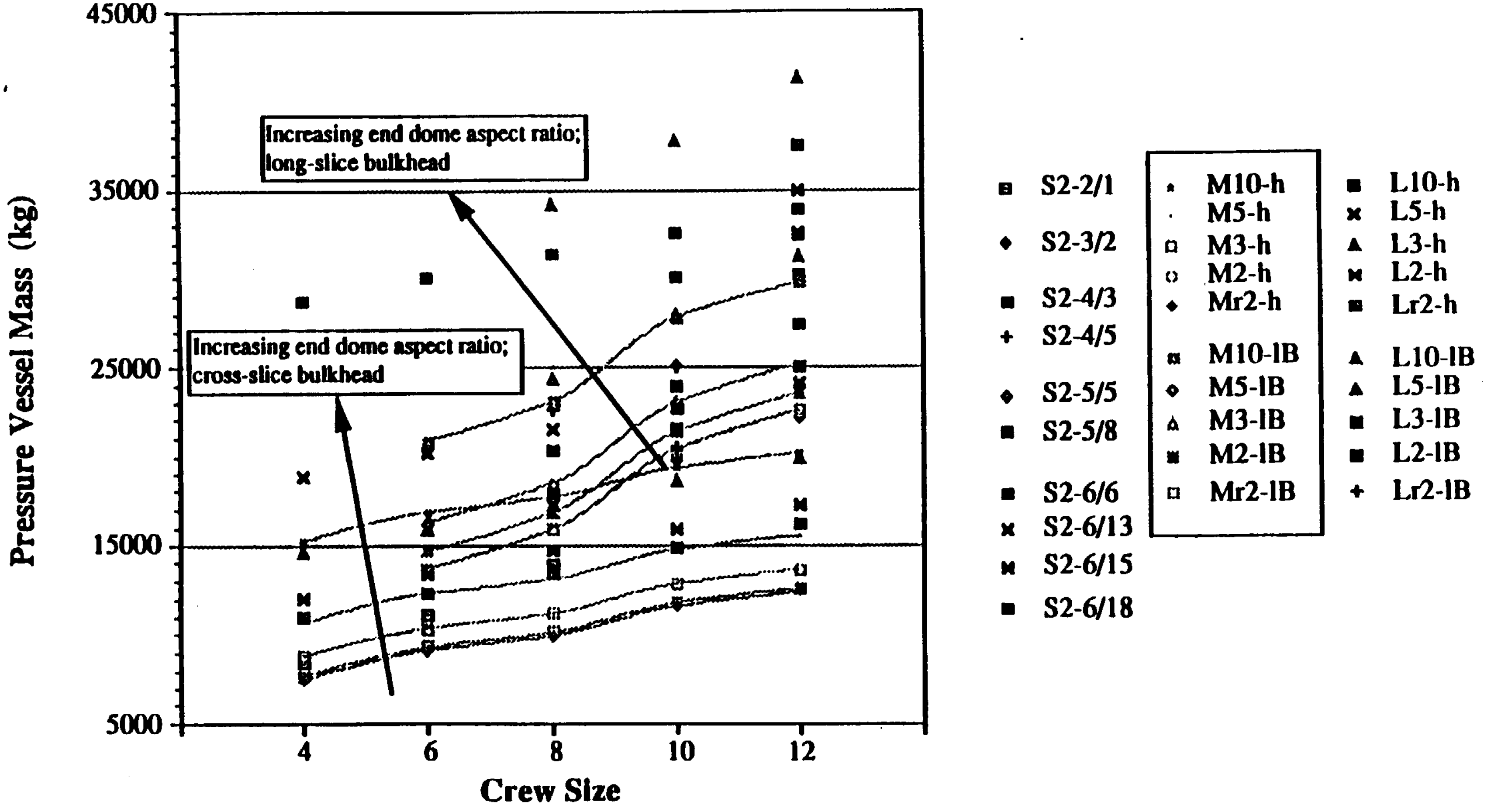
Plotted here are the medium diameter options, both stacked and tunnel-oriented, parametrized according to end dome aspect ratio. Flat end domes are extremely mass-expensive, as is the tunnel orientation (which according to study assumptions has the internal pressure bulkhead running longitudinally).

The curves are not linear because a unit module length increase is achieved by adding 2.8 m (one floor height plus floor structure), which does not correspond precisely to our specific volume assumptions as the crew size increments by one. The module concepts sized for 4 and 8 crew have thus been slightly volume-penalized.

7.[illegible]m-diameter Module Mass Analysis

Mass Sensitivity (Medium-diameter)

Parameters: End Dome Aspect Ratio
Pressure Bulkhead Orientation



Crew size is a non-linear independent variable for the medium-diameter module; flatter end-domes and longitudinal bulkheads increase mass dramatically

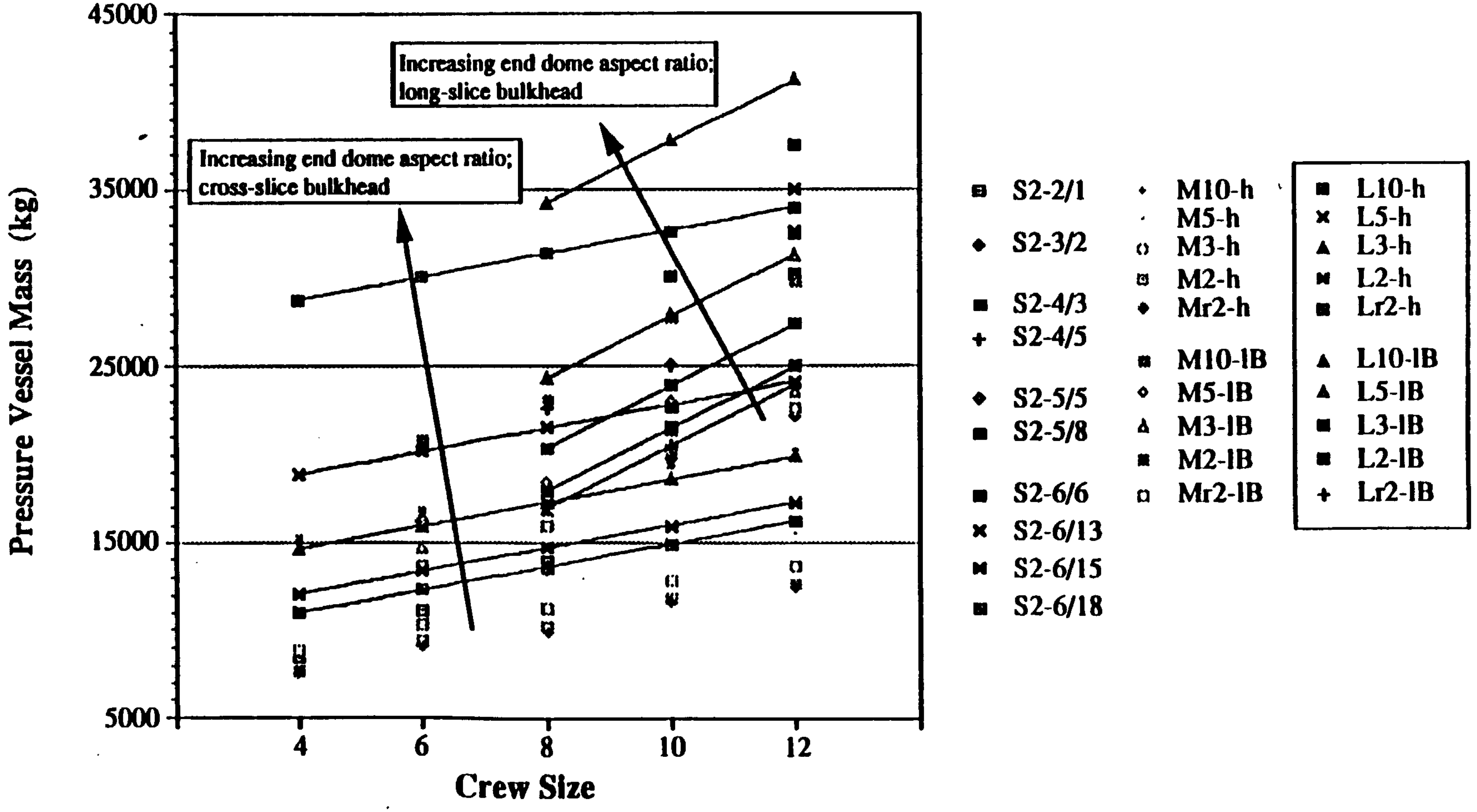
10m-diameter Module Mass Analysis

Plotted here are the large diameter option masses. The plot is analogous to the previous one; however, in this case a one-floor increment in module length corresponds to a one-person increment in crew size, so the curves are linear. The spread caused by various end-dome shapes is more pronounced than for the medium diameter options, because the larger diameter exacerbates the high stresses in the dome shoulder. Dome aspect ratios of 2 and $\sqrt{2}$ are quite close in mass performance.

10m-diameter Module Mass Analysis

Mass Sensitivity (Large-diameter)

Parameters: End Dome Aspect Ratio
Pressure Bulkhead Orientation



Flatter end domes have dramatically increased masses; the longitudinal bulkhead orientation is heavier and is more sensitive to crew size increases

Reference Concept Mass Analysis

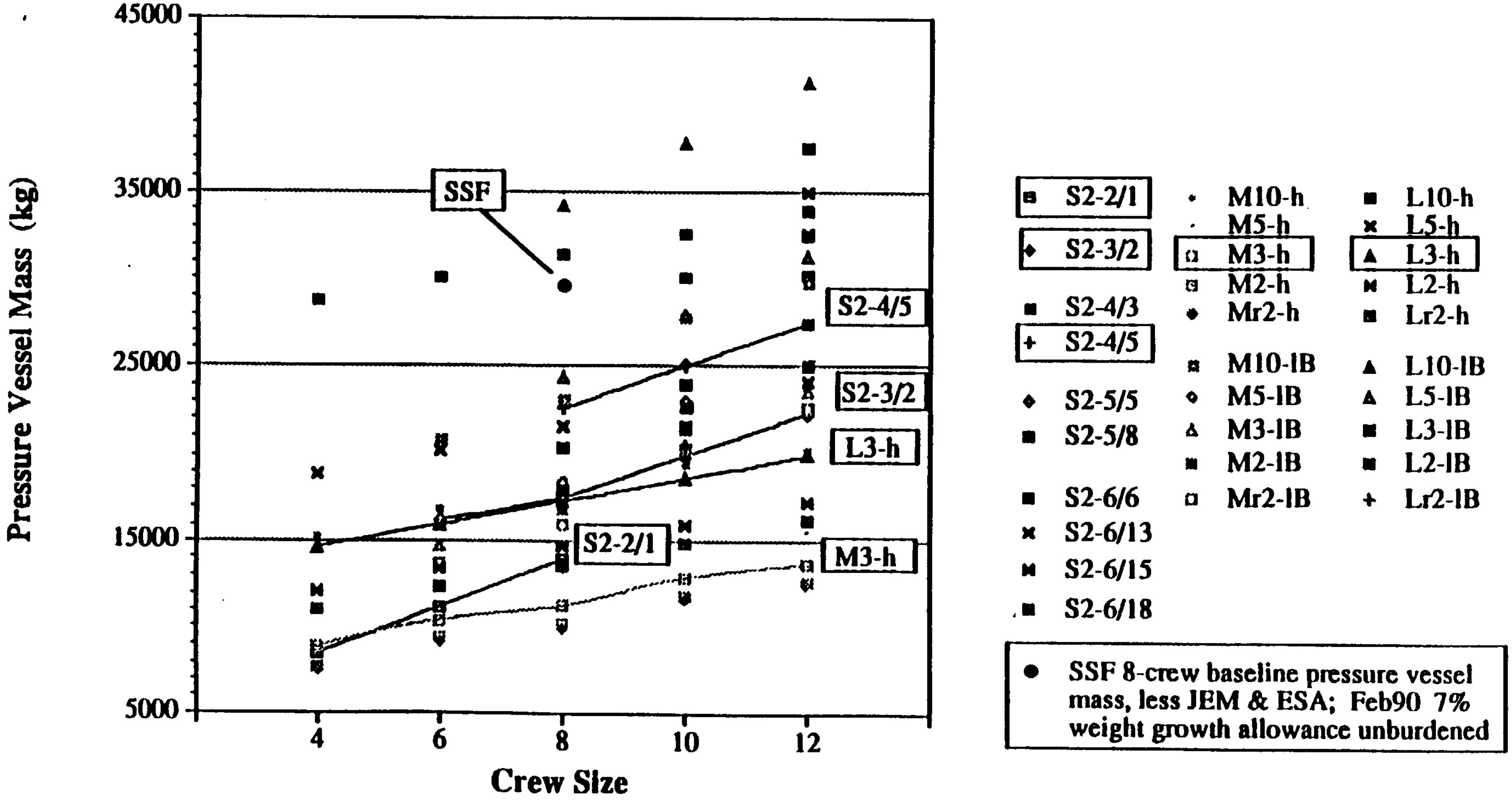
Plotted here is a comparison of reference concepts from all classes of module types. SSF mass is shown for comparison, calculated according to the same assumptions used for the traded module concepts. Although the value does not include the mass of the JEM and ESA modules, it exhibits a quite high weight region; this is primarily due to the heavy end cones of SSF modules, and the particularly mass-expensive topology it baselines (two modules plus four nodes, the rough equivalent of a five-module cluster in our trade study).

The stacked, medium diameter option is seen to trade quite favorably for small crew sizes, and to win handily for larger crew sizes.

Reference Concept Mass Analysis

Mass Sensitivity

Comparison of Reference Concepts



For crew sizes over 6, larger-diameter concepts have an increasing weight advantage over small-diameter cluster concepts

Outfitting Equipment Mass Estimation

An important consideration for larger unitary module concepts is their ability to be outfitted on the ground prior to launch. Orbital integration is a costly operational burden for an exploration architecture.

The next two charts list assumptions and sources used to develop a parametric outfitting mass estimation algorithm for a Mars-class mission.

Outfitting Equipment Mass Estimation

Nomenclature

F ≡ freezer mass
N ≡ number of crew
E ≡ number of ECLSS strings
M ≡ number of equivalent SSF module volumes
A_p ≡ partition area (m^2).
A_f ≡ floor area (m^2)
P ≡ power level (kW)

Power Levels

Crew	kW
4	25
6	30
8	35
10	45
12	50

Freezer Mass

Crew	kg
4	720
6	938
8	1139
10	1323
12	1491

Equipment	Parametric value (kg)	Comments
ECLSS	1909 * E	Derived from SSF mass, 10 % A&I (attachment & integration penalty)
Sample freezers	50 * N	Derived from SSF mass, 10 % A&I
Food freezers	F	Estimated through preliminary design
DMS/comm & A/V	1560 * M	SSF system mass augmented for long duration mission (LDM).
CHC/exercise	400 * N	SSF system mass augmented for LDM.
Science	667 * N	SSF derived mass: 1/2 of equiv. experimental equip. complement, 10 % A&I
Greenhouse	240 * N	Derived from SSF plant growth facility with 20 % A&I
Wardroom/galley/storage	200 * N	SSF derived mass including ovens, washers, etc, 10 % A&I
Personal hygiene	72 * N	SSF derived mass for shower, handwash, and waste mgt. equip
Storm shelter	2000	Shielding required in addition to configured consumables

(continued)

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Outfitting Equipment Mass Estimation (2)

Equipment	Parametric value (kg)	Comments
Windows	30 + 22.5 * N	2 windows (SSF type) + 1.5 windows/crew member
Crew quarters	250 * N	SSF derived mass, augmented for gravity configuration
Partitions	1.43 * Ap	Double thickness SSF derived partition
Chairs	10 * N * M	One chair per crew member per module
Tables	16.7 * N	Two "table places" per crew member total
Floors	13.3 * Af	*Skylab* -derived Al"waffle grid" floors with beam supports
Finishes & miscellaneous	1 * Af	Floor & wall coverings, hardware allowance for access doors
Power dist. and control sys.	17 * P	Scaled from SSF EPDS
Lighting	73 * M	SSF derived mass
External hatches & bulkheads	694	SSF derived mass

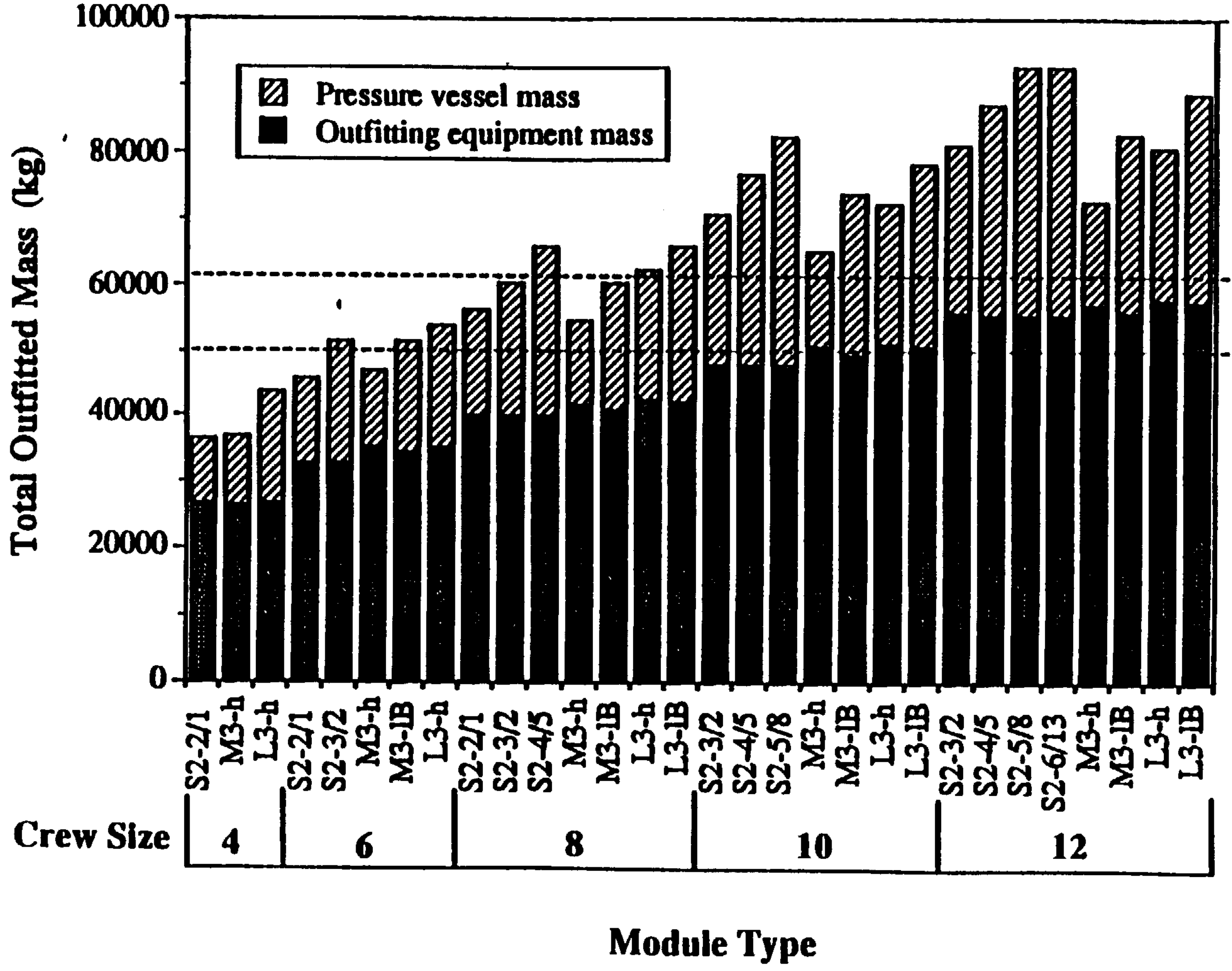
$$\boxed{M_{equip} = 2724 + (1909)E + (1919)N + F + (1633)M + (1.43)A_p + (14.3)A_f + (17)P + (10)NM}$$

Long-duration Module Outfitted Mass

Plotted here are the estimated total masses for the 30 habitat concepts brought through the mass analysis. As shown, these numbers do not include items easily integratable on orbit, but only those subsystems which require attachment, connection, test and checkout. The plot shows that within each crew size range, the equipment mass is roughly constant (equipment for stacked module options tends to be slightly heavier than that for tunnel options). Thus, the pressure vessel mass itself is the interesting discriminator.

Given a reference lunar down-cargo capacity in expendable mode of 50 t, we see that some unitary options for crew sizes of 6 can be landed already integrated on the Moon. With the application of detailed weight-reduction efforts, unitary modules for long-duration crews of up to 8 may be accommodated the same way. Clearly, the small-diameter options can be broken up into smaller pieces than their mass totals indicate, for piecemeal launch, landing and integration. Using and ETO launcher like an HEI Shuttle-C in expendable mode, and applying weight reduction efforts to the module concepts, we can see the possibility of launching a module for 10 crew, fully integrated, into orbit. Such a module could be landed on the Moon with some internal systems removed.

Long-duration Module Outfitted Mass



- Representative HLLV lift capacity
- 7.6m Shuttle-C lift capacity
- Representative lunar expendable down-payload capacity

Outfitting equipment mass includes only internal items which require subsystem integration, utility connection, or structural attachment. Specifically excludes:

- Crew personals
- Food
- Water
- ECLSS consumables
- Equipment spares
- External power system

Medium and large-diameter module options for crews of up to 6 can be launched from Earth, and landed on the Moon, already integrated

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Habitation Module Fabrication Technologies

Critical Requirements	• Thermal / mechanical stability • Radiation resistance • Corrosion and moisture resistance • High specific strength & stiffness • Producibility & inspectability • Damage resistance (toughness) • Vibration damping capability
Technology Options	• Conventional welded structure • Honeycomb core • Metal matrix composites • Organic matrix composites

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Habitation Module Fabrication Technologies

This chart lists the essential requirements for module materials, and the prime options available for advanced M&P application to space habitat manufacture.

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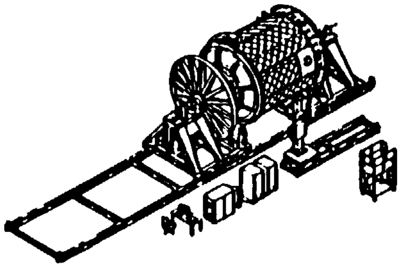
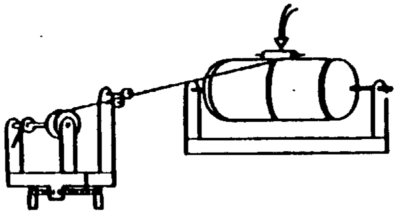
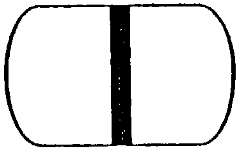
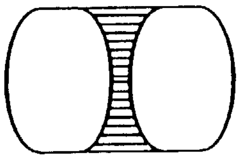
Habitation Module Fabrication Options

Illustrated and elaborated here are the prime candidates for making both pressure hulls, and internal bulkheads for larger diameter modules.

Habitation Module Fabrication Options

STCAEM/jeb/26Feb90

	Options	Suggested Methods of Fabrication
Pressure Vessels	Function: Provide safe habitable volume for crew Assumptions: Near term technologies; external environment exposure; stresses primarily tensile	
	Conventional design - Isogrid - Monocoque	Aluminum Alloy - Welded
	Composite design - Metal matrix - Honeycomb	- Filament wound SiC/ plasma sprayed aluminum, with compaction by hot mandrel - Al face sheet brazed to Al core
Interior Pressure Bulkhead	Function: Provide safe-haven capability in event of hull penetration Assumptions: Near term technologies; internal environment exposure; shear, bending, and tensile stresses	
	Conventional design - Flat panel - Monolithic	Aluminum Alloy - Welded
	Composite design - Concave panel - Honeycomb - Metal matrix face sheet - Organic matrix face sheet	Aluminum Alloy - Brazed or adhesive bonded - Hot pressed SiC / Al, brazed to Al core - Graphite/epoxy layup, adhesive bonded to Al core

Organic Matrix Composites
Metal Matrix Composites

The next two charts survey the features of various fibers and matrices for composite materials that might be considered for habitat construction.

Organic Matrix Composites

Fibers **(filament, fabric, or tape):**	
Graphite	Causes galvanic corrosion of aluminum. High strength & stiffness, poor vibration damping, low cost
Boron	Very high cost, high compressive strength
Kevlar	Limited compressive strength, good vibration damping, good compatibility with epoxy
Glass/quartz	Low cost, good strength, low modulus, low fatigue resistance, poor adhesion to matrix
Organic Matrix Resins:	
Epoxy	Low offgassing, moderate toughness, thermoset processing, low cost, low temp cure
Polyimide	Potential offgassing, good toughness, thermoplastic or thermoset processing, higher cost than epoxies, high temp cure
PEEK	Higher cost than polyimide & epoxy, thermoplastic processing, high toughness, high temperature strength, repairable by heating
Others	TBD

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Metal Matrix Composites

Fibers (filament, fabric, or tape):	
Graphite	Low cost, high strength & stiffness, potential reactivity with matrix alloy
Boron	Very high cost, high compressive strength
SiC	Low cost, compatible with matrix alloys readily processed
Al_2O_3	High cost, potential reactivity with matrix alloy, high temperature stability, lower impact strength than boron
Matrix Alloys:	
Aluminum	Lowest cost, moderate temperature capability, better environmental resistance than Mg.
Magnesium	Moderate cost, combustible, higher temp capability than Al, lower impact resistance than Al.
Beryllium	Very high cost, toxic products, limited supply, favorable thermal properties, low impact resistance
Titanium	High cost, high temperature strength, resistant to corrosion, lower strength:weight ratio than alternatives

Habitation Module Materials Technologies

Summarized and compared here are the synthesized results of our investigation into fabrication technologies and materials options for advanced habitat manufacture. The favored candidate is a composite with SiC-reinforced aluminum matrix. Its combination of desirable structural properties, damage tolerance and environmental inertness indicate the benefit of pursuing technology demonstrations at large scale to generate more data. Its potential for automation could result in cost-efficient production and testing, and its performance advantages over monolithic aluminum would reduce mass and thereby reduce transportation costs as well

④

Habitation Module Materials Technologies

Preliminary

Construction Option	Advantages	Disadvantages
Welded Monolithic Aluminum	Extensive service history Good transverse strength Low cost High damage tolerance Machinability	Highest weight Low specific strength and stiffness
Aluminum Honeycomb Core/ Aluminum Face Sheet	Good in shear & bending Lower weight than monolithic	Low tensile strength High volume penalty Complex design & fab.
Aluminum Honeycomb Core/ Graphite-Epoxy Face Sheet	Good in shear & bending lower weight than Al/Al honeycomb sandwich	Potential corrosion High volume penalty Complex design & fab. low damage tolerance
SiC Reinforced Al Matrix (plasma spray & hot press)	High automation possible Cost comparable to Gr/Ep High strength & stiffness Lower weight than monolithic Tailorable thermal properties Good damage tolerance Good environmental resistance	Limited data Intricate process Technology demonstration at large diameters
Graphite Reinforced Epoxy Matrix	High strength & stiffness Moderate cost Tailorable thermal properties Lowest weight	Sensitive to environment (radiation, temp, etc.) Low damage tolerance Requires metallic vapor barrier

Module Concept Selection

The next two charts record our final, preferred module concept and justify the choice according to the four discriminator categories outlined at the beginning of the trade study. The concept selected for further use in the STCAEM Study is not, *per se*, one of the candidates carried through the trade study, but rather represents a combination of the best features of all the leading candidates.

Several mass-reduction decisions have been incorporated in this new reference concept. First, the medium diameter module was the mass winner. It could clearly fit early HEI ETO launchers, and could either use welded-metal technology or drive more advanced, weight-saving M&P options. The bulkhead is turned the "light" way, crosscut through the module amidships, although the module itself is turned tunnel-oriented for use in gravity fields. The upper floor is located at the module diameter (an average of floor options "B" and "C"), introducing the possibility of using it as a diametral tension tie for further vessel mass reduction (commercial airplanes use this technique). The end dome aspect ratio is 2.

The concept enjoys potential for extensive commonality across exploration architectures, both for spacecraft and surface base applications.

The only perceptual reservation about this concept is that it consistently traded poorly for intrinsic pathway boredom and spatial unit option variety. This means that for long-duration missions, the interior outfitting configuration must compensate carefully, to mitigate perceptions of a severely limited habitable domain.

(continued)

Module Concept Selection

Selection

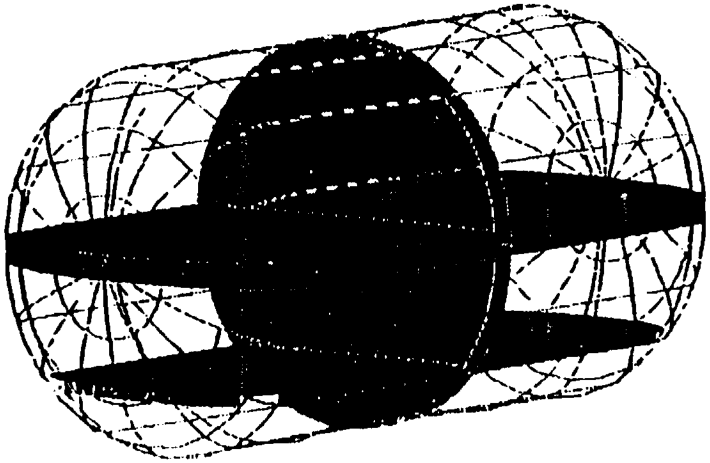
Modified Mg2-1 concept family selected for further reference use in the STCAEM study, for

- **concept development activities**
- **trade & sensitivity analyses**
- **more detailed habitat system definition**

Major Features

- **7.6 m diameter**
- **2:1 aspect ratio, unpenetrated end domes**
- **Cross - section, bisecting bulkhead**
- **Diametral tension - tie, deep floor**
- **Extensive commonality across architecture: g-field optimized**

6 Crew Configuration



Module Concept Selection

The proportions of this module type do not approximate that of SSF modules until crew sizes of about 12 are reached. Beyond that point, it is useful to think of clustering these 7.6m-diameter modules together in simple topologies to extend the habitable domain, for surface bases as well as for large-crew in-space transportation systems.

Finally, it is important to remember that the nature of the trade study has led us to generate a quite conservative habitat concept, which although it combines features demonstrated to be advantageous, still reflects a rather limiting set of assumptions. As a next step, concepts should be considered which combine this reference module type with the smaller diameter module types which we still see as widely applicable throughout all phases of the HEI. For advanced applications, clusters which mix module types and sizes promise good accommodation of functional requirements as well as interesting and stimulating psychological environments.

Module Concept Selection (2)

Functionality

- Unitary vessel minimizes leakage, parts count
- Permits wide variety of internal outfitting designs
- Diametral floor maximizes nominal floor area, facilitates weight-reducing tension tie
- Compact domain, good for access-time safety
- Best overall multi-floor efficiency in g-condition for a range of crew sizes
- Less wall area than smaller diameter; outfitting can compensate

Integration

- Minimizes orbital assembly operations required
- 7.6 m launch shroud likely available for early HEI
- Large crews can be accommdated through simple clustering
- Compact habitat facilitates aerobrake integration

Perception

- Survey results show technical people perceive larger diameter concepts as more spacious
- Barrel vault proportionately invariant with crew (module) size, better than dome
- Module width has better plan aspect ratio than smaller diameters
- Low intrinsic number of unique spatial units; outfitting can compensate
- Lowest score for circulation option boredom over long duration

Cost

- Lightest weight (transportation cost critical for exploration vehicles)
- Welded-metal technology feasible here, well-understood
- Prime opportunity for M&P improvements, however
- End dome complication less than for 10 m size
- Commonality in growth architectures more appropriate for surface system applications

Selected Module Concept
Estimated Total Mass

Listed are the estimated total masses, by crew size, for the selected module concept. These now include the pressure vessel and associated mass, the outfitting equipment mass, and the crew and provisions needed to make the module complete for a mission.

<u>Not</u> included are the power production system, airlocks, and airlock consumables (hatches and pressure bulkheads specifically associated with airlocks <u>are</u> included). For the purpose of this study these are regarded as external equipment.

Selected Module Concept

Estimated Total Mass

Crew Size	Mass (kg)
4 Crew	**44,260**
Pressure Vessel (module, bulkhead, hatches, and ECLSS equipment)	8,849
Crew Consumables & Spares (food, water, crew & effects, ECLSS consumables, & spares)	10,700
Outfitting Equipment Mass (per formula)	24,711
6 Crew	**59,256**
Pressure Vessel	10,746
Crew Consumables & Spares	16,050
Outfitting Equipment Mass	32,460
8 Crew	**72,089**
Pressure Vessel	11,694
Crew Consumables & Spares	21,400
Outfitting Equipment Mass	38,995
10 Crew	**87,456**
Pressure Vessel	13,591
Crew Consumables & Spares	26,750
Outfitting Equipment Mass	47,115
12 Crew	**99,880**
Pressure Vessel	14,538
Crew Consumables & Spares	32,100
Outfitting Equipment Mass	53,242

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Large Crew Size Impact Assessment

From the time of von Braun's *Das Marsprojekt* through the *90 Day Study*, acceptable crew sizes for Mars-class missions decreased from 70 to 4. That shift can be credited partly to our sobering experience with the true complexity of advanced space exploration; partly to advances in robotic science and the automation of formerly human jobs; partly to attention to real, modern space budgets; and partly to implicit changes in our conception of what the exploration of planets is all about. Nonetheless, STCAEM concluded that 4 is *too* small to be practical or safe. Twice that number, or 8, provides a realistic, minimum skill mix. Our long-duration habitat trade study generated a singular module concept capable of supporting 12, more if clustered together. Twice 8, or 16, is a number about right for a bevy of really specialized scientists to investigate Mars during conjunction-class surface stays. And twice that, or 32, approaches a crew size range appropriate for transporting settlers to the red planet. Since we do not yet know what visions will come to guide SEI as it grows, we need to apply our modern understanding of Mars mission technologies, consolidated so far in the STCAEM reference in-space transportation concepts keyed to just 4 crew, to larger crew sizes.

We performed an evolutionary Large Crew Size impact assessment for crew sizes of 8, 16 and 32, looking at Mars mission masses and vehicle strategies for all five prime propulsion candidates, aerobraking constraints, habitat system clustering and staging implications, vehicle configuration impacts, and life support strategies. As expected, advanced propulsion has high mass-limiting leverage for Mars missions, as does the use of conjunction profiles. Both CAP and CAB (the latter flying opposition profiles, of course) are not cost-effective for the large payload masses required by large crew sizes. Because SEP power level scales linearly with area, that option appears better suited to flotilla approaches than "large-vehicle" approaches, although this is sensitive to trip time requirements. NEP scales very well, as does NTR. Aerobraking was found to be theoretically feasible at Mars for vehicles of order 64 times heavier than our reference vehicles, assuming similar geometries; therefore, limitations on aerobraked vehicle size are *intrinsic*, having to do with assembly and trim during flight rather than atmosphere properties. Beyond a certain crew size, (assumed to be between 16 and 32 in this assessment), the use of RMEVs pays off. In settlement scenarios where 28 of 32 crew are left at Mars, staging a modular transfer habitat system both reduces return payload *and* leaves useful habitats at Mars. Clustering large habitat modules together highlights a trade

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between habitability (gravity-living) and safe-haven splitting (losing too large a percentage of habitable volume through "keystone" module loss). STCAEM vehicle archetypes were configured to be able to accommodate clustered habitat modules as well as multiple landers, even in artificial gravity modes. Like artificial gravity, if large crew sizes become a requirement, mission designs can be found using the concept vocabulary developed by STCAEM to handle them.

Interplanetary Habitation

- *Human* factor increasingly recognized as the tallest pole

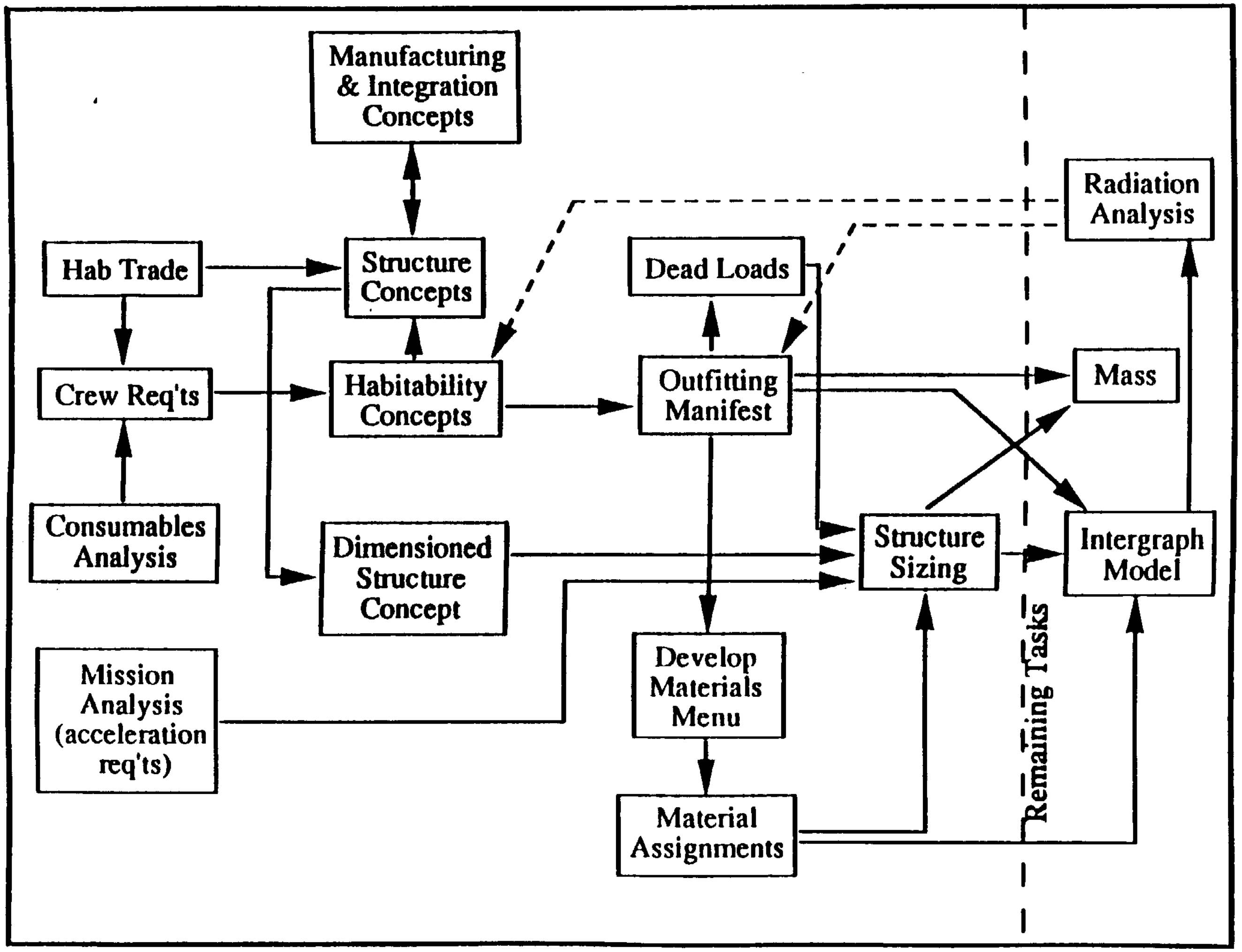
- Among spacecraft systems, habitation systems tend to have:
    - the highest relative cost
    - the greatest public interest and visibility
    - the highest leverage for human performance

- Habitation systems for SEI should therefore acceed to the highest possible standards of accommodation:
    - safety
    - utility
    - comfort

MTV Hab Module Internal Definition

Shown opposite are a list of assumptions made in order to derive design criteria for the internal layout of the MTV hab module. The diagram at left describes the design process.

MTV Hab Module Internal Definition



Manufacturing & Integration Concepts
Hab Trade
Structure Concepts
Crew Req'ts
Habitability Concepts
Consumables Analysis
Dimensioned Structure Concept
Mission Analysis (acceleration req'ts)
Dead Loads
Outfitting Manifest
Develop Materials Menu
Material Assignments
Structure Sizing
Radiation Analysis
Mass
Intergraph Model
Remaining Tasks

Process

Assumptions

- 6 Crew
- 1000 day duration
- Module concept from STCAEM Hab Trade
- Layout referenced to 1g conditions

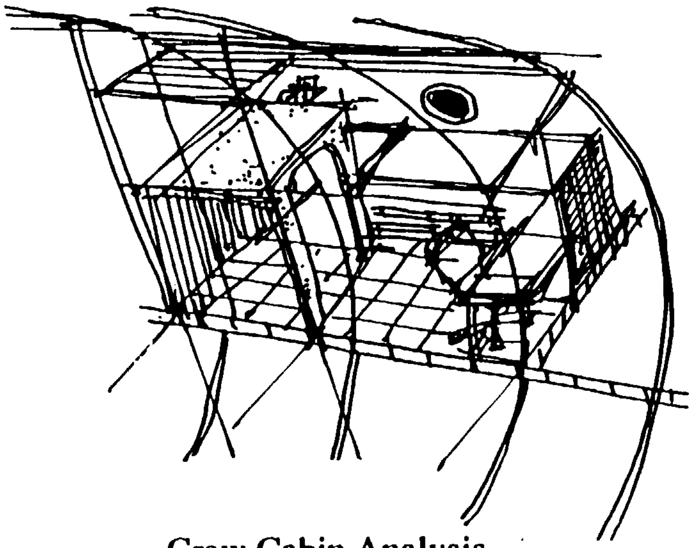
Functional Allocation Analysis

Long duration habitat design draws upon NASA standards, *Space Station Freedom* allocations, and area / volume requirements for terrestrial analogs (architectural design standards). "Grassroots" spatial analysis combines these three sources into preliminary sketch form to determine placement and orientation of elements, and to obtain a visual "feel" for the space.

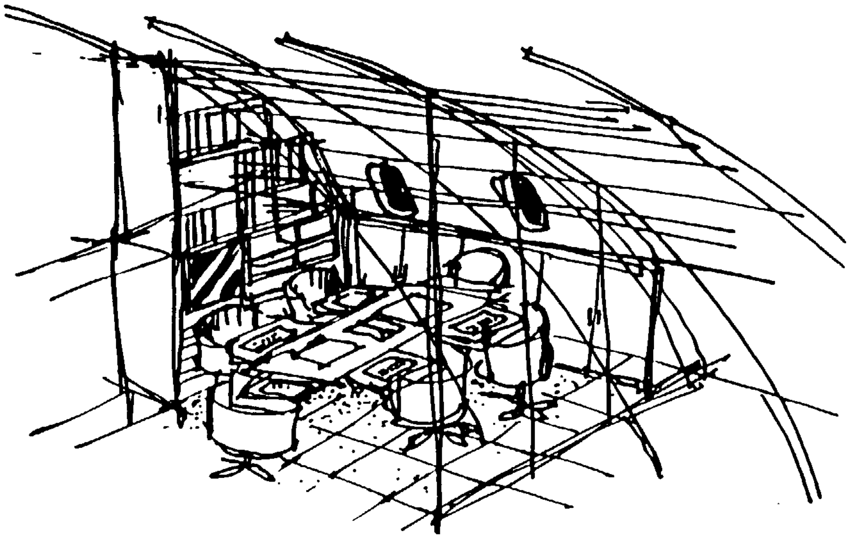
Functional Allocation Analysis

Sources

- NASA STD-3000
- SSF allocations
- "Grassroots analysis"
- Terrestrial analogs



Crew Cabin Analysis



Wardroom Study

MTV Habitation Functional Analysis

The relationship between functions within the hab module, and the area / volume requirements given to each are described below through the use of a scaled proximity "bubble" diagram. Functional areas are shown in close proximity, or are distant to each other, based on crew activities and operations.

MTV Habitation Functional Analysis

Area / Volume Allocations *

Function	Area (m^2)	Volume (m^3)
6 Crew Quarters	31.2	71.1
Laundry Facility	1.5	3.4
2 Hygiene/ Waste Management	4.7	10.7
Crew Health Care	6.5	14.8
Science Equipment	14.0	31.9
EVA Equipment	8.0	18.2
4 Workstations	12.0	27.4
Recreation/ Exercise	20.0	45.6
Operations	6.0	13.7
Wardroom	18.0	41.0
Galley	18.5	42.1
Greenhouse	26.0	59.3
Circulation	32.0	73.0
	198.4 m^2	452.2 m^3

* Analysis presumes 1g conditions.

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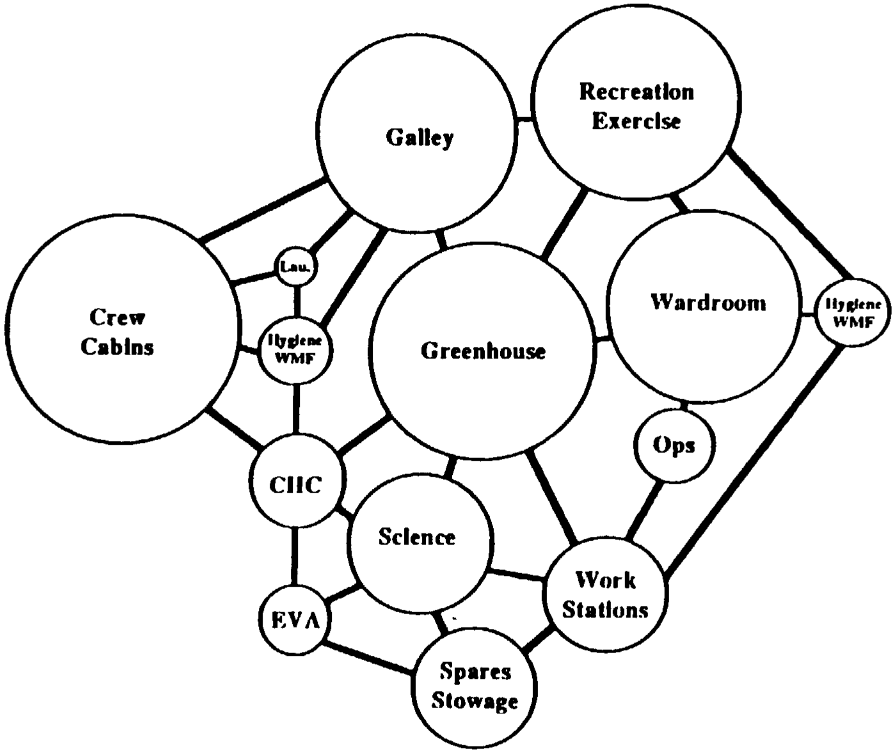

Scaled Proximity Diagram

Internal Structure Concept Development

Shown below are several sketches that trace the conceptual development of the internal supporting structure for the MTV habitat module. These concepts originated from a desire to utilize the bisecting internal bulkhead as a central load bearing member, from which to cantilever a deep framework of small, lightweight structural members. The sketches at lower left indicate a framework joined with moment resisting connections, and the sketches at upper left illustrate a concept using a combination of diagonal struts and a moment resisting frame to form a twin "spine". The drawing at right shows a hybrid of the two concepts.

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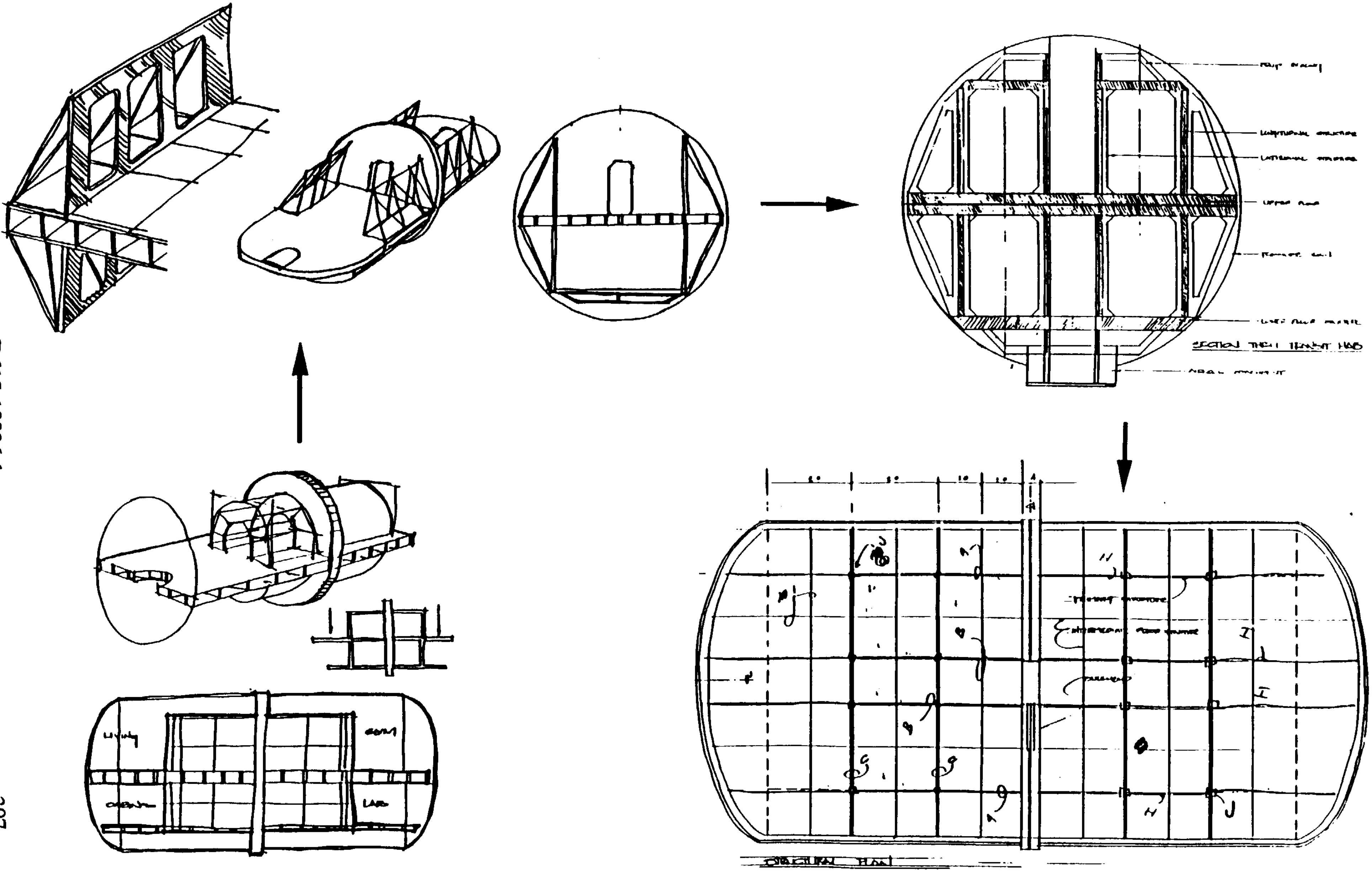
Internal Structure Concept Development

BOEING



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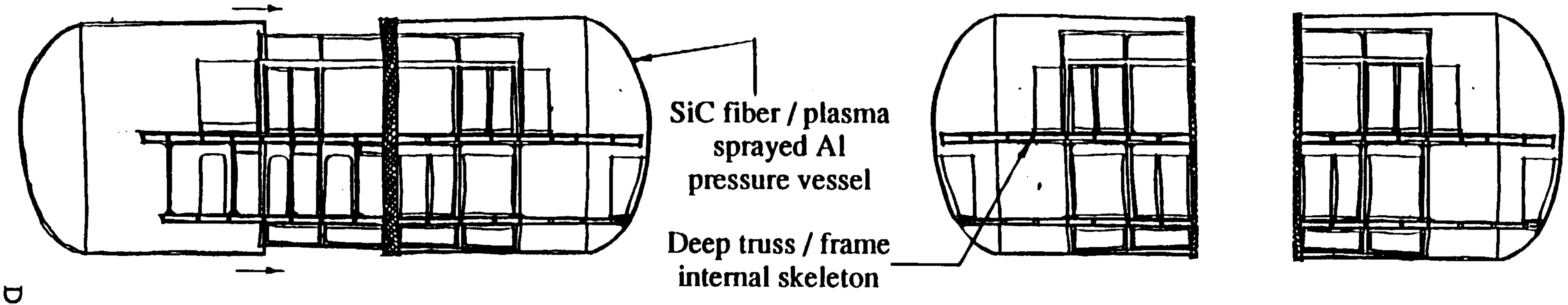
Large Habitat Manufacturing Option Evolution

Two separate methods for manafacturing the habitat module are described below.

Large Habitat Manufacturing Option Evolution

① Build internal and external structure separately

② Build internal and external structure together



- Integrate internal subsystems in open condition
- "Slip" skin over internal frame and weld to bulkhead
- Allows easy integration

- Integrate internal subsystems through bulkhead hatches
- Connect "halves" at bulkhead
- Allows specialization, growth options
- Allows winding bulkhead as part of external structure

- **Is the "twin-bulkhead" heavier than the separate, minimum guage pressure skin option?**
- **Can the skin be successfully fabricated via winding ?**
- **Can the skin be attached to the internal structure as well as the bulkhead ?**

MTV Habitat Module Layout (1)

Shown below is the internal layout for the MTV hab module upper deck, showing the living areas and galley to the left of the bulkhead, and recreation / health functions to the right.

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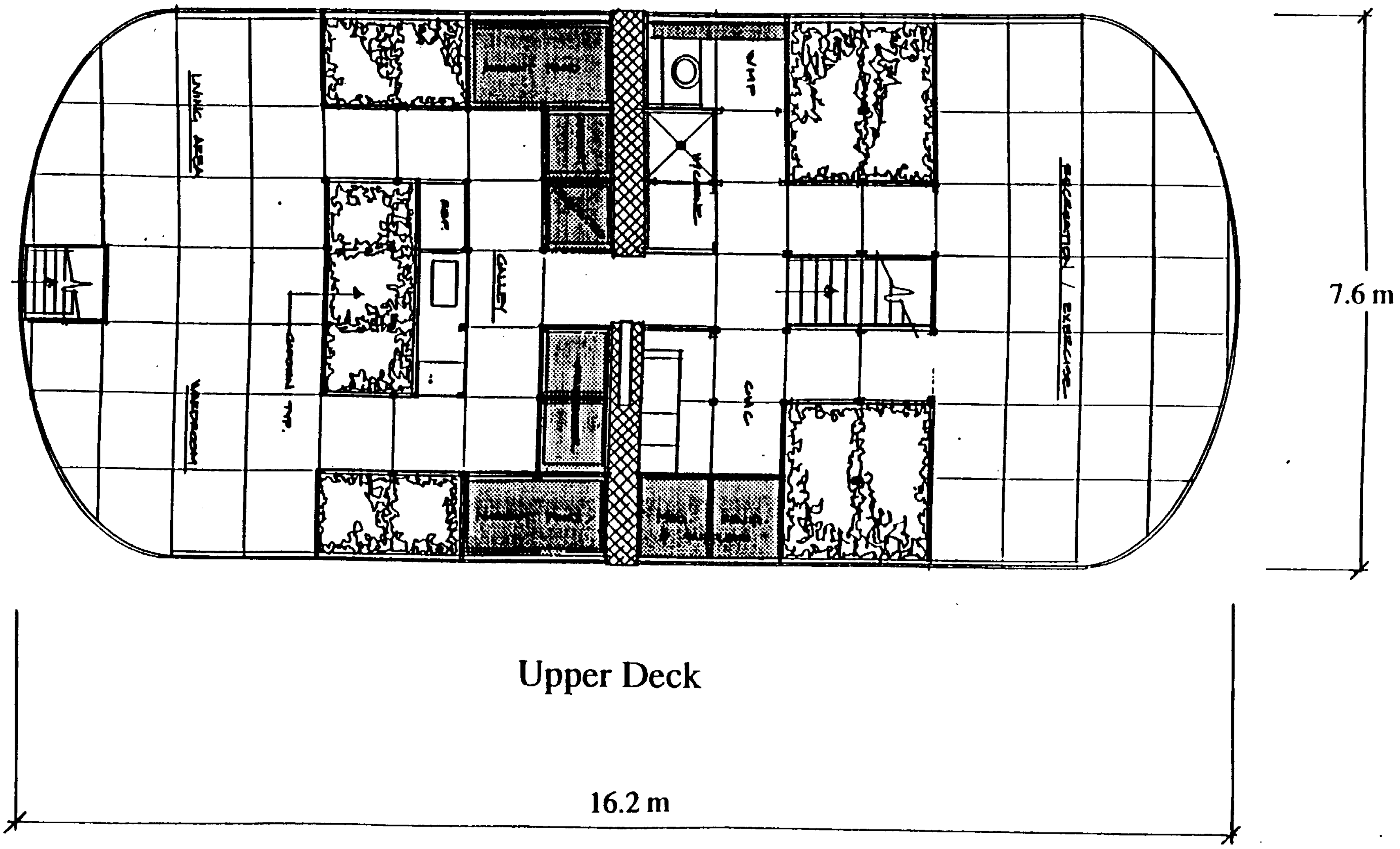
SPACE SYSTEMS

MTV Habitat Module Layout (1)

BOEING



Upper Deck

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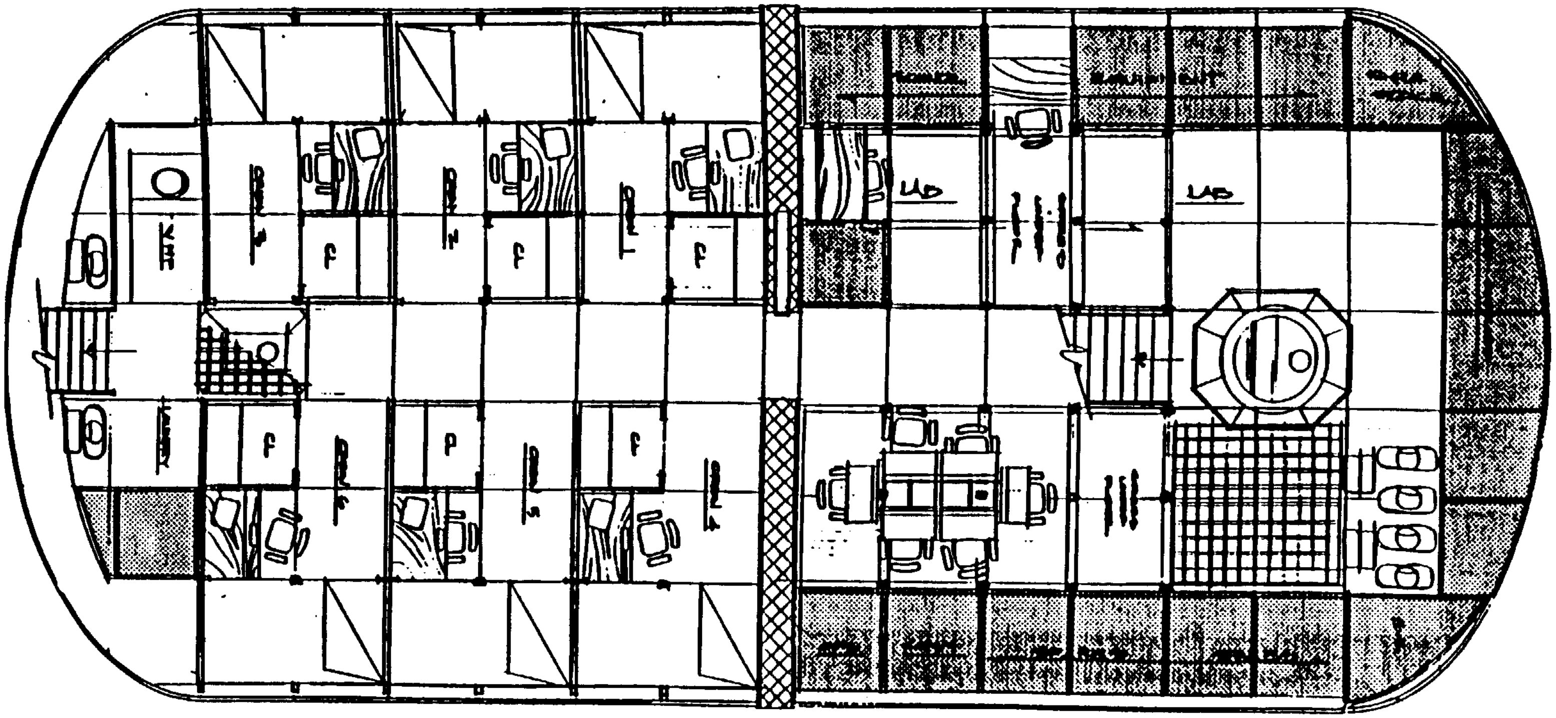
MTV Habitat Module Layout (2)

Shown on the facing page is the lower deck area of the MTV habitat module for 6 crew. To the left of the bulkhead are crew quarters and waste management, and to the right are the lab and work areas.

MTV Habitat Module Layout (2)



Lower Deck

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MTV Radiation Analysis Materials Menu

Shown below are reference hab module materials and their function. Densities of each material were calculated for use in radiation prediction analyses.

MTV Radiation Analysis Materials Menu

Application	Reference Material	Density (g/cc)
Pressure Shell	SiC/Al	2.85
Interior Insulation	Glass Batts	.035
Exterior Quartz Window	Double Wall Quartz	2.20
Interior Acrylic Window	Single Wall Acrylic	1.18
Internal Structure	Graphite/Epoxy Composite	1.60
Flooring / Walls	Graphite/Epoxy Composite	1.60
Padded Furniture	Composite; Steel, Foam, Al.	.140
Nylon Carpeting	Nylon, 60% dense	.780
Electronics	Composites, Cu, Al, plastics	.692
Electronics spares	Composites, Cu, Al, plastics	.725
ECLSS and Spares	Al, stainless steel, plastics	.466
Science Equip. and Spares	Al, stainless steel, plastics	.466
Consumables	Frozen Food	.76
	Ambient Food	.68
	Dehydrated Food	.45

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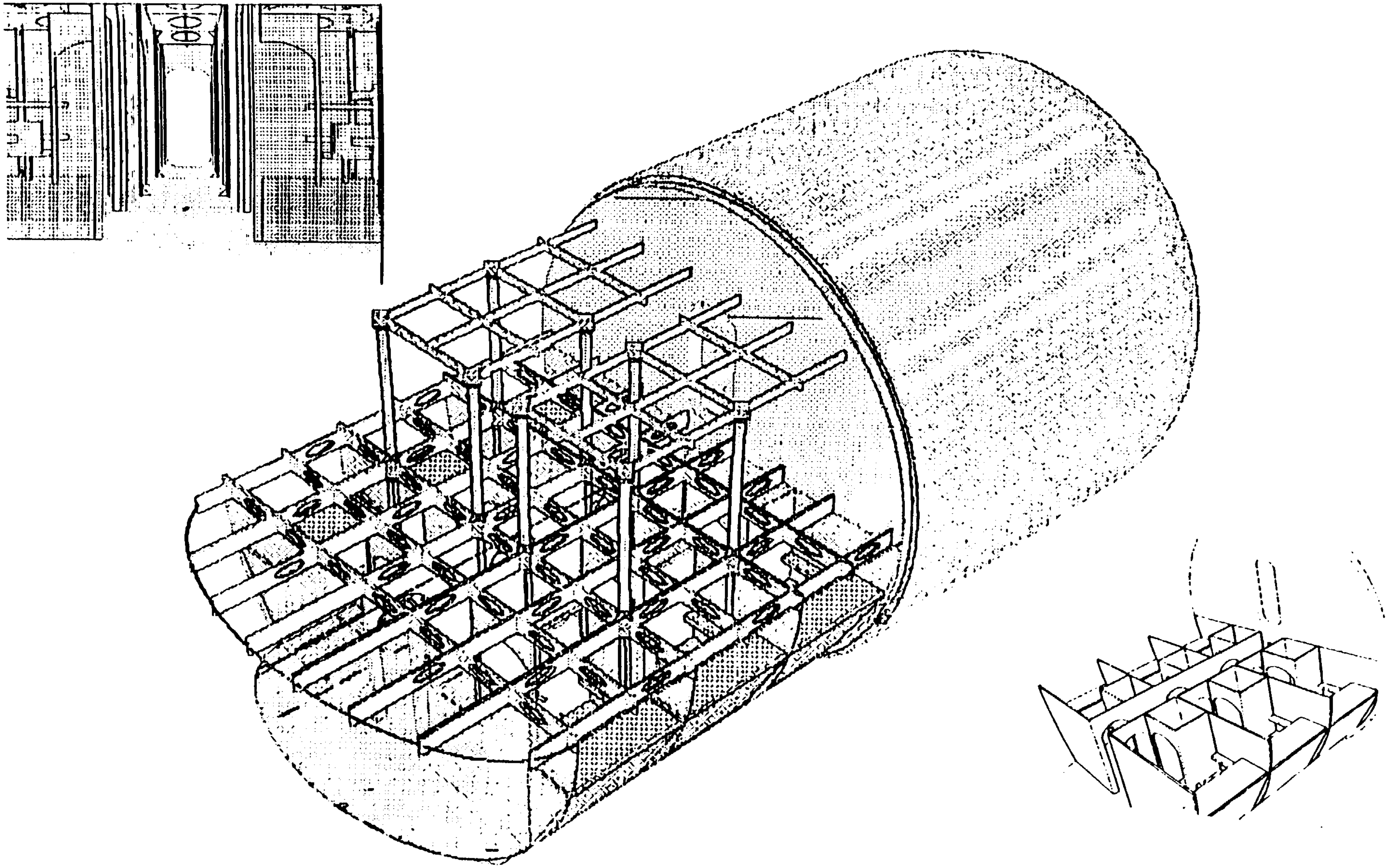
CADD Views of Module

Shown on the opposite page are CADD model views of the MTV habitat pressure vessel and the internal structure and equipment.

MTV Hab CADD Model



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Large Crew Size Impact Assessment

- Mars mission crew size during the *90 Day Study* went from 5 to 4
- STCAEM later concluded *minimum* Mars crew size is 6-7
- STCAEM Long-duration Habitat Trade Study examined crew sizes from 4-12
- STCAEM reference large-scale (settlement) program uses crew sizes of 18
- Von Braun considered Mars crew sizes as large as 70

What are the major impacts to modern vehicle concepts of requiring large crew sizes?

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LCS Chief Questions

- What mission strategy tendencies emerge from propulsion constraints (i.e. how does each system "want" to fly)?
- What are the mission mass impacts (IMLEO & resupply)?
- What are size limitations on Mars aerobraking vehicles?
- What are the results of clustering large habitat modules?
- What vehicle configuration requirements are generated?
- What life support strategies are appropriate?

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LCS Starting Assumptions

- Three Factors of 2 beyond *90 Day Study:*

  8 crew (roughly STCAEM minimum)

  16 crew (roughly "full science")

  32 crew (useful for "settlement")

- CAP, CAB, NTR, NEP, SEP archetypes

- Each propulsion method matched to most appropriate flight mode

  ("playing field" not fictitiously flat)

- Both large vehicles and flotillas permissible

- Cargo (except for crew consumables/spares) missions not considered

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LCS Input Data

Propulsion Method / Crew Size	CAP ②	CAB ③	NTR ②	NEP ②	SEP ②
8	Hab: 75.6	Hab: 66.0	Hab: 75.6	Hab: 75.6	Hab: 75.6
	2 MEVs: 146.2 (4)	2 MEVs: 168.8	2 MEVs:146.2	2 MEVs: 146.2	2 MEVs: 146.2
16	Hab: 151.2	Hab: 131.9	Hab: 151.2	Hab: 151.2	Hab: 151.2
	4 MEVs: 292.5	4 MEVs: 337.6	4 MEVs: 292.5	4 MEVs: 292.5	4 MEVs: 292.5
32 ⑤			Hab: 280.8 out 278.5 in 1 RMEV+3 down refills: 153.2 ④	Hab: 280.8 out 278.5 in 1 RMEV+3 down refills: 153.2	Hab: 280.8 out 278.5 in 1 RMEV+3 down refills: 153.2

1. All masses in tonnes
2. Conjunction low energy transfer (1020d)
3. 2016 opposition with Venus swingby (434d); hab mass reflects duration; MEV mass reflects aerobrake sized for capture
4. MEVs assumed standard 4-crew version; RMEVs assured used ≥ 4 times
5. 32-crew missions leave 28 crew at Mars (settlement)
6. 8 crew: 8-crew habs
7. 16 crew: 28-crew habs
8. 32 crew: 310-crew habs

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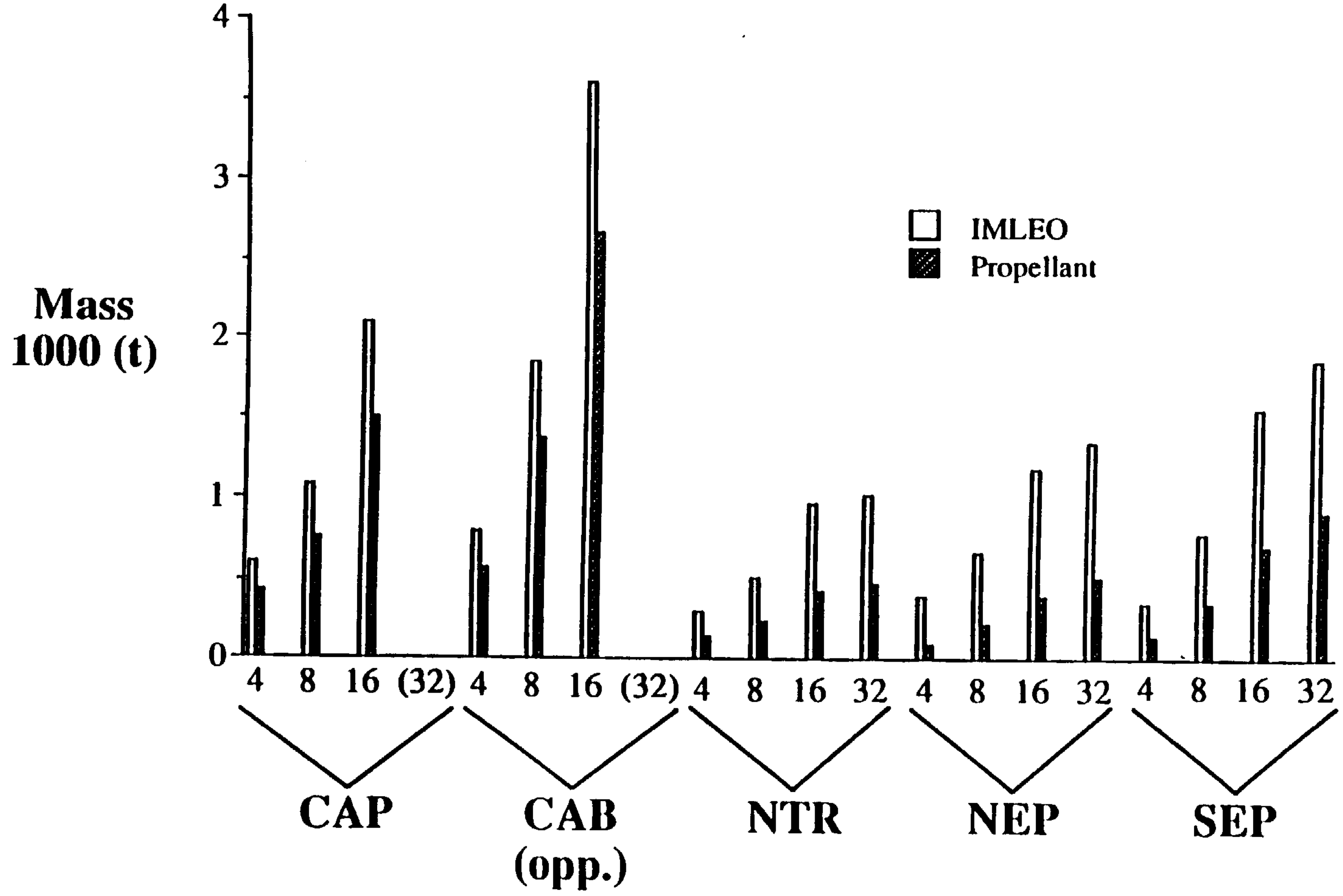
Large Crew Size Mass Results

Crew sizes for the reference vehicle concepts, and their associated IMLEO and resupply mass, are shown below. The 32 crew vehicle mass for the chemical options are not shown due to their extremely large size, and masses shown for the chemical aerobraking option, are for opposition missions only due to the extreme size of conjunction class vehicles for any size crew. The results indicate that advanced propulsion offers significant advantages for larger crew sizes.

Large Crew Size Mass Results



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Large Crew Size Mass Impacts

This chart shows the actual IMLEO and propellant mass for each case shown on the previous graph. Also listed are the parameters and assumptions used to generate the masses shown.

C-4

Large Crew Size Mass Impacts

Crew Size		CAP	CAB	NTR	NEP②	SEP③
	4	603 (IMLEO)	801	302	398	355
		434 (Propellant)	576	139	94	139
	8	1,078	1,840	513	663	773
		768	1,365	232	219	346
	16	2,103	3,596 ①	964	1,175	1,546
		1,491	2,661	435	397	691
	32	-----	-----	1,008	1,330	1,892
		-----	-----	483	505	898

① Sending down-cargo (MEVs) on unpiloted, low energy conjunction profile in a "split-opposition" mode saves only 101 t (3%)

	Crew	Power (MWe)	α (kg/kWe)	Trip Time (d)
② NEP assumptions:	4	25.0	7.5	335
	8	29.8	7.5	460
	16	51.6	6.5	460
	32	65.2	6.0	460
③ SEP assumptions:	4	10.0	9.0	430
	8	41.4	5.0	460
	16	82.7	5.0	460
	32	112.0	5.0	460

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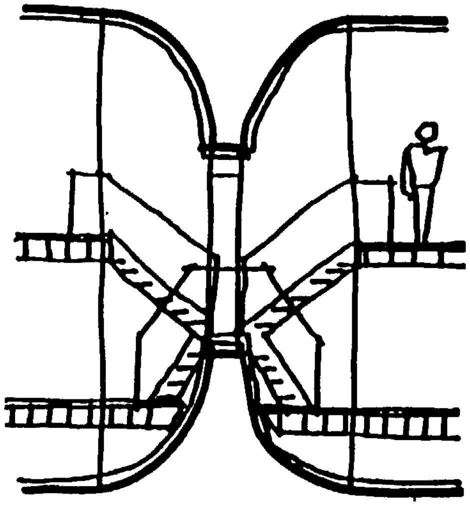
Large Module Clustering Issues

The sketches below were done to identify and illustrate some of the issues involved in grouping large hab modules for large crew sizes. Artificial gravity and / or surface application were assumed.

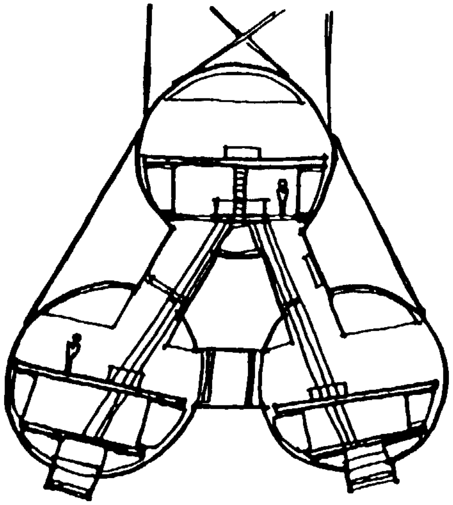
Large Module Clustering Issues



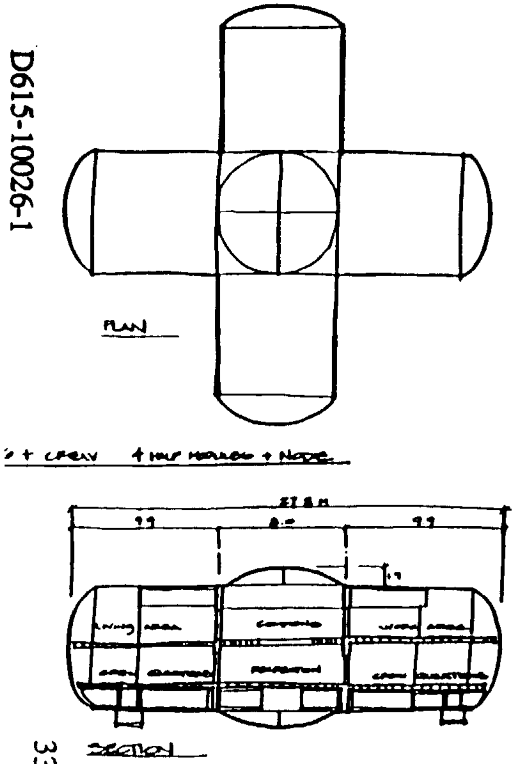
Module to module coupling

Issues

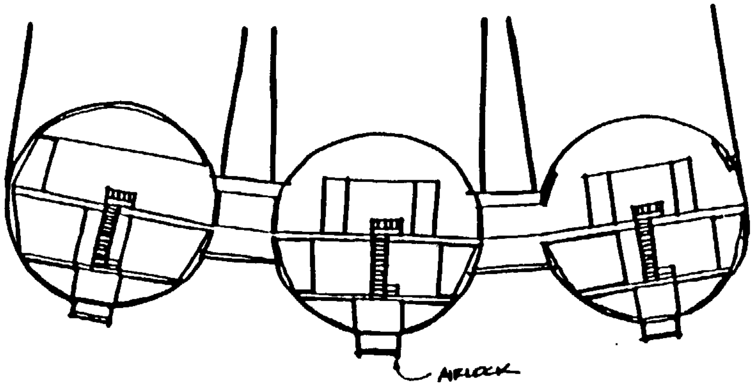
- Vertical circulation
- Planet surface commonality
- Integration with transfer vehicle
- Safe haven capability



32-crew artificial
gravity arrangements



Growth and
surface
application



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Large Crew Size Integration Results (1)

Mission Design Strategy For Large Crew Sizes

- Opposition profiles problematic
- Advanced propulsion strongly indicated
- EP specific mass can vary with vehicle size for NEP but not for SEP
- SEP α exceeds reasonably achievable technology with competitive trip time
- Large-vehicle approach favored for NEP
- Flotilla approach favored for SEP
- Either approach for high-thrust systems (driven by redundancy strategy, vehicle processing complexity)

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Large Crew Size Integration Results (2)

Aerobraking

- m/CdA ≈ 2000 kg/m^2 (4 times higher than baseline) possible at Mars
- Vehicles of order 4500 t could be braked at Mars using reference

  L/D ≈ 0.5 proportions

  (Size limit higher for L/D shapes)
- Limits to size of aerobraked vehicles are therefore *intrinsic:*
  - Ability to assemble, launch, process
  - Propulsive budget for flight control

Vertical Integration

- Artificial gravity configurations require "swinging room"

  for clustered hab modules and multiple MEVs
- STCAEM archetypes per-adapted for this growth

,STCAEM/crf/ 9Jan91

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Large Crew Size Integration Results (3)

Life Support

- Most equipment expected to be modular-scalable
- Large "buffer" size for consolidated approaches, potential safety enhancement
- Use of plants as P-C backup, with ancillary food production, is interesting approach

Modules Clustering

- Multiple modules have no need for bisecting bulkheads
- Side-by-side favored for gravity configurations
- Single-floor or multi-floor connections?
- Side-by side poor for "safe-haven split" contingency
- "Raft" arrangements for > 4 modules may be appropriate

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Reducing exposure and the protection of crew members from ionizing radiation will be a key issue confronting mission planners and vehicle designers involved in the Human Exploration Initiative. Understanding that radiation exposure to astronauts in space may be controlled, but never completely eliminated, the National Council on Radiation Protection and Measurements (NCRPM) has recommended both career and annual exposure limits for NASA to use in planning manned missions.

Radiation protection requirements initially set in the September 8, 1989, Level II document, "Human Exploration Study Requirements", were unchanged in a subsequent document dated March 14, 1990.

The radiation summary contained within this section includes basic units and terms used to describe human responses to ionizing radiation. Short term and career limits for the protection against nonstochastic effects recommended by the NCRP have also been included. The limits recommended by the the NCRP for astronauts are in direct contrast to those established for high risk occupations on Earth. The currently used risk assessment system relies on a quality factor, Q, which normalizes all forms of radiation to the same biological effectiveness. The quality factors have been established by the International Commission on Radiation Protection (ICRP). Quality factors in effect give an indication of how much biological damage will occur for various types of radiation as it traverses tissue and gives up energy. The amount of energy that is released by a charged particle as it passes through a medium is called linear energy transfer (LET). The majority of the data obtained on radiation effects to man are for low LET types of radiation. The current risk assessment system is being challenged as of late. Initial studies indicate that assigned quality factors may be far too excessive and in fact may cause solution over-engineering. For example, very little data exists regarding very high energy particles found in space. It is currently impossible to duplicate these high energies in a laboratory to determine their effects on man. For this reason we arbitrarily assign a quality factor of 20 to such particles. In fact, all particles that we have little information on are dropped into this same 'bin'. Research is progressing in the development of a new risk assessment method.

Radiation with energy levels in excess of 30 MeV are generally considered harmful to biological systems such as man. The natural radiation encountered by astronauts may be differentiated by its source and includes magnetically trapped radiation, galactic cosmic radiation (GCR), and solar proton event emissions (SPE). The Earth's magnetic field provides the mechanism for trapping and deflecting charged particles. Commonly referred to as the Van Allen belts, these somewhat overlapping and loosely defined inner and outer bands contain captured protons and electrons. The major contribution to crew exposure in Low Earth Orbit (LEO) will come from trapped protons, in fact, roughly 90% of the incurred dose. The primary portion of the proton dose will occur during passage through the South Atlantic Anomaly (SAA). Charged particles that would be normally trapped at higher altitudes are brought to lower altitudes over this region. A vehicle orbiting at 28.5° inclination and roughly 450 km altitude will traverse this region an average of six times in a twenty four hour period. At an altitude of roughly 2000 km, where the peak density of trapped protons occurs, exposure rates can get as high as 1000 rem/hr. Even though the Earth's magnetic field traps radiation in this way, it also protects the crew members from other forms of ionizing radiation. When astronauts leave the relative protection of the Earth's magnetic field it is practically impossible to specify the proton environment due to the unpredictable nature of solar proton events. Large solar proton events have the potential of exposing the crew to monumental doses in very short periods of time unless they have been adequately warned and protected. The third type of space radiation, GCR, arrives omnidirectionally. Major sources of GCR are found far beyond the reaches of the solar system and include distant stars and galaxies. Protons make up the primary portion of the GCR components. The heavy ion component amounts to less than 1% of the total composition but accounts for the largest dose contribution. Because of the high energies associated with the GCR it is far more penetrating than other forms of radiation and much more difficult to shield against.

The radiation environments that will be encountered are variable both temporally and spatially. This variability occurs for a number of reasons including fluctuations in solar activity associated with the solar cycle, altitude, inclination, and longitude in LEO, planetary mass, and general anisotropies existing in a particular radiation field.

It will be necessary to employ radiation countermeasures to reduce the exposure of possible radiation effects to crew members. Several methods continue to be investigated. The use of an induced magnetic field, a "mini-geomagnetic" field, has been suggested. However, serious questions regarding crew exposure to such concentrated fields remain. Studies of chemical inhibitors to suppress the effects of ionizing radiation are continuing with positive results, especially by the Department of Defense. NASA follows closely the method of ALARA - As Low As Reasonably Achievable - to limit unnecessary exposure to astronauts. Thus, crew members do not perform planned EVA's during passage through the SAA. In the past inherent shielding provided by the spacecraft structure and equipment has been adequate to protect crew members. However, future programs, such as manned Mars and Lunar missions, must rely on effective strategic placement of all forms of inert mass, from consumables to equipment, to provided added protection. This protection method is known as bulk shielding and allows spreading the burden between various subsystems to provide protection and reduce exceeding weight constraints. New and innovative methods and materials for shielding will be a critical technology issue in providing radiation protection.

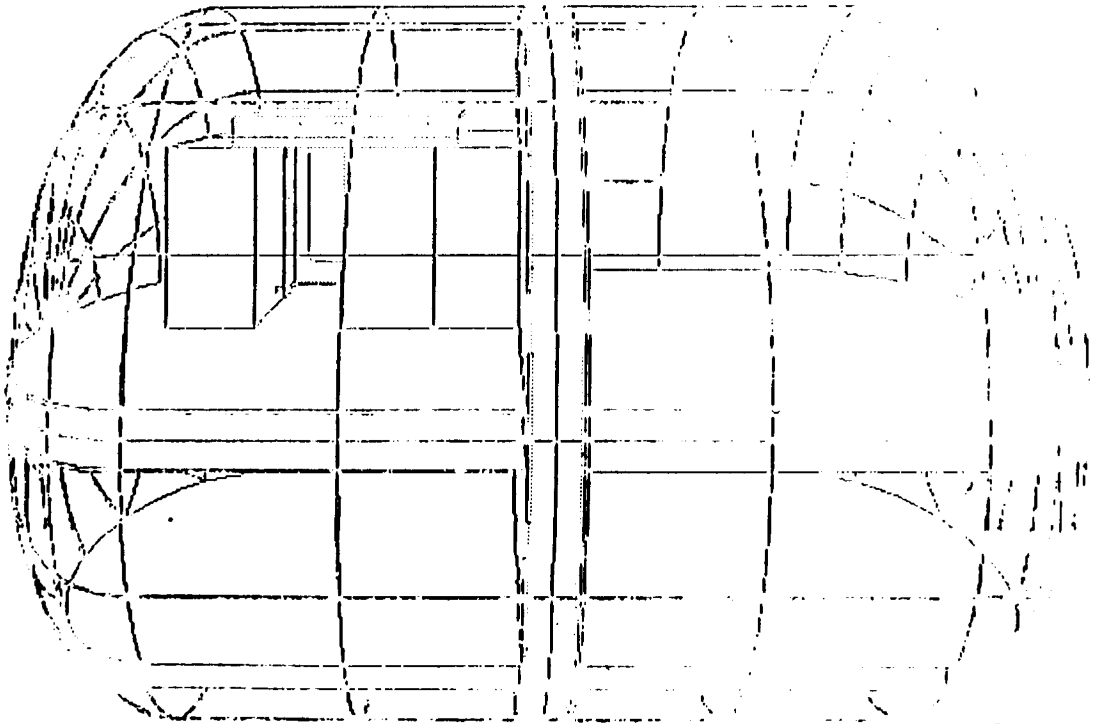
Several radiation research concerns exist today and require considerable investigation. These concerns include but are not limited to: (1) a reevaluation of the conventional risk assessment system as previously discussed, (2) the development of real-time SPE and dosimetry warning systems, (3) trade studies to realistically select and assess shielding material, mass, size, and structural integrity, (4) an evaluation of the potential for exacerbating the effects of radiation under weightless conditions, (5) evaluation of shielding technologies including: waste water, and lightweight composite materials, and (6) further analysis of the dependance of shielding and warning for various mission profiles greater and less than 1 AU.

In terms of the external and internal radiation environments, it will be essential to obtain further data and reliable descriptions of the fluxes and types of primary and secondary radiation. The Life Sciences Division is currently planning a reusable, free-flying biological satellite program (LifeSat), that will provide the capability to study the biological effect of radiation exposure and the effectiveness of various shielding materials. Accurate information will be provided on a unique spectrum of radiation that will be extremely valuable for risk assessment and protection methodology. It has been estimated that a 60-day mission in polar orbit would simulate 5% of a Mars mission in terms of the radiation environment.

Radiation Protection

Space Transfer Concepts and Analyses for Exploration Missions

Marshall Space Flight Center
Huntsville, Alabama



Future Studies Office

ADVANCED
CIVIL
SPACE
SYSTEMS

NASA Study Contract NAS8-37857
March 23, 1990

Boeing Aerospace and Electronics
Huntsville, Alabama

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Radiation Protection

Contents (1)

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Radiation Protection

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Introduction

The protection of crew members from the harmful effects of ionizing radiation will be a key issue confronting mission planners and vehicle designers involved in the Human Exploration Initiative. It is important to develop a full understanding of the natural radiation environments that will be encountered during exploration missions leaving the protective cover of the Earth's magnetic field. As a point of departure an extensive literature search was undertaken to develop this understanding and ascertain the current state of technology regarding radiation protection.

The brief summary of radiation protection requirements has been pulled from the Level II (MASB) September 8, 1989 document. A recent revision to this document, released March 14, 1990 entitled "Human Exploration Study Requirements", does not show any changes to the radiation protection requirements. A broad-brushed review has been provided referring to crew exposure guidelines. Relationships between mission phases and the radiation environments follows. Included in this section are descriptions of the natural environments. It will be important to develop new and innovative methods of protecting crew members during long duration missions. One such method, the combination storm shelter/galley configuration, now in an early stage of development, is shown.

A substantial number of questions and concerns remain about the hazards and methods of dealing with ionizing radiation. A list has been provided describing some of these concerns.

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Radiation Protection Requirements Summary

- Provide protection from natural and manmade radiation environments (ionizing and nonionizing) to crew members and spacecraft electronics

- Protection to be provided as stipulated in NASA STD-3000 section 5.7.2.2.2, paragraph *a*, with the addition of real-time monitoring

- Radiation protection shall not be provided in the MEV

- Radiation protection for crew members to conform to method of ALARA

- "Guidance on Radiation Received in Space Activity": NCRP Report No. 98 dated July 31, 1989 (Dosage levels permitted) shall be adhered to

rom Level II - Human Exploration Study Requirements: September 8, 1989

Units and Terms Used to Describe Human Responce to Ionizing Radiation

The quality factors, Q, have been set by the International Commission on Radiological Protection (ICRP) and accounts for the different biological effectiveness of various ionizing radiation. The values that have been assigned to the high atomic number and energy particles (HZE) associated with galactic cosmic radiation and solar proton events reflects a conservatism. This conservatism is dictated by a serious lack of knowledge about the biological effectiveness of the high LET radiations (radiations with LET's ≥ 175 keV/μm). Average Q values for these particles may be exaggerated. Recent research has raised questions as to whether such high quality factors are justified. For example, the dose equivalence (DE) which appears on the following chart, assumes a uniform distribution of energy throughout the tissue of interest. In reality less than half of the cells of an astronaut will be traversed by HZE particles. A more realistic approach may be to assign relative health risks per fluence (particles $/cm^2 \cdot s$) of given linear energy (or charge and velocity).

Units & Terms Used to Describe Human Response to Ionizing Radiation

Dose - (D)

- The amount of radiation energy absorbed by tissue
- Common unit of measure - rad (1 rad = 100 ergs per gram of material)
- SI unit for dose - gray (Gy)
- 1 Gy = 100 rads

Linear Energy Transfer - (LET)

- Denotes the rate of energy dissipation along the path of a charged particle
- Units expressed in energy/unit length (keV/μm)

Quality Factor - (Q)

- An artificial factor dependent on the LET of which biological effects from absorbed doses may be related to X- and gamma radiation (how much biological damage)
- Nondimensional factor
- Values are based on the most detrimental biological effects from continuous low dose exposure
- Values for many high rate exposures may be considerably lower

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Units & Terms Used to Describe Human Response to Ionizing Radiation

(continued)

Dose Equivalent - (DE)

- The amount of biologically damaging ionizing radiation
- Common unit of measure - rem (roentgen equivalent man)
- SI unit - sieviert (Sv)
- 1 Sv = 100 rem
- $DE = D \cdot Q$

Relative Biological Effectiveness - (RBE)

- Related but distinctly different from Q
- Based solely on experimentally determined effects of different types of radiation on biological systems
- Nondimensional quantity

Short Term Dose Equivalent Limits and Career Limits for Protection Against Nonstochastic Effects

This chart provides the latest recommended dose equivalent limits for astronauts contained in the NCRP Report No. 98, Guidance on Radiation Received in Space Activities. The NCRP (National Commission on Radiation Protection) recognizes the many inherent risks involved in exploratory class missions that leave the protective confines of the Earth's magnetosphere. "No specific limits are recommended for personnel involved in exploratory class missions, for example to Mars". The NCRP recommends in addition to the principal of ALARA (As Low As Reasonably Achievable), the career limits proposed be adhered to as guidelines rather than limits whenever possible. NASA has a radiation protection program for astronauts that limits the amount of radiation received deep in the body to what is judged an acceptable level. Ancillary standards to the eye and skin are also shown. In certain situations such as EVAs in the trapped radiation belts, the dose to the eyes or skin could be very high before the dose limits to the BFO (Blood Forming Organs) could be met. Thirty day limits are set to avoid immediate radiological impacts on a mission involving nausea, vomiting and the like. The career dose-equivalent limits are based upon keeping the life-time risk of excess cancer mortality to less than 3%, an excess risk judged to be acceptable. As can be seen the career limits differ according to sex and age.

Short Term Dose Equivalent Limits and Career Limits for Protection Against Nonstochastic Effects

STCAEM/mha/07March90

All values presented in Sv - (1 Sv = 100 rem)

Time Period	BFO*	Lens of Eye	Skin
30 day	0.25	1.0	1.5
Annual	0.5	2.0	3.0
Career	See table below	4.0	6.0

* Blood forming organs. This term has been used to denote the dose at a depth of 5cm

Career whole body dose equivalent limits based on a lifetime excess risk of cancer mortality of 3%

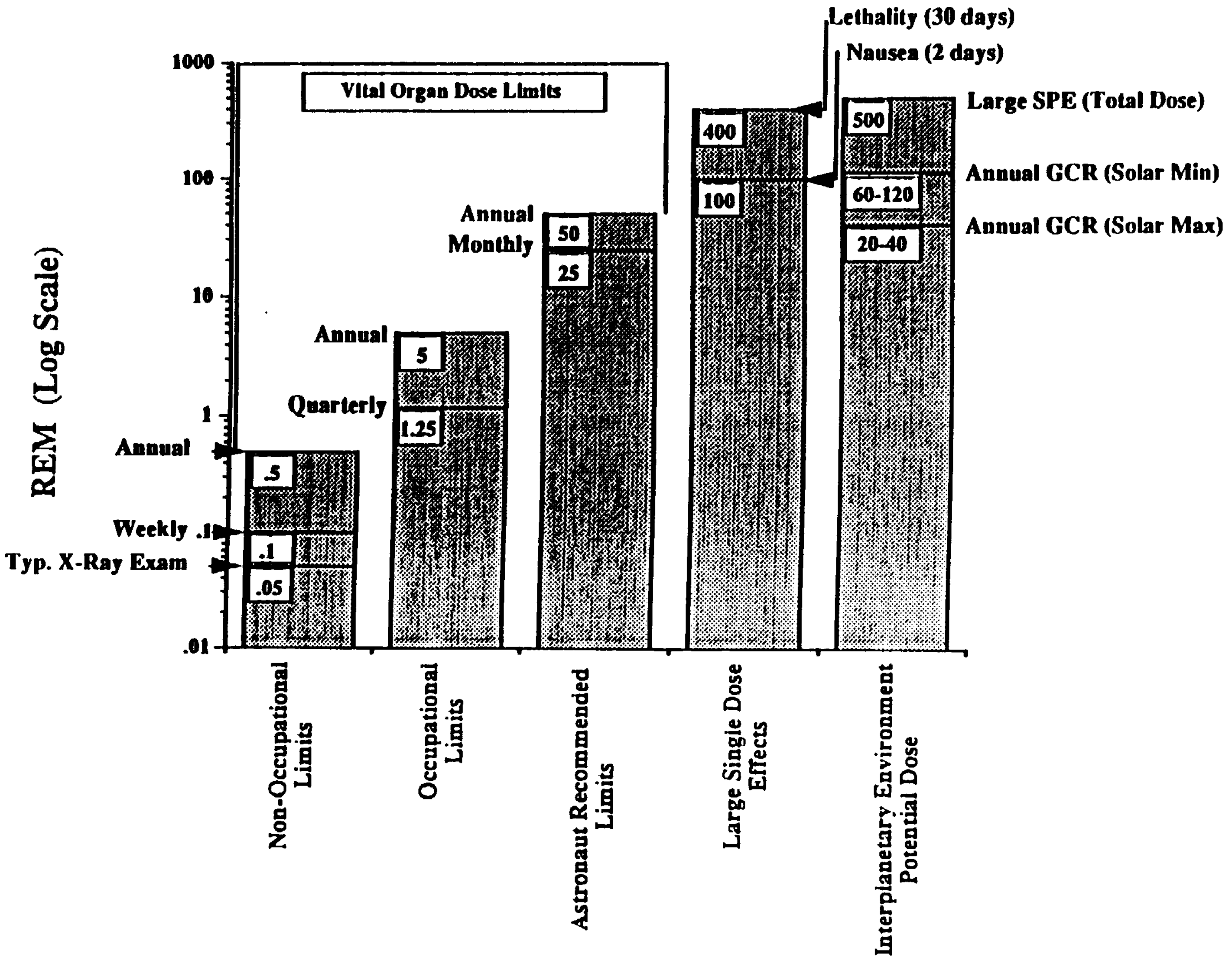
Age (years)	Female	Male
25	1.0	1.5
35	1.75	2.5
45	2.0	3.2
55	3.0	4.0

Data from Guidance on Radiation Received in Space Activities, NCRP Report No. 98

Radiation Dose Comparison - Cause/Effect/Limits

This chart presents a comparison of acceptable equivalent dose limits for terrestrial non-occupational and occupational workers and astronauts to the blood forming organs. Exposure of crew members in space may be reduced but cannot be completely eliminated and, therefore, must be considered an occupational hazard. However, for various reasons, occupational standards that are used on the ground should not be applied directly to situations in space. In the recommendation of the career exposure limits by the NCRP, cancer is considered the principal risk. Based on this consideration the NCRP recommends a career limit of 3% excess risk of cancer mortality for space activities for both sexes of all ages. In addition to the comparison between occupations, large single dose effects are represented and the potential interplanetary environment dosages that may be encountered on a trip to Mars. The chart may be somewhat deceptive due to logarithmic scale that is used. This is a reconstruction of a chart presented by Dr. S. Nachtwey in "Health Physics", August 1988.

Radiation Dose Comparison - Cause/Effect/Limits



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se comparison data from Dr. S Nachtwey, report from "Health Physics, August 1988

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Radiation Dose Examples and Effects

From Life on Earth	**Exposure**
• Transcontinental round trip by jet	0.004 rem
• Chest x-ray (lung dose)	0.010 rem
• Living one year in Houston	0.100 rem
• Living one year in Denver	0.200 rem
• Xeromammography (breast dose)	0.383 rem
• Barium enema (intestine dose)	0.875 rem
• Living one year in Kerala India	1.300 rem
• Max. allowable radiation worker/yr	5.000 rem
Manned Spaceflight	
• Skylab 3, 84 days (blood forming organs)	7.94 rem
(eye lens)	12.83 rem
(skin)	17.85 rem
• Max. allowable space worker/yr	50.00 rem

Effect in Healthy Adults	**Acute Dose**
• Blood count changes common	50 rad
• Vomiting, "effective threshold"	100 rad
• Mortality, "effective threshold"	150 rad
• LD_{50} minimal medical treatment	320-360 rad
• LD_{50} supportive medical treatment	480-540 rad
• LD_{50} bone marrow/blood stem cell transplant	1000 rad
Effects on Reproductive Systems	
• 50% temporary sperm count reduction	15 rad
• 100% sperm loss lasting a few months	100 rad
• Male sterility lasting 3 or more years (if subject survived high dose)	600 rad
• Possible menopause in 40 yr.-old woman	300 rad
• Possible temporary menstrual suppression in 20 yr.-old woman.	300 rad

From the SICSA Outreach Journal; Vol. 2,No. 3, July-September, 1989, Stuart Nachtwey

Quality Factor for Various Types of Radiation

These two charts show the relation between quality factor (Q) and linear energy transfer (LET). The Q values are those which are currently used for various types of radiation. As a given particle degrades in tissue, the quality factor will rise as its energy transfer per micron rises. For a beam of protons having a wide range of energies, the average Q tends to drop with increasing depth in tissue as the lower energy component tends to be removed with increasing depth and the high-energy component continues its traversal.

The standard Q values are based on the most detrimental chronic biological effects for continuous low-dose rate exposure that may be met in industrial situations.

Quality Factor for Various Types of Radiation

Type of Radiation	Quality factor, Q
X-rays	1
Gamma rays & bremßtrahlung	1
Beta particles, electrons, 1.0 MeV	1
Beta particles, 0.1 MeV	1
Neutrons, thermal energy	2.8
Neutrons, 0.0001 MeV	2.2
Neutrons, 0.005 MeV	2.4
Neutrons, 0.02 MeV	5
Neutrons, 0.5 MeV	10.2
Neutrons, 1.0 MeV	10.5
Neutrons, 10.0 MeV	6.4
Protons, greater than 100 MeV	1 - 2
Protons, 1.0 MeV	8.5
Protons, 0.1 MeV	10
Alpha particles (helium nuclei),5 MeV	15
Alpha particles, 1 MeV	20

LET - Q relationship

LET - in water (keV/µm)	Q
≤ 3.5	1
7	2
23	5
53	10
≥ 175	20

Nature and Location of Electromagnetic and Particle Ionizing Radiation

Ionizing radiations vary greatly in energy. Electromagnetic radiation have energy quanta determined by their wavelength or frequency. The energy of particulate radiation depends on the mass and velocity of the particles. This chart summarizes the main types of ionizing radiation including their charge, mass, and location. Crew members will be subjected to radiation emanating from two primary sources, those that are manmade and those originating from natural sources. Naturally occurring radiation is comprised of charged particles and accompanying electromagnetic radiation attributable to a number of distinct sources.

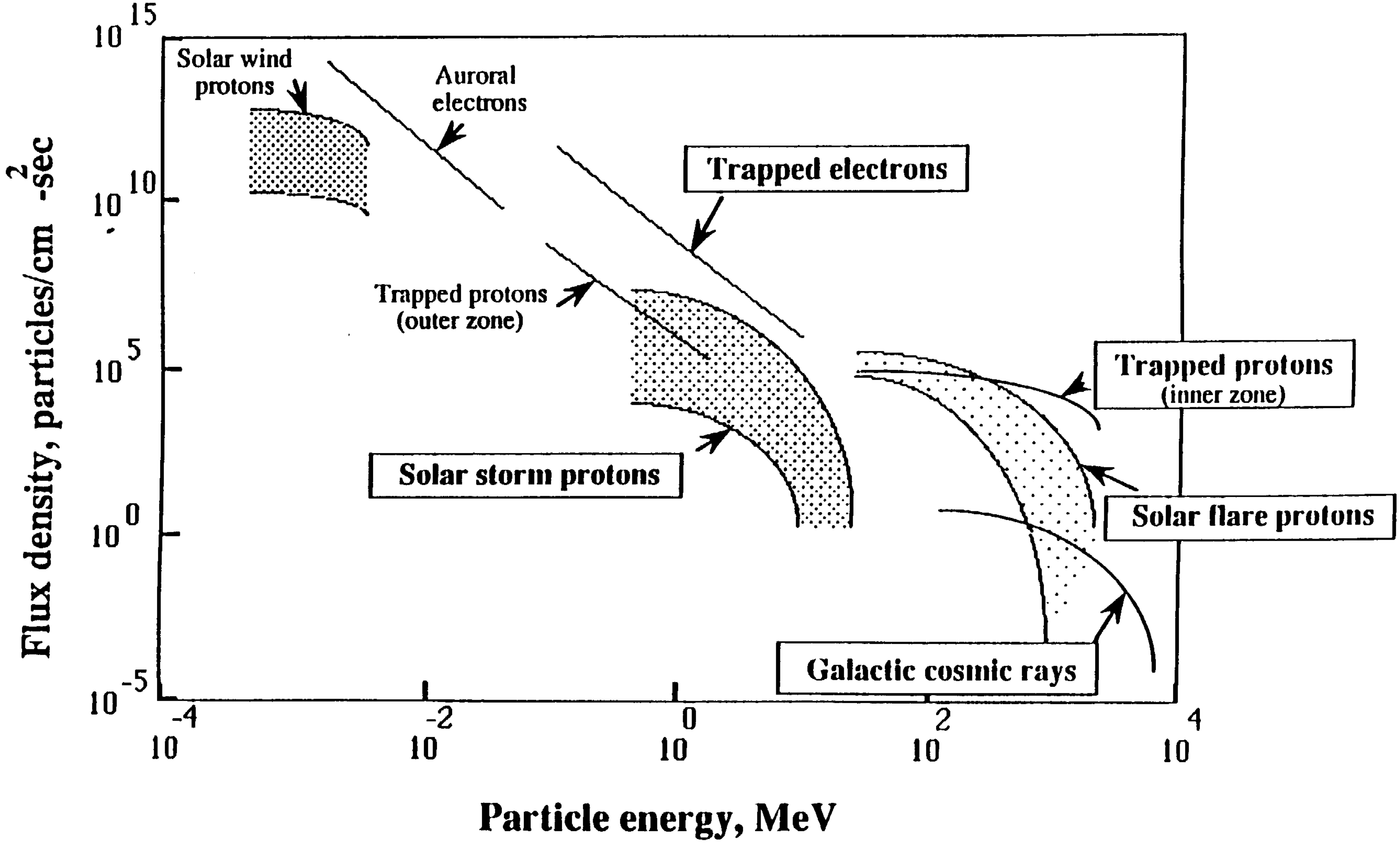
Nature and Location of Electromagnetic and Particulate Ionizing Radiation in Space

Name	Charge	Nature of radiation	Mass	Location/source
X-ray	0	Electromagnetic	0	Radiation belts, solar radiation (produced by nuclear reactions and by stopping electrons) Bremsstrahlung radiation (-e deflection by Coulomb field at atomic nuclei of target material)
Gamma ray	0	Electromagnetic	0	Everywhere in space (disintegration of atomic nuclei)
Electron	-e	Particle	1 m_e	Radiation belts and elsewhere
Proton	+e	Particle	1840 m_e or 1 am	Galactic and solar cosmic rays, radiation belts
Neutron	0	Particle	1841 m_e	Secondary particles produced by nuclear interactions involving primary particle flux
Alpha particle (helium nucleus)	+2e	Particle	4 am	Galactic and solar radiation
HZE particle (heavy primary)	≥+3e	Particle	≥ 6 am	Galactic and solar radiation

Space Radiation Environments

Particle radiations that occur in space are summarized in this chart. The various radiation environments occur with both temporal and spatial variations. Trapped particles exist only in the geomagnetosphere, the auroral electrons are observed only in the polar regions, and solar flare protons are emitted at dangerous levels infrequently and highly unpredictably. Radiations with energies below 100 keV and protons below approximately 10MeV are important primarily from a materials standpoint and are considered to be biologically unimportant. Galactic cosmic radiation, trapped protons and electrons within the radiation belts, and solar flare protons are all biologically very important. Even though the galactic cosmic radiation has a very low flux density many questions surrounds it because of its particular composition and high energies.

Space Radiation Environments



om Workshop on the Radiation Environment of the SPS, J.W. Wilson

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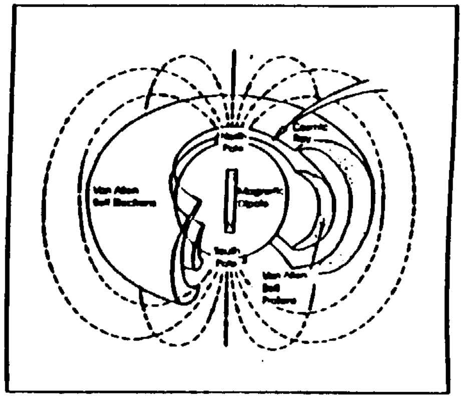
Radiation Environments



- Trapped Radiation Belts

  ~ Belts geomagnetically trapped around Earth

  ~ Inner proton belt
    - Inner belt consists of protons and electrons
    - Flux density and energy of protons dominate as source of ionizing radiation
    - Inner belt densities respond to temporal variations in solar activity
    - Extends out to an altitude of approximately 12000 km
    - Proton density peaks at an altitude of 2000 km

  ~ Outer electron belt
    - Consists primarily of trapped electrons
    - Secondary radiation (Bremßtrahlung) dominates as source of ionizing radiation
    - Outer belt also responds to temporal variations in solar activity
    - Extends from an altitude of approximately 16000 to 36000 km
    - Density peak on average at 20000 km

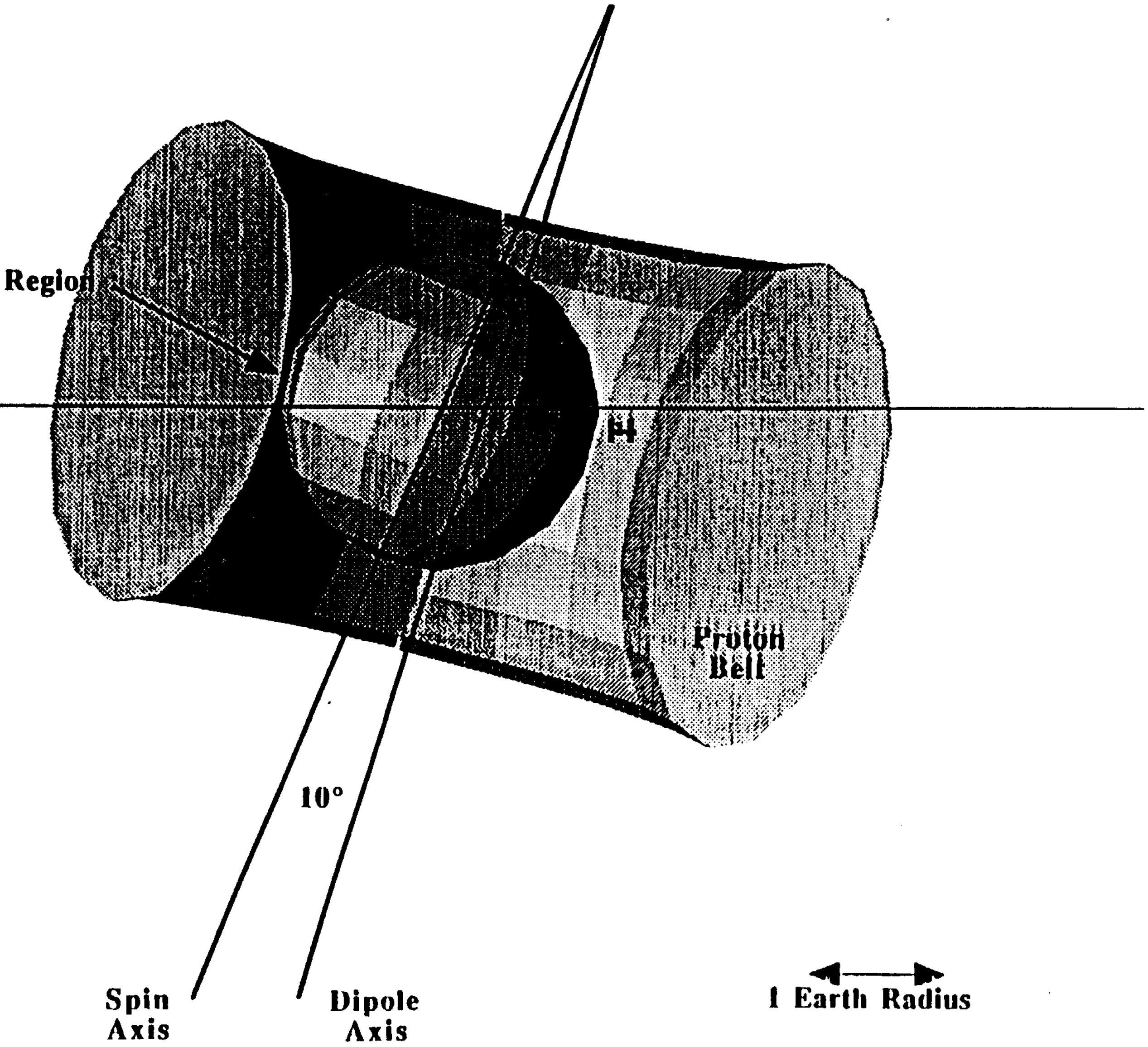
  ~ South Atlantic Anomaly
    - Caused by combination of [1] anomaly in geomagnetic field over South Africa and [2] slight displacement of dipole axis (10°) from Earth's rotational axis
    - Proton intensity for energies >30MeV are observed at altitudes between 200 and 400 km, approximately 1100 to 1300 km below normal

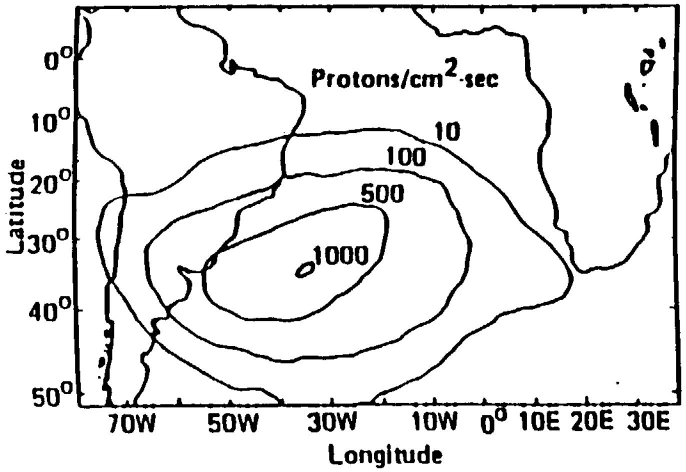
South Atlantic Anomaly

A feature of spatial distribution which has and is attracting much interest is the South Atlantic Anomaly (SAA). Caused by a combination of an anomaly in the geomagnetic field over South Africa and a slight displacement of the dipole axis (the magnetic central axis) from the Earth's rotational axis, the fluxes of the trapped particles are larger at low altitudes over the South Atlantic Ocean. The SAA extends from from 20° East to about 70° West longitude and 10° to 55° South latitude. The contours shown are trapped proton intensities for energies ≥ 30 MeV at an altitude of 200 km.

For trajectories of space vehicles of ~30° inclination, there will be five or six traverses through this region each day. Experience with Earth orbital missions to date indicates that nearly all of the accumulative radiation exposure has been attributable to passage through this zone. During the period of vehicle assembly and checkout this will be a concern to both crew and electronics.

South Atlantic Anomaly





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Radiation Environments (continued)

- **Galactic Cosmic Radiation (GCR)**

  ~ Originates outside of the solar system

  ~ Radiation consists of atomic nuclei ionized and accelerated to very high energies

  ~ Present isotropically in space (comes from all directions)

  ~ Decrease in flux caused by increase in strength of interplanetary magnetic field (below 100 Mev/nucleon)
  - flux density at solar: maximum = 2 prot/cm ^{2} ; minimum = 4 prot/cm ^{2} frontal area

  ~ In the energy range from 100 MeV/nucleon to 10 GeV/nucleon, where fluence is greatest the baryonic component consists of:
  - 87% protons (H^+)
  - 12% alpha particles (H^{++})
  - $\leq$ 1% HZE particles (high Z, high energy)

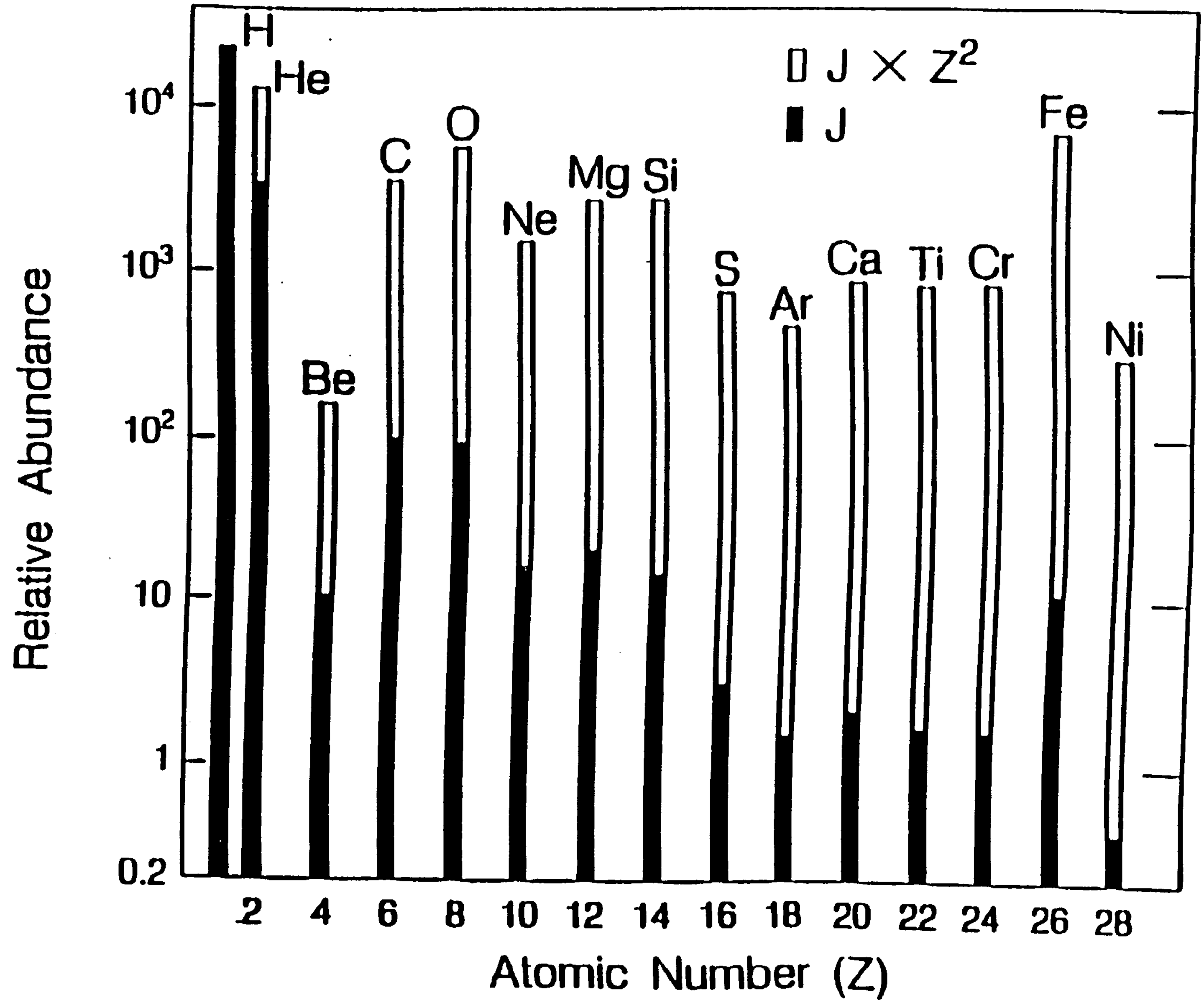
  ~ Main contribution to the radiation dose equivalent comes from the HZE particles and not from protons

  ~ Energies of particles extend to values of 10^{20} eV/nucleon

Relative Abundances of GCR Nuclei and a Measure of Their "Ionizing Power"

This histogram shows the relative abundances of the even numbered GCR nuclei (solid bars, J) compared to their abundances weighted by the square of the particles charge (Z) ^{2}to give a measure of the "ionizing power" of each element (open bars). The ions that are heavier than helium are generically termed HZE particles. Although iron ions are only one-tenth as abundant as carbon or oxygen ions, their contribution to the GCR dose is substantial as indicated.

Relative Abundances of GCR Nuclei and a Measure of Their "Ionizing Power"



Radiation Environments (continued)

- Solar Proton Events (SPE)

  ~ Highly unpredictable in nature (frequency, intensity, duration)

  ~ Large emissions of charged particles [primarily: protons (95-98%), alpha (1-3%) and HZE (<1%)]

  ~ Large fluences of charged particles emitted from the sun primarily associated with solar flare activity

  ~ Occurrences of flares is associated directly with the 11 year solar cycle

  ~ Flares tend to occur more frequently during the declining portion of the 11 year cycle

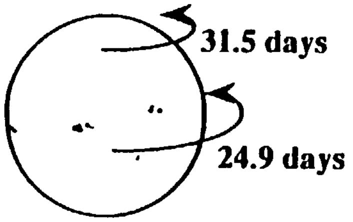
  ~ Solar proton events fall into two broad categories
    - "ordinary" events
    - anomalously large events (ALSPE); on average may occur 2 or 3 times during 4 to 6 year period of high sun spot activity

  ~ Large solar flares can have fluences greater > 10^{10} protons/cm^{2} with energies > 10 MeV

  ~ Potential of delivering extremely high dosages in short period of time

  ~ Small percentage of flares will be of sufficient intensity to emit large proton fluences

The Active Sun



Solar Differential Rotation

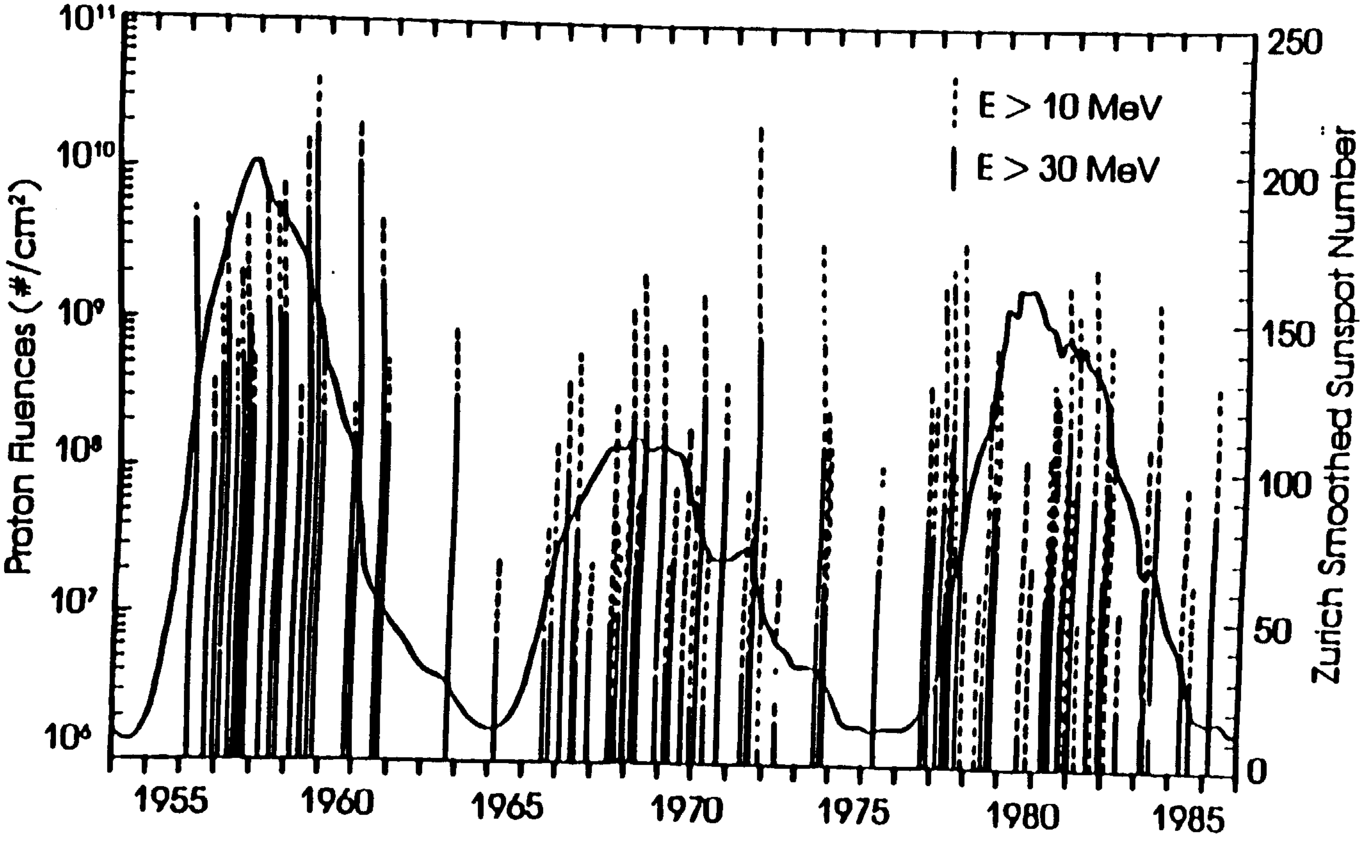
- Solar intensity will fluctuate rapidly due primarily to distortion of the Sun's large scale magnetic field
- Distortion of the magnetic field comes from differential rotation of gaseous body
- Magnetic field becomes twisted and contracted into specific regions such as facula, plage, spicules, prominences, sunspots, and flares
- Energy is often times released explosively in the form of a solar flare appearing as sudden local brightening in the chromosphere
- Stored magnetic energy is released as kinetic energy as field relaxes back to initial state (total energy released may be $10^{21}$to 10^{25}joules integrated over three flare phases)
    - ~ precursor - slight enhancement of observed soft x-rays
    - ~ flash - increase in optical and x-ray emission by 50% above background
    - ~ main phase - bulk of energetic particle emission
- Radiation from solar flare extends from radio to x-ray wavelengths
- Most flare events last about an hour. ALSPE, highly lethal occurrences are relatively rare but will last for hours or even days

Solar Activity and Flare Proton Fluence

It is important to note the effect of the solar cycle, or sunspot cycle, on the occurrence of solar-proton events. The sunspot number has been observed for approximately 200 years and varies with an average period of 11 years. During the upper half of the cycle, when the sunspot number is the largest, solar-proton events occur. They generally do not occur during the lower half of the cycle. Attempts to obtain detailed relationships between sunspot numbers and the frequency have shown that nothing can be said with the assurance beyond the fact that events tend to occur during the upper half of the cycle.

This chart provides a basis of comparison between the Zurich Smoothed Sunspot Number, the proton fluences and the time of occurrence for solar cycles 19, 20, and 21 from left to right. Currently cycle 19 is considered one of the most extreme cases in terms of sunspot number. The occurrence of several SPE's during this cycle (i.e., 2/56 and 11/60) were used as the basis for modeling protective measures for early manned missions. These events were used in fact as "worst case flares" until the occurrence of the 1972 event in August. Occurring during a cycle that was initially thought to be "average", it became apparent as to the lack of understanding we had in making predictions of such events. We are aware today that had the 8/72 event occurred at a more "favorable" location on the solar surface relative to the Earth this event would have been substantially larger.

Solar Activity and Flare Proton Fluence



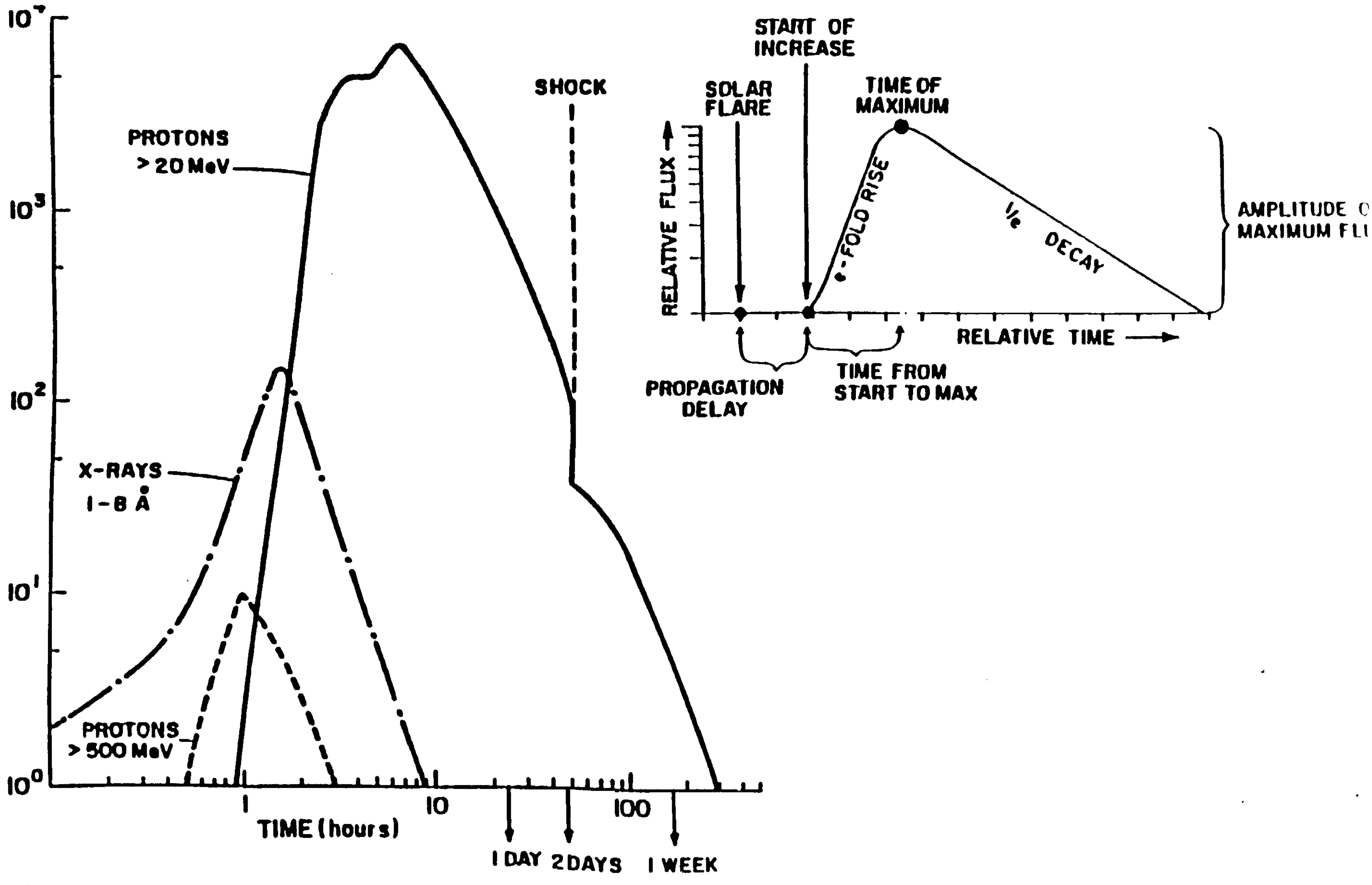
* From the SICSA Outreach Journal; Vol. 2,No. 3, July-September, 1989, Stuart Nachtwey

Relative Time of Solar Particle Emissions at 1 AU

It is important for the purpose of evaluating potential radiation protection schemes to understand solar particle propagation. Energetic solar particles will reach the orbit of Earth in a few short minutes if the particles have high energies, or within hours if possessing lower energies. This chart presents a relative time scale of solar emissions at 1 AU.

The inset graph shows the general time behavioral characteristics of a solar proton event. The propagation delay time is defined as the time from the maximum of the visible flare intensity to the particle arrival at the detector. The delay time will vary considerably from event to event with variations from several minutes to hours. The fold rise is the time interval between the first arrival of the particles of a particular energy and the time at which the flux of these particles reaches its maximum intensity. The fold rise is also strongly event and energy dependent, the high-energy having a shorter rise time, again times vary from minutes to hours. Finally the decay time is that time between maximum flux intensity and the disappearance of particles of a given energy.

Relative Time of Solar Particle Emissions at 1 AU

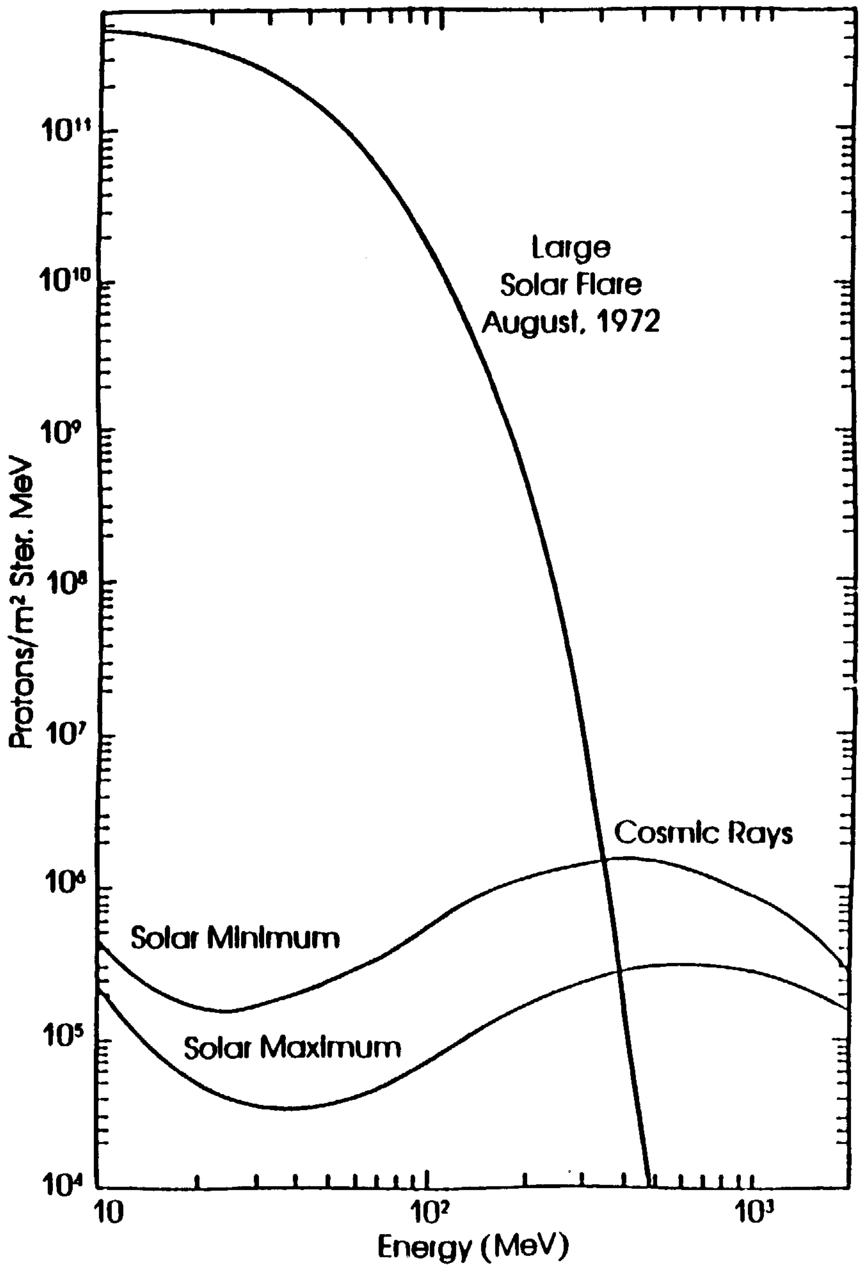


rom "Proton Events During the Past Three Solar Cycles", Smart, D.F., and Shea, M.A

Proton Energy Spectrum

This chart provides a comparison of the time-integrated spectrum for the solar proton event of August, 1972 with the galactic cosmic ray proton spectra accumulated in one week during solar minimum and maximum. The spectral distribution at Earth changes as a function of time because high energy particles tend to arrive before those with lower energy. The angular distribution of the particles also varies from event to event. During some of the high energy events the particles tend to be directional early in the event. The arrival of the lower energy particles tends to be more isotropic in nature.

Proton Energy Spectrum



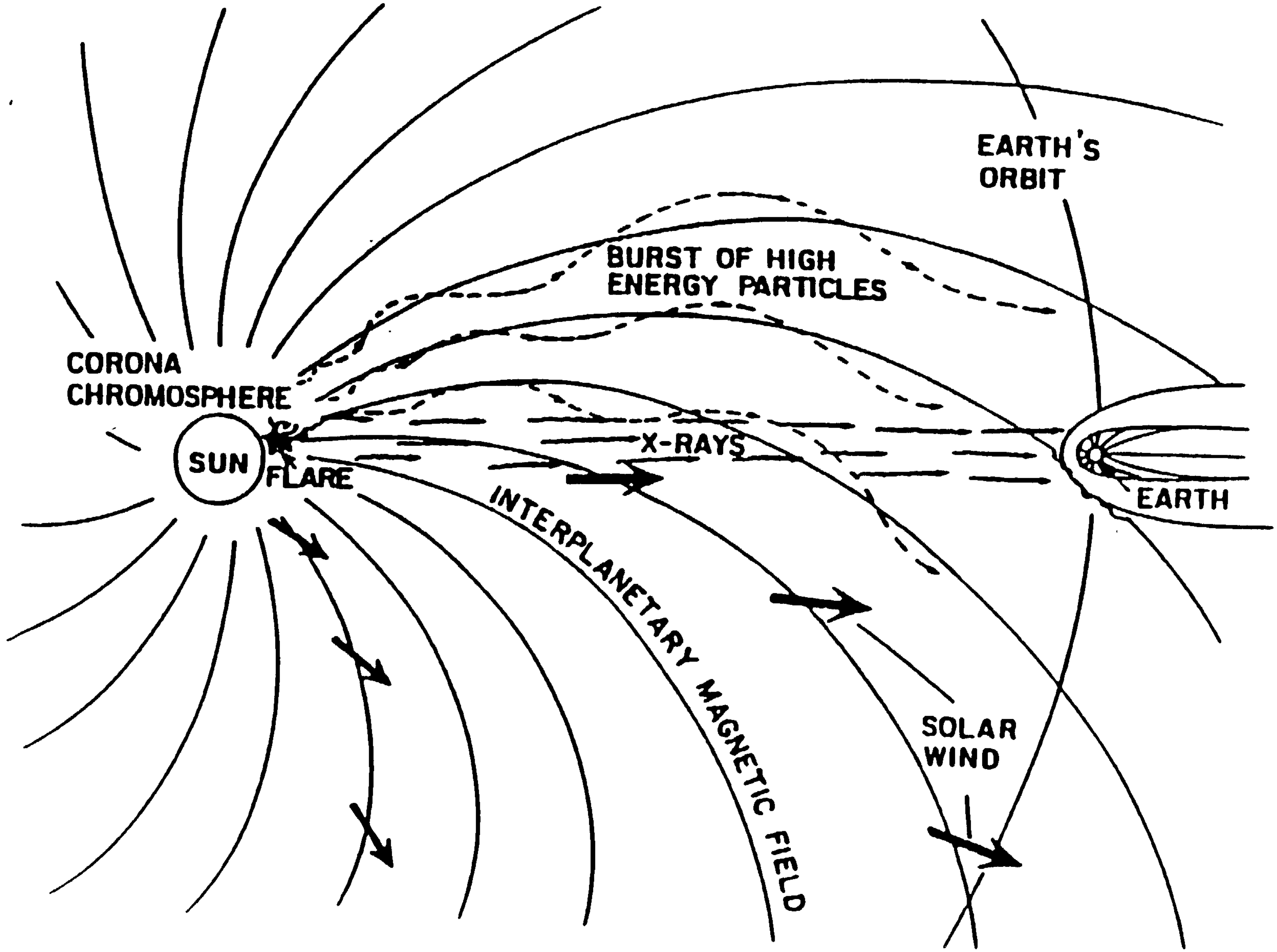
From the SICSA Outreach Journal; Vol. 2,No. 3, July-September, 1989, J.R. Letaw, R. Silberberg and C.H. Tsao

Characteristics of the Idealized Structure of the Interplanetary Medium

Unlike solar electromagnetic radiation, both the onset time and the maximum intensity of the solar particle flux depend on the heliolongitude of the flare with respect to the detection location in space. The directionality results because particles will move more easily along the interplanetary magnetic field direction. The interplanetary magnetic field topology is determined by the solar wind outflow and the rotation of the sun which during "quiet" conditions can be approximated by an Archimedian spiral shown in the figure.

The charged particles emitted during a solar flare consists of a nonequilibrium plasma cloud which expands to several solar diameters as it migrates away from the Sun. The particle fluxes observed by a detector inside this plasma cloud are essentially isotropic and these particles constitute the larger portion of the total flare radiation.

Characteristics of the Idealized Structure of the Interplanetary Medium

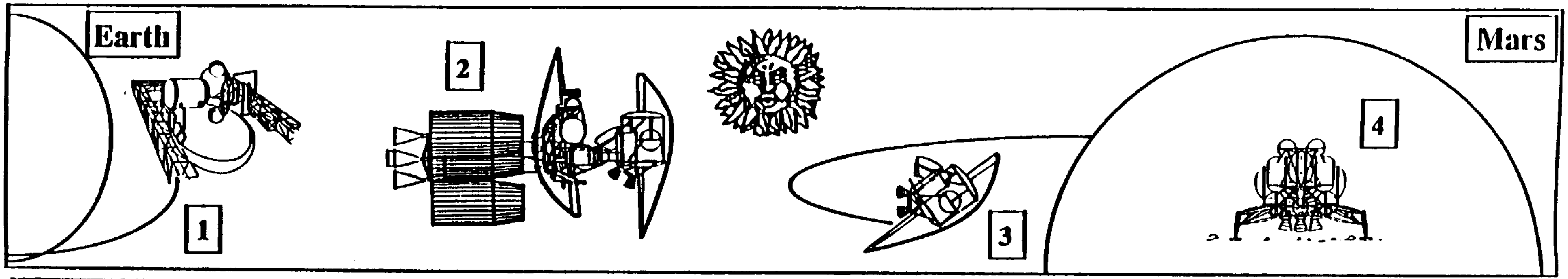


om "Proton Events During the Past Three Solar Cycles", Smart, D.F., and Shea, M.A.

Radiation Environments for Mars Mission Phases

This chart describes radiation environments which are of most concern to mission planners and vehicle configurators. The Mars exploration class mission has been divided into four phases; (1) on-orbit assembly and checkout, (2) vehicle transfers, (3) the Mars orbital sequence, and (4) surface stay time referenced at 30 days. Galactic cosmic radiation and the highly unpredictable (time of occurrence and magnitude of event) solar proton events constitute the overwhelming threat to crew and vehicle except in those areas that fall under the protective coverage of the magnetosphere. In this regime the trapped radiation (Van Allen Belts) and in particular the South Atlantic Anomaly (SAA) pose the largest concern during vehicle assembly and checkout. During the surface exploration phase of the mission additional protection is provided by the Mars atmosphere. The environments are variable both temporally and spatially. This variability occurs for a number of reasons including fluctuations in solar activity associated with the solar cycle, altitude and inclination in LEO, mass of the planetary body, and diurnal and seasonal variations which influences atmospheric density on Mars and consequently changes its attenuating properties. Natural and man-made shielding that will be influencing design work for each of the phases has also been listed in the final column.

Radiation Environments for Mars Mission Phases



Reference	Mission Phase	Environment	Variability	Shielding
1	On Orbit Assembly and Checkout	Trapped Radiation SAA	Solar Activity Altitude, Inclination Earth Proximity	• Geomagnetic Field • Vehicle Structure, fluids, stores equipment and spares
2	Vehicle Transfer Earth/Mars/Earth	Trapped Radiation GCR, SPE	Solar Activity	• Interplanetary magnetic field • Vehicle structure, fluids, stores equipment, spares, & waste • Vehicle orientation
3	Mars Orbit	GCR, SPE	Solar Activity Mars Proximity	• Planet mass • Vehicle structure, fluids, stores equipment, spares, & waste • Vehicle orientation
4	Mars Surface	GCR, SPE	Solar Activity Diurnal and Seasonal Variations	• Mars atmosphere • Planet mass • Mars surface material • Vehicle structure, fluids, stores equipment, spares • Deployable "zenith" shield

Mission Opportunity Stay Time Coincidence with Predicted Solar Maximum and Minimum Years

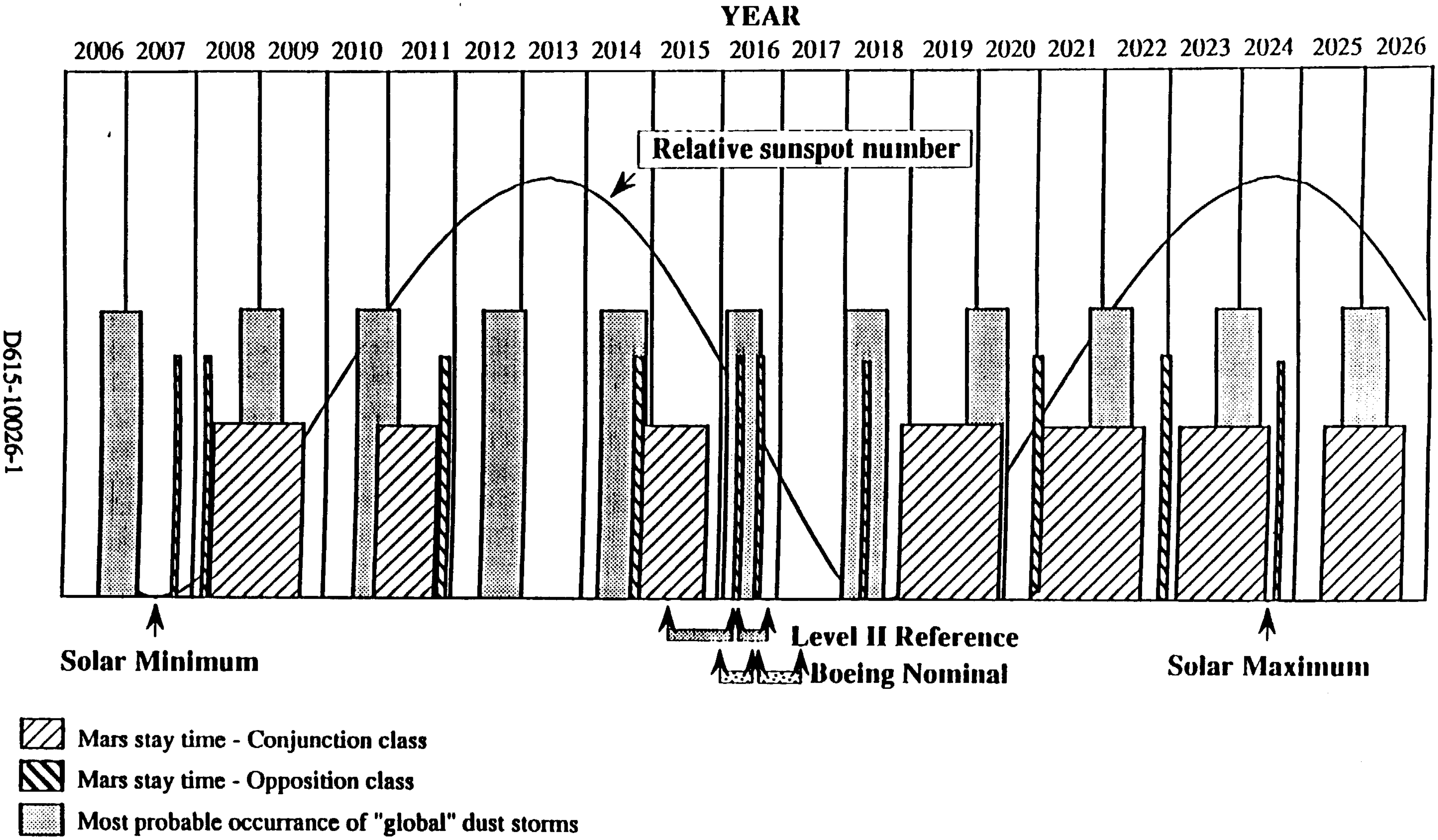
As expected solar flares are often the greatest source of radiation dosage received on a long duration mission outside the protective shield of the Earth's magnetosphere. The timing of a mission can have a major effect upon the expected dose. In addition to showing Mars stay times and the most probable occurrences of "global dust storms" a curve representing the relative sunspot number is shown. Records of sunspots have been kept for over two centuries. The 11 year cycle of sunspots is only approximated but the cyclic behavior is unmistakable.

During solar maximum, when the interplanetary magnetic field strength is greatest, cosmic ray particles are attenuated more effectively producing a GCR flux minimum (the Fornbush decrease), conversely, GCR flux is largest during solar minimum.

Solar proton events change in frequency and size during the 11 year sunspot cycle, reaching maximums before and after sunspot maximum. This chart shows that greater concern for occurrences of SPE's is not to be directed only at those years of predicted solar maximums but also in the regions on the curve surrounding the solar maximum.

Mission Opportunity Stay Time Coincidence With Predicted Solar Maximum and Minimum Years

STCAEM/mha/07March90

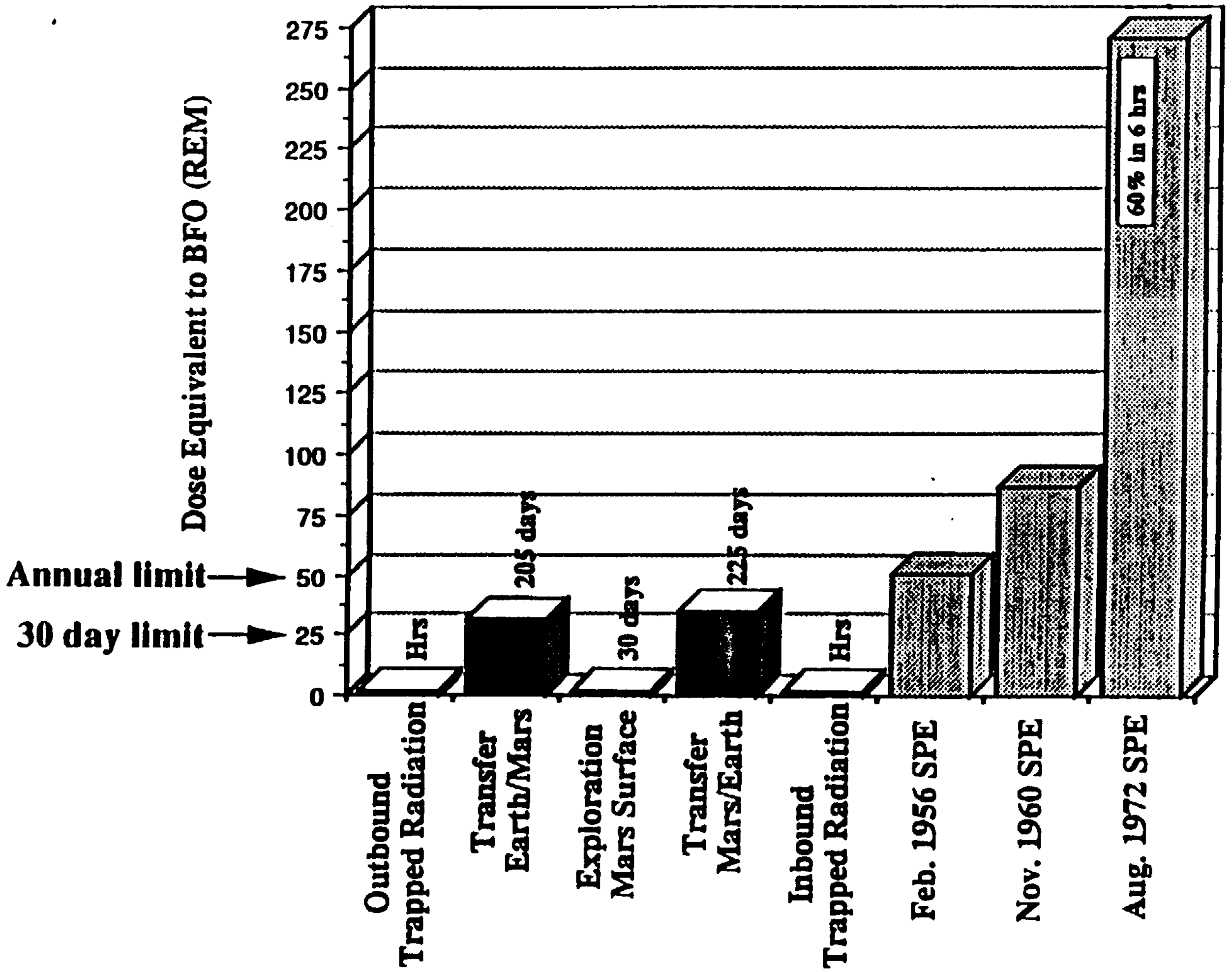


Dose Equivalents to the BFO for Various Mission Phases to Mars with Representative SPE's

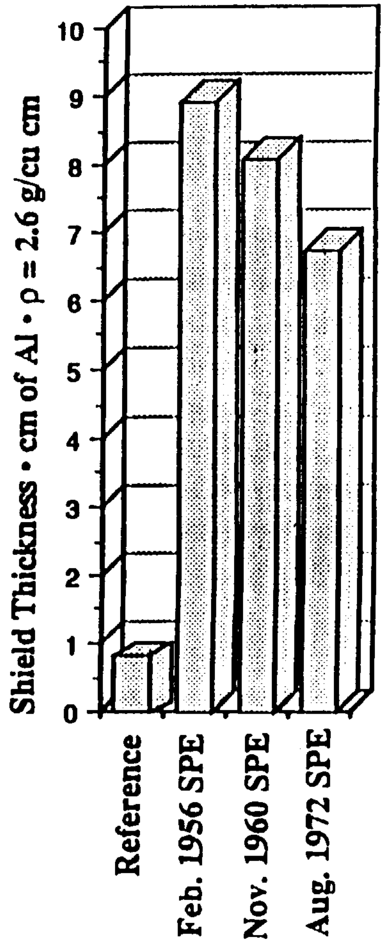
This chart is meant to provide reference data to complement the chart titled "Radiation Environments for Mars Mission Phases". A representative opposition mission with total transit time of 430 days and stay time of 30 days on the Martian surface was used to determine the respective dose equivalents to the blood forming organs during the various mission phases. The black bars show these dosages and also indicate the duration in which the crew remain in the particular environment. In addition to these "constant" forms of radiation three representative flares are also presented to show the potential hazard of these unpredictable events. The August 1972 event occurred toward the end of solar cycle 20 previously thought to be "stable". Prior to this event the 1956 and 1960 events shown were described as the worst recorded cases. It is important to note that the 1972 flare could have been worse if it had occurred at a more "favorable" position on the sun relative to the Earth. The chart clearly shows the immense dosage that can be received during such a short duration event. The data on this chart assumes the protection to the crew would come from 0.77cm of aluminum shielding except when the crew is on the surface in which the Martian atmosphere adds an additional 3.85cm (aluminum) of effective shielding. Another important point to note about the chart is that the 1956 flare was more energetic than the 1972 flare. The high fluence associated with the 1972 flare and longer duration give it greater "ionizing power". In order to reduce the received dosage below the 30 day limit an effective net shielding would have and areal density of ~24g/cm^2 or shield thickness of ~8.9cm.

Dose Equivalents to BFO for Various Mission Phases to Mars with Representative SPEs

- All shielding assumed to be 2 g/sq cm except for Mars surface where additional 10 g/sq cm for atmosphere



Thickness to Reduce Below 30 Day Limit



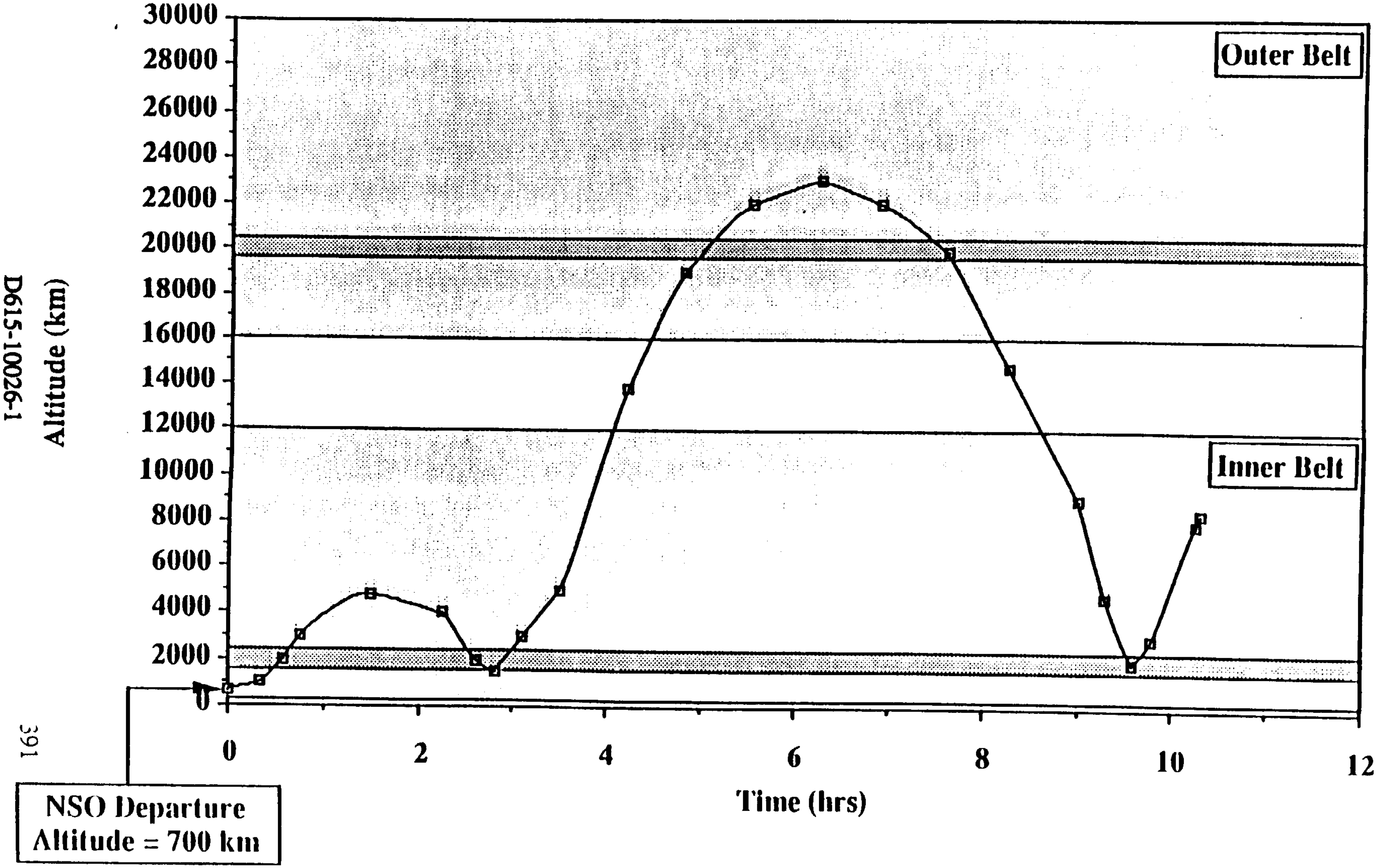
Data from S. Nachtwey, JSC, NASA and J.E. Nealy, Langley, NASA

Low Thrust NTR - 3 Burn TMI - Altitude vs. Time

Radiation exposure times for multiple burn trans-Mars injections are inherently higher than those using single burn strategies. If our low thrust NTR vehicle departs from a nuclear safe orbit (NSO) of 700km and orbital inclination of 28.5 degrees, the total elapsed time to perform the three burns through the inner and outer belts will be approximately 9.75 hours. The upper altitude, lower altitude, and peak regions (indicated by the thin darker band) of the belts vary with solar activity. Conservative values for these altitudes based on literature research have been selected to allow determination of radiation exposure to crew members under .75cm (2g/sq cm) of aluminum. This altitude vs. time plot used in conjunction with the JSC AP-8 and AEI-7 codes were used to determine an approximate dose to the blood forming organs. Preliminary analysis indicated that crew members would receive on the order of 4 rems, not a significant amount but much higher than that received in a straight passage through the belts as the Apollo lunar missions.

Low Thrust NTR - 3 Burn TMI - Altitude vs. Time

- Altitude parameters of inner and outer radiation belts will vary with solar activity and changes in latitude

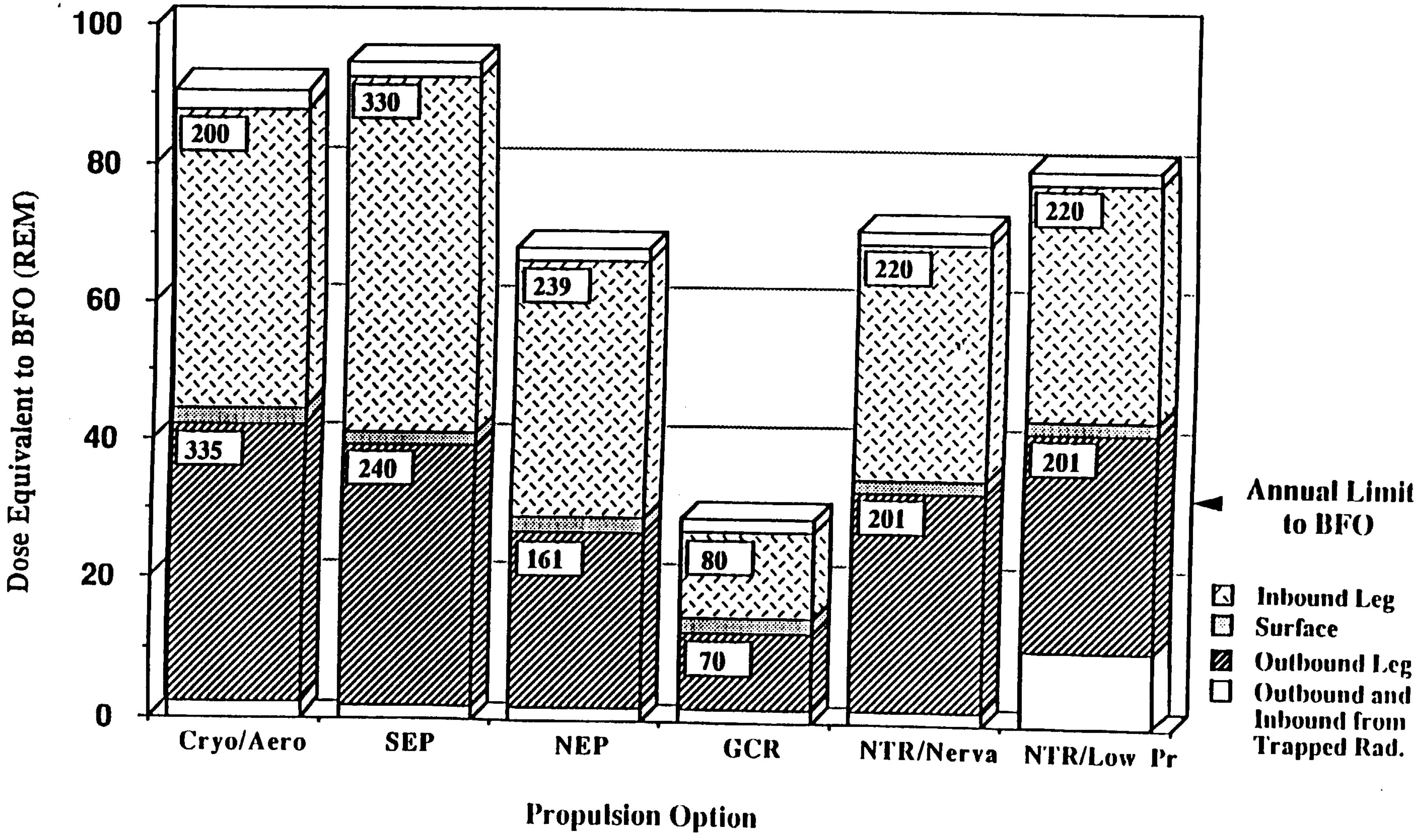


Dose Equivalents to the BFO for Various Propulsion Options

It is possible to predict the amount of ionizing radiation a crew member will receive from the "constant" forms of radiation such as the trapped and galactic cosmic radiation. The exposure to the blood forming organs (assumed to be the limiting system for total dose) will be determined from the bodies own shielding capabilities, the amount of bulk vehicle shielding, total time of exposure,and the energy associated with the charged particles. If we assume a constant vehicle shield thickness of .75cm (2g/cm^2), a Mars stay time of 30 days under a conservative areal density of 10 g/cm^2 , and various trip times through the trapped radiation belts (propulsion option and mission profile dependent), the following dosages would result. Crew members will not be on board SEP and NEP vehicles as they spiral out from a nuclear safe orbit. Transportation to the slowly accelerating vehicle will be accomplished by an OTV, consequently radiation exposures to crew members will constitute a single pass through the trapped radiation belts. The only variation shown to the accumulated dose passing through the trapped belts comes on the outbound leg of the low thrust NTR. The value shown here reflects the use of the previous chart and the JSC AP-8 and AEI-7 codes. Radiation exposures do not include that which may come from nuclear propulsion options or that incurred from solar proton events.

Dose Equivalents to BFO for Various Propulsion Options

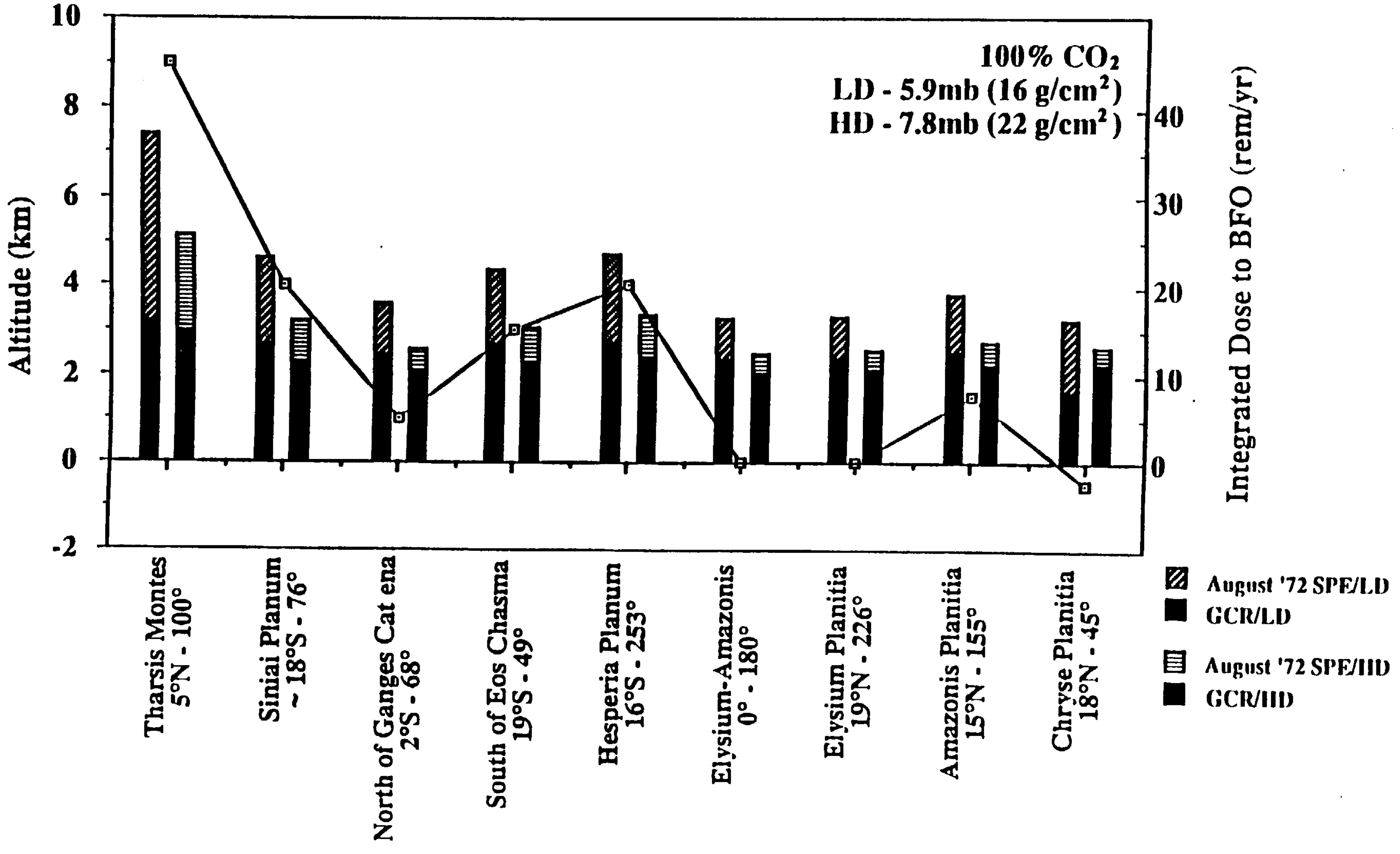
- All shielding assumed to be 2 g/sq cm except for Mars surface where additional 10 g/sq cm for atmosphere
- Stay time on Mars - 30 days



Altitude and Dose Comparison for Mars Using High and Low Density Atmospheric Models

One major concern to mission planners and vehicle designers will be the damaging effects of ionizing radiation from high energy galactic cosmic radiation (GCR) and solar proton events. Crew members will encounter the most harmful radiation exposures during the transit phase of the mission. Once on the surface the tenuous Martian atmosphere should provide significant protection from the harmful radiative fluxes. Variations in the amount of this protection will be the result of changes in the altitude, pressure (seasonal), and the angle from the zenith of the incoming high energy particles. This chart indicates two of those variations, altitude and pressure. Assuming that the composition of the Martian atmosphere is one-hundred percent CO2 (actually ~ 95%), high density (HD) and a low density (LD) models were used to determine the effective shielding provided by the atmosphere. As the pressure increases so to does the potential shielding. In addition to the continuous radiation coming from the GCR flux , one large representative solar flare (August 1972) was added to the integrated GCR exposure over one year to give the annual dose to the blood forming organs. These models assume that incident particles are coming from straight overhead. The chart shows the amount of radiation that would be received at various potential landing sites. As one would expect the greater the altitude of the site, the greater the exposure. The low and high density models indicate the variations that may be encountered with changing season and the movement of the CO2 to the polar regions. The line graph indicates the relative variations in the altitude.

Altitude and Dose Comparison for Mars Using High and Low Density Atmospheric Models



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Ionizing Radiation Protection Design Considerations

Four primary protection methodologies considered

- **Bulk mass shielding**
- **Active electromagnetic shielding**
- **Use of chemical protectors**
- **Avoidance of high radiation fluxes**

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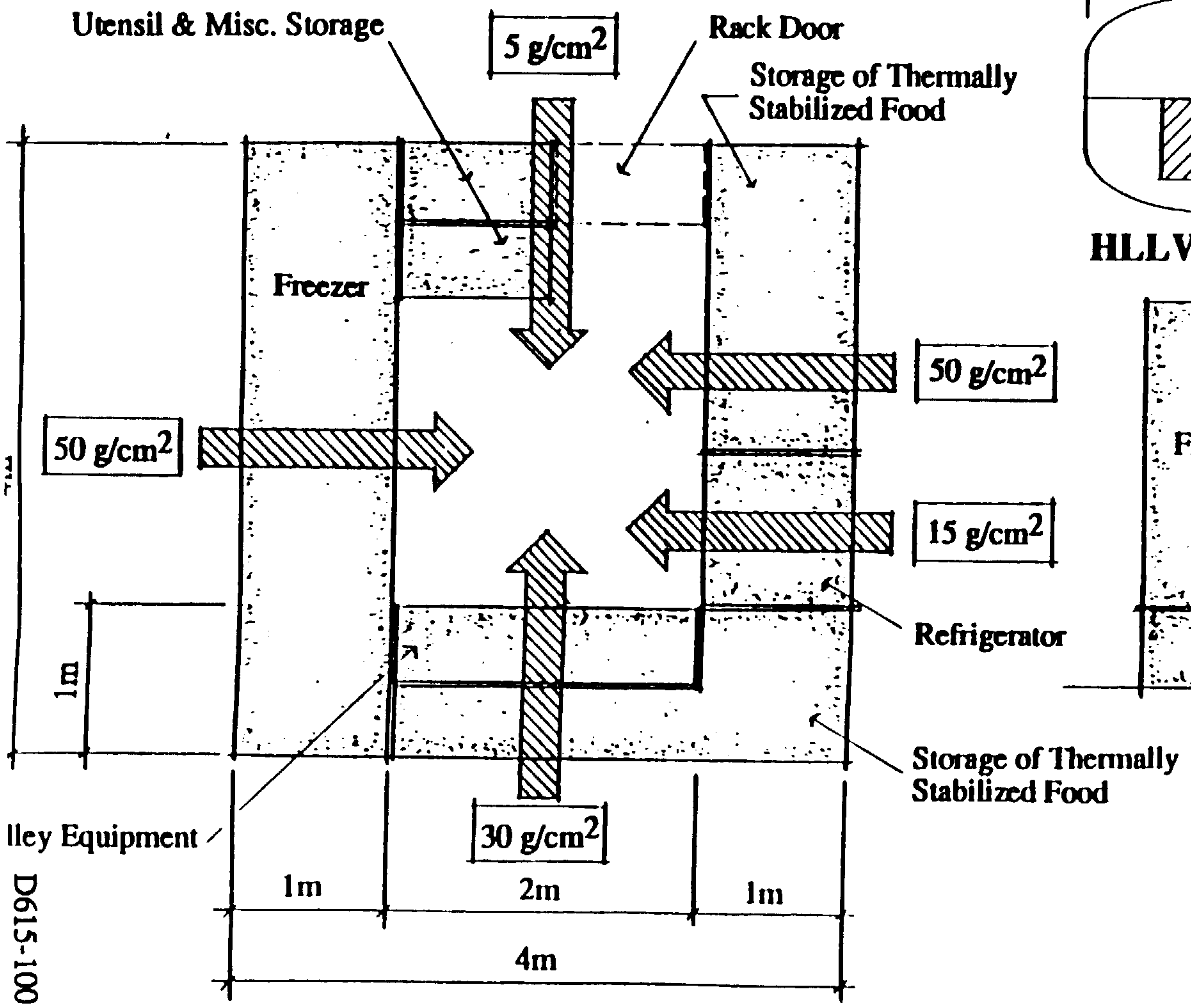
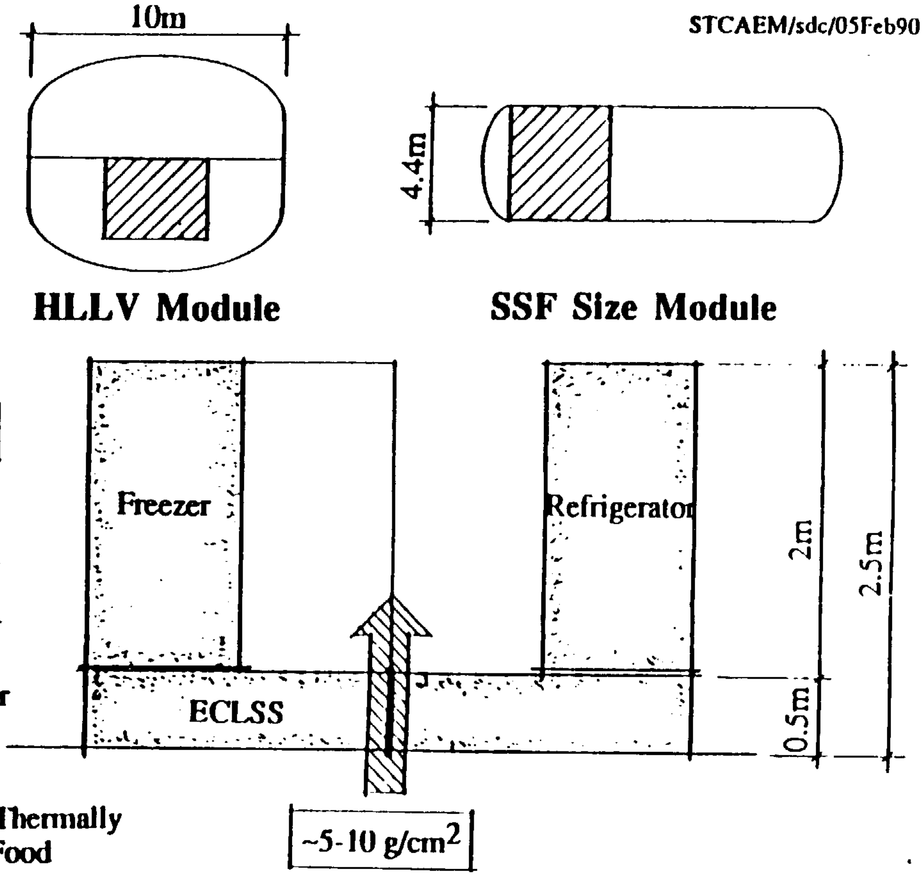
MTV Habitat Galley/Storm Shelter

A primary design consideration for radiation protection is the use of bulk shielding. One novel concept now being evaluated is the use of a storm shelter/galley configuration. The next four charts show a plan and section view of this concept and then provide back drop information. As a first order approximation areal densities are extremely good but do not provide 4 pi protection. It will be necessary to explore further the use of "composite" walls and innovative means of packaging and storing equipment and consumables. Analyses of various potential protection concepts will be key upcoming work.

MTV Habitat Galley/Storm Shelter



Plan



Section

- 40m3 total volume
- 9m3 free volume
- Floor and ceiling require dedicated composite shield panels.

Consumables Provisioning for MTV

- **Stored food** (2.0 kg/crew/d brought from Earth)

0.62	food solids
1.15	65 % water in wet food (water "surplus")
0.23	packaging

- **Storage density**

0.6 t/m3	frozen or thermally stabilized
0.2 t/m3	fresh

- **Potable water** (2.35 kg/crew/d provided recycled by ECLSS)

1.59	drinking
0.76	food prep

MTV Food

Mission constraints

- Maximum mission time 1020 d

- Food preparation and consumption is one of the most critical means available to boost morale and stabilize groups in hazardous, long-duration confinement.

Derived requirements

- At least SSF quality; some actual cooking advisable

- 5 % fresh (controlled atmosphere storage; 1 yr lifetime possible)
  50 % frozen (limited-access deep freeze)
  40 % thermally stabilized
  5 % dried (beverages, soups)

- 3 % supplemental may be grown onboard (not mission-critical)

Synergistic Usage of Consumables

With wet food, advanced water recovery is not required

- Avoids development cost, operational risks of high-energy water systems
- SSF ECLSS with enhanced long-duration reliability is satisfactory for MTV

Consumables are valuable for radiation shielding

- 8.2 t of packaged food available on a 4 crew, 1020 d mission
- Only 530 kg is unrecoverable with SSF ECLSS (fecal solids and water)
- Brine requires minimal biological stabilization
- Food packages stored in blocks; empty blocks become brine containers, filled by ECLSS; manually replaced into storage frame; shield wall continually maintained throughout mission

Combined galley / storm shelter reduces shielding penalties

- Dramatically limits dedicated shielding mass otherwise required
- Temporary sleep accommodations rigged for flare duration
- Separate shelter provisioning not required

Radiation Research Concerns

- Reevaluating the tradeoff between simpler radiation schemes and potentially increased career cancer risks
- Reevaluation of the "conventional" risk assessment now being used
- Development of SPE and dosimetry warning systems
- Trade studies required for realistically selecting and assessing shielding questions such as material, mass, size, and structural integrity
- Evaluate the potential of exacerbating the effects of radiation with a weightless or reduced 'g' environment
- Effectiveness of chemical inhibitors and nutritional supplements
- Evaluation of shielding technologies including: waste water, lightweight composite materials, electromagnetic shielding and propellants
- Analysis of trajectories that may come as close as 0.6 AU to the sun

LifeSat

In terms of the external and internal radiation environments, it will be essential to obtain further data and reliable descriptions of the fluxes and types of primary and secondary particles. The Life Sciences Division is currently planning a reusable, free-flying biological satellite program (LifeSat), that will provide the capability to study the biological effect of radiation dosages and the effectiveness of various shielding materials. Accurate information will be provided on a unique spectrum of radiaton that will be extremly valuable for risk assessment and protection methodology. It has been estimated that a 60-day mission in polar orbit would simulate 5% of a Mars mission in terms of radiation exposure.

LifeSat

Goal:

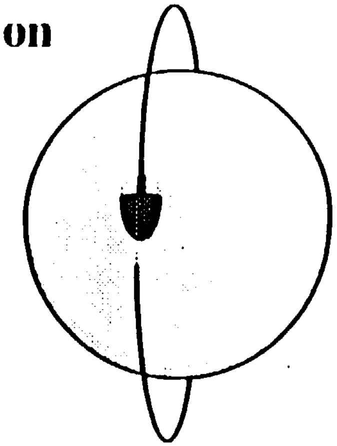
- Understand the effects of microgravity and radiation in the space environment through the capabilities of a reusable free-flying satellite

Objectives:

- Provide accessibility to range of orbits, including polar
- Provide long-duration missions of approximately 30-60 days
- Provide capability to perform research at artificial gravity levels between 0 and 1.5 g

Status:

- Recently awarded Phase B contracts
- Budget estimates assume significant international collaboration
- Schedule support potential FY 1992 New Start



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Boeing Huntsville IR&D Transport Code Development Objectives

STCAEM/mha/07March90

- Research codes available to model primary and secondary radiation effects
- Obtain codes satisfying engineering requirements of various environments and mission profiles
- Become code proficient and generate subroutines that will model a variety of radiation environments
- Develop subroutine to model variety of materials and vehicle geometries
- Code verification
- Document code modifications in users guide

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All manned space missions have inherent risks associated with them. From the time astronauts enter their vehicle to the time they egress, NASA must plan for any number of contingency operations. Provisions for rescue and abort must be made for safety of the crew. As missions depart from low Earth orbit and venture out to distant destinations such as the moon and Mars, new dimensions to rescue and abort procedures must come to bare.

The experience of Apollo 13's mishap has left mission planners and vehicle designers a test of what can happen. This situation has also left us with an understanding of what good contingency planning can do. It is virtually impossible to predict all scenarios that may occur during the course of a mission requiring emergency action to be taken by the crew. Even though emergency operations were defined for the Apollo program, much of what was done during the Apollo 13 mission to save the crew was done so in real-time. The United States prides itself on its ability to contend with problems that arise during a mission. Many high risk scenarios this side of a catastrophic failure or total vehicle destruction must be defined. Astronauts at some point may need to seek shelter or enter a 'safe-haven' region of the vehicle. From this position crew members may be better able to contend with situations that threaten their lives or the successful completion of the mission.

The advantage of lunar missions is that the Moon is closer to Earth than Mars. Should the need arise to make an abort return to Earth, as it did on Apollo 13, the transfer time will be relatively short. On the other hand, interplanetary missions to Mars in which missions durations may be measured in years, will raise some very serious concerns about rescue and abort operations. Some considerations for abort scenarios may be built into the design of the missions, such as a free return or powered swing-by around the target body as part of the flight mechanics. This luxury is not provided during the nominal inbound and outbound legs of the mission however. During these mission phases the crew is more-or-less on their own. In addition to abort procedures that would require crew return to Earth and a scrubbed mission, planners and vehicle designers must also define more 'moderate' contingencies. These would encompass emergency situations that may result from a power failure confined to certain subsystem, for example. Crew members may be required to seek shelter during the course of this situation. From this safe-haven crew members would then deal with the situation at hand. Provisioning, equipment, and tools would be made available. Such a situation may not require mission abort unless the problem was unable to be corrected. In effect, a mission of the magnitude as one to Mars, would require that the crew and vehicle be made as self reliant as possible.

Close examination of abort, rescue, maintenance, and safety operations is important during the course of vehicle development and design. For example, during the Mars descent operations, the Mars Ascent Vehicle (MAV) will have the capability to abort to orbit. This operation would require special considerations be made regarding the separation of the aerobrake. Dynamic maneuvers related to aeroshell separation during a Mars descent abort must provide adequate crew safety. In effect, developing an understanding of potential problems early in a program allows for equal definition of possible actions that may be taken to insure a safe crew return.

Rescue and Abort

While it is acknowledged that a space mission to distant targets such as Mars involves inherent risk, provisions for rescue and abort must be made for the safety of the crew. Some of these considerations may be built into the design of the missions, such as free return or powered swingby around the target planet as part of the flight mechanics. This, however, can not be done during the outbound or inbound transit legs of the mission. The crew is on their own and must be provided with the tools and operations to cope with contingency situations during this time. Such considerations must be "built in" to the design of the vehicle from the allowance for passage in EVA suites to the presence of an in-transit repair and maintenance capability. They will have to be made as self reliant as possible.

Rescue and Abort

Flight Dynamics:

- Mars swingby and return capability
  - Powered or Unpowered return swingby at Mars
  - Consumables for from 443- 1020 days will include the projected abort time

Structure:

- A "storm shelter" will be provided for Solar Proton Events
- Major circulation points in the vehicle will be large enough to allow two EVA suited personnel to pass
- Two means of egress from any one area will be provided
- Habitat repressurization capability
- There shall be at least 2 functionally independent pressurized areas for emergency conditions with 2 EVA suites stored in each of these areas
- ECCV can serve as a "lifeboat" close to Earth

Operations:

- MAV will have abort-to-orbit capability during landing descent
- MEV, MTV, MAV will have autonomous rendezvous and docking capability
- A limited in-flight repair capability will be provided
- Components on life critical systems will be redundant and common to similar systems through out the vehicle

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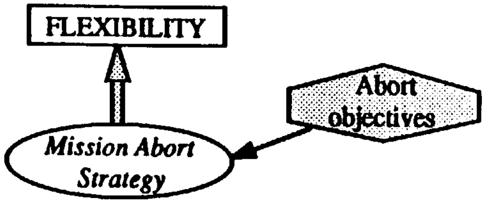
Generalized Abort Strategy Priorities
for Man Rated Vehicles



Statement of objectives of abort strategies in generalized terms

1. Primary objective:

Safe Crew Return to Earth

"Immediate action to place crew into best possible position to accomplish a safe return to Earth or SSF"

2. Secondary objective: (addresses mission continuation after some degree of recoverable vehicle failure)

Retention of Full Safe Crew Return Capability

" Change of mission goals and/or tasks so as to retain the capability for a safe crew return to Earth at any point during the mission" (Continuation on the condition that capability to perform primary abort objective is never compromised)

3. Plausible exception to primary objective for a long term SEI program: given establishment and maturity of frequent Lunar missions and operations:

Safe Haven Option

"Immediate action to place crew into best possible position to reach a prepared safe haven to await crew rescue by independent spacecraft system"

Mac Chart/Disk #7/generalized abort strategy priorities/12-12-90

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Classification of Man Rated Spacecraft Failure Modes

1. **Indicated Failure or Potential Failure (False indication):** *monitor or controller erroneously indicates failure of healthy, properly functioning system*

2. **Potential Failure (True indication):** *legitimate system or subsystem indication of failure, or imminent failure*

   Level of Urgency
   a. anticipated to be of insignificant nature to warrant any near term (duration of present mission) action
   b. anticipated to be of significant nature, but considered within the capability of performance margin/redundant system
   c. anticipated and averted/resolved by inflight repair/correction of faulty system

3 **Recoverable failure:** *nature of failure does not compromise capability for safe crew return abort*

   Level of Urgency
   a. mission abort averted by adequate performance margin of primary operating systems
   b. mission abort averted by use of dedicated backup or redundant system
   c. mission change necessitated; performance margin/redundant system inadequate for continuation of mission as originally planned; necessitates reduced, or eliminates vehicle activity to some degree in order to retain the capability of the vehicle system for safe crew return to Earth or SSF
   d. mission abort; nature of failure requires immediate and complete redirection of mission to safe crew return abort mode. The Apollo 13 failure that precluded the LEM surface mission, would fall under this category. That failure was 'recoverable' from the point of view of the combined Command module/Service module/LEM system, though probably 'nonrecoverable' for the Command/Service module without the aid of the LEM

4. **Non-recoverable failure:** *nature of failure precludes successful accomplishment of safe crew return to earth* necessitates independent spacecraft for rescue where possible

Mac Chart/Disk #7/classification of failure modes/12-12-90

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Cryo/Aerobraked Vehicle

I. Introduction.

The reference chemical Mars mission utilizes a cryogenic H_2-O_2 fueled vehicle which employs high energy aerobraking for capture at Mars, and an ECCV for crew capture at Earth. High energy aerobraking is required of both the MTV and MEV aerobrakes. The cryo/aerobraked vehicle served as the baseline for the NASA 90 day study, completed in October, 1989.

II. Reference vehicle design and operations.

The reference cryogenic/aerobraked vehicle is assembled in a SSF orbit. The TMI stage is assembled with a core stage consisting of a single H_2/O_2 tankset, advanced space engines (4), structure, and plumbing. Four modular tanksets are spaced radially around the core tankset to form the remainder of the TMI stage. The Stage is jettisoned after the TMI burn. At 50 to 60 days before Mars arrival, the MEV executes a separation burn adequate to ensure its arrival and capture at Mars 1 day before the MTV. After the vehicles capture into the same orbit, the MTV and MEV rendezvous , and the crew transfers to the MEV for descent to the Mars surface. The MEV descent into the Martian atmosphere is slowed by the aerobrake, which is jettisoned shortly prior to landing. The descent engines fire through an opening in the aerobrake created by jettisoning the engine bay doors on the aerobrake. After a 30 day surface stay, the crew boards the ascent vehicle, which ascends to rendezvous with the MTV. After crew transfer to the MTV, the ascent ship is jettisoned, the TEI burn executed, and the TEI stage jettisoned. About 1-2 days before Earth arrival, the crew transfers into the ECCV, along with any science or surface sample payload. The ECCV then either captures into a SSF orbit, or executes an Apollo style direct entry at Earth.

The reference vehicle configuration, shown in detail in the following charts, consists of the MTV, MEV, and TMI stage. The MTV consists of the transfer habitat, Mars departure propulsion stage, ECCV, airlock, and Mars capture aeroshell. The MTV is packaged to ensure that all MTV components are behind the wake protection envelope provided by the aeroshell. The MEV consists of the descent stage, ascent stage, surface cargo, and landing aeroshell. The MEV is similarily packaged to ensure placement of components inside the wake protection envelope of the landing aeroshell. The TMI stage consists of 5 LH_2-LOX tanksets, 4 advanced space engines, and associated plumbing and structure. Overall vehicle dimensions are ~30 m diameter x ~50 m length. The 2018 cargo vehicle consists of a TMI stage, and two MEV's loaded with ~93 mt of surface cargo.

III. Reference Cryo/Aerobraked Vehicle Mass Statement

The remainder of the information in this section consists of summary mass statements for the reference cryo/aerobraked piloted and cargo vehicles, and a detailed mass statement for the reference piloted vehicle. The detailed mass breakdown includes rationale and design assumptions used in constructing the model. Similar assumptions and rationale were used in constructing the cargo vehicle model, but are not presented here. These assumptions, where applicable, were similar to those made for the piloted vehicle.

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Cryogenic/Aerobrake (CAB) Reference Configuration

Introduction

The cryogenic/aerobrake (CAB) concept was used as the NASA 90-day Study reference vehicle. It offers conceptual continuity with the mainstream Mars transportation studies performed over the last several years. Its only major new technology development is high energy aerobraking (HEAB) for planetary capture, but the concept also requires a high-thrust cryogenic space engine. Being able to land on Mars using the CAB concept requires a successful rendezvous between separately captured vehicles in Mars orbit.

Nominal Mission Outline

- The vehicle is assembled, checked out and boarded in LEO
- The TMI burn occurs and the TMIS is jettisoned
- MTV/MEV coasts to Mars
- MTV and MEV separate 50 days prior to Mars capture
- The MEV aerocaptures robotically a day ahead of the MTV, providing last-minute verification of atmospheric conditions and targeting
- The MTV captures, followed by rendezvous in the parking orbit with the MEV
- The landing crew transfers to the MEV and checks it out
- The MEV descends to the surface, jettisoning its aerobrake prior to landing
- After surface operations, the ascent vehicle (MAV) leaves its descent stage and surface payloads, ascends to orbit and docks with the MTV for crew transfer
- The MAV is jettisoned in Mars orbit, and the TEI burn occurs
- The MTV coasts back to Earth
- The crew transfers to a modified ACRV (MCRV), jettisons the MTV and performs a direct entry at Earth (optional: the entire MTV aerocaptures into a LEO parking orbit for refurbishment and re-use)

Vehicle Systems

The vehicle consists of three main elements: the Mars Excursion vehicle (MEV), the Mars Transfer Vehicle (MTV) and the Trans-Mars Injection Stage (TMIS).

Mars Transfer Vehicle (MTV)

The MTV configuration shown consists of a transit habitat sized for 4 crew, an aerobrake, and a TEI propulsion system. The transit hab is located centrally in the aerobrake with an external airlock and an MCRV attached to the top (in the configurations shown, an Apollo-style ECCV was used to represent the MCRV). The airlock allows access to the MEV crew cab and surface habitat during all phases of the transfer mission until the MEV separation 50 days prior to Mars arrival. The MCRV is used for mission scenarios featuring direct-entry crew return; these scenarios expend the entire MTV upon return to Earth. In a reusable mode, the entire MTV would be aerocaptured back at Earth for refurbishment and re-use; a second airlock would be located in place of the MCRV. The aerobrake is of identical geometry and construction as the MEV aerobrake, but is stronger and heavier due to its larger payload mass, and does not require any engine doors. The propulsion system (TEI) is divided symmetrically into two tank-stacks straddling the transit hab, like the MAV tankset configuration. The propulsion system is oriented at an angle relative to the aerobrake axis, with the two engines aimed out the rear of the aerobrake, to avoid TPS penetrations while still permitting mass-balanced operation during the burn.

Trans-Mars Injection Stage (TMIS)

The TMIS consists of a core unit with four advanced space engines (ASE), avionics and cryogenic propellant tanks, and provision for up to four "strap-on" propellant tanksets. This configuration allows propellant cross-feeding in the case of engine-out, and modular accommodation of the entire stage's performance according to the mission opportunity requirements. Keeping the engines close together on the core stage allows tracking the CM during an engine-out condition via gimballing. This strategy avoids either opposite-shutoff (leading to long burn times and greater gravity losses), or a requirement for extra structure (a 125m truss) between the propellant tanks and engines to allow CM tracking. The TMIS accounts for about 75 % of the total IMLEO, a substantial per-mission resupply cost.

Mars Excursion Vehicle (MEV)

The reference MEV is a manned lander that can transport a crew of 4 to the surface. It consists of a surface-stay habitat module (roughly SSF-module size), an airlock, 5 t of surface-science payload, a cryogenic descent propulsion system with four engines and bus structure, and the ascent vehicle (MAV). The MAV consists of a short-duration crew cab, and cryogenic ascent propulsion system with two engines. All propellant tanks are mass-balanced around their maneuver CMs so that no lateral CM shifting occurs. The entire MEV is packaged in a rigid, truncated-hyperboloidal aerobrake with L/D = 0.5, to which it is attached at eight points (four bus-frame corners and four landing-gear footpads). The aerobrake is fitted with doors which open to allow the descent engines to extend and ignite prior to aerobrake separation (allowing full benefit of the brake's drag). The brake is then jettisoned as the landing gear extend prior to terminal approach and hovering touchdown.

Dominant configuration constraints for the MEV are as follows:

- Payload manifesting
- Surface access
- Crew visibility
- Contigurous crew volumes
- Short vehicle stack
- Engine-out capabilities
- On-orbit assembly

Payload manifesting is mainly a proximity and mass balance issue. The surface habitat and airlock, which is the bulk (80%) of the payload, require access to the ascent crew cab and the surface, as well as being mass balanced for proper flight. The science payload requires surface access for ease of unloading. Docking is facilitated by placing the crew cab high in the vehicle stack. The flight deck window is located to provide viewing to the surface for landing as well as to the upper hatch for docking. Keeping crew volumes contiguous allows access during flight for check-out procedures and simulation training. The vehicle stack is kept as short as possible for aerobrake wake protection, which tends to conflict with having the center of mass (CM) as high as possible, desirable for a small engine gimbal-angle to provide minimal steering loss in an engine-out scenario. A high CM within a short stack is accomplished by placing the dense ascent LOX high in the configuration. Finally, although the dominant constraints for the MEV derive from its performance at Mars, consideration has been given to its ETO launch. It is configured to be launched in a few, large, pre-integrated systems for minimal on-orbit assembly. For example, the ascent vehicle can be launched intact in a 10 m diameter shroud, while the descent structure can be launched in 2 sections for fairly simple on-orbit assembly and integration.

Cryo/Aerobraking

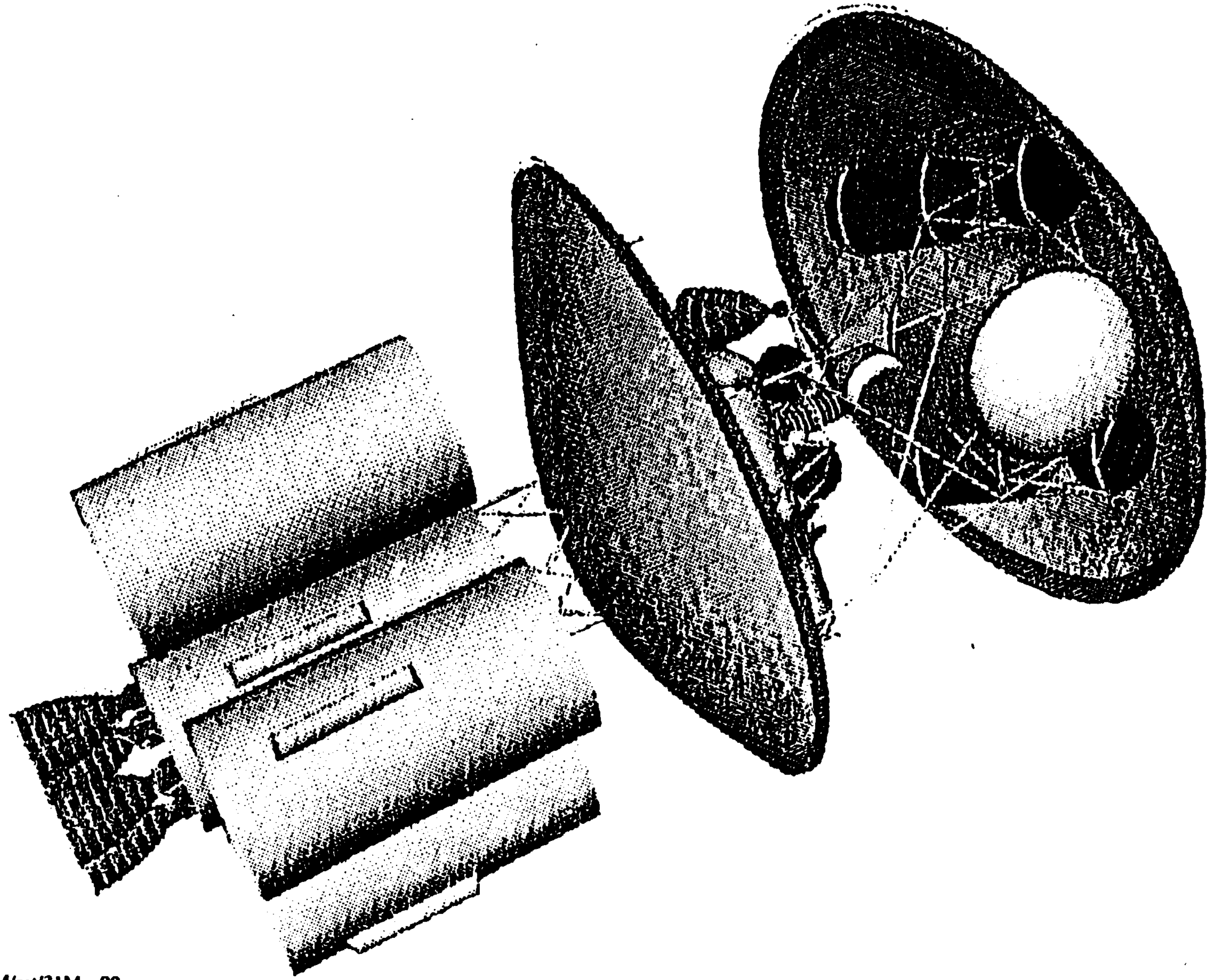
Trades and Rationale

- A TMI core stage with four 200 klbf class advanced engines and four "plug-in" propellant tanks. Tanks and core stage rendevous and dock automatically. Core stage provides simple plumbing and good engine out performance.
- High energy aerobraking for MTV & MEV capture at Mars

Mission Modes And Operations

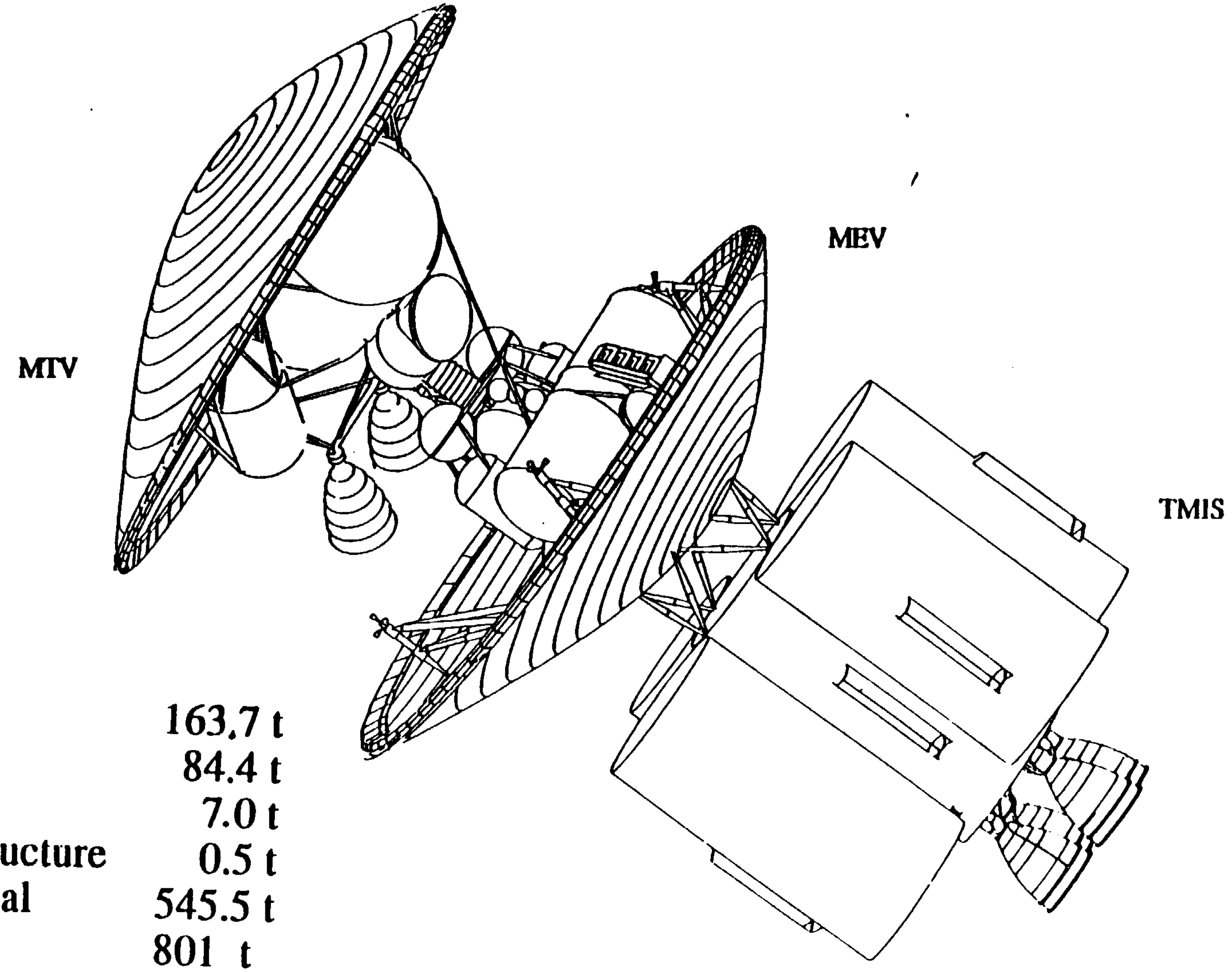
- NASA 90 day study baseline.
- Vehicle assembled in SSF orbit.
- TMIS jettisoned after TMI burn.
- MEV/MTV separate prior to Mars aerocapture.
- Crew transfer to MEV/Aerobrake after MTV/MEV rendevous.
- MEV/Aerobrake entry. Aerobrake jettisoned prior to landing.
- Crew cab ascent after surface mission, leaving lander and surface hab.
- Crew cab left in Mars orbit after rendevous, docking and crew transfer.
- TEI burn.
- Crew transfer to ECCV shortly before Earth arrival
- ECCV capture and SSF rendezvous or direct landing

Cryo / Aerobraking
3D CAD MODEL



STCAEM/crt/31May90

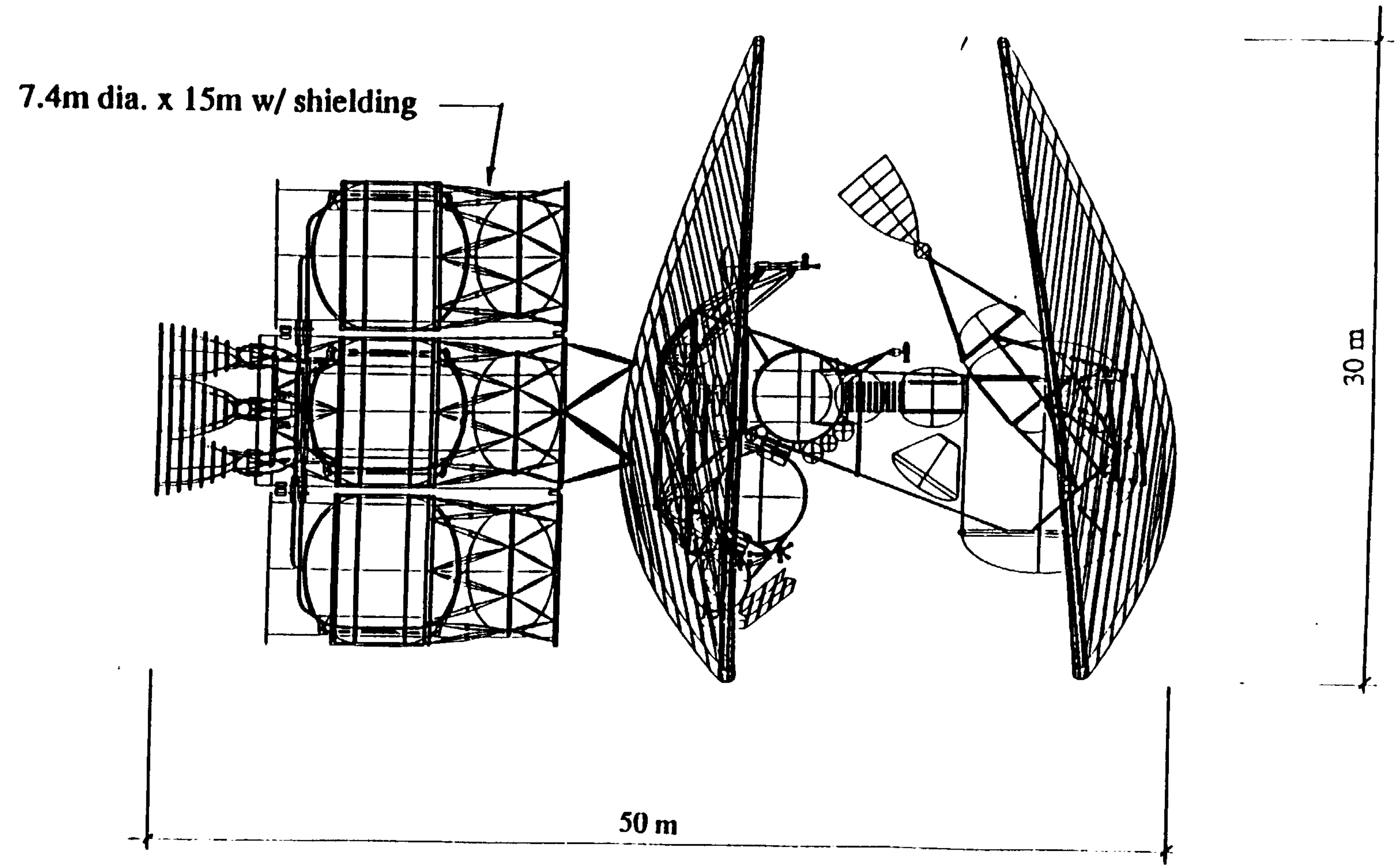
Cryo/AB Reference Configuration



MTV total	163.7 t
MEV total	84.4 t
ECCV	7.0 t
Interstage Structure	0.5 t
TMI stage total	545.5 t
IMLEO	801 t

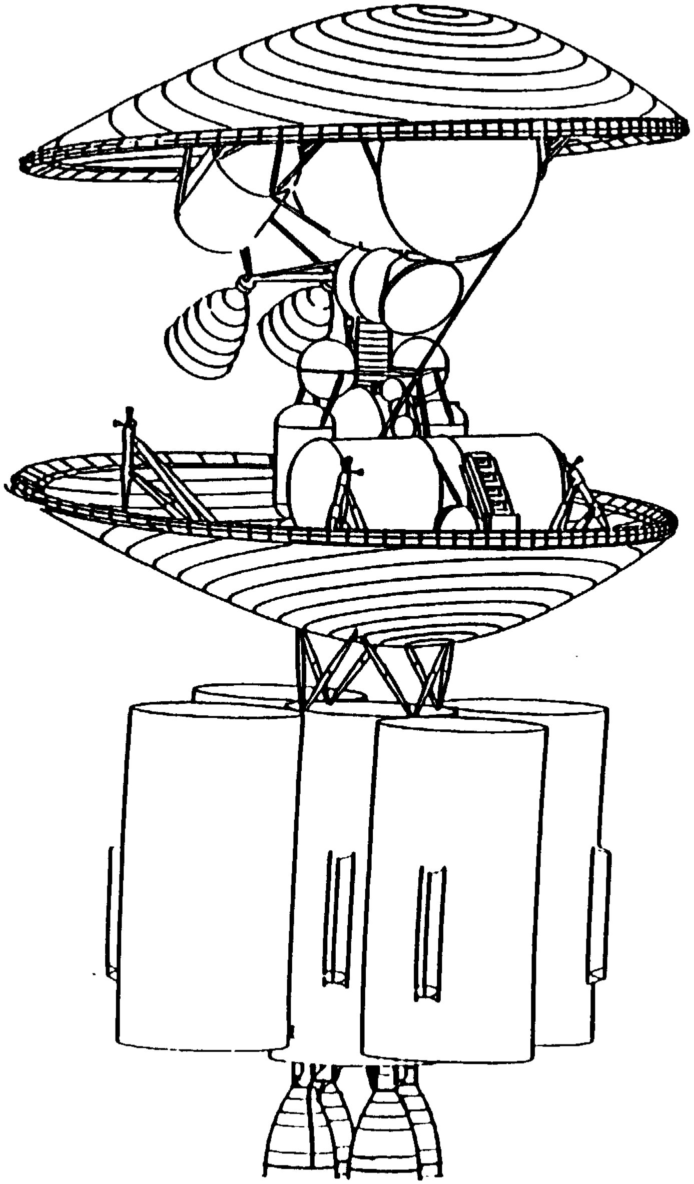
STCAEM/sdc/31May90

Cryo/AB Reference Configuration



STCAEM/sdc/01June90

Reference Cryo/aerobrake Mass Statement



Element	mass (kg)
MTV Mars aerobrake	23758
MTV crew hab module 'dry'	28531
MTV consumables & resupply	7096
MTV science	1000
MTV propulsion stage	18206
MTV propellant load	85141
MTV total	163732
MEVMars capture & desc aerobrake	15138
MEV ascent stage	22754
MEV descent stage	21457
MEV surface cargo	25000
MEV total	84349
ECCV	7000
Cargo to Mars orbit only	0
MTV-TMI interstage wt	500
TMI inert stage wt	54560
TMI propellant load	490950
TMI stage total	545510
IMLEO	801090

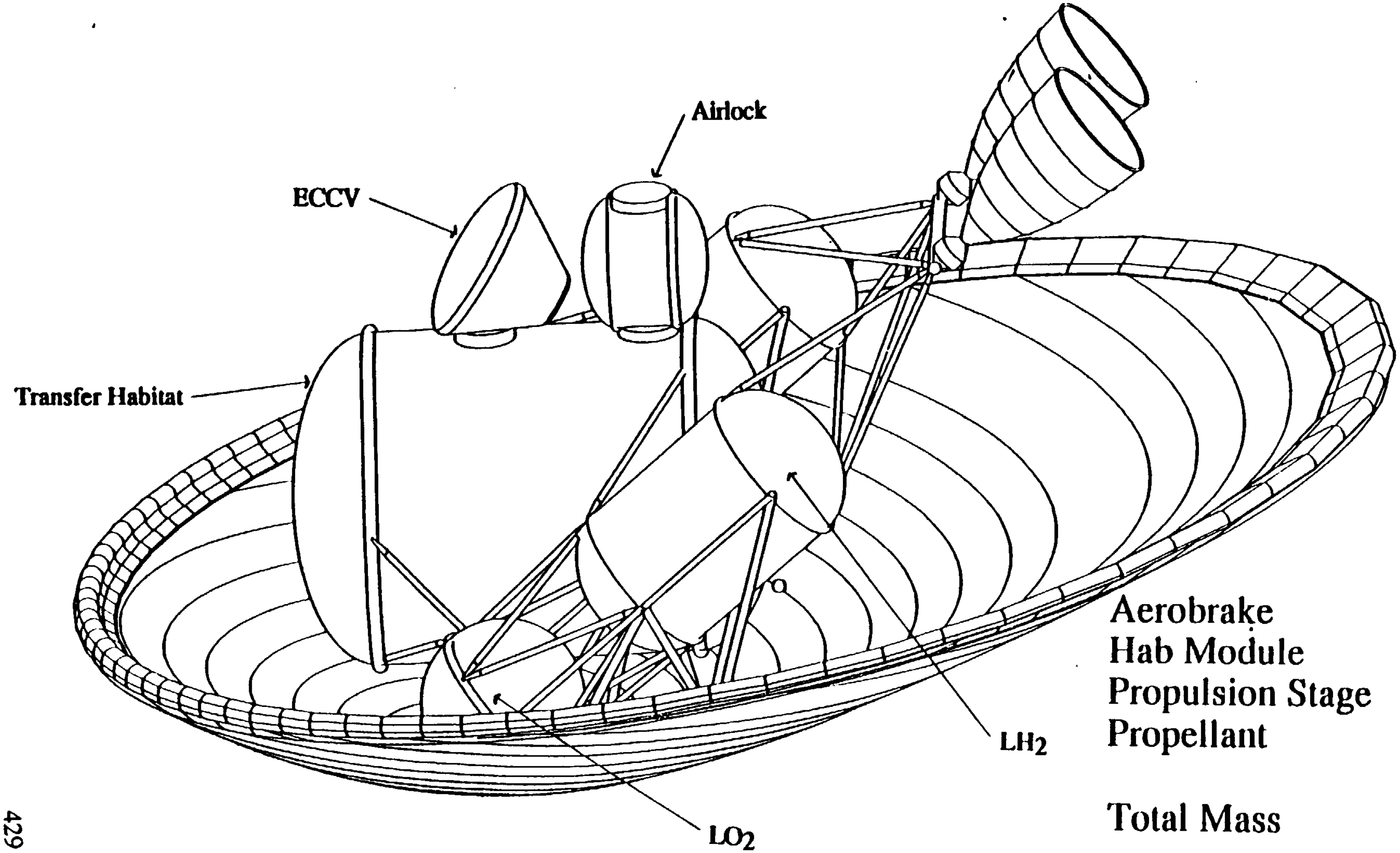
Mac chart: M Ref chem/ab cover pg
synthesis model run# marschemmtv.dat;21

STCAEM/crf/31May90

MTV Reference Configuration

The facing page illustrates the reference MTV / Aerobrake configuration. Shown are the transfer hab, ECCV, airlock and TEI propellant tanks and engines. The tank and engine structure is configured as shown to allow docking access by the MEV, and to allow the assembly to remain within the protected wake region.

MTV Reference Configuration



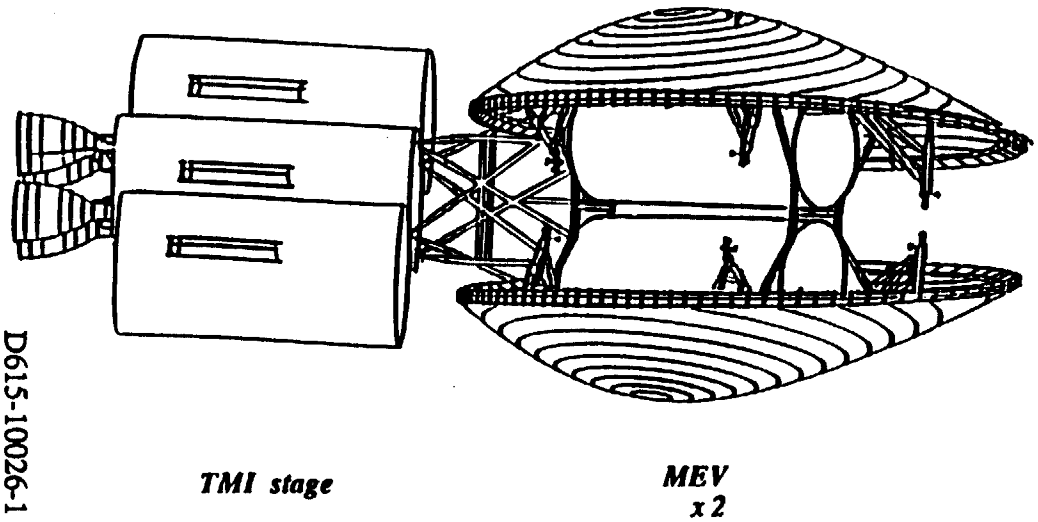
Aerobrake	23.8 t
Hab Module	36.6 t
Propulsion Stage	18.2 t
Propellant	85.1 t
Total Mass	163.7 t

STCAEM/sdc/31May90

Cargo Chem/aerobrake Veh for one way 2018 Conjunction Mission

Unmanned, 2 cargo landers (46.5 t surf cargo each), 10 t navigation set, no MTV propulsion stg, TMI stg Isp=475

Revision 2 7/30/90



Element	*mass (kg)*
MTV Mars aerobrake	0
MTV crew hab module 'dry'	0
MTV consumables & resupply	0
MTV science	0
MTV propulsion stage	0
MTV propellant load	0
MTV total	0
MEV Mars orbit capture & desc aerobrake	15138
MEV descent stage	21457
MEV surface cargo	46457
MEV total	84349
x 2	**168968**
ECCV	**0**
Cargo to Mars orbit only (navigation set)	10000
TMI inert stage wt	25770
TMI propellant load	231920
TMI stage total	**257690**
IMLEO	**436658**

Mac chart: Cargo chem/ab 2018 wt cover pg
Veh synthesis model run #: marschemmtv.dat;37

TMI stg - Reference MTV for 2015 Chem/Aerobrake Veh

ECCV Return, 4 x 200k lbf advanced space engines; Isp = 475 sec

Revision 2 5/22/90

		Element	*mass (kg)*	*Rationale*
TMI Pay load	[556]	Tot MTV Mars dep stg	103347	See mars dep stage wt statement
	[380+179]	MTV Crew hab mod sys	36627	See MTV crew hab module wt statement
	[230]	ECCV	7000	4 man apollo type entry vehicle; MTV expended
	[106]	MEV	84349	4 man, 30 day stay, 25 t surface cargo
	[159]	Outb 'to-Mars-orbit' cargo	0	communication sat's taken on precurser mission
	[1292]	Mars Site Recon Vehicle	0	Not taken for Ref 2015 mission
	[163]	MTV-TMI interstage wt	500	Structural member joining TMI to MTV
		MTV Mars capture aerobrake:		Structural design assumptions:
		• Primary spar weight	4239	200ksi spar strength
		• Secondary spar wt	3434	22.5 inch spar depth
		• Honeycomb wt	12785	note: 200ksi may require additional material technology development efforts
		• TPS wt	3300	
	[169]	Total:	23758	
	[168]	***Tot TMI stg 'Payload wt'***	255581	TMI propulsive stg injects this wt into hyperbolic trajectory
TMI stage	[172-173]	TMI stage inert	54560	0.9 propellant fraction
	[173]	TMI propellant load	490950	TMI stage tanks topped off before ignition, no boiloff accounted for
	[172]	***TMI stage total mass***	545510	4 x 200k lbf advanced spaace engines, Isp=475 sec
	[171]	***IMLEO*** ***initial mass in low Earth orbit***	801090	

STCAEM/bhl/31May90

synthesis model run #: marschemmtv.dat;21
Mac chart M Ref TMI wt-rationale

Crew hab mod - Reference MTV for 2015 Chem/aerobrake Vehicle

Crew of 4, 565 day total trip time *Revision 2 5/22/90*

	Element	*mass (kg)*	*Rationale*
[360]	Structure	8351	Cyl length: 9 m, dia: 7.6 m, ellip ends, 3 levels; tri grid w beam supports. Tens. ties
[363]	ECLSS	4256	SSF derived with same degree of closure, sized for crew of 4 for 565 days
	Command/Control/Power		
[364]	• Internal	1159	ECWS, DMS, batteries, other avionics/computing/monitoring eq, conditioning equip
[1281]	• External Power	1539	Solar array,boom, power distribution,power management,fuel cell system
[368-316]	Man systems	4121	Wts -all sys:SSFderived(as a funct. of crew size&occupancy time)for Mars missions
[316]	Crew & effects	440	110 kg per person including personal belongings
[373]	Spares/Tools	1496	Subsys component level spares. Life crit sys are 2 fault tolerent (approach of SSF)
[247]	Radiation shelter	1802	Provides 10 g/cm2 protection + 3-5 g/cm2 provided by vehicle structure and equip
[377]	Weight growth	2973	15% weight growth for dry mass excluding crew & effects and radiation shelter
[378]	Airlocks	1530	2 x 765 kg external airlocks (shuttle type airlocks modified for MTV mission)
[330]	EVA suits	0	EVA suits weight counted in MEV ascent cab weight statement
[160]	TTNC & GN&C platforms wt	863	3 platforms
Sum	***MTV 'dry' crew hab mod wt***	28531	'dry' hab module represents structure and support systems equip & hardware that are dependant on crew size and independant of mission duration
[371]	*On board equip resupply	1304	Based on adjusted SSF resupply reqts for pot w, hyg w, ARS,TCS/THC &WMS
[398]	*Consumables	5792	Crew of 4 for 565 days; food:2.04 kg/man/day, food pkg:0.227, pharmaceuticals: 0.25 other: 0.291 Clothes: 42 kg/man. food vol: 0.0055 m3/man/day, other: 0.0018.
[380]	***MTV crew mod 'wet' wt***	7096	
[165]	*Transfer science equipment	1000	Inb and outb MTV science hardware and supplies
	Remote Manipulator-arm Sys	0	all large external self assembly hardware left in LEO
380+179	***MTV crew mod & support systems weight***	36627	This wt refects the Boeing ref crew of 4 mod loaded for the 2015 opposition mission. The mod 'dry' wt represents a SSF type closed ECLS Sys (air >99%, water >95%) that serves the crew with 2 fault tolerence on all life critical sys except structure. Its wt varies primarly with crew size. consumables wt varies with crew size and mission duration.

** MTV hab mod consumables, resupply, and transit science dependant on mission duration, and free abort requirement. i.e. crew mod 'wet' wt will vary for different missions*

Mac chart: M Ref MTV mod wt-rationale
synthesis model run# marschemmtv.dat;21

STCAEM/bbd/31May90

Mars departure stg - Reference MTV for 2015 Chem/Aerob Veh

Crew of 4, 2 advanced space engines; Isp = 475 sec

Revision 2 5/22/90

		Element	*mass (kg)*	*Rationale*
TEI stage inert	[154]	Fuel tank	5424	2 SIC/Al metal matrix tanks, 37 ksi working stress, tank MEOP=175kPa
	[155]	Oxygen tank	3100	2 SIC/Al metal matrix tanks, 37 ksi working stress, tank MEOP=175kPa
	[158]	MLI/meteor shield	1082	MLI: density = 32 (kg/m3); 100 layers at 20 layers/cm. Meteor Shield: 2 (kg/m2)
	[159]	Frame structure	5132	5% of MTV propellant + 5% of MTV stg inert mass
	[518]	Main propulsion	794	2 x 30k lbf advanced space eng's: Isp=475 s, high AR nozzle not extendible
	[183]	RCS inert	300	Scaled from RCS propellant
	[160]	Mass growth	2374	15% growth for inert stage
	[161]	*Mars dep stg 'dry' wt*	18206	
TEI prop, boil-off & totals	[118]	MTV RCS propellant	699	Storable: N2O4/MMH propellant, Isp=280 sec, MTV RCS dV=30 m/sec
	[122]	MTV inb midcourse burn prop	1256	delta V: 90 (m/sec); burn done with MTV Mars dep main propulsion
	[128]	Mars dep usable prop	71525	LH2/LO2, MR=6:1, *Mars dep dV: 3400 m/sec* usable=prop req after outb & inorbit
	[545+546]	In orbit Mars dep prop boiloff	426	boiloff; 30 day boiloff period; calculated with Boeing's 'CRYSTORE' program
	[554]	*tot onboard prop at Mars arr*	73906	
	[121]	Outb midcourse burn prop	6709	midcourse maneuver delta V: 120 (m/sec); burn done w MTV main propulsion
	[498+499]	Outb Mars dep prop boiloff	4526	335 day outbound trip time.
	[555]	*MTV propel expended outb*	11235	
	[556]	*Tot M dep propulsive stg wt (at time of E dep burn)*	103347	

synthesis model run #: marschemmtv.dat;21
Mac chart M Ref MTV vch wt-rationale

STCAEM/bbd/31May90

Desc stage - Reference MEV for 2015 Chem/Aerobrake Vehicle

Crew of 4, 30 day stay, 4 advanced space engines; Isp=475 sec, 25 t surf cargo

Revision 2 5/22/90

			Fuel / Oxidizer	
Desc stage inert	[98/99]	Single tank wt	242/126	2 SiC/Al metal matrix tanks for each, 37ksi wk stress, MEOP=175 kPa, min t=3.5mm
	[124/125]	Meteoriod Shield	31/16	One 0.40 mm sheet of Al
	[122/123]	MLI	47/24	MLI: density = 32 (kg/m3); 100 layers at 20 layers/cm.
	[126/127]	Vapor Cooled Shields	37/19	1 VCS at 2 x 0.13mm Al outer sheet w 0.57 kg/m2 honeycomb core
	[0/0]	Vacuum shell	0/0	not on desc tanks
	[2x1316]	Propel line wt	50/50	50 kg per tank
	[132/133]	Tank wt growth	41/23	15% wt growth
	[128/129]	Sum single tank inerts	448/258	Total single tank + tank inert wt
	[130/131]	***Tot: Fuel & Ox tanks:***	**896/516**	2 LH2 & 2 LO2 tanks
	[501]	Main propulsion	1127	4 x 30klbf Adv eng's: Isp=475 sec, w extendible/retractable nozzles
	[102]	Asc frame & struc wt	567	4% of desc stage stg wt + 2% of surf crew mod mass
	[103]	Landing legs	1487	3% of total landed mass
	[1273,526]	RCS inert	331	Estimate from RCS prop load
	[104]	Propul, frame wt growth	490	15% of total inerts
	Sum	***Desc propul & frame inert***	**4002**	
Prop loads	[91+92]	Desc usable Prop	13477	*Desc propulsive veh dV= 931 (m/sec) from 250 km periapsis alt. by 1 sol orbit.*
	[0]	Desc boiloff	0	
	[101]	Desc RCS prop	2566	
	Sum	***Total Desc propellant load***	**16043**	N2O4/MMH prop, Isp=280 sec, *desc RCS dV=100 (m/sec)*
Aero brake wt		***MEV aerobrake:***		Structural design assumptions:
		• Primary spar wt	2484	200ksi spar strength
		• Secondary spar wt	2596	22.5 inch spar depth
		• Honeycomb wt	6758	note: 200ksi may require additional material technology developement efforts
		• TPS wt	3300	
	[78]	Total:	15138	
	[77]	***Surface crew hab module***	**25000**	Level II Requirement: surf modulw, surf science & surf stay consumables
	[61]	***Asc veh total mass***	**22754**	from 'Asc stage' wt statement page
	[106]	***MEV mass***	**84349**	all masses in kg

synthesis model run#: marslander dat. l

STCAEM/bbd/23May90 Mac chart: M Ref MEV decs veh wt-ratio

STCAEM/bbd/31May90

Ascent Cab - Ref MEV for 2015 Chem/Aerobrake Vehicle

Crew of 4, 3 day occupancy time Revision 2 5/22/90

	Element	mass (kg)	Rationale
Cab ECLSS	Atmospheric Revitization Sys/ Trace contaminant control assembly	123	CO2 adsorption unit, expendable LiOH cartridge Pre & postsorbent beds,catalytic oxidizer for particulate & contaminant control
	Atmosphere Control System	62	Total & partial press control; valves,lines & resupply/ makeup O2 & N2 and tanks
	Atmos. Composition & Monitor Assem.	55	O2 & N2 monitor for ACS, particulate & contaminant monitor for ARS
	Thermal Control Sys	40	Temp control: sensible liq. heat exchanger, ext radiator wt included in 'secondary structure' mass
	Temp. & Humidity Control	240	Condensing heat exchanger, fans, ducting
	Water Recovery and Management	45	Stored Potable water only
	Fire Detection & Suppression Sys.	113	Automatic sys w manual extinquishers as backup
	Waste Management Sys and Storage	-	Considered part of 'Man Systems'
	Asc cab ECLSS mass	***678***	Apollo style open ECLSS system
Cab Structure	Primary/Secondary Structure	519	Overpressurized (20 psia) on launch for structural integrity. Stiffening rings added at cylinder/endcap interface for added strength. Skylab derived triangular grid floor with beam supports on 6" centers. Support ring interface on pressure vessel to carry loads imposed by the floor and equipment during launch to aerocapture.
	Berthing ring/mechanism (1)	139	
	Berthing interface plate (1)	90	
	Windows	90	
	Couches	80	
	Hatches (2)	80	
	Asc cab Structure mass	**998**	

synthesis model run number: marsntr.dat
Mac chart:M Ref MEVasc cab wt rati

Asc stage - Reference MEV for 2015 Chem/Aerobrake Vehicle

Crew of 4, 30 day stay, 2 advanced space engines; Isp=475 sec *Revision 2 5/22/90*

Ascent Cab		Structure	998	SSF dia center cyl section w ellip ends. Stiffening rings added. See 'Structures pg'
		ECLSS	678	Open sys:CO2 adsorption unit, stored H2O,O2,N2, no airl., no hyg w. see 'ECLSS pg'
		Command/Control/Power	330	Power: fuel cells
		Man systems	82	Waste management sys/waste storage/medical equp.
		Spares & tools	192	Subsystem component level spares
		Wt growth	376	15% growth for dry mass
		Asc 'dry' mass	2656	Total cab dry mass
		Consumables (food & water)	62	Minimum; food and water only; 3 occupancy
		Crew/effects/EVA suits	760	Crew of 4, 100 kg EVA suit per crew member
	[22]	*Ascent cab gross mass*	3478	
			Fuel / Oxidizer	
Ascent stage inert	[45/46]	Single tank wt	312/140	2 SiC/Al metal matrix tanks for each, 37ksi wk stress, MEOP=175 kPa, min t=3.5mm
	[68/69]	Meteoriod Shield	40/18	One 0.40 mm sheet of Al
	[50/51]	MLI	59/26	MLI: density = 32 (kg/m3); 100 layers at 20 layers/cm.
	71/72/112/113	VCS & Vacuum shell	47/21	1 VCS and 1 Vac shell: both 2 x 0.13mm Al outer sheet w 0.57kg/m2 honeycomb core
	[2x1316]	Propel line wt	50/50	50 kg per tank
	[116/117]	Tank wt growth	62/31	15% wt growth
	[73/74]	Sum single tank inerts	617/307	Total tank & tank inert wt
	[114/115]	*Tot: H2 & O2 tanks:*	1234/614	2 LH2 & 2 LO2 tanks
	[500]	Main propulsion	564	3 x 30klbf Adv eng's: Isp=475 sec, w extendible/retractable nozzles
	[118]	Asc frame & struc wt	478	3% of total asc stg propellant wt
	[1274,525]	RCS inert	122	Estimate from RCS prop load
	[54]	Propul, frame wt growth	174	15% of total inerts
	Sum	*Asc propul & frame inert*	1338	
Prop loads	[60]	Asc usable propellant	15500	*Asc veh dV= 5319 (m/sec) to 250 km periapsis alt. by 1 sol orbit.*
	[56+58]	Asc boiloff	418	50 day sep from MTV before M arr+ 30 day surf stay;calc:Boeing 'CRYSTORE' program
	[52]	Asc RCS prop	172	N2O4/MMH prop, Isp=280 sec, *Asc RCS dV =35 (m/sec)*
	Sum	*Total Asc propellant load*	16090	
	[63]	*Asc veh total mass*	22754	all masses in kg

synthesis model run#: marslander.dat;108

Mac chart: M Ref MEV asc veh wt-rationale *STCAEM/bbd/22May90*

STCAEM/bbd/31May90

Cryogenic All-Propulsive Vehicle

I. Introduction.

The all-propulsive chemical Mars mission utilizes a cryogenic H2-O2 fueled vehicle which employs low energy aerobraking only for MEV descent, and an ECCV for crew capture at Earth. High energy aerobraking is not required for any mission phase. All missions are conjunction-class, with a ~300 day stay at Mars, 30-90 days of which is spent on the surface.

II. Reference vehicle design and operations.

The cryogenic all-propulsive vehicle is assembled in a SSF orbit. The TMI stage is assembled with a core stage consisting of a single H2/O2 tankset, advanced space engines (4), structure, and plumbing. Three modular tanksets are placed in line with the core tankset to form the remainder of the TMI stage. The Stage is jettisoned after the TMI burn. At Mars arrival, the Mars Orbit Capture/ Trans-Earth Injection (MOC/TEI) engines are fired to provide a propulsive capture of the vehicle. After capture and orbit insertion, the crew transfers to the MEV for descent to the Mars surface. The MEV descent into the Martian atmosphere is slowed by the aerobrake, which is jettisoned shortly prior to landing. The descent engines fire through an opening in the aerobrake created by jettisoning the engine bay doors on the aerobrake. After a 30-90 day surface stay, the crew boards the ascent vehicle, which ascends to rendezvous with the MTV. After crew transfer to the MTV, the ascent ship is jettisoned, the TEI burn executed, and the MOC/TEI stage jettisoned. About 1-2 days before Earth arrival, the crew transfers into the ECCV, along with any science or surface sample payload. The ECCV then either captures into a SSF orbit, or executes an Apollo style direct entry at Earth.

The all-propulsive vehicle configuration, shown in the following mass summary charts, consists of the MTV, MEV, TMI stage, and MOC/TEI stage. The MTV consists of the transfer habitat, ECCV, and airlock. The MEV consists of the descent stage, ascent stage, surface cargo, and landing aeroshell. The MEV is packaged to ensure placement of components inside the wake protection envelope of the landing aeroshell. The TMI stage consists of 3 LH2-LOX tanksets, 4 advanced space engines, and associated plumbing and structure. Overall vehicle dimensions are ~30 m diameter x ~65 m length.

III. Reference Cryo/Aerobraked Vehicle Mass Statement

The remainder of the information in this section consists of summary mass statements for four all-propulsive cryogenic fueled piloted vehicles. The first mass breakdown is for a landed crew of 3 with a 90 day surface stay time, and all cryogenic stages. Also included is a Lunar vehicle, which is essentially an offloaded Mars vehicle. The second mass statement is for a similar mission with a lower surface cargo payload (5 mt vs. 30 mt), and storable ascent stage propellant. Storable propellant for the ascent stage allows the extended stay time (90 days) with little risk of propellant storage system failure. The propulsion system is also much simpler, and more reliable. The final two mass summaries are for cryogenic conjunction class all-propulsive vehicles for 2009 and 2010 respectively. The primary difference between the two missions are the ΔV budgets (5781 m/s vs. 6916 m/s).

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All Propulsive Cryogenic Vehicle

Trades and Rationale

- Addition of MOI/TEI stage eliminates the need for a high energy Aerocapture at Mars.

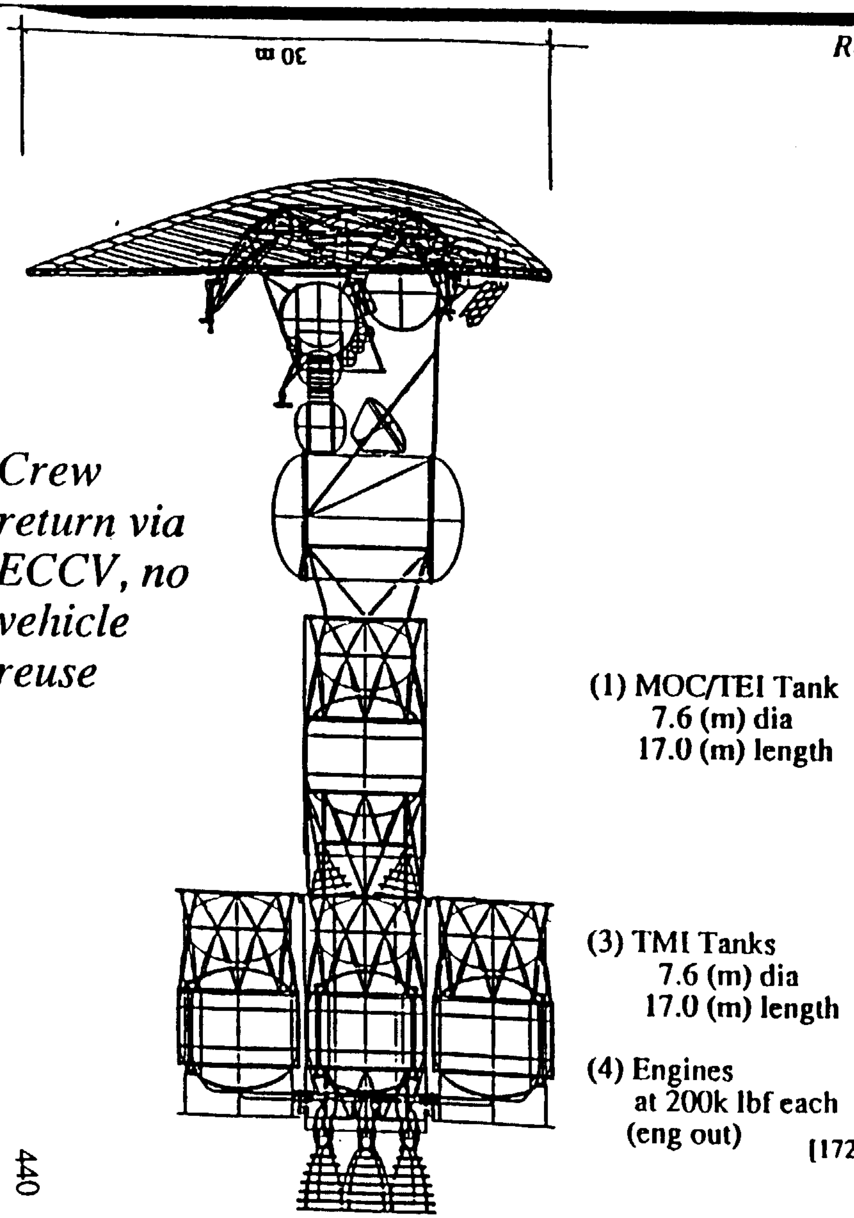
Mission Modes And Operations

- Vehicle assembled in SSF orbit.
- TMIS discarded after TMI burn.
- MOI burn and capture prior to MEV/Aerobrake entry.
- Aerobrake separates from MEV prior to landing.
- Crew cab ascent after surface mission, leaving lander, surface hab.
- Crew cab left in Mars orbit after rendevous, docking and crew transfer.
- TEI burn.
- Crew return to earth via ECCV.

MOI /TEI = Mars Orbit Insertion/ Trans Earth Injection

All Propulsive Chemical Vehicle for Conj Mission - *Human Expedition*

Single MEV; 30t surf cargo, crew of 3 for 90 days on surf, 1 crew member left in orbit, Common tank sets for MTV stgs dV's: TMI dV= 3900 m/s, MOC=1530 m/s, TEI= 860, E arr Vinf=3200,TMI, MOC,TEI eng Isp=475, MEV eng Isp=460



Crew return via ECCV, no vehicle reuse

Revision 5 9/14/90

	<u>Element</u>	*Mars*	**Lunar* *offloaded TMI*
[378]	MTV crew hab module 'dry'	39000	39000
[398+371]	MTV hab consumables & resupply	14000	500
[179]	<u>MTV hab mod science</u>	0	0
sum	***MTV crew hab module***	**53000**	**39500**
[128]	TEI usable propellant	15713	
[551]	TEI outbound boiloff	1455	
[545+546]	<u>TEI inorbit boiloff</u>	2716	
sum	***Total TEI propellant***	**19884**	23659
[541]	MOC usable propellant	72214	
[538+539]	<u>MOC outbound boiloff</u>	3752	
sum	***Total MOC propellant***	**78910**	47745
	<u>*EOC propel (Lunar case: returnto LEO)*</u>	n/a	55166
sum	***Tot MOC/TEI/EOC propellant in common tank set***	98794	126570
[118]	RCS propellant	852	917
[121]	Outb midcourse correction prop	2274	0
[122]	Inb midcourse correction prop	663	0
[161]	<u>MOC/TEI propul stg inert</u>	16335	16335
sum	***MTV propulsion stg total***	**115974**	**143822**
[313]	MEV descent only aerobrake	12000	12000
[63]	MEV ascent stage *Cryo/Isp=460*	24262	11716
	MEV descent stage *Cryo/Isp=460*	23190	34739
[166]	<u>MEV surface cargo</u> (3 crew for 90 days)	30000	30000
[106]	***MEV total***	89452	88455
[230]	*ECCV for crew return to LEO*	7000	0
[172-173+547]	TMI inert stage wt	45800	45800
[173]	<u>TMI propellant load</u>	407680	311245
[172]	***TMI stage total***	453480	357045
[171]	***IMLEO** (all masses in kg)*	718910	628822

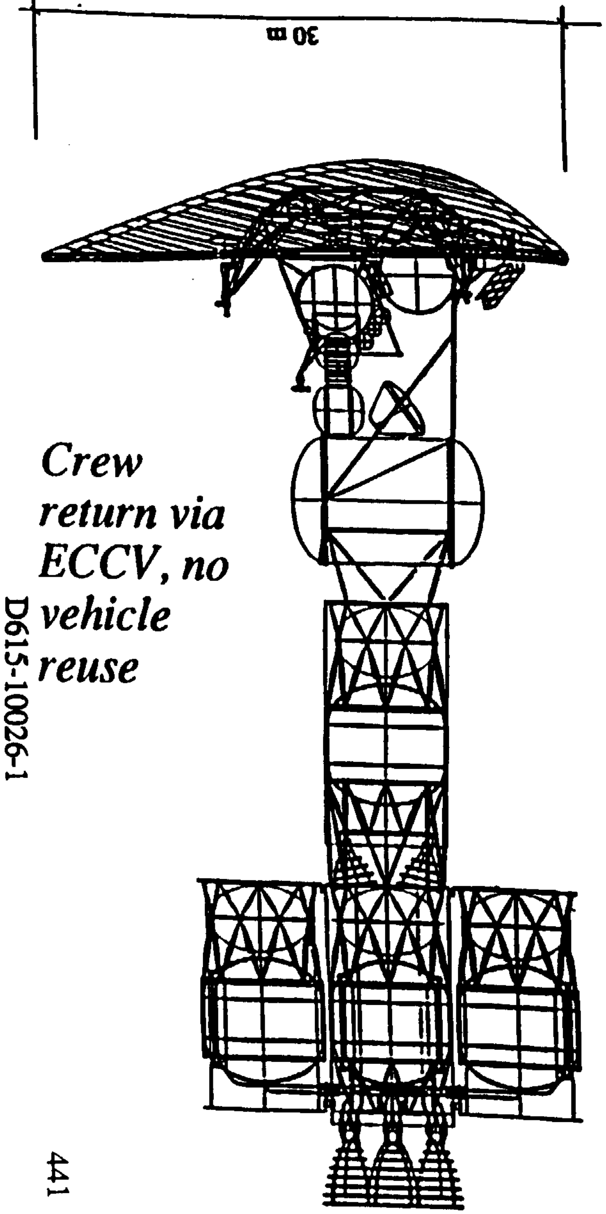
* same vehicle with offloaded TMI tanks for vehicle check/test mission to the Moon.
Vch synthesis model run #: marschemmtv.dat;64

All Propulsive Chemical Vehicle for Conj Mission - *Evolution Emphasis*

Single MEV; 5t surf cargo, crew of 4, Common tank sets for MTV stgs

dV's: TMI dV= 3900 m/s, MOC=1530 m/s, TEI= 860, E arr Vinf=3200,TMI, MOC,TEI eng Isp=475, MEV: cryo/storable

8/7/90



Crew return via ECCV, no vehicle reuse

(1) MOC/TEI Tank
7.6 (m) dia
14.5 (m) length
w/o engines

(3) TMI Tanks
7.6 (m) dia
14.5 (m) length
w/o engines

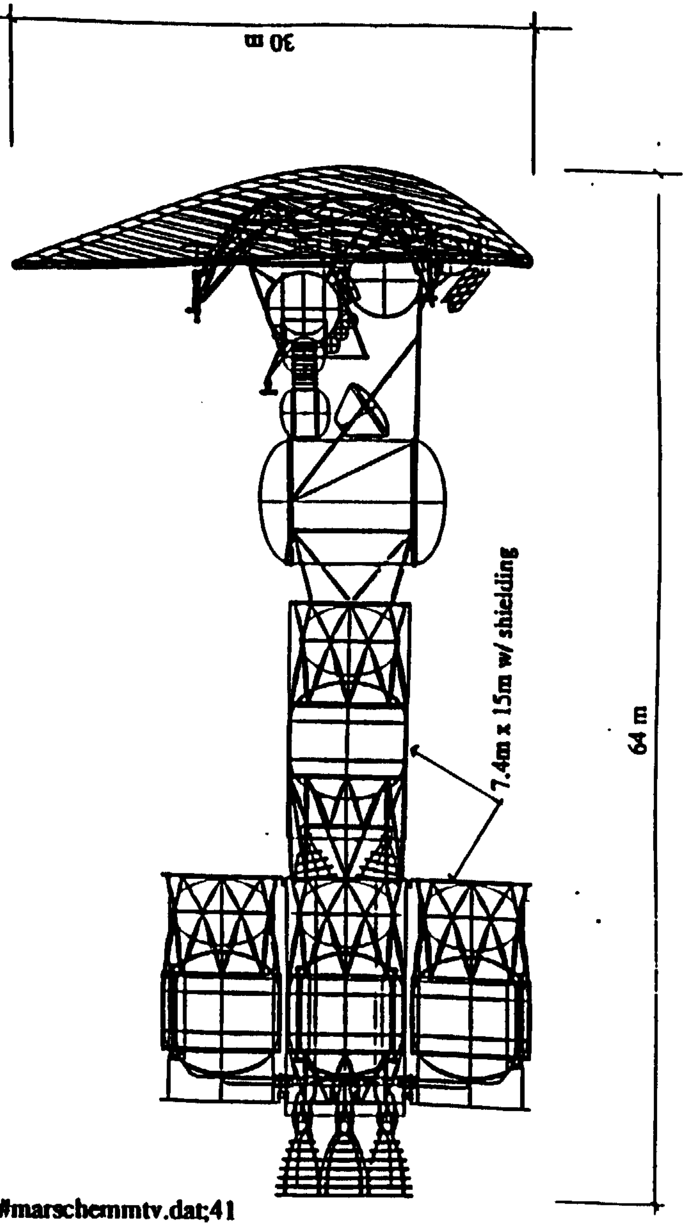
(4) Engines
at 200k lbf each
(eng out)

	Element	*Mass, kg*
[378]	MTV crew hab module 'dry'	39000
[398+371]	MTV hab consumables & resupply	14000
[179]	MTV hab mod science	0
sum	***MTV crew hab module***	**53000**
[128]	TEI usable propellant	15545
[551]	TEI outbound boiloff	1446
[545+546]	TEI inorbit boiloff	2698
sum	***Total TEI propellant***	**19689**
[541]	MOC usable propellant	62474
[538+539]	MOC outbound boiloff	3420
sum	***Total MOC propellant***	**66894**
sum	***EOC propel lant***	**n/a**
[118]	RCS propellant	842
[121]	Outb midcourse correction prop	1971
[122]	Inb midcourse correction prop	656
[161]	MOC/TEI propul stg inert	15510
sum	***MTV propulsion stg total***	**105562**
[i313]	MEV descent only aerobrake	6000
[63]	***MEV ascent stage***	***37406***
	Propellant / Isp	***Storable/340***
	MEV descent stage	***17019***
	Propellant / Isp	***Cryo/475***
[i66]	MEV surface cargo (3 crew for 90 days)	5000
[106]	***MEV total***	**65425**
[230]	***ECCV for crew return to LEO***	7000
	TMI inert stage wt	39770
[173]	TMI propellant load	353360
[172]	***TMI stage total***	**393130**
[171]	***IMLEO (all masses in kg)***	**624117**

Mac chart: M 2016 conj Evol all propul
Veh synthesis model run #: marschemmtv.dat;61(cryo) & ;62(storable)

All Propulsive Chem Veh for *2009 Conj *Crew of 4, ECCV return, 959 day trip*

(dV's from 1988 Boeing OTV study; TMI dV=4026 m/s, MOC 837 m/s, TEI 918 m/s)*



Element	*mass (kg)*
MTV crew hab module 'dry'	28531
MTV hab consumables & resupply	11458
MTV hab mod science	1000
MTV crew hab total	**40989**
TEI usable propellant	14157
TEI outbound boiloff	1400
TEI inorbit boiloff	1719
Total TEI propellant	**17276**
MOC usable propellant	30972
MOC outbound boiloff	2252
Total MOC propellant	**33224**
RCS propellant	689
Outb midcourse correction prop	5042
Inb midcourse correction prop	1239
MOC & TEI propul stg inert	13002
MTV propulsion stg total	**70472**
MEV Mars capture & desc aerobrake	7000
MEV ascent stage	22464
MEV descent stage	18659
MEV surface cargo	25000
MEV total	**73118**
ECCV	7000
Cargo to Mars orbit only	0
MTV-TMI interstage wt	500
TMI inert stage wt	36650
TMI propellant load	329910
TMI stage total	**366560**
IMLEO	558639

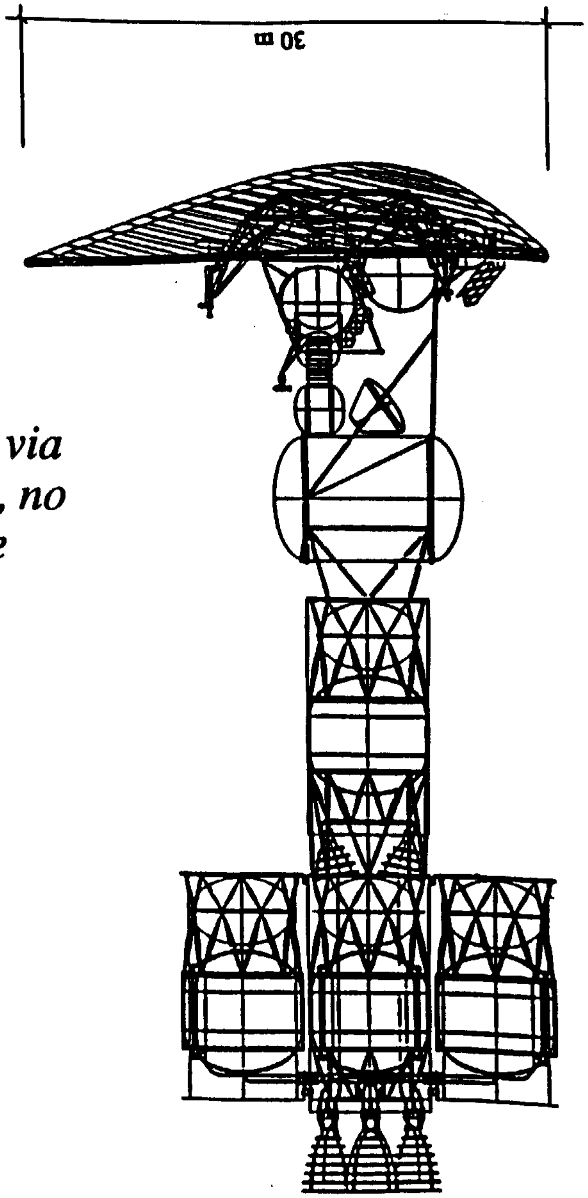
7/24/90

synthesis modelrun#marschemmtv.dat;41

Mac chart: M 2009 conj all propul chem cover pg

All Propulsive Chemical Vehicle for 2010 Conjunction Mission

Single 73t MEV carries 25t to surf, Common tank set for MOC/TEI stg
dV's: TMI dV= 4570 m/s, MOC=1160, TEI= 1186, Adv space eng's: Isp=475



Crew return via ECCV, no vehicle reuse

8/9/90	*Element*	*Mass, kg*
[378]	MTV crew hab module 'dry'	28531
[398+371]	MTV hab consumables & resupply	10560
[179]	MTV hab mod science	1000
[381]	***MTV crew hab module***	**40091**
[128]	TEI usable propellant	17677
[551]	TEI outbound boiloff	1415
[545+546]	TEI inorbit boiloff	2263
sum	***Total TEI propellant***	**21355**
[541]	MOC usable propellant	43602
[538+539]	MOC outbound boiloff	2468
sum	***Total MOC propellant***	**46070**
sum	***EOC propel***	n/a
[118]	RCS propellant	670
[121]	Outb midcourse correction prop	1744
[122]	Inb midcourse correction prop	521
[161]	MOC/TEI propul stg inert	12944
sum	***MTV propulsion stg total***	**83304**
[i313]	MEV descent only aerobrake	7000
[63]	***MEV ascent stage***	***22464***
	Propellant / Isp	*Cryo/475*
	MEV descent stage	***18659***
	Propellant / Isp	*Cryo/475*
[i66]	MEV surface cargo (4 crew for 30 days)	25000
[106]	***MEV total***	**73118**
[230]	***ECCV for crew return to LEO***	7000
	TMI inert stage wt	46870
[173]	TMI propellant load	417300
[172]	***TMI stage total***	**464170**
[171]	***IMLEO***	**667683**

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Nuclear Thermal Rocket (NTR)

I. Introduction.

The NTR offers a higher Isp than any currently defined chemical system. While high energy aerobraking is an option for any mission, there are alternatives. The high Isp of the NTR propulsion system provides the opportunity to fly both opposition and conjunction class missions. The reference NTR vehicle exhibits a mass savings over the reference cryo/aerobraked vehicle (~66 mt). The main advantage of the NTR, however, is not the IMLEO savings, but the reuse capability of the NTR (only drop tanks and MEV non-reusable), and the absense of a need for a high energy aerobraking maneuver. The disadvantage of the need for technology advancement in the area of nuclear propulsion is at least partially offset by the savings resulting from the lack of technology development needs for high energy aerobraking (although descent aerobraking development will still be needed).

II. Reference vehicle design and operations.

The NTR vehicle is assembled in a SSF orbit. Two LH2 tanksets is jettisoned after the TMI burn. At Mars arrival, the vehicle propulsively captures, and two LH2 tanks are jettisoned. After capture and orbit insertion, the crew transfers to the MEV for descent to the Mars surface. The MEV descent into the Martian atmosphere is slowed by the aerobrake, which is jettisoned shortly prior to landing. The descent engines fire through an opening in the aerobrake created by jettisoning the engine bay doors on the aerobrake. After a 30 day surface stay, the crew boards the ascent vehicle, which ascends to rendezvous with the MTV. After crew transfer to the MTV, the ascent ship is jettisoned, and the TEI burn executed. At Earth arrival, the vehicle propulsively captures into a high Earth orbit (nuclear safe), and the crew returns to SSF.

The NTR vehicle configuration, shown in detail in the following charts, consists of the reactor and shield, MTV, MEV, and 4 LH2 drop tanks. The MTV consists of the transfer habitat, main structure, core LH2 tank, reactor/shield, and airlock. The MEV consists of the descent stage, ascent stage, surface cargo, and landing aeroshell. The MEV is similarily packaged to ensure placement of components inside the wake protection envelope of the landing aeroshell. The 4 strap-on LH2 tanksets are used for TMI(2), and MOC(2), while the core tank is used for the TEI and EOI burns. Overall vehicle dimensions are ~30 m diameter, by ~110 m length.

III. Reference NTR Design History/Structure Trade

The history of the Boeing reference NTR vehicle is presented. The nuclear engine greatly influences the overall physical configuration of any NTR vehicle. The necessity for radiation attenuation between the engine source and the crew as well as the placement and staging of very large hydrogen propellant tanks are two major considerations that are unique to NTR systems. The following factors are applicable in this regard:

(1) Radiation dosage received by crew = 1/(separation distance) squared

Separation distance between the crew and reactor is a key parameter in reducing the amount of reactor generated radiation that reaches the crew habitat module. Since the reactor radiation dosage that eventually reaches the hab module is equal to the inverse of the separation distance squared, grouping the lengthy propellant tanks into a axial alignment rather than a radial cluster maximizes radiation attenuation by maximizing the separation distance provided by the tankage/structure without unduly penalizing the vehicle with

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structure dedicated solely to extending separation distance. Doubling the separation distance reduces the received dosage by a factor of 4.

(2) Axial alignment of tanks rather than radial clustering also allows the reactor radiation shadow shield protected cone half angle to be smaller since there would be less projected tank area around the reactor that could scatter direct radiation and thus become a secondary source. Any reactor shadow shield would include a very dense layer of material such as tungsten or berylium, dedicated solely to gamma ray attenuation. Minimizing the shield size is important in keeping the mass down.

(3) Axial alignment provides more hydrogen propellant to be utilized as a secondary thermal neutron shield in the direct line between the crew cab and the reactor.

The configurations shown are representations of various tank size and tank placement options. It is beneficial from a shielding viewpoint to keep the Earth arrival propellant in an 'inline' tank just behind the reactor shield. It is beneficial from an IMLEO standpoint to:

(a) jettison the tanks after each burn
(b) use as large a tank size as the launch vehicle(s) can deliver
(c) use advanced materials such as metal matrix composites to keep the tank fraction as low as possible

Other issues include: Providing for tank release and jettison; minimizing and facilitating on-orbit assembly; anticipating meteor shielding requirements (with or without a protection hanger at SSF); vehicle return for reuse refurbishment/resupply issues; artificial g accommodations.

IV. Reference and Three Lander NTR Vehicles Mass Statements

The remainder of the information in this section consists of summary mass statements for the reference opposition-class and three lander conjunction-class NTR vehicles, and a detailed mass statement for the reference vehicle, including rationale. The reference vehicle mass breakdown is for a landed crew of 4 with a 30 day surface stay time, while the three MEV option includes mass summaries for 3 full size MEV's with 20 mt and 1 mt payload delivered to Mars orbit, and 3 "mini" MEV's, each capable of a 7 day surface sortie.

V. NTR vehicle Mass vs. Opportunity Year and Reference Delta V Parametric Data

The reference vehicle configuration was used to produce parametric vehicle data of vehicle mass vs. mission phase delta-V. This data can easily be used to estimate an approximate vehicle mass for different mission opportunities than those presented here. The IMLEO for an advanced particle bed reactor NTR vehicle was determined over a range of mission years. The 2016 opportunity proved to be the most difficult, although it still was almost 85 mt lower in mass than the reference NERVA-derived NTR vehicle.

C-5

Nuclear Thermal Rocket Vehicle Reference Configuration

Introduction

The nuclear thermal rocket (NTR) concept offers advantages of higher I_{sp} than cryogenic concepts, fully propulsive capture at Mars and Earth to avoid high energy aerobreaking, and the potential for recovery and re-use of the expensive transfer habitation system. NTR represents a proven technology; early versions were extensively tested in the 1960s and early 1970s.

Nominal Mission Outline

- The vehicle is assembled, checked out, and boarded in LEO
- The TMI burn occurs, and two empty LH_2 tanks are jettisoned (opposition case)
- The MTV coasts to Mars
- MOI burns capture the MTV into Mars orbit
- Two LH_2 tanks are jettisoned
- The MEV is checked out, separates from the MTV and descends
- The MEV aerobrake is jettisoned prior to final approach
- The MEV touches down, and surface operations ensue
- The MAV ascends for rendezvous with the MTV, leaving the descent stage, surface habitat and science equipment
- The MAV is jettisoned in Mars orbit after crew transfer
- The TEI burn occurs, and the MTV coasts back to Earth
- In expendable scenario, crew return is accomplished with modified ACRV (MCRV), MTV is jettisoned at Earth
- In re-usable scenario, MTV captures propulsively into high parking orbit (500 km by 24 hr) for 30 d cool-down period
- Crew returns to SSF using LEV-class taxi
- Post-cooldown, MTV is refurbished in SSF orbit

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Vehicle Systems

Crew Systems

The crew portion of the vehicle consists of a transfer habitat (common with other concepts), deployable PV power plant, and an MEV (common with other concepts). All habitable volumes are contiguously connected, and located at the opposite end of the vehicle from the reactors. The ends of the vehicle are separated by a lightweight truss spine.

Propulsion System

The reactor/engine is a technology-upgrade from the NERVA reactor of the 1970s. A composite shadow shield limits both direct and secondary-particle-scattered dosage to the crew and sensitive electronics. LH_2 propellant is used. Four cryogenic storage drop-tanks are located on the truss. Another, in-line propellant tank is for TEI and EOI; remaining full for most of the mission enables it to provide extra radiation protection to the crew systems. All propellant from the drop-tanks is flowed through the in-line tank, so that its supply remains relatively un-irradiated throughout the mission.

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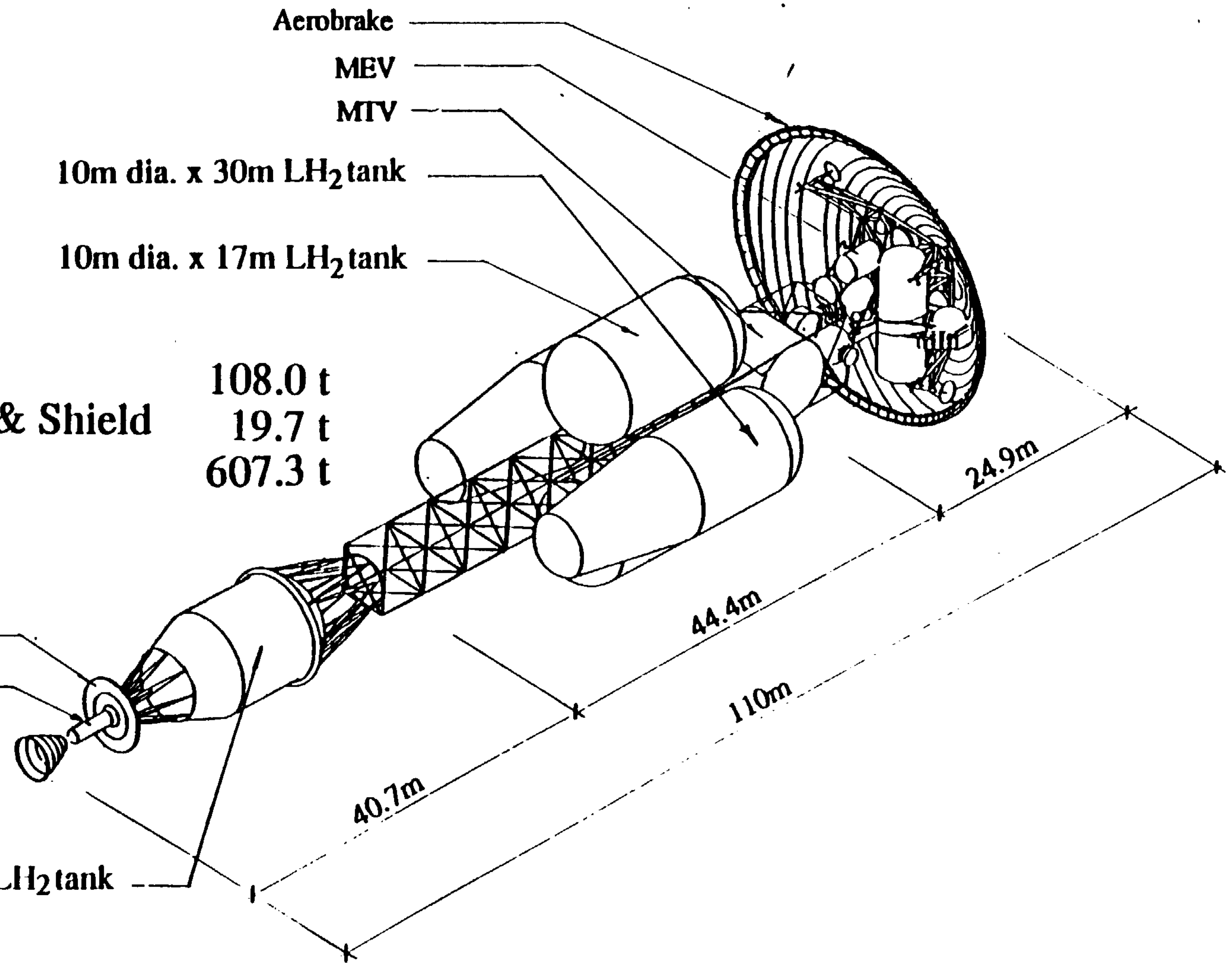
2016 Advanced NERVA NTR Reference Vehicle Configuration

The 925 Isp NERVA derivative engine was chosen by NASA MSFC as the reference propulsion system for the NTR vehicle studies. The performance of the 925 Isp system corresponds to an 'intermediate' reactor fuel element material. Composite fuel elements (see fuel element chart) operating such that the hydrogen propellant reaches approximately 2700°K at 450 psia chamber pressure would provide this Isp, given a large expansion ratio nozzle, and would require no redesign of the NERVA reactor beyond that necessary for integration of these higher temperature fuel elements (cooling and element corrosion are such factors). An Isp of 925 is approximately 85 sec higher than that obtained by the Phoebus 2A reactor in 1967. Such a level of enhancement entails no high risk new technology development, rather it would be an extension of the advanced fuel element analysis that was already underway in the early 1970's when the NERVA program was canceled. Materials development and fabrication techniques in general have seen a lot of advancement in the last 20 years. Thereference vehicle was built around this performance level using the *Boeing Vehicle Synthesis Model* , a sophisticated computer code that outputs vehicle performance figures and weight breakdowns based on very specific vehicle configuration and requirements inputs. The vehicle, as illustrated has four 10 meter dia. hydrogen propellant tanks with a tank fraction of 14%. Two tanks for Earth departure propellant, that are jettisoned after TMI burn, one Mars arrival propellant tank jettisoned after Mars capture and one tank that remains with the vehicle that holds both the Mars departure and Earth arrival propellant. A 2 meter by 35 meter SSF type truss is shown as connecting the in line Mars departure / Earth arrival tank to the 33t, 4 crew habitat module and MEV. The MEV hasa single, low energy, Mars descent only aerobrake - this is not a high energy aerobrake designed for Mars orbit capture. The vehicle does propulsive burns for orbit capture both at Mars and Earth.

NTR Configuration



MEV/MTV	108.0 t
Propulsion, Frame & Shield	19.7 t
Propellant & Tanks	607.3 t

IMLEO	735.0 t

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NTR

Trades and Rationale

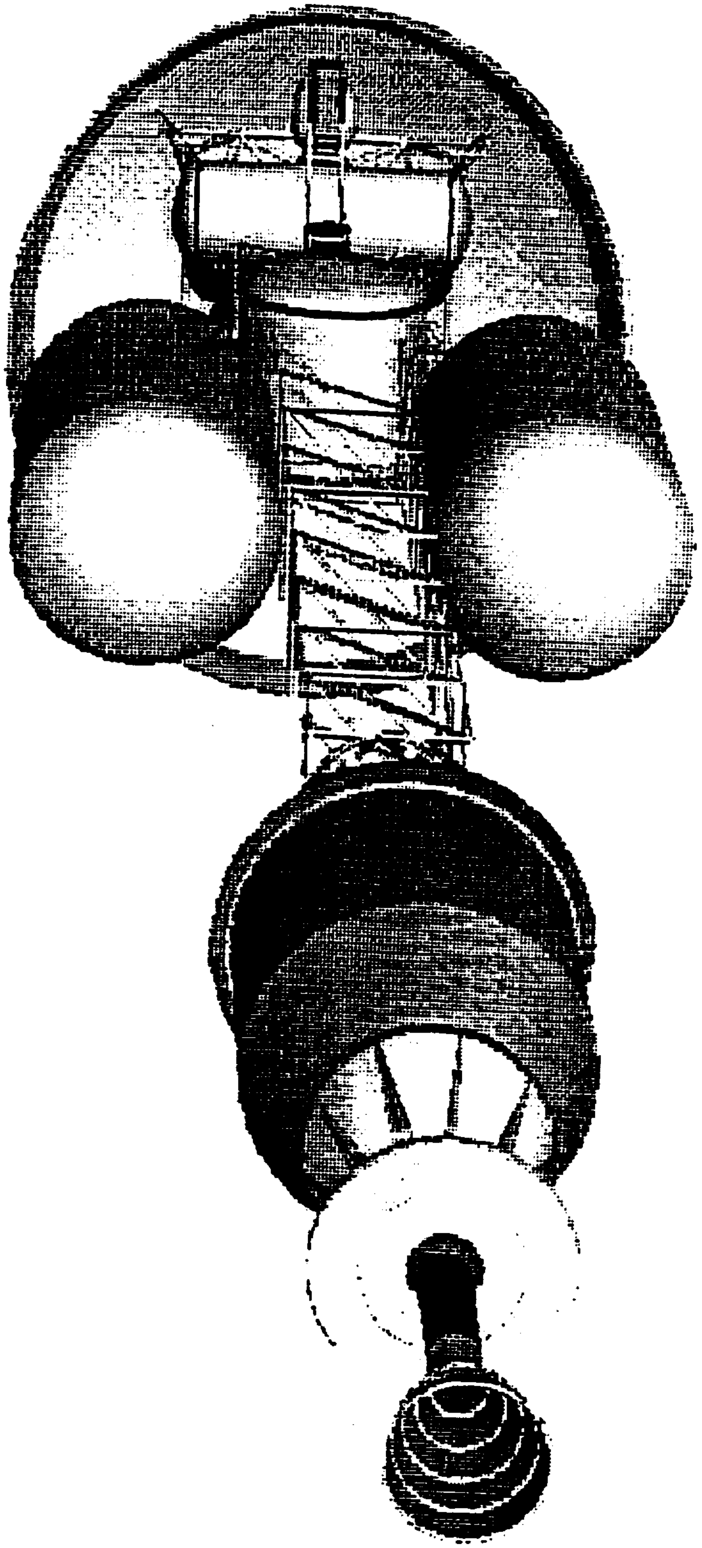
- High Isp compared to chemical engine
- Fully propulsive capture at Mars and Earth avoids high energy aerocapture.

Mission Modes And Operations

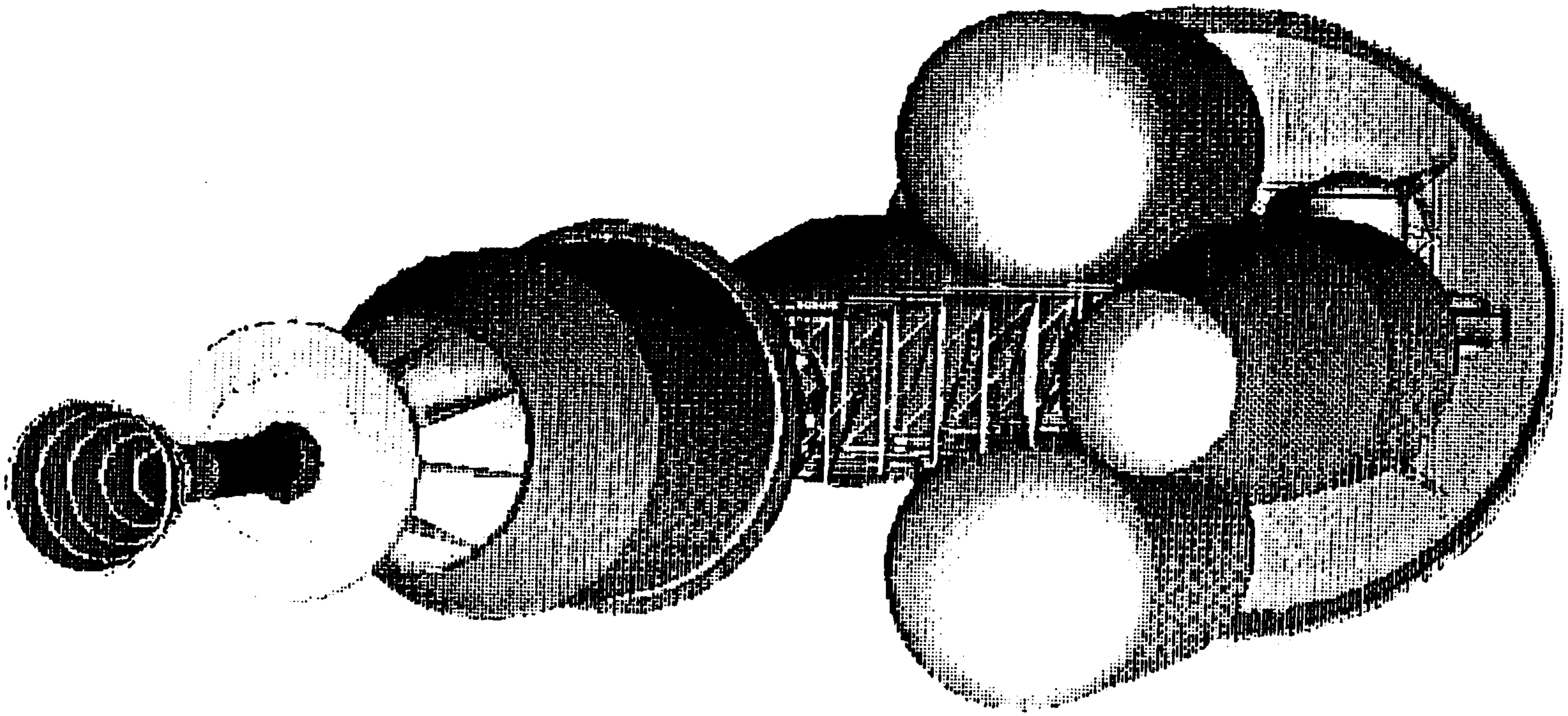
- Vehicle assembled in SSF orbit
- Two LH_2 tanks jettisoned after TMI burn
- Two LH_2 tanks jettisoned after MOI burn
- MEV/Aerobrake separate from vehicle prior to entry and landing
- Aerobrake separates from MEV prior to landing.
- Crew cab ascent after surface mission, leaving lander, surface hab.
- Crew cab left in Mars orbit after rendevous, docking and crew transfer.
- TEI burn
- EOI burn and crew return to SSF.

/STCAEM/crf/31May90

Reference Configuration: *Outbound and Mars Capture*



Reference Configuration: *Earth Departure*



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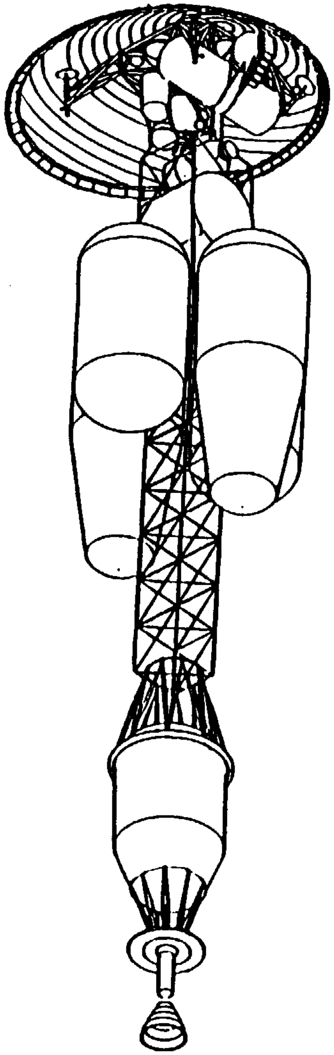
Reference NTR Vehicle for 2016 Opposition Mission

Veh return to Earth for Reuse, no ECCV, Crew of 4, 434 day trip time *Revision 5 5/22/90*



Element	*NERVA T/W=3.5*	*PBR T/W=15*
MEV desc aerobrake	7000	7000
MEV ascent stage	22464	22464
MEV descent stage	18659	18659
MEV surface cargo	25000	25000
MEV total	**73118**	**73118**
MTV crew hab module 'dry'	28531	28531
MTV consummables & resupply	5408	5408
MTV science	1000	1000
MTV crew hab sys tot	**34939**	**34939**
MTV frame, propulsion, & shield wt	19777	12086
Earth Orbit Capture (EOC) prop	27756	24296
Trans Earth Inject (TEI) prop	59245	51727
EOC/TEI common tank wt	13845	12424
Mars Orbit Capture (MOC) prop	151680	138800
MOC tanks	25572	23962
Trans Mars Inject (TMI) prop	286146	262100
TMI tanks	43092	39973
ECCV	0	0
Cargo to Mars orbit only	0	0
IMLEO	**735190**	**673425**

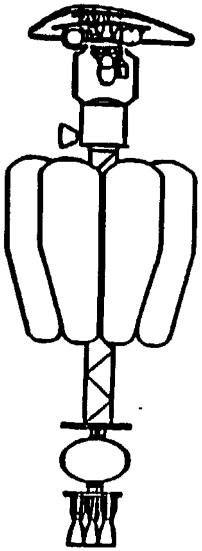
all masses in kg's

Mac chart: M Ref NTR cover pg
synthesis model run# marsntrmtv.dat;161,183

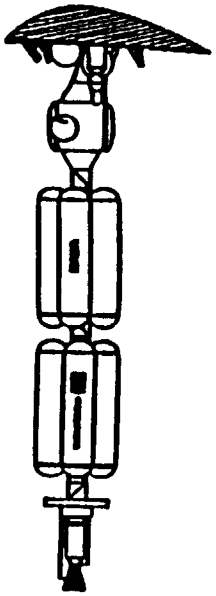
2015 reference NTR Design History

The following diagrams show the progression of preliminary designs for the 2015 NTR vehicle. The final configuration has the Earth departure and Mars capture tanks forward on the truss completely within the 'shadow shield protected cone. This is necessary so that radiation from the reactor and nozzle can not reach the tanks to produce secondary gamma's.

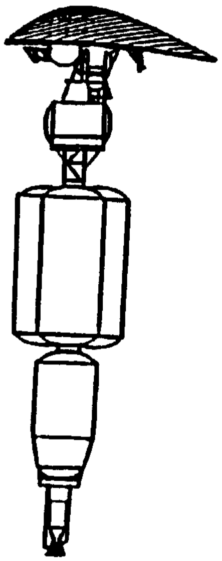
2015 Reference NTR Design History



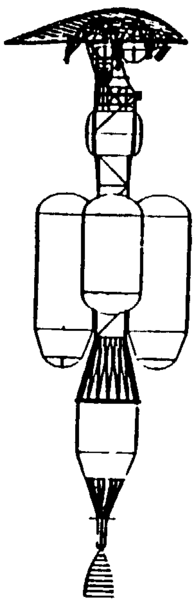
Earliest concept



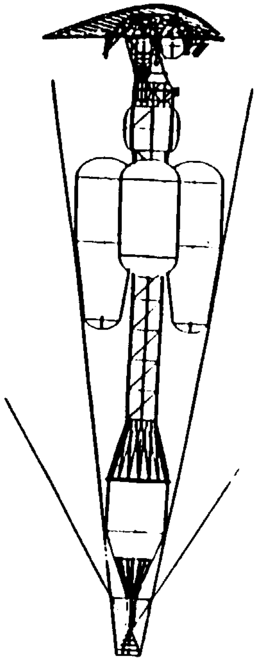
Shuttle-C Tanks



10 meter dia tanks



CAD-CAM 5 tanks



SAIC Radiation assessment

STCAEM/mha/31May90

Truss vs. Tank for Structure Trade - 2016 Ref. NTR Vehicle

The NTR vehicle configurations that utilize propellant tanks as structural members for the duration of the mission, instead of dropping them after they are emptied from a specific burn, impact the design with a major weight penalty. These large hydrogen tanks that are covered with MLI, vapor cooled shields and meteor shields are relatively heavy, seeing that a typical tank fraction is about 14 % (tank fraction = tank weight/ total tank and propellant weight). It is a disadvantage for any NTR vehicle to have to carry a tank, used and emptied on the outbound leg, back to Earth that could have been dropped off earlier. A tank, if serving also as a structural member on the vehicle, could be replaced in that secondary capacity with a much lighter truss member, it could offer a significant IMLEO cost savings since the truss may be as much as an order of magnitude lighter than an empty tank for one of these NTR vehicles. A trade was done on the Boeing 2016 reference design to determine the effect of keeping or dropping the large Mars Orbit Capture (MOC) propellant tank.

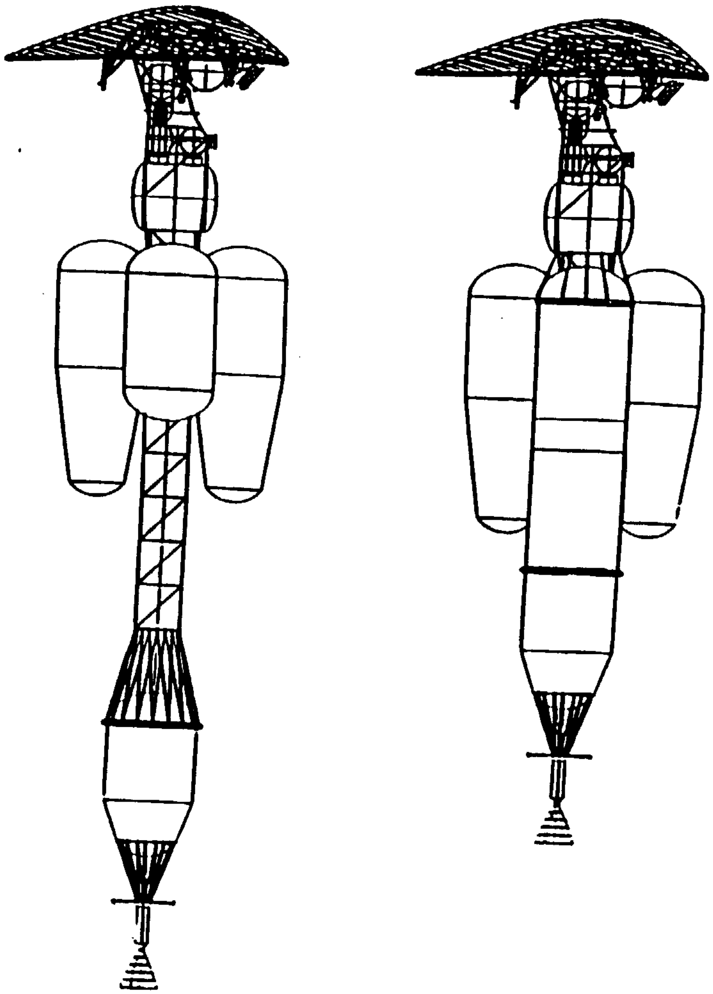
(1) reference vehicle: the 2 MOC tanks were jettisoned immediately after the MOC burn. They did not serve as structural members - a SSF type truss served to connect the engine and aft tank to the crew habitat module and the TMI and MOC tanks were attached to it. The truss weighted about 2400 kg using standard SSF truss bays that weighted 160 kg per 5 m by 5 m bay. The two MOC tanks together weighted 25572 kgs, at a tank fraction of approximately 14%.

(2) alternate vehicle: the truss system structurally linking the engine and the aft tank to the crew hab. module has been replaced by using a single MOC tank. By necessity, the tank can not be dropped after the MOC burn but must be carried back to Earth since its secondary job is to be part of the structure. This single MOC tank weights 25301 kgs.

Results: Alternate vehicle: Having to carry the 25301 kgs of empty MOC tank inbound causes an increase in IMLEO of 95010 kg over the reference design IMLEO. The reference vehicle carries the 2400 kg truss inbound (about 10% of the MOC tank weight of the alternate). All designs that do not drop the larger tanks (Earth departure or Mars capture) suffer this same disadvantage to some degree.

Truss vs Tank for Structure Trade - 2016 Ref NTR Veh

keep or drop Mars Orbit Capture (MOC) tanks 6/8/90



Mac chart:truss vs tank structural trade

Vehicle synthesis model run #: marsntrmtv.dat;161,184

/STCAEM/bbd/8June90

Ref Vehicle:

- **Truss**: length 45 m (7 standard truss bays)
  width: 5 m by 5m SSF standard
  weight: 2400 kg

- **2 expendible MOC tanks;** empty wt: 25572 (kg)
  includes MLI,VCS, meteor shield, jetison hardware

- *Tot IMLEO of Reference Vehicle: 735,190 (kg)*

Operational concerns:
Provides radiation attenuating sep distance
Each of the 2 MOC tanks jetisonned after burn
2400 (kg) truss carried outbound & inbound

Alternate Vehicle:

- **Truss**: none; 0 (kg)

- **One 'in line' MOC tank** used as structural member
  empty wt: 25301 kg; includes MLI,VCS, meteor shield

- *Total IMLEO of Alternate Vehicle: 830,200 (kg)*

Operational concerns:
MOC 'inline' tank displaces truss & provides sep distance
25301 (kg) MOC tank carried outbound & inbound
95,010 (kg) IMLE0 wt penalty for carring empty tank
back to Earth

Desc stage - MEV for 2016 Reference NTR Vehicle

Crew of 4, 4 adv eng's; Isp=475, 25 t surf cargo, descends from 250 km alt *Rev 5 5/22/90*

			Fuel / Oxidizer	
Desc stage inert	[98/99]	Single tank wt	225/117	2 SiC/Al metal matrix tanks for each, 37ksi wk stress, MEOP=175 kPa, min t=3.5mm
	[124/125]	Meteoriod Shield	29/15	One 0.40 mm sheet of Al
	[122/123]	MLI	43/22	MLI: density = 32 (kg/m3); 100 layers at 20 layers/cm.
	[126/127]	Vapor Cooled Shields	34/17	1 VCS at 2 x 0.13mm Al outer sheet w 0.57 kg/m2 honeycomb core
	[0/0]	Vacuum shell	0/0	not on desc tanks
	[2xi316]	Propel line wt	35/35	25 kg per tank + 10 kg for tank instrumentation
	[132/133]	Tank wt growth	41/23	15% wt growth
	[128/129]	Sum single tank inerts	407/229	Total single tank + tank inert wt
	[130/131]	***Tot: Fuel & Ox tanks:***	**814/458**	2 LH2 & 2 LO2 tanks
	[501]	Main propulsion	1127	4 x 30klbf Adv eng's: Isp=475 sec, w extendible/retractable nozzles
	[102]	Asc frame & struc wt	562	4% of desc stage stg wt + 2% of surf crew mod mass
	[103]	Landing legs	1540	3% of total landed mass
	[i273,526]	RCS inert	428	Estimate from RCS prop load
	[135]	Propul, frame wt growth	493	15% of total inerts
	Sum	***Desc propul & frame inert***	**4150**	
Prop loads	[91+92]	Desc usable Prop	12061	*Desc propulsive veh dV= 931 (m/sec) from 250 km periapsis alt. by 1 sol orbit.*
	[0]	Desc boiloff	0	
	[101]	Desc RCS prop	1173	
	Sum	***Total Desc propellant load***	**13234**	N2O4/MMH prop, Isp=280 sec, *desc RCS dV=50 (m/sec)*
Aero brake wt		***MEV aerobrake:***		
		• Primary spar wt	1149	Structural design assumptions: 200ksi spar strength, 22.5 inch spar depth
		• Secondary spar wt	1200	NTR vehicle does propulsive braking of MTV & MEV into Mars orbit. This MEV
		• Honeycomb wt	3125	aerobrake is used only for descent to surface. It does not do aerocapture, which accounts
		• TPS wt	1526	for the wt difference between it and the MEV aerobrake wt (15138)for the Chem/AB veh
	[78]	Total:	**7000**	
	[77]	***Surface crew hab module***	**25000**	Level II Requirement: surf modulw, surf science & surf stay consumables
	[61]	***Asc veh total mass***	**22462**	from 'Asc stage' wt statement page
	[106]	***MEV mass***	**73118**	all masses in kg

synthesis model run#: marslander.dat;114
STCAEM/bbd/23May90 Mac chart: M Ref MEV decs veh wt-rationale

Asc stage - MEV for 2016 Reference NTR Vehicle

Crew of 4, 30 day stay, 2 adv eng's; Isp=475 , Ascends to 250 km alt *Revision 6 5/22/90*

Ascent Cab		Structure	998	SSF dia center cyl section w ellip ends. Stiffening rings added. See 'Structures pg'
		ECLSS	678	Open sys:CO2 adsorption unit, stored H2O,O2,N2, no airl., no hyg w. see 'ECLSS pg'
		Command/Control/Power	330	Power: fuel cells
		Man systems	82	Waste management sys/waste storage/medical equp.
		Spares & tools	192	Subsystem component level spares
		Wt growth	376	15% growth for dry mass
		Asc 'dry' mass	2656	Total cab dry mass
		Consumables (food & water)	62	Minimum; food and water only; 3 occupancy
		Crew/effects/EVA suits	760	Crew of 4, 100 kg EVA suit per crew member
	[22]	***Ascent cab gross mass***	3478	
			Fuel / Oxidizer	
Ascent stage inert	[45/46]	Single tank wt	301/138	2 SiC/Al metal matrix tanks for each, 37ksi wk stress, MEOP=175 kPa, min t=3.5mm
	[68/69]	Meteoriod Shield	38/18	One 0.40 mm sheet of Al
	[50/51]	MLI	57/26	MLI: density = 32 (kg/m3); 100 layers at 20 layers/cm.
	71/72/112/113	VCS & Vacuum shell	90/42	1 VCS and 1 Vac shell: both 2 x 0.13mm Al outer sheet w 0.57kg/m2 honeycomb core
	2x[1316+1318]	Propel line wt	35/35	25 kg per tank + 10 kg tank instrumentation
	[116/117]	Tank wt growth	57/28	15% wt growth
	[73/74]	Sum single tank inerts	578/287	Total tank & tank inert wt
	[114/115]	***Tot: H2 & O2 tanks:***	1156/574	2 LH2 & 2 LO2 tanks
	[500]	Main propulsion	564	3 x 30klbf Adv eng's: Isp=475 sec, w extendible/retractable nozzles
	[118]	Asc frame & struc wt	469	3% of total asc stg propellant wt
	[i274,525]	RCS inert	222	Estimate from RCS prop load
	[54]	Propul, frame wt growth	188	15% of total inerts
	Sum	***Asc propul & frame inert***	1443	
Prop loads	[60]	Asc usable propellant	15482	*Asc veh dV= 5319 (m/sec) to 250 km periapsis alt. by 1 sol orbit.*
	[56+58]	Asc boiloff	157	30 day surf stay; calc: Boeing 'CRYSTORE' program
	[52]	Asc RCS prop	172	N2O4/MMH prop, Isp=280 sec, *Asc RCS dV =35 (m/sec)*
	Sum	***Total Asc propellant load***	15811	
	[63]	***Asc veh total mass***	22462	all masses in kg

synthesis model run#: marslander.dat;114

Mac chart: M Ref NTR MEV asc veh wt *STCAEM/bbd/22May90*

Ascent Cab - for Reference MEV Vehicle

Crew of 4, 3 day occupancy time Revision 2 5/22/90

	Element	*mass (kg)*	*Rationale*
Cab ECLSS	Atmospheric Revitization Sys/ Trace contaminant control assembly	123	CO2 adsorption unit, expendable LiOH cartridge Pre & postsorbent beds,catalytic oxidizer for particulate & contaminant control
	Atmosphere Control System	62	Total & partial press control; valves,lines & resupply/ makeup O2 & N2 and tanks
	Atmos. Composition & Monitor Assem.	55	O2 & N2 monitor for ACS, particulate & contaminant monitor for ARS
	Thermal Control Sys	40	Temp control: sensible liq. heat exchanger, ext radiator wt included in 'secondary structure' mass
	Temp. & Humidity Control	240	Condensing heat exchanger, fans, ducting
	Water Recovery and Management	45	Stored Potable water only
	Fire Detection & Suppression Sys.	113	Automatic sys w manual extinquishers as backup
	Waste Management Sys and Storage	-	Considered part of 'Man Systems'
	Asc cab ECLSS mass	**678**	Apollo style open ECLSS system
Cab Structure	Primary/Secondary Structure	519	Overpressurized (20 psia) on launch for structural integrity. Stiffening rings added at cylinder/endcap interface for added strength. Skylab derived triangular grid floor with beam supports on 6" centers. Support ring interface on pressure vessel to carry loads imposed by the floor and equipment during launch to aerocapture.
	Berthing ring/mechanism (1)	139	
	Berthing interface plate (1)	90	
	Windows	90	
	Couches	80	
	Hatches (2)	80	
	Asc cab Structure mass	**998**	

synthesis model run number: marsntr.dat.
Mac chart:M Ref MEVasc cab wt-ratio

Crew habitat module - MTV for 2016 NERVA NTR Ref Vehicle

Zero-g, Crew of 4, 434 day total trip time *Revision 6 5/22/90*

		Element	*mass (kg)*	*Rationale*
Crew Hab sys	[360]	Structure	8351	Cyl length: 9 m, dia: 7.6 m, ellip ends, 3 levels; tri grid w beam supports. Tens. ties
	[363]	ECLSS	4256	SSF derived with same degree of closure, sized for crew of 4 for 565 days
		Command/Control/Power		
	[364]	• Internal	1159	ECWS, DMS, batteries, other avionics/computing/monitoring eq, conditioning equip
	[i281]	• External Power	1539	Solar array,boom, power distribution,power management,fuel cell system
	[368-316]	Man systems	4121	Wts -all sys:SSFderived(as a funct. of crew size&occupancy time)for Mars missions
	[316]	Crew & effects	440	110 kg per person including personal belongings
	[373]	Spares/Tools	1496	Subsys component level spares. Life crit sys are 2 fault tolerent (approach of SSF)
	[247]	Radiation shelter	1802	Provides 10 g/cm2 protection + 3-5 g/cm2 provided by vehicle structure and equip
	[377]	Weight growth	2973	15% weight growth for dry mass excluding crew & effects and radiation shelter
	[237]	Airlocks	1530	2 x 765 kg external airlocks (shuttle type airlocks modified for MTV mission)
	[330]	EVA suits	0	EVA suits weight counted in MEV ascent cab weight statement
	[i60]	TTNC & GN&C platforms wt	863	3 platforms
	[378]	***MTV 'dry' crew hab mod wt***	28531	'dry' hab module represents structure and support systems equip & hardware that are dependant on crew size and independant of mission duration
	[371]	*On board equip resupply	986	Based on adjusted SSF resupply reqts for pot w, hyg w, ARS,TCS/THC &WMS
	[398]	*Consumables	4422	Crew of 4 for 434 days; food:2.04 kg/man/day, food pkg:0.227, pharmaceuticals: 0.25 other: 0.291 Clothes: 42 kg/man. food vol: 0.0055 m3/man/day, other: 0.0018.
	Sum	***MTV crew mod 'wet' wt***	5408	
Crew mod supp. sys	[i65]	*Transfer science equipment	1000	Inb and outb MTV science hardware and supplies
	[227]	Art-g RCS spin up propel	0	zero g environment
	[i61+126	Art-g tether mass	0	zero g environment
	[i57]	Remote Manipulator-arm Sys	0	all large external self assembly hardware left in LEO
	[734]	***Hab mod support sys wt***	1000	
	124-230	***MTV crew mod & support systems weight***	34939	This wt refects the Boeing ref crew of 4 mod downloaded by 1370 kg of consumables and 318 kg of onboard resupply because the shorter 2016 opposition mission (434 days vs 565 chem/AB ref). The mod 'dry' wt represents a SSF type closed ECLS Sys (air >99%, water >95%) that serves the crew with 2 fault tolerence on all life critical sys except structure. Its wt varies primarly with crew size. consumables wt varies with crew size and mission duration.

* *MTV hab mod consumables, resupply, and transit science dependant on mission duration, and free abort requirement. i.e. crew mod 'wet' wt will vary for different missions*

Mac chart: M Ref MTV mod wt-rationale
synthesis model run# marschemmtv.dat;21

2016 NERVA NTR Reference Vehicle:

Frame, propulsion sys, & shield wt: Isp = 925 *Rev 5 5/22/90*

[159] Spacecraft frame (truss) struct	3000	Truss struct: Graphite epoxy, Ec= 16 Msi, Den=0.06 lb/in3, 2 m by 35 m SSF type
[183] RCS inert wt	800	estimate scaled from RCS propellant load
[709] Main prop line wt	451	Main line from tank lines to reactor, L=35m,d wall s steel;dens=7833kg/m3,t=0.8mm
[160] Mass growth	638	15% mass growth
[518] Engines wt (1)	9684	*75k lbf Thrust*, wt estimate: NASA/LeRC propul task order (Westinghouse, others)
[543] Engine shield wt (1)	4500	4500 lbf shadow shield wt from LeRC propul task order
[118] RCS prop wt	704	Transfer RCS dV = 20, Isp = 300, storable biprop
[696] ***Frame & propul 'dry' wt***	19777	

Mars dep & Earth capt stages - 2016 NERVA NTR Ref Veh

14% tank fraction, 3% cooldown penalty, 2% prop margin, Isp = 925 sec Rev 5 5/22/90

		Element	*Mass (km)*	*Rationale*
Mars dep	[128]	Mars dep usable prop load	53168	***Mars dep dV= 3900 m/s; eng Isp=925 sec,*** H2 density =70.8
	[703]	Mars dep prop residuals	1063	2% residuals/reserve left after boiloff,burn and cooldown
	[699]	Mars dep burn 'cooldown' prop	1595	3% post burn prop for reactor cooldown; no thrust/Isp counted in this approximation
	[498]	Mars dep stg outbound boiloff	1792	Out b boiloff for given MLI & VCS insul.;no refrig,based on Boeing 'CRYSTORE'
	[545]	Mars dep stg inorbit boiloff	361	31.5 day inorbit stay time
	[122]	Inbound midcourse prop	1266	Inb midc maneuver dV=90 m/s; done by main propulsion system
	[711]	**Tot Mars dep stg prop load**	**59245**	total at time of TMI burn
Earth capt	[561]	Earth arr stg usable prop tot	23638	***Earth arr dV=2629 m/s; propulsive burn capture into 500 km by 24 hr ellip orbit***
	[704]	Earth arr stg prop residuals	472	2% residuals/reserve left after boiloff,burn and cooldown
	[700]	Earth arr stg 'cooldown' prop	709	3% post burn prop for reactor cooldown; no thrust/isp counted in this approximation
	[562]	Earth arr stg outbound boiloff	2937	434 day b.off period; additional b.off from this tank also accounted in M dep p b.off
	[563]	**Total Earth arr stg prop load**	**27756**	Total at time of TMI burn
	[570]	**Total combined prop load**	**87001**	M dep/E arr prop: put in 1 tank along veh centerline aids NTR radiation attenuation
Common tank				***1 continuous reinforced Silicon Carbide/Al metal martrix tank:*** dia:10.m, L:19.0m,
	[683]	Single M dep/E arr tank wt	5986	filament wound; dens= 2436 kg/m3; 37ksi Wk. stress; tank skin thickness = 4.0 mm
	[684]	MLI wt	1978	MLI: density = 32 (kg/m3); 200 layers at 20 layers/cm. wt=SA x no. layers x dens
	[685]	Vapor cooled shield wt	1563	2 VCS - 2 x 0.13 mm Al sheets with 0.57 kg/m2 honeycomb core each
	[686]	Meteoriod shield wt	1323	One 0.80 mm sheet of Al; comparsion: SSF plans 0.8 mm, Mariner 9 used 0.4 mm
	[687]	Propel line/valves wt	225	length =10 m,double wall stainless steel H2 prop line; density= 7833 kg/m3, t=0.8mm
	[689]	Mass growth wt	2770	25% wt growth for tank shell, MLI,VCS,meteor shield,prop line & attachment
	[565]	Sum of inerts:single tank	13845	Total for single tank with all tank related inerts.
	[566]	Total for 1 tank	13845	Overall tank fraction [571] = 13.7 %
	[572]	***Combined Mars dep/Earth arr tank set & propellant load***	**100846**	Total for ' Mars dep/Earth arr tank set ' at time of TMI burn
	[171]	***IMLEO***	**735190**	

Mac Chart: M NTR E arr v/r

Boeing vehicle synthesis model run #: marsntrmtv.dat;55

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Earth dep & Mars capt stages - 2016 NERVA NTR Ref Veh

14 % tank fraction, 3% cooldown penalty, 2% prop margin, Isp = 925 Rev 5 5/22/90

Earth dep stg wt	[586]	Earth dep usable propel tot	272520	*Earth dep dV: 4182 m/s (includes 200 m/s gloss for 2 burn E dep) ; Isp = 925 sec*
	[701]	Earth dep prop residuals	5450	2% residuals/reserve left after boiloff, burn prop, and cooldown
	[697]	Earth dep burn 'cooldown' prop	8176	3% post burn prop for reactor cooldown: no thrust/Isp counted for this estimated %
	[705]	**Tot Earth dep stg prop load**	**286146**	
				2 continuous reinforced Silicon Carbide /Al metal matrix tanks: dia: 10.0 m,
	[668]	Single tank wt (cyl/ellip ends)	9198	L:30.0m, dens= 2436 kg/m3; 37ksi wk. stress, thickness = 4.0 mm, root 2 ellip ends
	[669]	MLI wt	3040	MLI: density = 32 (kg/m3); 200 layers at 20 layers/cm. wt=SA x no. layers x density
	[670]	Vapor Cooled Shield wt	2401	2 VCS: at 2 x 0.13mm Al outer sheets with 0.57 kg/m2 honeycomb core each
	[671]	Meteoriod shield wt	2039	One 0.80 mm sheet of Al; comparison: SSF uses 0.8 mm, Mariner 9 used 0.40 mm
	[i312]	Tank/frame attachment	400	Tank attachment mounting brackets & hardware as well as tank release mechanism
	[672]	Tank feed prop line wt	159	Short prop line from tank to main prop line:double wall, stainless steel: 10 meter
	[674]	Mass growth wt	4309	25% wt growth for tank inert,MLI,VCS,meteor shield,prop lines, tank/veh attachment
	[588]	Sum of single tank inerts	21546	Total single tank inert wt;
	[592]	Total for 2 tanks	43092	Total for ' Earth dep tank set ': inert wt; Overall tank fraction [593] = 13.1 %
	[597]	***Earth Dep stage tot wt***	**329238**	Total Earth dep stg weight at time of Trans Mars Injection burn
Mars arr stg wt	[610]	Mars arr usable prop tot	137450	*Mars arr dV: 3870 m/s; eng Isp=925,* H2 density = 70.8
	[702]	Mars arr prop residuals	2749	2% residuals /reserveleft after boiloff, burn prop, and cooldown
	[698]	Mars arr burn 'cooldown' prop	4124	3% post burn prop used for reactor cooldown; prelim;based on Westingh. estimate
	[611]	Mars arr stg outbound boiloff	3349	Boiloff for given MLI,VCS and Outb trip time; based on Boeing's 'CRYSTORE' prog
	[121]	Outbound midcourse prop	4008	Outb midc maneuver dV = 120 m/s; done by main propulsion from M arr tanks
	[612]	**Tot Mars arr stg prop load**	**151680**	
	[633]	Single M arr tank wt	5346	*2 SiC/Al metal matrix tanks;* dia:10.0 m, L:17.0 m, dens=2436 kg/m3;thick=4.0mm
	[634]	MLI wt	1737	MLI: density = 32 (kg/m3); 200 layers (4 inches) at 20 layers/cm
	[635]	Vapor cooled shield wt	1396	2 VCS: at 2 x 0.13 mm Al sheets with 0.57 kg/m2 honeycomb core each
	[636]	Meteoriod shield wt	1185	One 0.80 mm sheet of LiAl: assumption-LEO assembley in protective hanger
	[i312]	Tank/frame attachment	400	Tank attachment mounting brackets & hardware as well as tank release mechanism
	[637]	Tank feed prop line wt	159	Double wall, stainless steel 10 meter H2 propellant line;dens= 7833 kg/m3, t=0.8mm
	[639]	Mass growth wt	2563	25% wt growth for tank inert,MLI,VCS,meteor shield,prop line&tank/veh attachment
	[614]	Sum of single tank inerts	12786	Total for single tank, with all inerts.
	[615]	Total for 2 tanks	25572	Overall tank fraction [620] = 14.5 %
	[617]	***Mars Arr stage wt***	**177252**	wt at time of Earth departure

Reference NERVA NTR design delta V parametric data

Vehicle IMLEO is plotted vs Earth departure dV (m/s) and Mars capture dV (m/s) for a range of Mars departure dV's (m/s). The Boeing reference NERVA NTR vehicle configuration was used with the following vehicle characteristics and assumptions:

(1) ECCV crew return, vehicle expended

(2) MTV crew of 4 habitat module consists of the following:
  a. dry crew hab wt = 28531 kg. includes: 1802 kg rad shelter, 1530 kg external airlocks (2) 1539 kg external solar array power system
  b. consumables = 4422 kg (4 crew for 434 days) The data does not account for consumables weight variation with changes in mission duration
  c. on board 'resupply' mass = 986 kg
  d. transfer science equipment, hardware, supplies etc = 1000 kg
  Total MTV crew module mass = 43939 kg

(3) LO2/LH2; 2 TMI tanks, 2 MOC tanks, single TEI tank at an approximate tank fraction of 14%
(4) N2O4/MMH storable RCS system, Isp=280 sec
(5) no artifical-g
(6) 120 m/sec outbound midcourse correction burn, 90 m/sec inbound
(7) MEV is propulsively captured at Mars with main NTR stage
(8) propellant boiloff was calculated for a 434 day mission; i.e. this data takes no account of boiloff variation with changes in mission duration

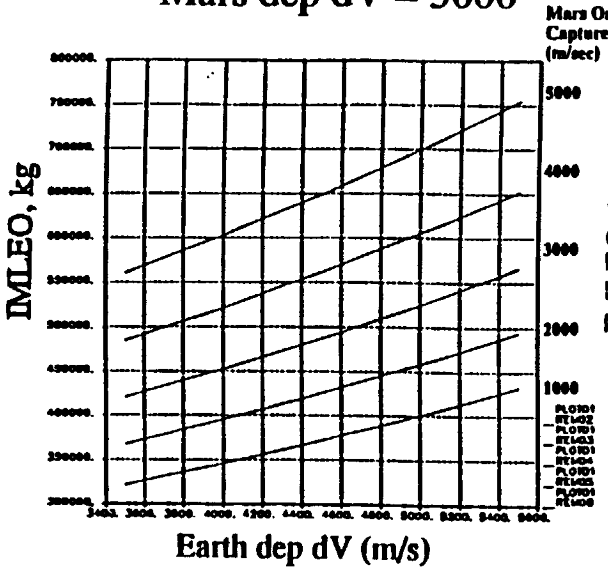
Reference NERVA NTR Design Delta V Parametric Data

Vehi IMLEO vs E dep dV, Mars capt dV & Mars dep dV, ECCV Ret

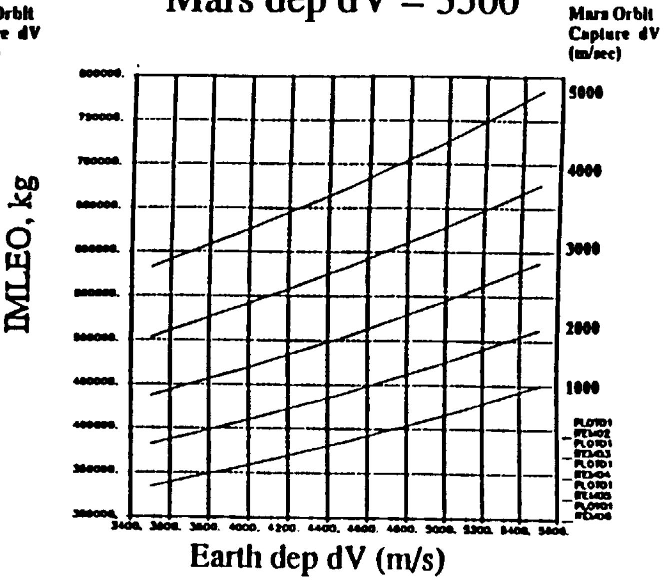
Vehicle Characteristics

Isp=925
MEV w heat shield= 73118 kg
MTV module total = 34939 kg
NERVA engine wt = 9684 kg
Shadow shield wt = 4500 kg
Truss & tank struts=3000 kg
Five 10 m dia tanks, 14% t.f.
ECCV crew return =7000 kg
2 midcourse corr burns
Zero-g, storable RCS prop
Departs from LEO
Mars orbit: 250 km by 1 sol

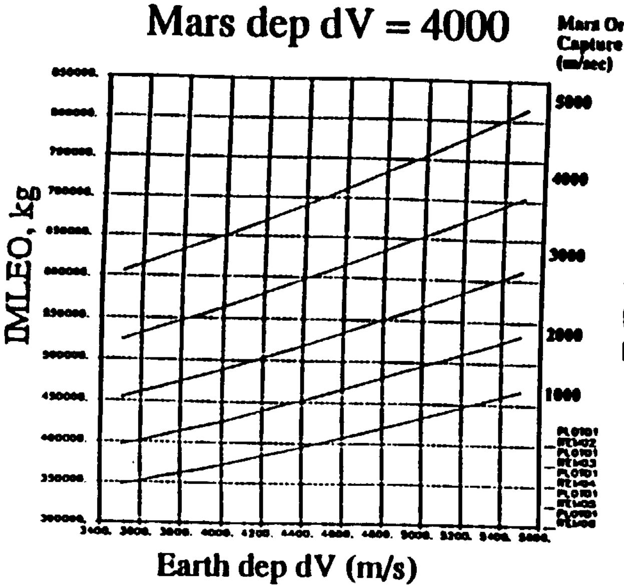
Mars dep dV = 3000



Mars dep dV = 3500



Mars dep dV = 4000



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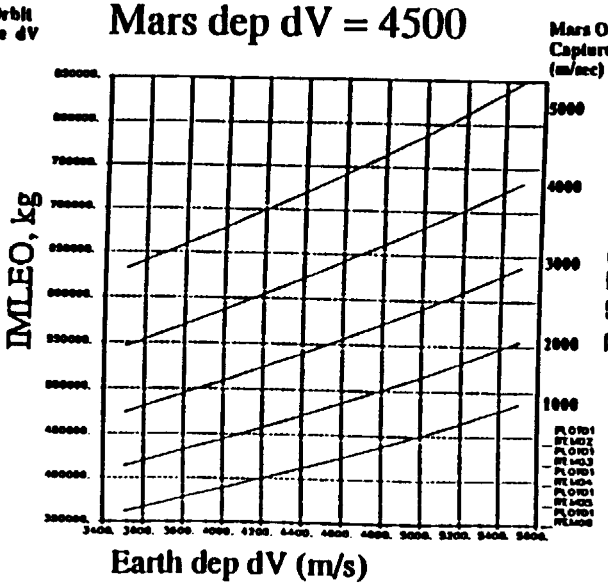
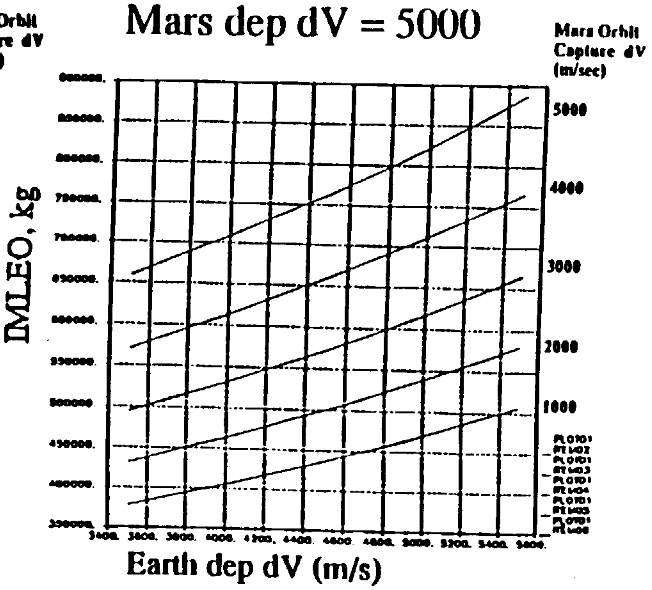


Mars dep dV = 5000



consumables variation with trip time not accounted

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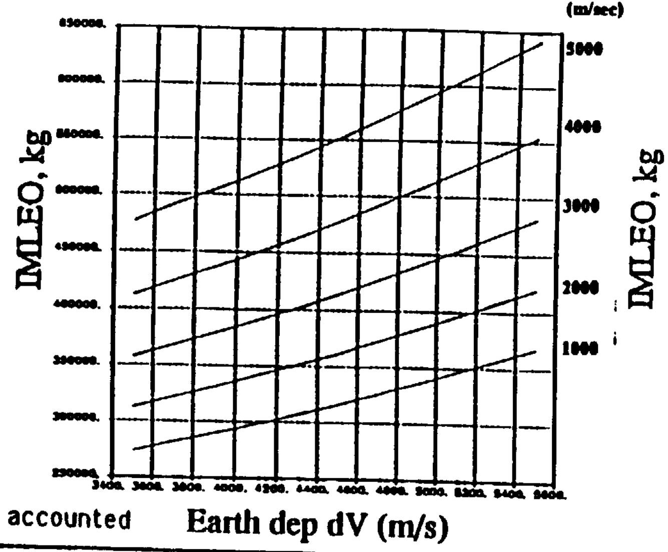
Reference NERVA NTR Design Delta V Parametric Data

Vehi IMLEO vs E dep dV, Mars capt dV & Mars dep dV, ECCV Ret

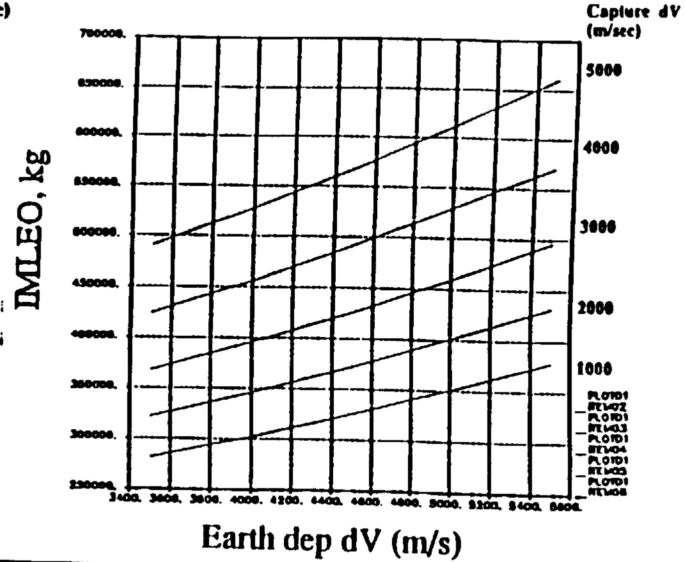
Vehicle Characteristics

Isp=925
MEV w heat shield= 73118 kg
MTV module total = 34939 kg
NERVA engine wt = 9684 kg
Shadow shield wt = 4500 kg
Truss & tank struts=3000 kg
Five 10 m dia tanks, 14% t.f.
ECCV crew return =7000 kg
2 midcourse corr burns
Zero-g, storable RCS prop
Departs from LEO
Mars orbit: 250 km by 1 sol
consumables variation with trip time not accounted

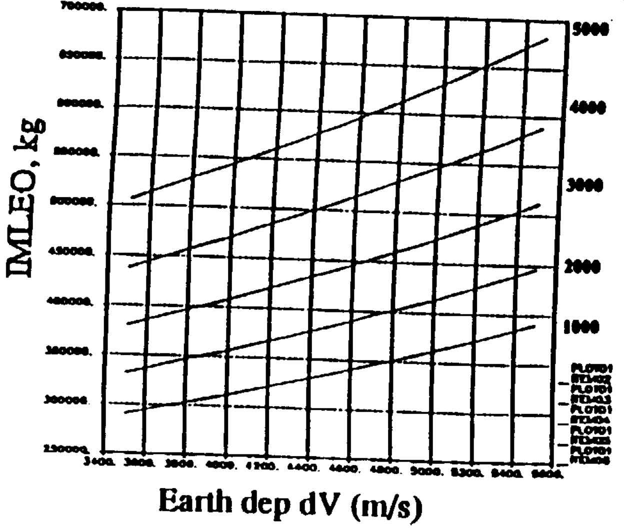
Mars dep dV = 500



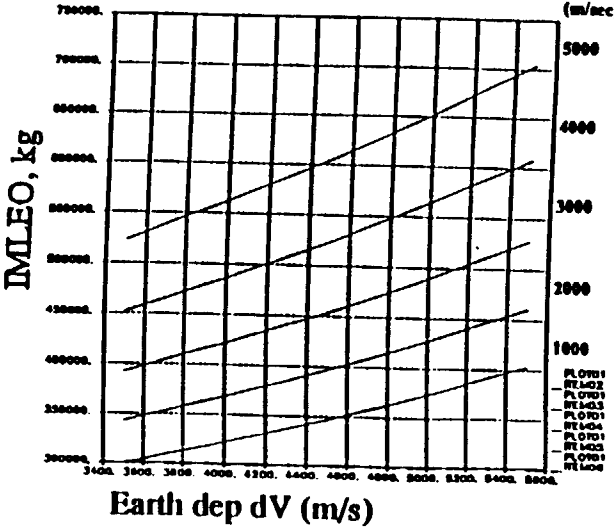
Mars dep dV = 1000



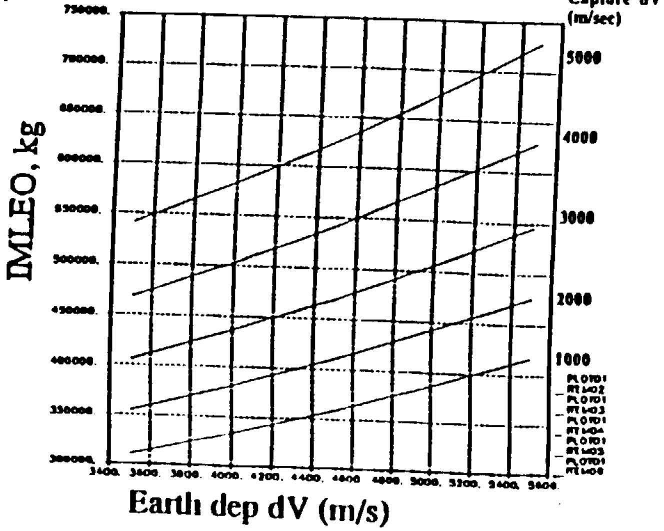
Mars dep dV = 1500



Mars dep dV = 2000



Mars dep dV = 2500



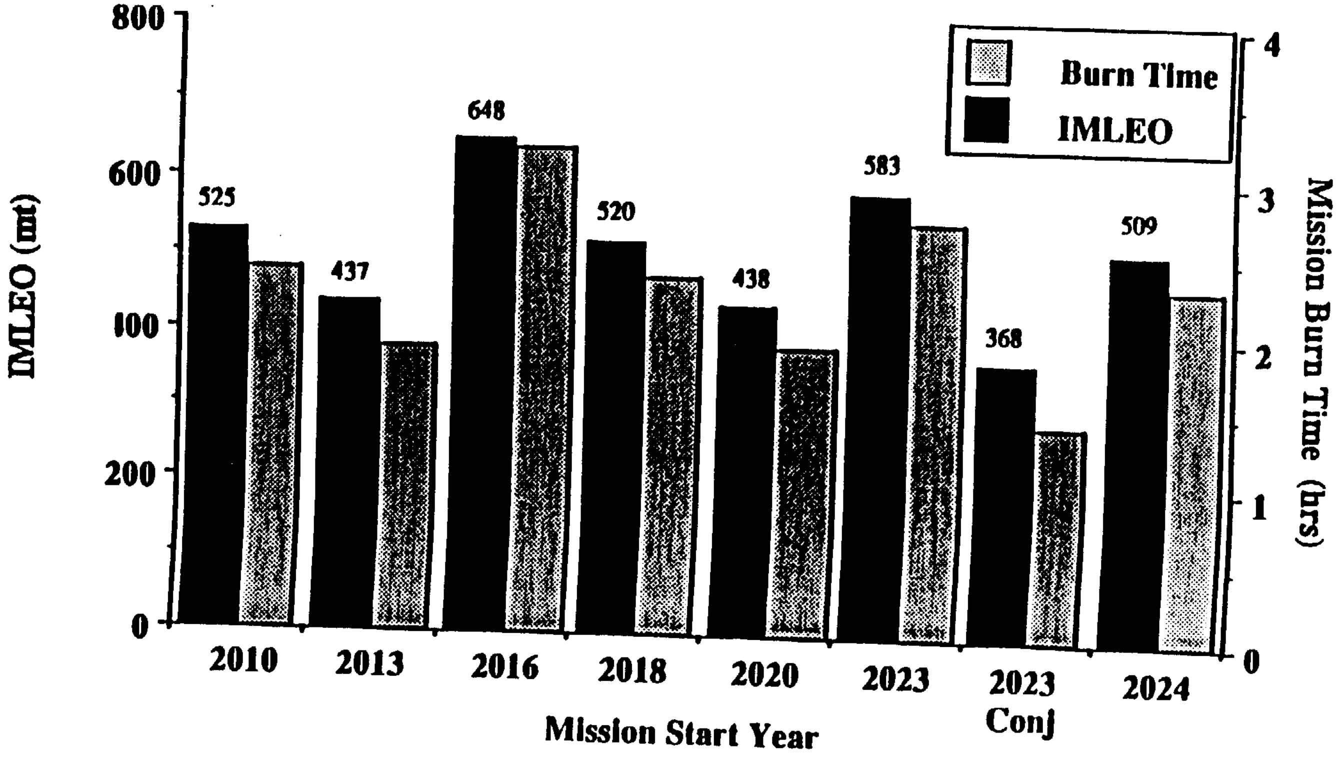
NTR vehicle weight vs opportunity year

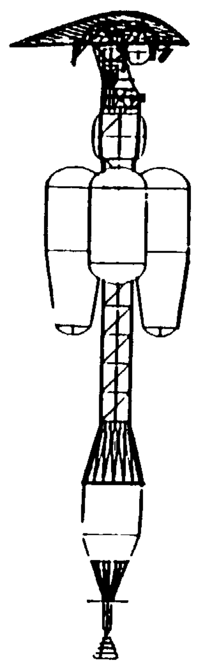
An earlier Boeing reference 2016 NTR vehicle (slightly lighter than the present 735 t reference) that utilized a NERVA derivative engine of 75000 lbf thrust, and a mass of 9684 kgs (t/w=3.5) was, for this particular trade, replaced with a Particle Bed Reactor (PBR) of the same thrust but with a t/w=10 (mass = 3401 kg). IMLEO figures for this modified craft were determined for 8 missions in the 2010 to 2024 time frame using the Boeing vehicle synthesis model. The results show that the 2016 reference mission trajectory proved to require the most propellant of all those evaluated. Engine burn time in hours is also listed on the chart with the IMLEO figure.

NTR Vehicle Weight vs. Opportunity Year

for Reactor/Engine T/W = 10

Revision 2 5/15/90





Engine Characteristics

- Eng T/W=10 Eng weight=3402 kg
- Engine Thrust = 75,000 lbf
- Engine Isp = 925 sec
- Reactor shadow shield wt = 4.5mt

!STCAEM/mha/31May90

Vehicle Characteristics

- Opposition missions except 2023 conjunction
- 3 burn Earth departures: g-loss: 300 m/s
- Crew of 4, 33 t MTV, 76 t MEV includes 25mt surf p/l
- Propulsive capture at Earth into 500km 24hr elliptical orbit

synthesis model run #: marsntrmtv.dat;123-130
Mac chart: NTR IMEO/burn time/yr 5/15/90

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Nuclear Electric Propulsion (NEP)

Contained within this section are a vehicle description, operation mode, and vehicle mass statement. Further details can be found in the NEP IP&ED document.

The NEP vehicle uses thrust obtained as a result of charged particles accelerated through an electric field. Argon propellant is first ionized in the thruster discharge chamber. The propellant, which is in a plasma state, is contained within the discharge chamber by a magnetic field. The propellant then "drifts" towards the accelerating grid where the charged particles are repelled out at an extremely high velocity. The charged particles must then be neutralized to prevent them from coming back to the spacecraft, which would negate thrust. An issue confronting the propulsion system involves the expected lifetime of the thrusters due to cathode and grid erosion. Expected thruster lifetime is 10,000-20,000 hrs.

The NEP creates electrical power necessary for the propulsion system with a nuclear reactor power system. The reactor power system is composed of twin uranium fast reactors. The reactors heat a working fluid which is used to drive turboalternators. The expansion of the working fluid drives the alternators, producing electricity. The working fluid must then be cooled for reuse through a radiator subsystem. The electrical power is then conditioned for transmission and sent to the thruster system on the distribution bus. Expected power plant lifetime is 10 years. Disposal locations of the spent reactors are TBD.

Mission analysis for various vehicles has revealed that high power levels (20-40 MWe) coupled with low vehicle alpha's (4-7 kg/kW) offer fast trips and low associated IMLEO (400-600 t) for most mission opportunities. Alpha is defined as the specific mass of the vehicle and has the units of kg/kW. Since vehicle alpha's play such an important role in vehicle performance, this technology area must be given serious attention early in the development program.

Certain gravity assists offer significant benefits for electric propulsion, without imposing launch window restrictions. The gravity assists that offer benefits are a Lunar fly-by, Mars fly-by, and an Earth fly-by. During Earth escape, the vehicle swings by the moon to gain a velocity boost on the order of 600-1000 m/s. During a Mars fly-by, the vehicle approaches Mars with excess velocity, drops the MEV off, and continues in heliocentric space in close proximity to Mars. When the vehicle decelerates enough to capture at Mars, the vehicle enters a highly elliptic orbit to allow the MEV multiple attempts to rendezvous with the transfer vehicle. The time frame for vehicle deceleration and Mars capture is calculated to be the same as the surface stay time. An Earth fly-by is similar to a Mars fly-by in the sense that the vehicle starts the deceleration phase of the mission leg, later than it normally would. As the transfer vehicle approaches the Earth with excess velocity, the crew is dropped off and the vehicle continues in heliocentric space. When an Earth fly-by is employed, the transfer vehicle cannot rendezvous back with the Earth for a considerable length of time (~200 days). This length of time may be detrimental to thruster lifetime. Therefore, the recommended gravity assists are Lunar and Mars fly-bys. These fly-bys can offer trip time reductions on the order of 40 days total.

A major operational issue confronting the NEP is departure and refurbishment orbits. Due to differential nodal regression, severe debris environments, and Van Allen belt radiation, the NEP is forced to operate from LEO (400 km) or GEO (35,000 km) and higher. A LEO operational node would offer the greatest advantages for the NEP, if nuclear safety operational issues can be resolved. Preliminary analysis from Bolch *et al*, Texas A&M [A Radiological Assessment of Nuclear Power and Propulsion Operations Near Space Station Freedom, NAS3 25808, March 1990], indicates that a multi-megawatt vehicle **can** operate safely in LEO. Electric propulsion, unlike ballistic trajectories, spirals in and out of Earth Orbit in a circular path. This type of circular spiral eliminates the risk of accidental Earth atmosphere re-entry.

Nuclear Electric Propulsion Vehicle Reference Configuration

Introduction

The Nuclear Electric Propulsion (NEP) Mars transfer concept offers advantages of a reusable, extremely high-I_{sp} (10,000 sec) system; a fully propulsive capture at Mars and Earth which avoids the need for high energy aerobraking; great mission flexibility (relative insensitivity to mission opportunity, capture orbit astrodynamics, or changes in payload mass) and low resupply mass (the argon propellent required amounts to roughly a third of total vehicle mass). Disadvantages of the concept are its high technology development cost; complex, high-performance power system and large, liquid-metal radiator system.

Nominal Mission Outline

- The NEP vehicle is assembled and checked out in LEO
- TMI is a slow spiral out of Earth's gravity well
- Just prior to Earth escape, the crew transfers onboard using an LTV
- Thrust continues throughout the interplanetary transfer, first accelerating relative to Earth and then decelerating relative to Mars, except for a 45 - 60 day no-thrust hiatus enroute.
- MTV flies by Mars with low relative encounter velocity
- MEV separates from MTV for aeroentry
- MEV descends to surface, jettisoning aerobrake prior to landing
- Surface operations ensue
- MTV continues decelerating into loosely captured, highly elliptical orbit
- Ascent vehicle leaves descent stage and surface payload on surface
- MAV rendezvous occurs at MTV periapsis; berthing and crew transfer
- MAV jettisoned in Mars orbit
- Reversal of interplanetary acceleration / coast / deceleration sequence
- Crew departs MTV for direct entry at Earth
- MTV spirals back to LEO for refurbishment (optional loose capture at L2 is attractive, if refurbishment infrastructure is available there and if resupply trips

from LEO use EP or beamed power propulsion for high efficiency)

Vehicle Systems

Primary vehicle systems are: **power plant** at the bow; **radiators** amidships; main **propulsion** astern; vehicle **bus**; and **crew systems** near the stern.

Power plant - The power plant consists of reactors, shadow shields, boiler (heat exchanger), electromagnetic pumps, and turbo-alternators. Two fast-spectrum (UN-W/25Re) reactors are used for redundancy. The reactors are positioned in line with the main vehicle axis to maximize mutual shielding of the rest of the vehicle. A radiation shield (WBe_2C/B_4C composite) is required aft of the reactors to protect the crew and sensitive electronic equipment from direct and scattered neutron and gamma fluxes. The shield is shaped to produce a shadow-cone with rectangular cross-section, tailored to the reactors' view of the rest of the vehicle. Lithium is the primary coolant, pumped by redundant electromagnetic pumps through the boiler. The secondary, potassium loop, also pumped electromagnetically, carries heat from the boiler to the turbo-alternator assembly. There are 5 pairs of turbo-alternators (3 primary and 2 backup pairs), which generate 40 MWe for propulsion. Each turbo-alternator pair counter-rotates to cancel its gyroscopic acceleration. This machinery is configured to permit straightforward robotic maintenance access when the reactors are not running, but the entire turbo-machinery assembly can be launched as one unit in a 10 m launch shroud, already integrated with the pumps, boiler and dormant reactors. The potassium runs through the condenser pipes which form the vehicle spine along the length of the radiator system. Reduced-diameter, armored pipes return the low-quality (mostly liquid) potassium to the boiler to complete the loop.

Radiators - The radiator system consists of a primary assembly, an alternator assembly and an auxiliary assembly. A typical assembly consists of several hundred individual, identical, sodium-containing, carbon/carbon heat pipes, whose evaporator ends are bonded mechanically to the secondary-loop condenser pipe. Their radiator fins are oriented in the plane of the overall array, and are bonded mechanically together for overall structural stiffness. The primary assembly cools the secondary-loop potassium; the alternator assembly cools the dynamic power conversion system (turbo-alternators); the auxiliary assembly provides cooling to the electromagnetic pumps during normal operations, as well as to the reactors during shutdown.

Propulsion - The propulsion system includes engine assembly, propellant storage subsystem, and plumbing. The engine assembly has 40 individual ion thrusters (including 10 spares) in a 5 x 8 rectangular array. Each thruster is 1 m wide by 5 m long; beam neutralizers are located between the thrusters. The argon propellant is stored cryogenically in insulated, spherical tanks, mounted on the forward side of the engine assembly via structural and fluid quick-disconnects. Including tanks, the propellant storage system masses 185 t (~ 35% overall vehicle IMLEO). This low propellant mass is a strong resupply advantage.

Vehicle bus - Thrust loads are extremely low for the EP system. Probable maximum loading is from impulses like ACS firings, berthing operations, and construction and maintenance activity. The primary vehicle structure is the armored, liquid-metal-carrying condenser pipes of the conversion and radiator systems. Additional lightweight, out-of-plane stiffening structure for the large, flat radiator panels is not shown. Astern of the radiators, an SSF-type truss continues the vehicle spine. The crew systems are attached to this, and the power feeds for the engines are deployed within it. Two communications satellites are embedded in the truss near the crew systems, to be deployed in Mars orbit for maintaining communication with Earth. Also mounted to the truss and not shown are deployable solar arrays which provide habitat and vehicle power when the nuclear power system is shut down (during LEO operations and interplanetary coast).

Crew systems - The crew systems consist of a long-duration transit habitat and one or more MEVs (the reference design shows one MEV). All habitable volumes are contiguous throughout each mission. The crew systems are wrapped around and hung on the vehicle bus, as far from the nuclear sources as practical without propulsion interference. The separation shown reflects an initial radiation shadow shield designed for crew system separation exceeding 100 m. Electric propulsion has the least sensitivity to increased payload mass, so an important option is provision for multiple MEVs. A multiple docking adapter (not shown), would

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allow several MEVs to be used without altering the vehicle configuration (additional propellant tanks would be required).

NEP

Trades and Rationale

- Extremely high Isp
- Fully propulsive capture at mars and earth , avoids high energy aerocapture.

Mission Modes And Operations

- Vehicle assembled in SSF orbit.
- Crew transfer and departure from SSF orbit.
- Propulsive capture at Mars.
- MEV/Aerobrake separate from vehicle prior to entry and landing.
- Aerobrake separates from MEV prior to landing.
- Crew cab-ascent after surface mission, leaving lander, surface hab.
- Crew cab left in Mars orbit after rendevous, docking and crew transfer.
- TEI
- Propulsive capture at earth and crew transfer to SSF.

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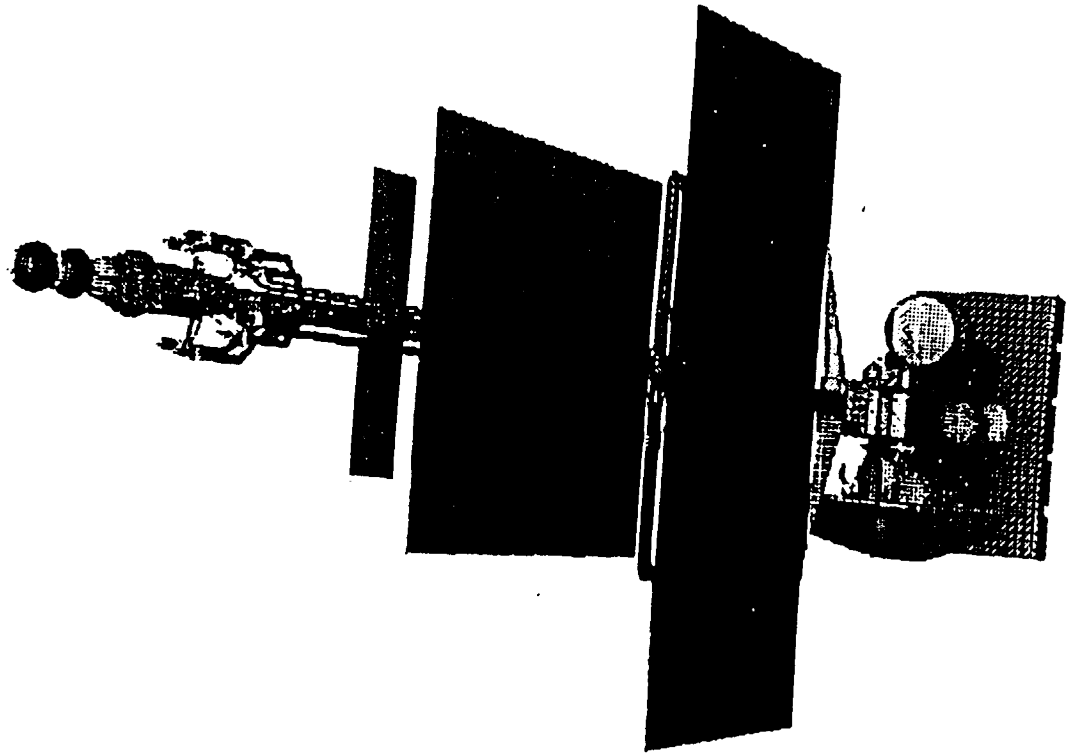
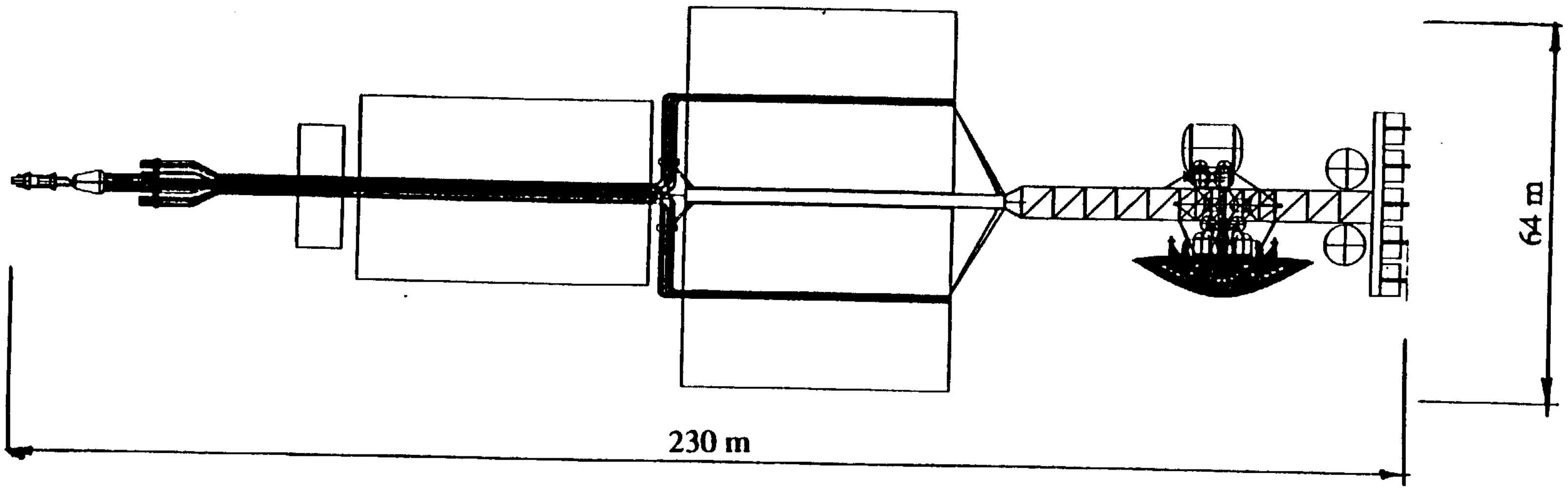
μg NEP Interim Configuration

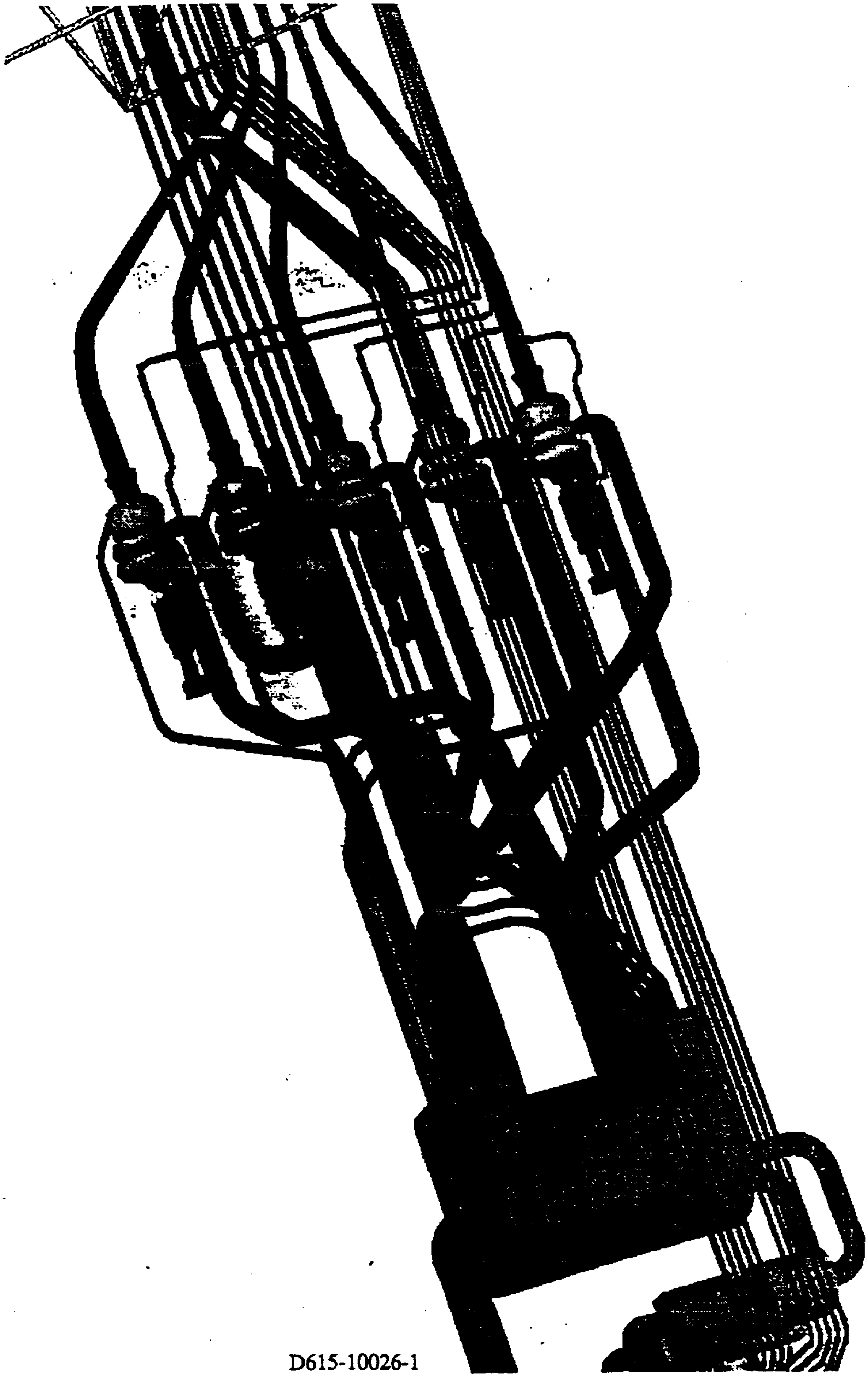


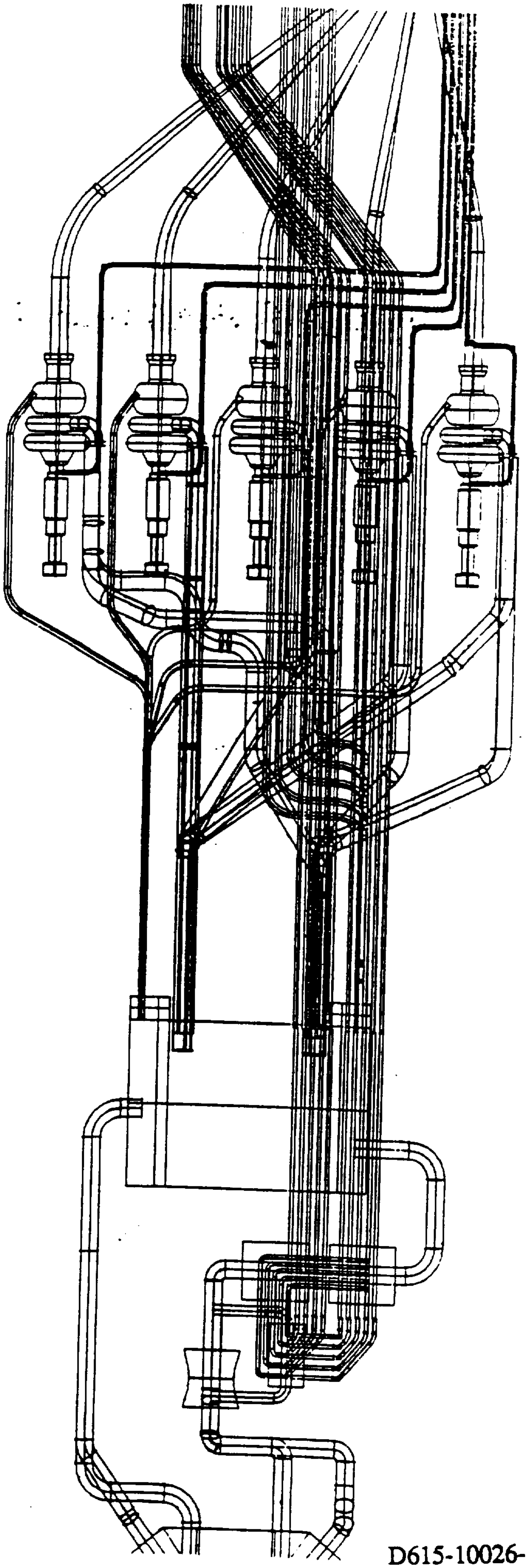
NEP Configuration

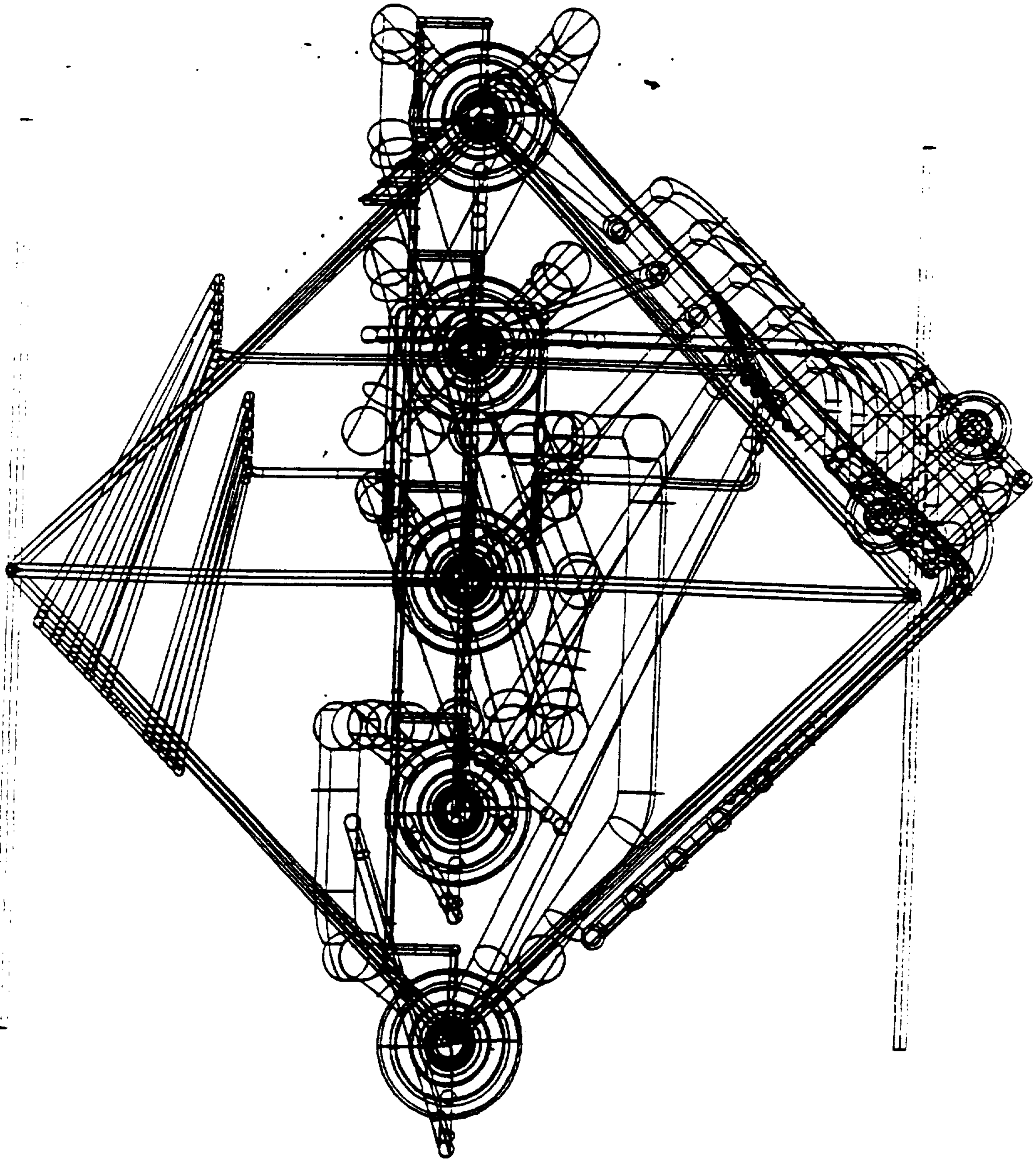
The following charts depict the reference nuclear electric propulsion vehicle that has been modeled on the Intergraph CAD workstation. Many views are shown to provide the detail that the vehicle has been designed to. The vehicle model has verified conceptual design.

Micro Gravity NEP









Micro-Gravity NEP Mass Statement

Payload	Mass in metric tonnes
Descent Aerobrake	7.0
MEV Descent Stage	18.7
MEV Ascent Stage	22.5
Surface Equipment	25.0
Transit Hab Module	44.3
	117.5

Propulsion	
Reactor 1	7.4
Reactor 2	7.4
Shield	8.6
Primary Heat Transport System	20.1
Auxiliary Cooling Subsystem	2.2
Boiler	21.6
Turboalternators	16.3
Alternator Radiator	2.6
Turbopumps	.4
Rotary Fluid Management Device	3.1
Main Cycle Radiator	10.6
Main Cycle Condenser	1.3
Main Cycle Plumbing	5.0
Auxiliary Cycle Radiator	.5
Auxiliary Cycle Condenser	1.3
Auxiliary Cycle Plumbing	6.0
Power Conditioning Radiator	1.1
Plumbing Insulation	4.1
Engine Assembly	23.5
Power Management & Distribution	68.0
	211.1

Structure	
5 Meter Bay Graphite-Epoxy Truss	4.5
Pressurized Berthing Adaptor	6.6
	11.1

Utilities	
Communications	.6
Attitude Control	5.7
Avionics	2.5
Houskeeping Power Distribution	.5
PV/RFC Power Subsystem	2.3
Robotics	3.6
	15.2
Tanks	3.3
Feed Lines	0.1
Propellant	167.2

Total	525.5
15% growth	35.6
IMLEO	**561.1 t**
Resupply	**339.2 t**

Trip Time = 490 days , alpha = 6.8 kg/kW

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Solar Electric Propulsion (SEP)

Contained within this section are a vehicle description, operation mode, and vehicle mass statement. Further details can be found in the SEP IP&ED document.

The SEP vehicle uses thrust obtained as a result of charged particles accelerated through an electric field. Argon Propellant is first ionized in the thruster discharge chamber. The propellant, which is in a plasma state, is contained within the discharge chamber by a magnetic field. The propellant then "drifts" towards the accelerating grid where the charged particles are repelled out at an extremely high velocity. The charged particles must then be neutralized to prevent them from coming back to the spacecraft, which would negate thrust. An issue confronting the propulsion system involves the expected lifetime of the thrusters due to cathode and grid erosion. Expected thruster lifetime is 10,000-20,000 hrs.

The SEP creates electrical power necessary for the propulsion system by converting energy from the sun into electricity through the use of solar arrays. The solar array is configured in multiple strings to insure redundancy. The loss of individual cells to debris and degradation damage is taken into account within the design. Direct screen drive enables the elimination of high voltage power processors. Low voltage power processors are still needed for heaters, ionizing potential, and other vehicle housekeeping tasks. The power generated from the arrays is piped to the thruster pods where the ion engines are located. Expected power plant lifetime is 10 years.

Mission analysis for various vehicles has revealed that power levels around 8-15 MW offer reasonable trip times and low IMLEO. Increasing power raises the thrust level, but the vehicle alpha (vehicle specific mass, kg/kW) goes up as well. When both the power plant mass and the power level increase you enter the dilemma of more power to push more mass. In other words, there is a point where increasing power level doesn't buy much since the mass has gone up as well. Since the vehicle is dominated by solar arrays, structure, and ion engines, the vehicle alpha doesn't decrease as it does for the NEP. Typical vehicle alpha's associated with SEP are in the 8-12 kg/kW for multi-megawatt vehicles. Typical trip times for these types of vehicles are on the order of 540-620 days.

Certain gravity assists offer significant benefits for electric propulsion, without imposing launch window restrictions. The gravity assists that offer benefits are a Lunar, Mars, and Earth fly-bys. During Earth escape the vehicle swings by the moon to gain a velocity boost on the order of 600-1000 m/s. During a Mars fly-by, the vehicle approaches Mars with excess velocity, drops the MEV off, and continues in heliocentric space in close proximity to Mars. When the vehicle decelerates enough to capture at Mars, the vehicle enters a highly elliptic orbit to allow the MEV multiple attempts to rendezvous with the transfer vehicle. The time frame for vehicle deceleration and Mars capture is calculated to be the same as the surface stay time. An Earth fly-by is similar to a Mars fly-by in the sense that the vehicle starts the deceleration phase of the mission leg later than it normally would. As the transfer vehicle approaches the Earth with excess velocity, the crew is dropped off and the vehicle continues in heliocentric space. When an Earth fly-by is employed, the transfer vehicle cannot rendezvous back with the Earth for a considerable length of time (~200 days). This length of time may be detrimental to thruster lifetime. Therefore, the recommended gravity assists are Lunar and Mars fly-bys. These fly-bys can offer trip time reductions on the order of 40 days total.

A major operational issue confronting the SEP involves the Earth escape spiral. The baseline operational mode calls for crew rendezvous with the SEP a few days prior to

Earth escape via Lunar Transfer Vehicle. The Earth escape spiral takes 50-100 days in the 10 MW range, spending too much time in the Van Allen belts for possible crew exposure. Radiation associated with the Van Allen belts causes considerable damage to the solar array while the SEP passes through the belts. Due to this degradation, the SEP must somehow get through the belts without the interplanetary array. Three possible solutions to this dilemma are (1) transfer by chemical boost stage, (2) transfer array scenario, or (3) transfer by a beamed power EOTV. A chemical boost stage would effectively double the IMLEO of the SEP, and is not recommended as a solution. The SEP truss structure is also not sized for the loads of a high thrust system. A promising solution is to carry 2 arrays; one array for the interplanetary transfer and one array for the Earth escape spiral. Once the vehicle has passed through the belts, it drops the transfer array at a location where the array could possibly be used by another operation (beamed power) and deploys the main array. On subsequent missions, the SEP can stage at L2 and have resupply requirements furnished by a beamed power EOTV.

Solar Electric Propulsion Vehicle Reference Configuration

The solar electric propulsion (SEP) Mars transfer concept is the only non-nuclear advanced propulsion option. It offers advantages of the lowest IMLEO of the four reference vehicles; a reusable, extremely high-I_{sp} (5,000 sec) system; a fully propulsive capture at Mars and Earth which avoids the need for high energy aerobraking; great mission flexibility (relative insensitivity to mission opportunity, capture orbit astrodynamics, or changes in payload mass) and low resupply mass (the argon propellent required amounts to roughly a third of total vehicle mass). Disadvantages include uncertainty about how economical the production of acres of solar arrays can become, and the need to deploy and control a relatively fragile vehicle, which is bigger than six football fields, in space.

Nominal Mission Outline

- The SEP vehicle is assembled and checked out in LEO
- TMI is a slow spiral out of Earth's gravity well
- Just prior to Earth escape, the crew transfers onboard using an LTV
- Thrust continues throughout the interplanetary transfer, first accelerating relative to Earth and then decelerating relative to Mars, except for a 45 - 60 day no-thrust hiatus enroute.
- MTV flies by Mars with low relative encounter velocity
- MEV separates from MTV for aeroentry
- MEV descends to surface, jettisoning aerobrake prior to landing
- Surface operations ensue
- MTV continues decelerating into loosely captured, highly elliptical orbit
- Ascent vehicle leaves descent stage and surface payload on surface
- MAV rendezvous occurs at MTV periapsis; berthing and crew transfer
- MAV jettisoned in Mars orbit
- Reversal of interplanetary acceleration / coast / deceleration sequence
- Crew departs MTV for direct entry at Earth
- MTV spirals back to LEO for refurbishment (optional loose capture at L2 is attractive, if refurbishment infrastructure is available there and if resupply trips from LEO use EP or beamed power propulsion for high efficiency)

Vehicle Systems

Primary vehicle systems are: **power plant**; main **propulsion**; vehicle **bus**; and **crew systems.**

Power plant - The power plant consists primarily of a field of solar arrays kept normal to the sun line at all times. The solar array area required to produce 10 MWe of power is ~ 35,000 m^2 and is maintained sufficiently rigid and in position by a deployable area truss (spaceframe) one bay deep. Details of deployment of the lightweight solar cell blankets across the structure are not yet worked out.

Propulsion - The propulsion system includes engine assembly, propellant storage subsystem, and plumbing components, split into two identical modules located at distal ends of the vehicle bus. Each engine assembly has 5 individual ion thrusters (the total of 10 includes 2 spares) in a 5 x 8 rectangular array. Each thruster is 1 m wide by 5 m long; beam neutralizers are located between the thrusters. The argon propellant is stored cryogenically in insulated, spherical tanks, mounted on the forward sides of the engine assemblies via structural and fluid quick-disconnects. Including tanks, the propellant storage system masses ~ 35% overall vehicle IMLEO. This relatively low propellant mass is a strong resupply advantage.

Vehicle bus - Thrust loads are extremely low for the EP system. Probable maximum loading is from impulses like ACS firings, berthing operations, and construction and maintenance activity. The primary vehicle bus structure has two components: the area truss covered by the solar array field, and truss outriggers extending sufficiently bar beyond the edge of the solar array that the ion engine plumes do not impinge on, and therefore erode, the power system. The crew systems are attached to the underbelly of the area truss (in the center for mass balance). Two communications satellites are also attached to the truss near the crew systems, to be deployed in Mars orbit for maintaining communication with Earth. Also mounted to the truss near the habitation system are thermal radiators for the power conditioning equipment.

Crew systems - The crew systems consist of a long-duration transit habitat and one or more MEVs (the reference design shows one MEV). All habitable volumes are contiguous throughout each mission. Electric propulsion has the least sensitivity to increased payload mass, so an important option is provision for multiple MEVs. A multiple docking adapter (not shown), would allow several MEVs to be used without altering the vehicle configuration (additional propellant tanks would be required).

Trades and Rationale

- Extremely high Isp
- Non-nuclear option
- Lowest IMLEO.
- High efficiency of solar electric propulsion

Mission Modes And Operations

- Vehicle assembled in SSF orbit.
- Vehicle spirals out to GEO using transfer array.
- Crew transfers to SEP via LTV.
- Vehicle executes lunar swingby prior to TMI.
- Vehicle executes Mars flyby during 30 day surface mission.
- MEV/Aerobrake separate from SEP for entry and landing.
- Aerobrake jettisoned prior to landing.
- Crew cab ascent after surface mission, leaving lander and surface hab.
- Crew cab left in Mars orbit after rendevous, docking and crew transfer.
- Crew depart vehicle via STV during earth flyby.
- Vehicle spirals in from HEO to GEO.

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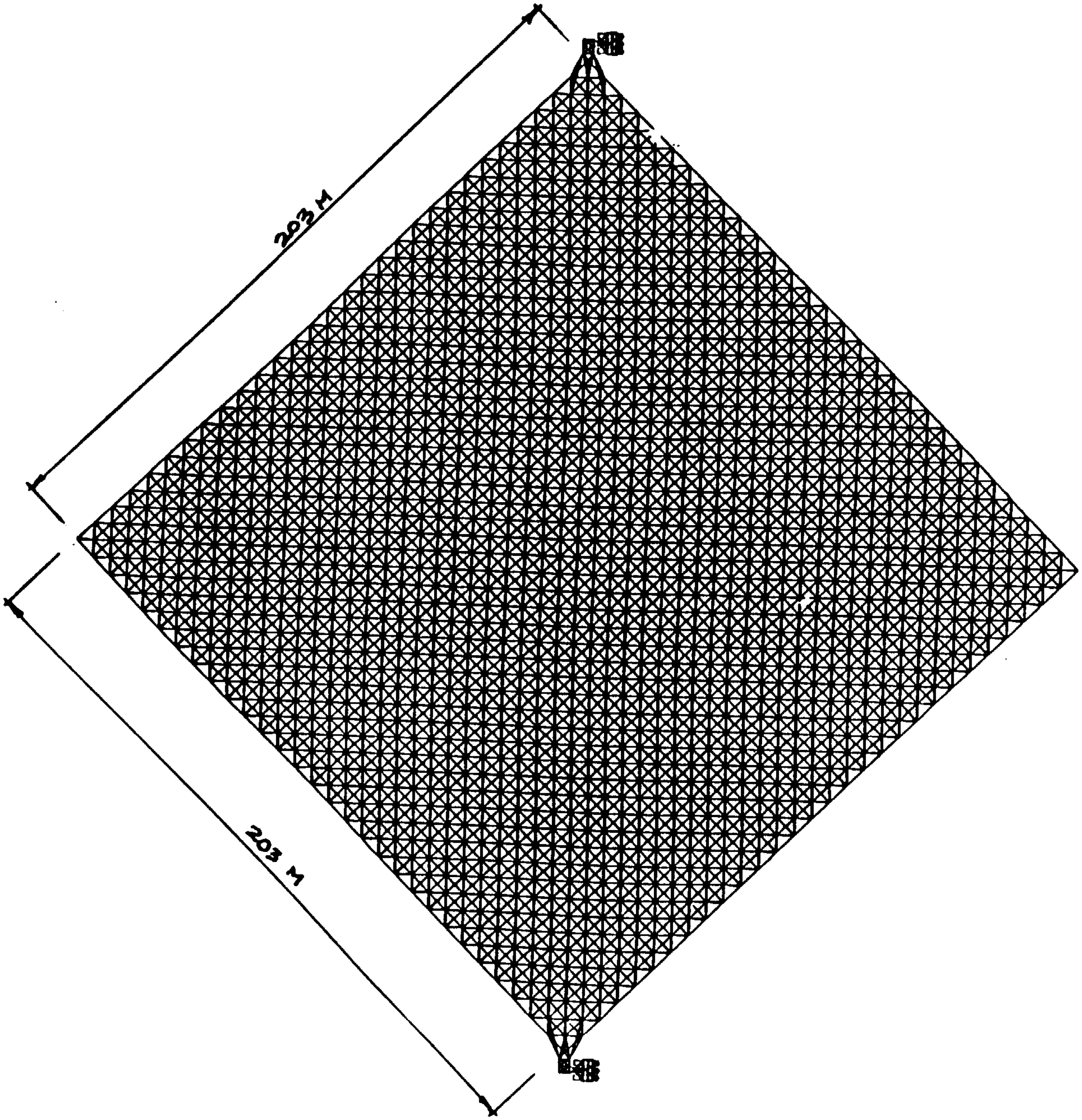
SEP Configuration



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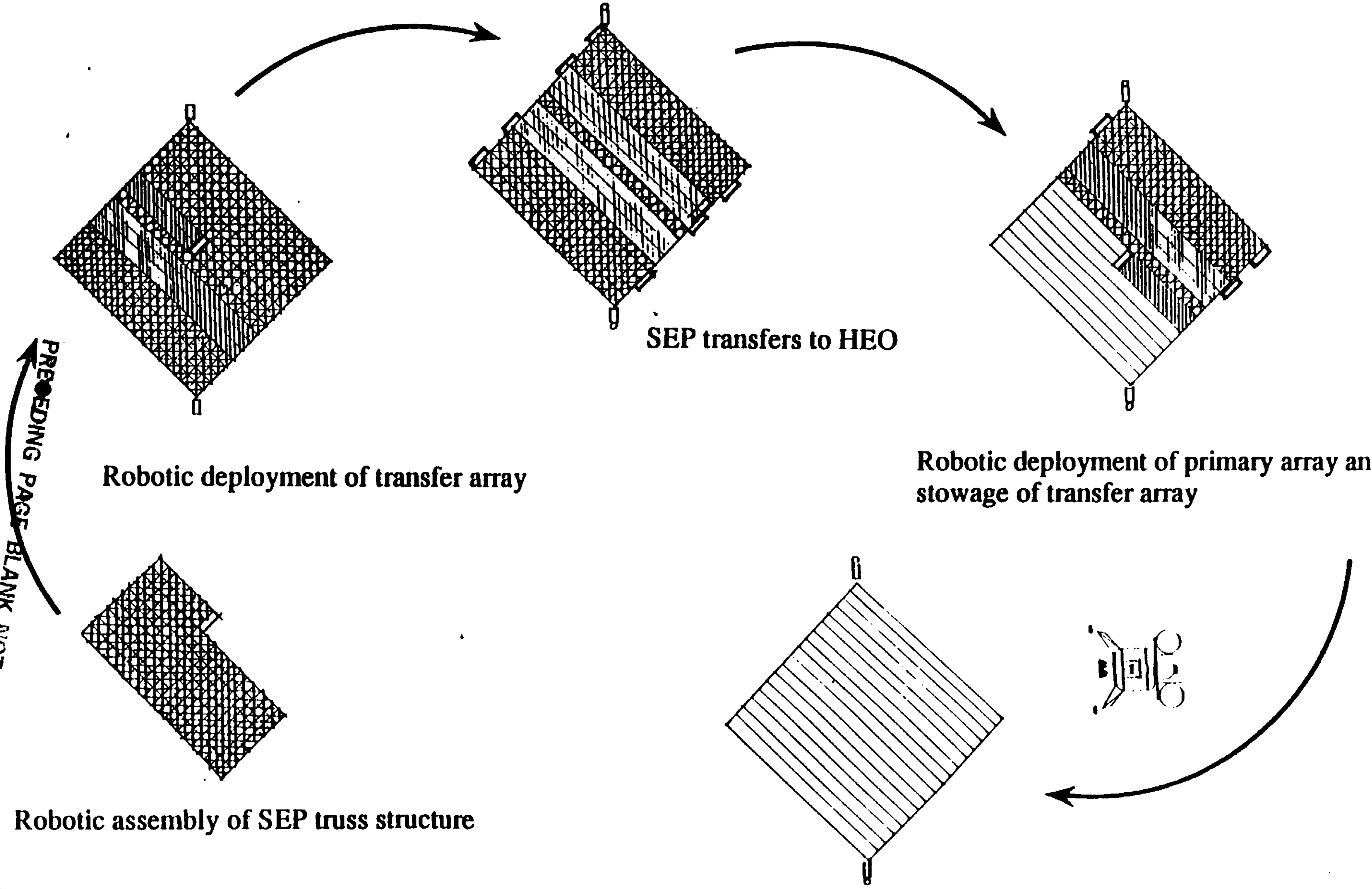
SEP Mission Sequence



Robotic assembly of SEP truss structure

Robotic deployment of transfer array

SEP transfers to HEO

Robotic deployment of primary array and stowage of transfer array

SEP *en route* to Mars

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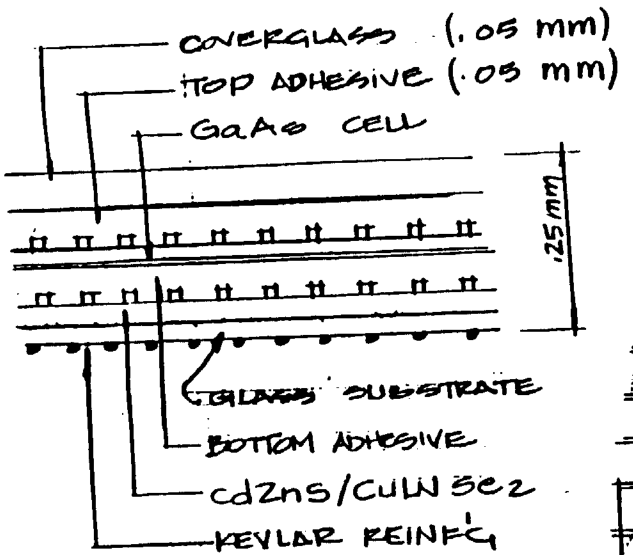
SEP P.V. Array Details



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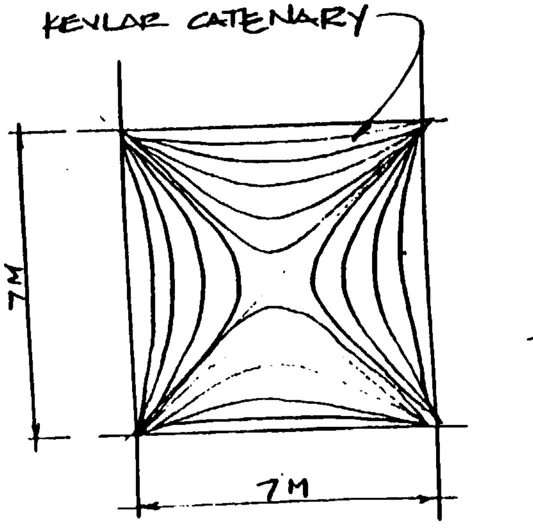
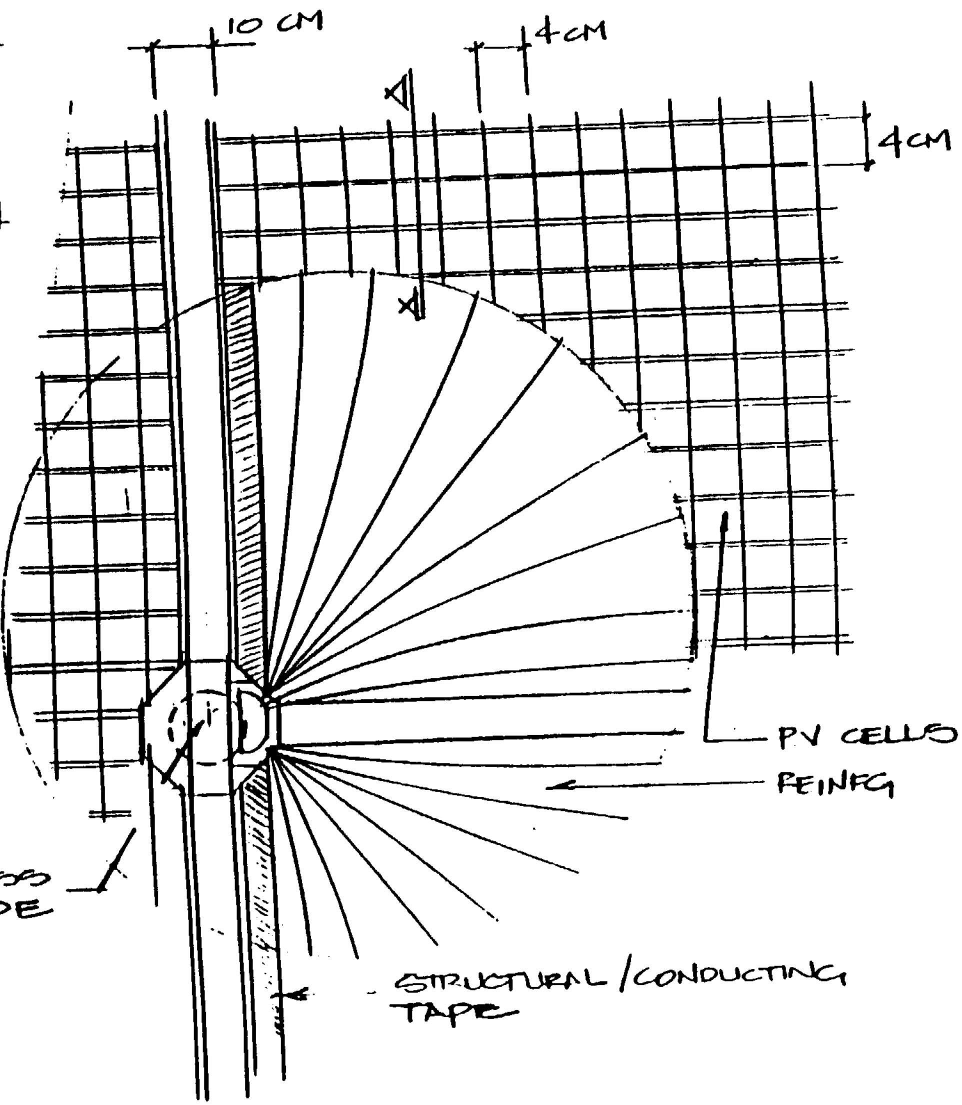


ARRAY REINFORCING



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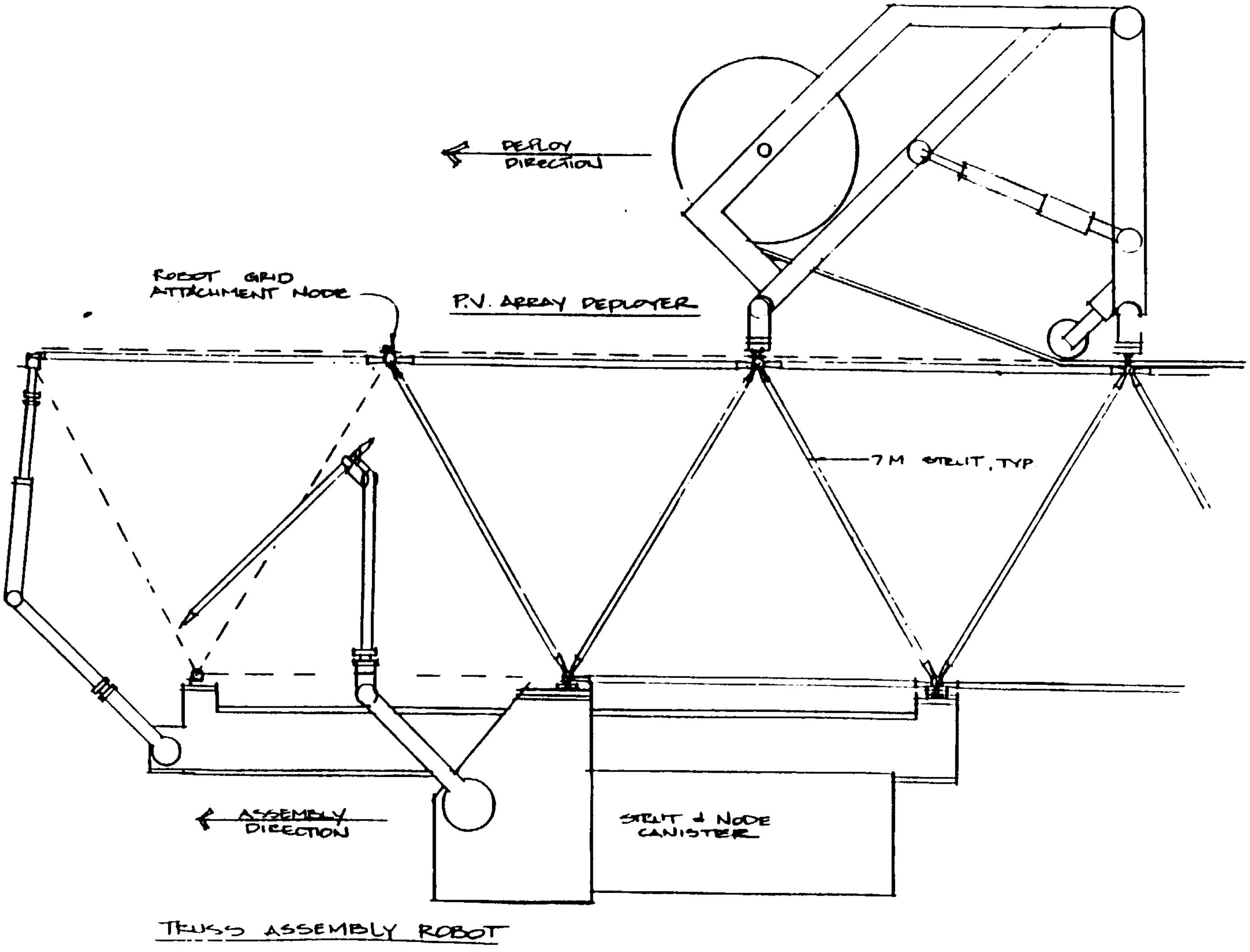
Robotic SEP Assembly



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Micro-Gravity SEP Mass Statement

Payload

Mass in metric tonnes

Descent Aerobrake	7.0
MEV Descent Stage	18.7
MEV Ascent Stage	22.5
Surface Equipment	25.0
Transit Hab Module	44.3
	117.5 t

Propulsion

Thruster Assembly	8.0
Power Management & Distribution	20.0
	28.0 t

Solar Array Blanket

Photovoltaic Cell	840mg/cell
Reinforced SiO2	211mg/cell
adhesive (fiber/cell)	60mg/cell
Kevlar Support Structure	68mg/cell

* 23,464,336 cells
=27.7 t

Structure

(6728) Graphite epoxy Struts	7.7
(1741) Nodes	2.4
.051mm Aluminum Cladding	1.2
	11.3 t

Utilities

Communications	.6
Attitude Control	5.7
Avionics	2.5
Houskeeping Power Distribution	.5
PV/RFC Power Subsystem	2.0
Robotics	3.6
	14.9 t

Tanks	3.8
Feed Lines	0.1
Propellant	190.0

Total	**393.3 t**
10% growth (structure & array)	3.9 t
15% growth (propulsion & misc.)	6.4 t
IMLEO	**403.6 t**
Resupply	**335.2 t**

Trip Time = 550 days, alpha = 9.2

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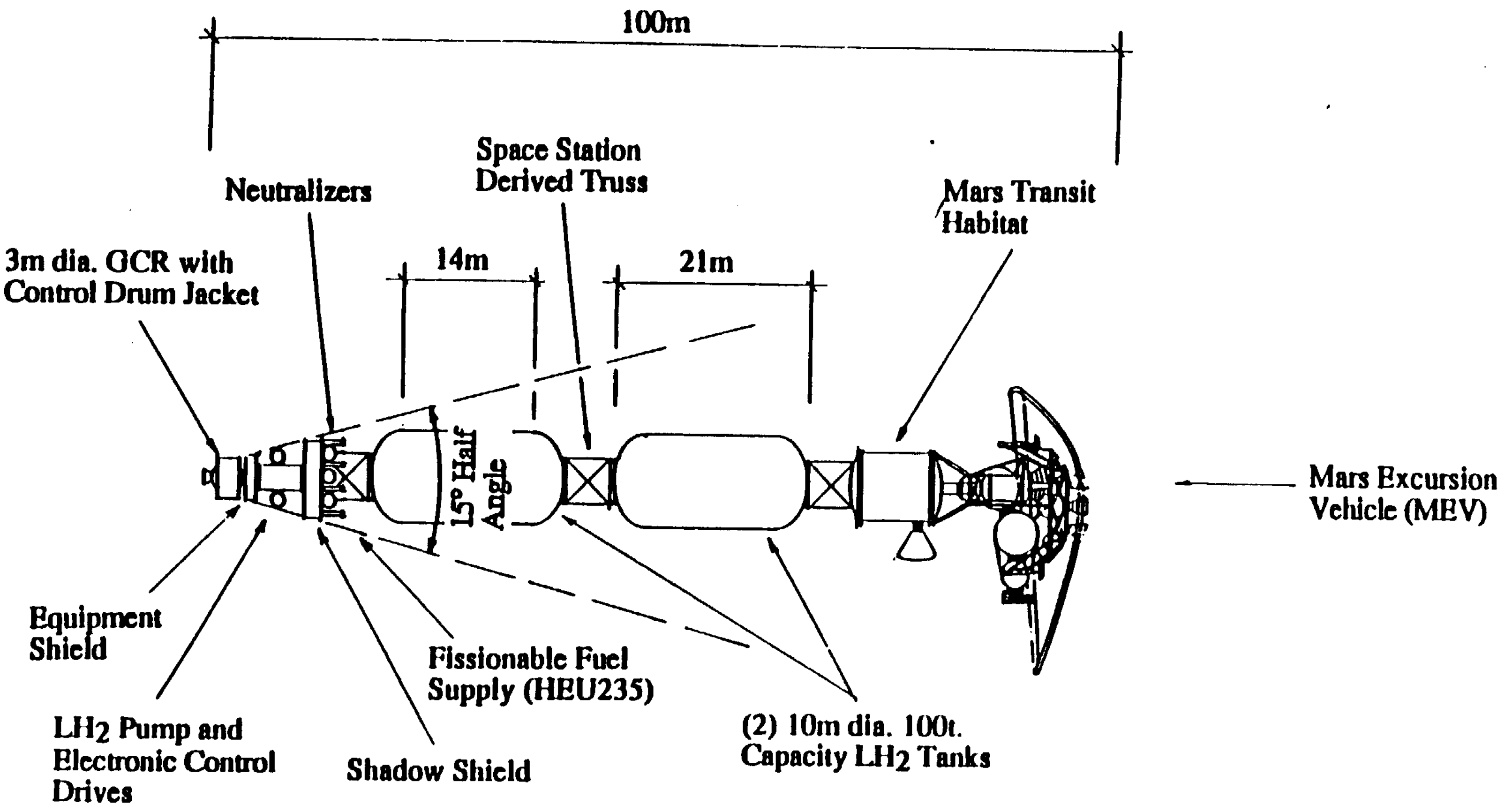
Gas Core Reactor (GCR)

The Gas Core Reactor is, at present, a theoretical model with many developmental questions to be answered before a working system can be developed . While it holds great promise for high performance and short trip times, it is still the furthest from development. In one estimate from NASA Lewis using an optimistic development schedule, the first test flight a GCR vehicle would occur in the 2016 time frame. This would not meet the national goals for a manned Mars mission in the first quarter of the next century. However as an evolutionary concept vehicle for follow on or continuing human presence architectures it should not be dismissed out of hand, but requires better identification of a working system.

GCR

GCR is an evolutionary system whose DDT&E will put man rated first use beyond the projected 2016 mars mission opportunity. However, GCR is a candidate for future mission architectures.

GCR Configuration



IMLEO 120 day mission	1030 t
IMLEO 180 day mission	600 t

- Technology will not be available during 2016 mission opportunity.
- Candidate for future mission architectures.

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Vehicle Comparisons

Shown on the next chart are the operating envelopes of trip time versus initial mass in low Earth orbit (IMLEO) opposition mission opportunities from 2010 through 2025 for all the vehicle types No Venus swingbys are used that might modify the outline of some of these envelops, particularly the Cryo/Aerobrake and NTR. These include "hard" years, when the position of the planets (relative angle between planet positions is large and opposition occurs away from Mars perihelion) makes intercept costly in ΔV and "easy " years when the positions are favorable. This is only a set of operations needs envelops, it does not consider the restrictions of vehicle development time or readiness to meet these envelope dates, the reusability of the vehicle or augmentation by ISRU (lunar or Mars sources).

The comparison between the various types of NTR (NERVA or Advanced) and the Cryo/Aerobrake is continued in more detail through the next several charts some of which includes the use of swingbys to amend the data.

All the vehicles are comprised of propulsion elements that have been identified for four mission vehicle configuration mixes (involving Moon, Mars zero-g, and Mars artificial gravity configurations) and an estimate made of the development effort to bring the individual elements up to man-rated flight ready status. The numbers obtained are a preliminary estimate of this effort needed to bring into operation a set of working elements for a lunar- Mars network. It is subject to change as the goals and objectives of the program become more clear and the architecture framework is better defined.

It becomes critical to understand the outstanding technology issues that each vehicle configuration presents in order to evaluate the development, costs and scheduling impacts that may be inherent in the vehicle design. The top issues and their constrains are listed in this section as well as the operational issues that must be solved in each of these configurations before readiness is achieved.

The summary of this section presents a comparison of the advanced propulsion system masses in Earth orbit for the initial mission as vehicle resupply (rebuild/ refurbish) and the payload resupply mass for the reference 2015-2016 mission time frame. The advantages and disadvantages of each configuration are outlined as a quick reference to the trades and issues that must be evaluated and defined. For every opportunity and every mission scenario at least one vehicle design that will fulfil the objectives can be identified.

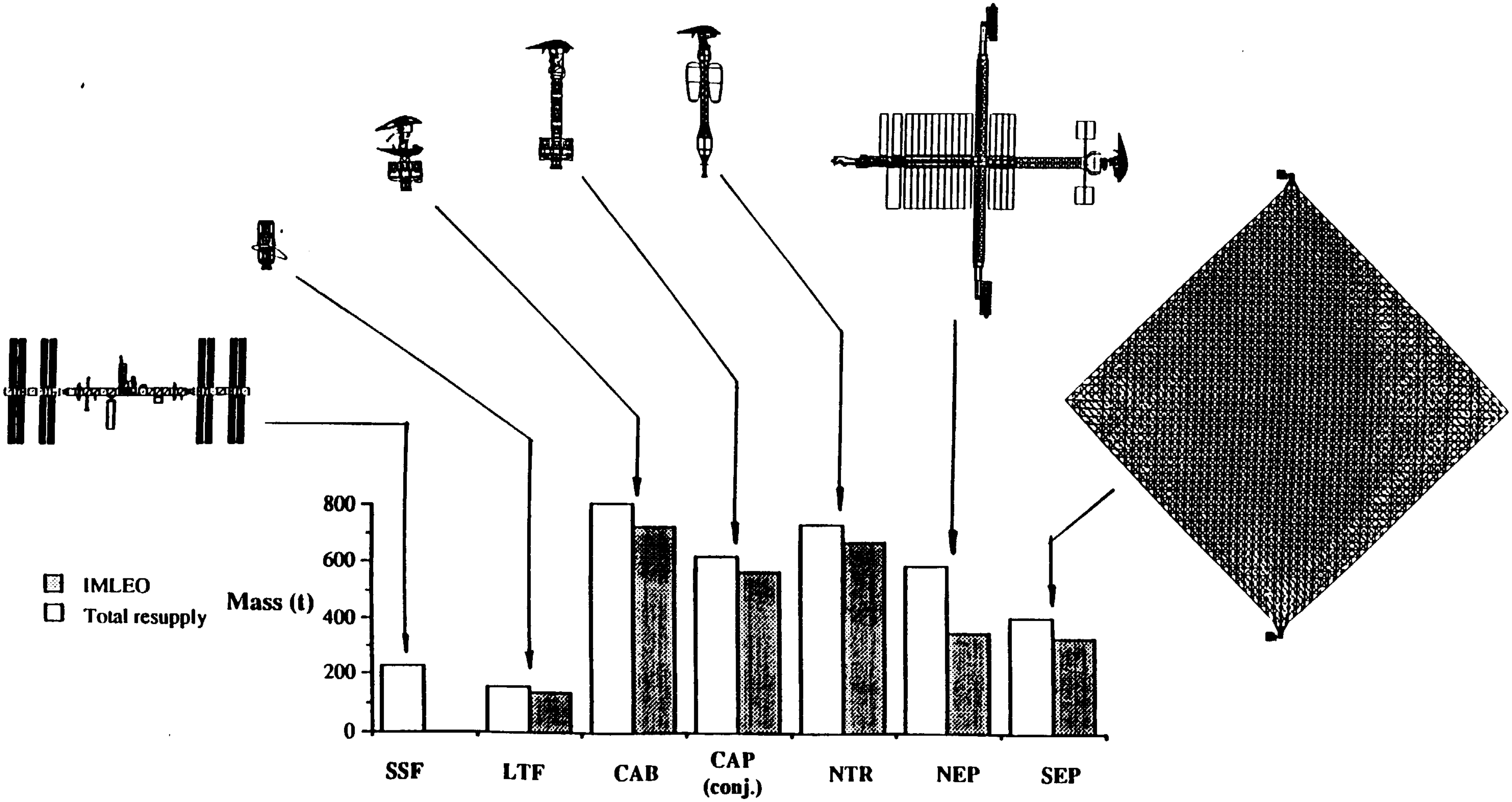
Part of the Work Breakdown Structure has been included to show the other advanced propulsion options that exist, but were not used after the first neckdown of possibilities for near term development and inclusion in the trades.

STCAEM Concept Size Comparison

Shown below are the reference transportation concept configurations to scale, compared with respect to size and mass. Shown for scale is the current *Space Station Freedom* configuration.

The vehicles are compared to SSF to show that in most cases, they are physically larger and more massive, making it difficult to assemble them on station.

STCAEM Concept Size Comparison



/STCAEM/crf/ 9Jan91

Mars Propulsion Options Compared

This is an updated version of the propulsion options matrix presented in October. The matrix compares the five prime candidates, from the standpoint of the key architecture drivers of flexibility and multi-use design(tasks 2 and 6), as well as from the top level standpoint of tasks 4 (integration), 5 (support), 7 (technology), and 8(programmatics) in the STCAEM study.

As previously stated, there is no clear winner among the options. A "preferred" option will depend entirely on which criteria are most valued.

Mars Propulsion Options Compared

		NTR	NEP	SEP	CAP	CAB
Flexibility (robustness, resiliency, evolution; preservation of options)	Launch window size	×	●	●	×	×
	Insensitivity to opportunity	○	●	●	×	○
	Insensitivity to variable payload mass	○	●	●	×	×
	Insensitivity to parking orbit	×	●	●	×	×
	Intrinsic system redundancy	×	○	●	×	×
	Capability for trans-Mars missions	○	●	×	×	×
Multi-use design (commonality, re-use; establishment of infrastructure)	High-power surface system commonality	×	●	×	×	×
	Lunar crew transportation commonality	●	×	×	●	●
	Lunar cargo transp. commonality	●	●	○	○	●
	Lunar return/MEV aerobrake commonality	×	●	●	×	×
	MTV re-usability	○	●	●	×	○
	In-space infrastructure buildup	○	●	●	×	×
Integration	Trip time	●	○	×	○	○
	Ease of MTV launch packaging	●	○	×	●	●
	Simplicity of in-space dry-dock operations	○	×	×	●	●
	Safety for EVA crews	×	×	●	○	○
Programmatics	Capability for early Mars	○	×	×	●	○
	Low acquisition cost	○	×	×	●	○
	Technology readiness	○	×	○	●	○
	Commercial technology potential	○	○	●	○	○

Key: × **comparatively poor** ○ **moderate** ● **good**

/STCAEM/crf/ 9Jan91

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Propulsion Option Comparison Assumptions

Chemical/AB

- Expendable - ECCV return
- Isp = 475 sec
- AB weight = 15 % for comparison, 'bands' on IMLEO vs trip time range from 15% to 30%

NTR-NERVA

- Expendable - ECCV return
- Isp = 925 sec, Tc=2700 K, Composite , Pc = 1000, nozzle AR = 500:1
- Engine T/W = 3.5 (NERVA: 9684 kg eng wt for 75,000 lbf thrust)
- 4.5 ton radiation shadow shield (also uses residual propellant as shield)
- Tank fraction = 14%

NTR-Advanced

- Expendable - ECCV return
- Isp = 1050 sec, Tc=3100 K, Carbide, Pc = 1000 psia, nozzle AR = 500:1
- Engine T/W = 20:1 (PBR: 1701 kg eng wt for 75,000 lbf thrust)
- 4.5 ton radiation shadow shield (also uses residual propellant as shield)
- Tank fraction = 14%

NEP

- Reusable
- Varied Power from 10 MW to 120 MW
- Alpha's varied from 8 kg/kW to 3 kg/kW respectively
- Isp ~10,000 sec
- Lunar and Mars flyby employed
- Crew rendezvous via LTV prior to Earth Escape

SEP

- Reusable
- Varied Power from 7 MW to 18 MW
- Vehicle Alpha = 8.5 kg/kW
- Isp ~5,500 sec
- Lunar and Mars flyby employed
- Crew rendezvous via LTV prior to Earth Escape

STCAEM/brc/16Aug90/disk7

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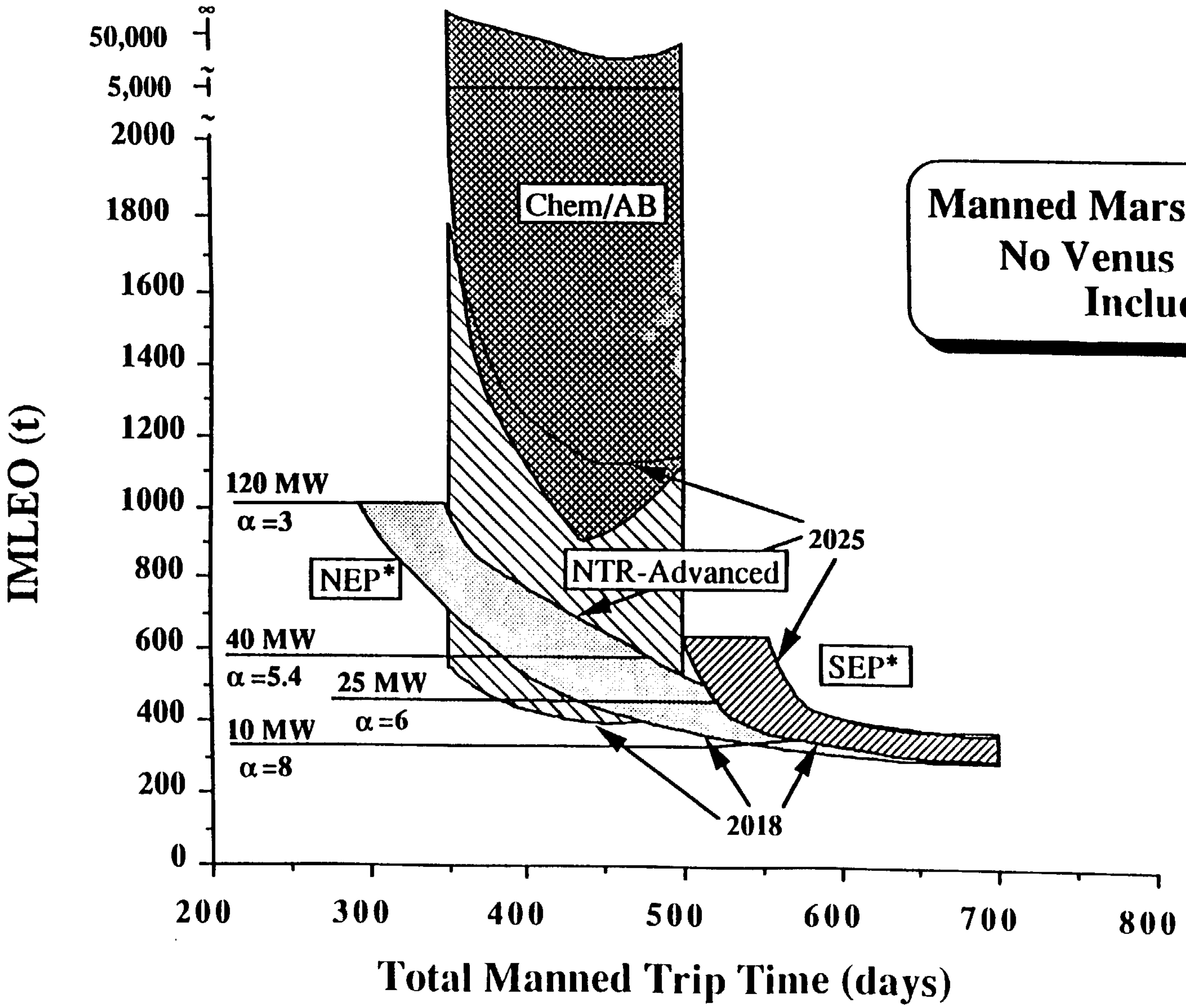
Propulsion Option Comparison for Opposition Missions

Comparison with a 120 t Payload Opposition Opportunities



Manned Mars 2010-2025
No Venus Swingbys
Included

* Reusable Vehicles

/STCAEM/brc/05Sept90

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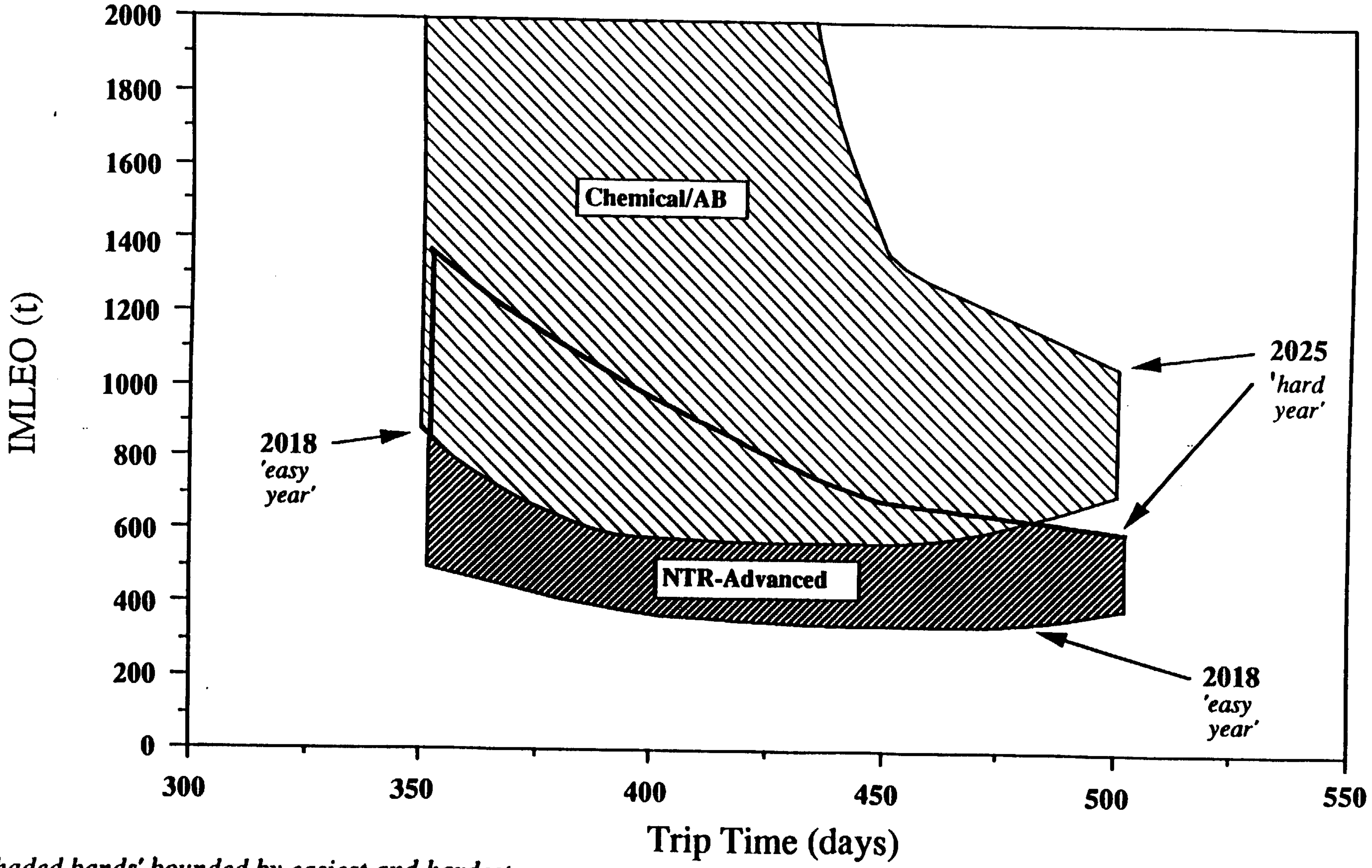
Chem/AB vs NTR MarsVehicle IMLEO Comparison

for Non-Swingby Opposition Missions



'Shaded bands' bounded by easiest and hardest years covers (contains) all non-swingby opposition mission opportunities

Mac Chart/Disk #8/Chem/AB vs NTR IMLEO vs trip time/11-28-90

Chem/AB & NTR Vehicle IMLEO Comparision Data

2018 & 2025 Non-Venus Swingby Opposition Missions

	year	trip time days	Cryo/Aerobrake *Isp=475, 15% A/B*	NERVA NTR *Isp=925, eng T/W=3.5*	Advanced NTR *Isp=1050, eng T/W=20*	Reusable *Advanced NTR*
'easy year'	2018	350	882 t	667 t	507 t	670 t
	2018	400	598 t	469 t	380 t	501 t
	2018	450	589 t	416 t	342 t	423 t
	2018	500	719 t	489 t	394 t	501 t
'hard year'	2025	350	∞ t	2,155 t	1,353 t	1,637 t
	2025	400	3,804 t	1,399 t	963 t	1,080 t
	2025	450	1,357 t	921 t	684 t	727 t
	2025	500	1,091 t	776 t	590 t	613 t

Mac Chart/disk #8/adv propul tabular data/11-28-90

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Mission Delta V's and Departure Dates-

2018 & 2025 Non-Swingby Opposition Missions

dep & arr dates | Vhp limit at Mars capture=7 km/s, ECCV Earth entry Vhp limit=9.7 km/s
Δ = diff between arr Vhp & required arr Vhp limit

	total trip time	Earth dep	Mars arr	Mars dep	Earth arr	(1) TEI dV	Outb Deep Space burn	MOC Vhp	(2) Δ	MOC dV	TEI dV	EOC Vhp	(3) Δ	EOC dV
2018 'easy year'	350	*8280	385	415	630	4.101	0	6.857	0	3.772	5.460	9.524	0	4.027
	400	8270	400	430	670	3.741	0	5.260	0	2.528	3.763	10.217	0.517	4.506
	450	8230	400	430	680	3.610	0	4.521	0	1.857	3.646	11.000	1.300	3.747
	500	8140	375	405	640	4.489	0	4.590	0	2.098	3.603	9.523	0	4.046
2025 'hard year'	350	**0390	595	625	740	10.228	0	7.020	0.020	3.984	6.223	8.050	0	3.119
	400	0365	579	609	765	7.805	1.613	7.156	0.156	4.018	4.193	4.744	0	1.428
	450	0300	540	570	750	5.187	2.761	7.229	0.229	4.079	2.030	6.020	0	2.000
	500	0275	391	421	775	4.374	2.925	6.761	0	3.693	1.946	3.749	0	1.541

(1) g-losses not accounted for (2) all aerocapture veh's arriving Mars with Vhp>7 (km/s) use cryo chemical propulsion to slow veh down to Vhp=7 (km/s) for aerocapture
(3) ECCV's arriving Earth with Vhp>9.7 (km/s) use cryo chemical propulsion to slow ECCV capsule down to Vhp=9.7 for entry

* 245xxxx -, ** 246xxxx Julian dates

all missions 30 day Mars orbit stay time

disk #8/dVs 2018-2025 non swby

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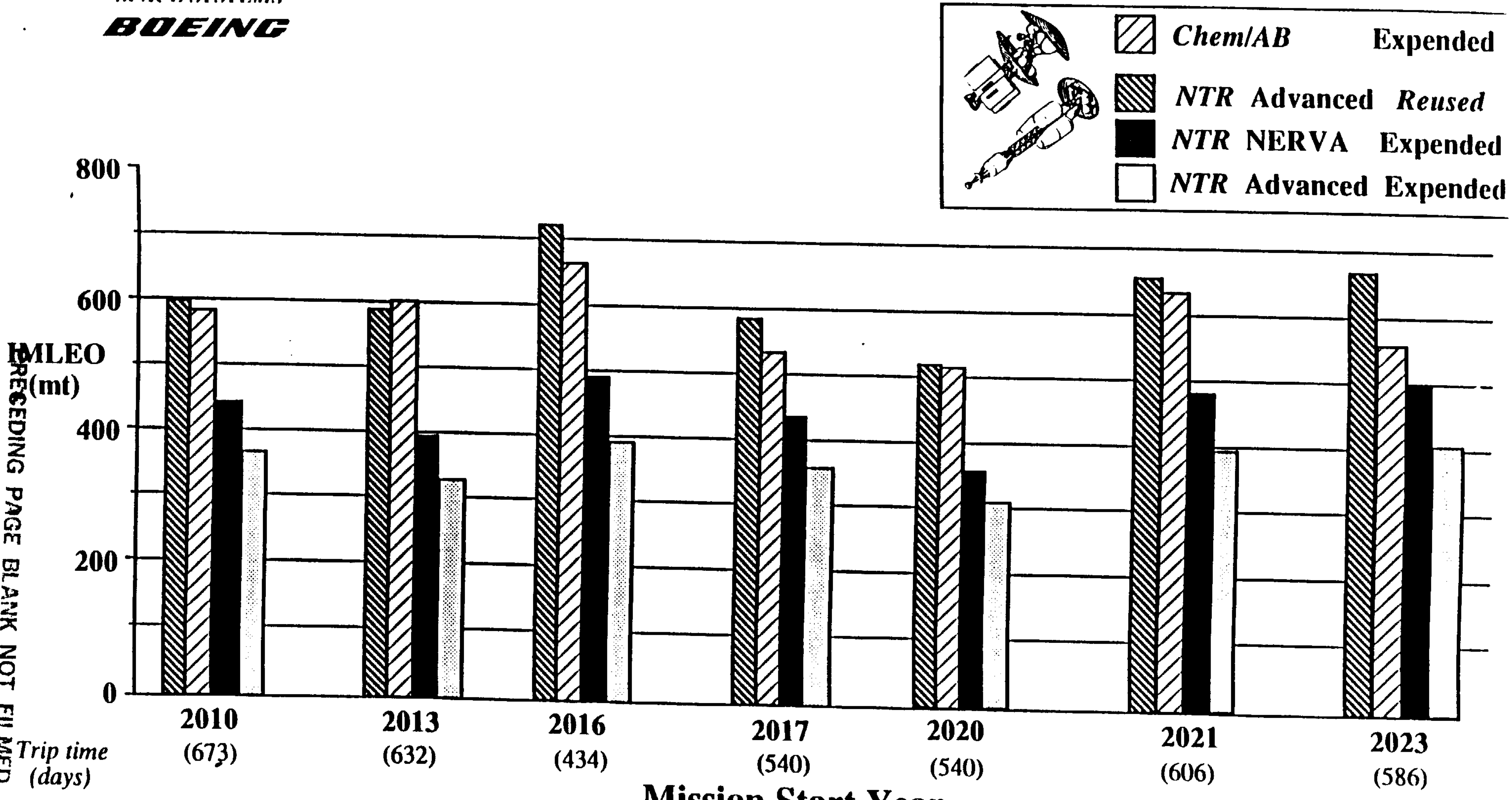
ADVANCED CIVIL SPACE SYSTEMS
BOEING

Chem/AB & NTR Vehicle IMLEO vs. Opportunity Year

Venus Swingby Opposition Missions



Engine & Vehicle Characteristics

- *Chem/AB: AB 15% of capt wt, eng Isp =475*
- *Adv NTR eng T/W= 20, eng wt = 1701 kg, Isp=1050 s*
- *NERVA eng T/W= 3.5, eng wt = 9684 kg, Isp= 925 s*
- *NTR engine Thrust = 75,000 lbf, Rad shield wt = 4.5 t*
- *NTR postburn cooldown prop = 3%, reserves = 2%*
- *NTR:tank fraction=14%, frame truss & t struts=2880 kg*
- *Crew of 4, 34 t MTV, 73 t MEV includes 25mt surf p/l*
- *reuse case: Propul capt at Earth: 500 km by 24 hr orbit*

All surface stay times are 30 days

synthesis model run #: marsntrmtv.dat;345-353,369-376; marschemmtv.dat;143-151 Mac chart: Disk #5/IMLEO/yr bar chart 11/9/90

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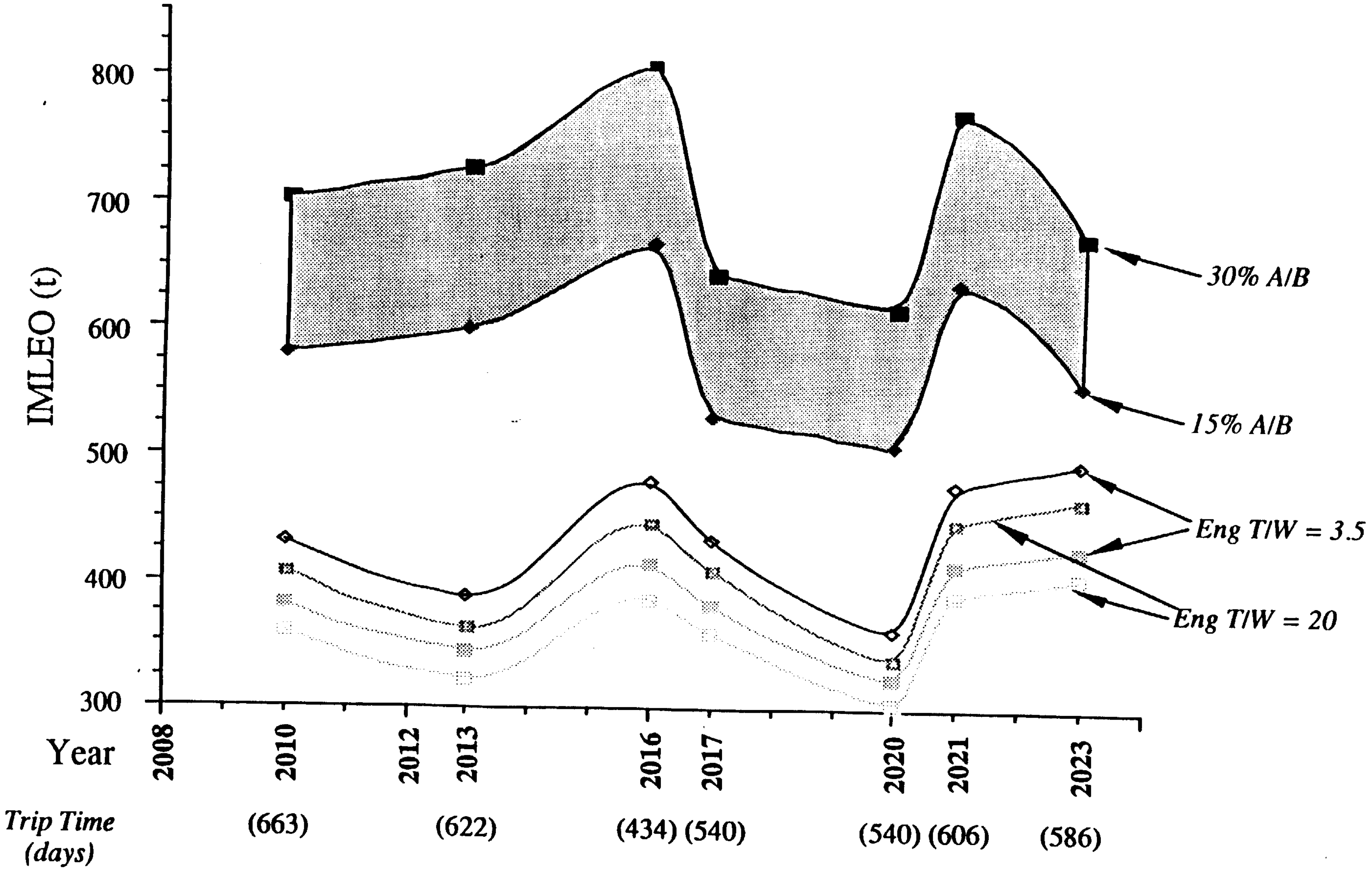
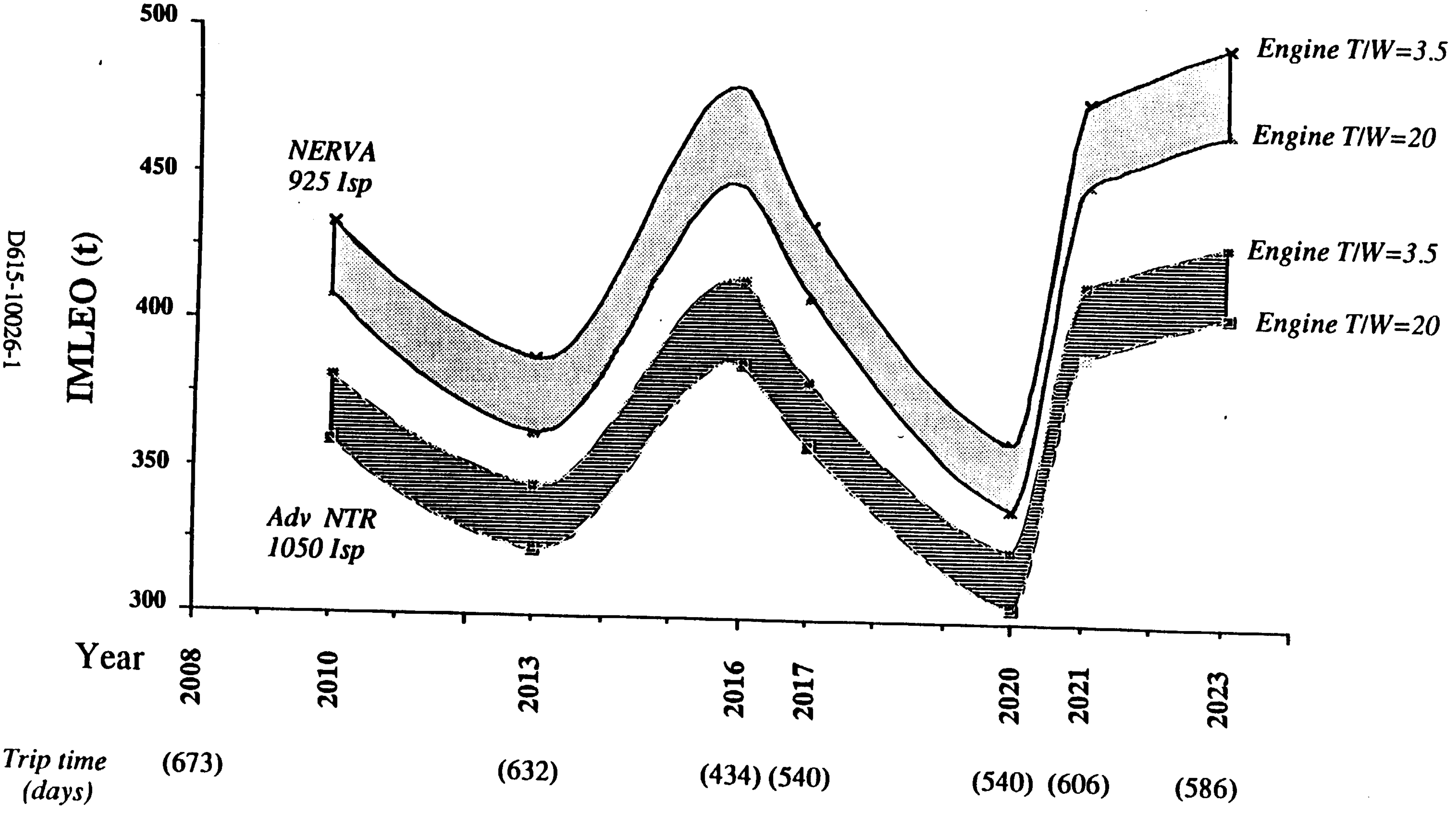
NTR vs Chem/AB vehicle IMLEO Comparison Opposition Missions With Swingbys



ADVANCED CIVIL
SPACE SYSTEMS

Opposition Missions with Swingbys
Adv NTR and NERVA NTR Vehicles

BOEING

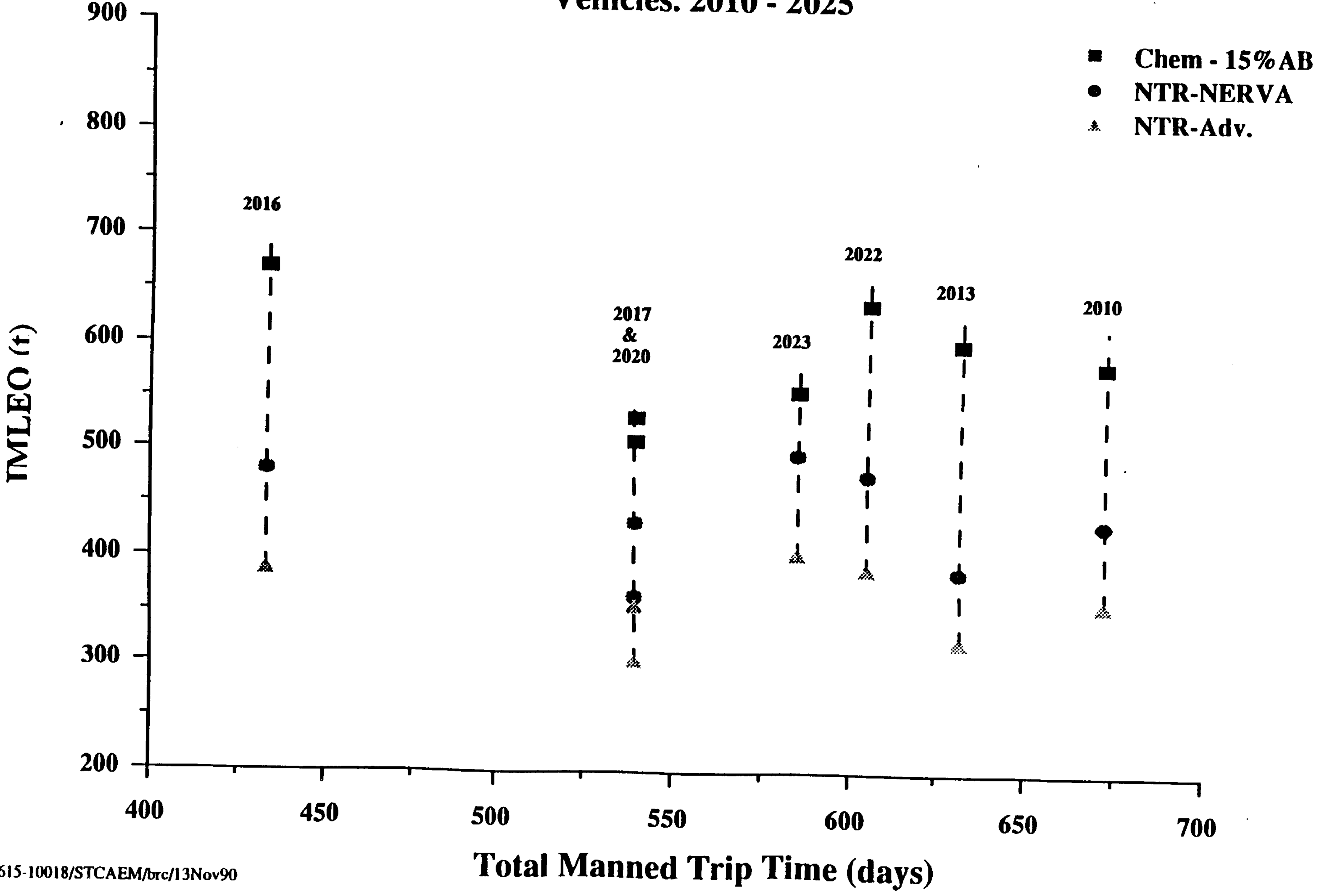


Propulsion Option swingby Opportunities for Opposition Missions

Optimum Mars trajectories with Venus swingbys are presented in the form of IMLEO vs trip time. The swingby points are given for three high trust vehicle options: Chem/AB, NERVA NTR, and advanced NTR. The swingby points are represented here as discrete points, not as a "band" . The two preceding IMLEO charts presented the swingby points as being part of a band, which is in a sense misleading since there is no continuity between two swingbys or a swingby and non swingby reference trajectory. The majority of swingby opportunities occur in the 530 - 675 day regime. Although most swingby opportunities have a longer trip time than the fast trip non-swingby opportunities, they offer a reasonably low IMLEO for all mission opportunities. An important point to note is that some years contain an outbound swingby opportunity, while some years contain an inbound swingby. the situation imposes less that an 18 month departure time between consecutive opportunities. This restrictive time frame could interfere with time restraints on the launch of HHLVs and assembly for the next mission.

Propulsion Option Swingby Opportunities

Swingby Opportunities for Expendable High Thrust Vehicles. 2010 - 2025



D615-10018/STCAEM/brc/13Nov90

Chem/AB & NTR Vehicle IMLEO Comparision Data

Opposition Missions *with Venus swingby*

Year	Mass (t) Chem/AB		NTR - NERVA		NTR - Advanced		NTR -Adv Reusable
	15 % AB	30 % AB	T/W=20	T/W=3.5	T/W=20	T/W=3.5	T/W=20
2010	581	704	409	433	360	381	597
2013	601	728	363	388	323	345	588
2015							
2016	669	811	448	481	387	415	720
2017	532	644	410	434	360	381	580
2020	510	619	339	361	305	324	518
2022	638	773	450	478	391	415	659
2023	558	676	468	497	405	429	669

STCAEM/brc/21Aug90/disk7

Mac Disk #8/Adv propul tabular data 11/8/90

High Thrust Trajectory Assumptions for Propulsion Option Comparison

1) Maximum Earth, Mars arrivial Vhp = 7 km/s. When Mars aerocapture Vhp exceeds 7 km/s, a cryogenic propulsive burn using propellant stored in the TEI stage (Isp=475 s) is done to slow down vehicle to Vhp=7 km/s before Mars aerocapture. For ECCV entries at Earth where Vhp exceeds 9.7 km/s, a separate cryogenic propulsive stage (on ECCV) slows capsule down to Vhp= 9.7 before entry.

2) Mars Parking Orbit - 250 km periapsis altitude by 1 sol period

3) Correction for DLA Loss at Earth is performed at maximum apoapsis altitude during a 3-burn departure maneuver.

4) g-loss not accounted for

5) Corrections for DLA and apsidal misalignment at Mars will be performed at the optimal true anomaly, inclination, and period; note that an inclination will be chosen that allows for daylight landing.

5) No analysis will be performed at this time to evaluate if deep space burn maneuvers are the optimal correction for Mars departure DLA losses and Mars departure apsidal misalignment.

6) For 2025, deep space burn will be analyzed as a mode of minimizing IMLEO

7) Non Venus-Swingby cases will be analyzed for 350, 400, 450, and 500 day round trip times for 2018 & 2025 missions.

8) Midcourse corrections for legs w/o swingby = 50 m/s; with swingby = 100 m/s.

9) Aerocapture, expendable - optimize TMI and TEI Delta V's.

10) All propulsive, expendable - optimize TMI, MOC, and TEI delta V's.

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Mac Disk #8/traj assumptions

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Mission Delta V's and Departure Dates
Venus Swingby Cases

Δ = diff between arr Vhp & required arr Vhp limit

departure & arr dates | *Vhp limit at Mars capture=7 km/s, ECCV Earth entry Vhp limit=9.7 km/s*

Year	Total trip time	Earth dep	Mars arr	Mars dep	Earth arr	(1) TEI dV	Outb Deep Space burn	MOC Vhp	(2) Δ	MOC dV	TEI dV	EOC Vhp	(3) Δ	EOC dV
2010	673	*5529	5859	5889	6192	4.426	0	4.927	0	2.318	1.310	7.550	0	2.831
2013	632	6618	6899	6929	7240	3.692	0	3.374	0	1.280	3.235	4.406	0	1.134
2016	434	7463	7621	7651	6897	3.805	0	5.308	0	2.562	3.979	5.562	0	1.799
2017	540	7850	8201	8231	8390	4.249	0	5.480	0	2.703	1.115	3.834	0	1.110
2020	540	9055	9216	9246	9595	3.867	0	3.761	0	1.523	1.826	4.235	0	1.251
2021	606	9518	9820	9850	**0124	4.258	0	5.795	0	2.944	2.520	8.172	0	3.138
2023	586	**0194	0494	0524	0780	4.264	0	6.391	0	3.466	1.464	2.820	0	0.811
2022	986	9811	**0167	0507	0797	3.882	0	2.820	0	1.012	1.168	3.611	0	1.037
2020	860	9086	9196	9821	9946	4.792	0	5.656	0	2.823	2.996	6.164	0	2.087

Oppositions

Conjunctions

(1) *g-losses __not__ accounted for* (2) all aerocapture veh's arriving Mars with Vhp>7 (km/s) use cryo chemical propulsion to slow veh down to Vhp=7 before aerocapture
(3) ECCV's arriving Earth with Vhp>9.7 (km/s) use cryo chemical propulsion to slow ECCV capsule down to Vhp=9.7 for entry

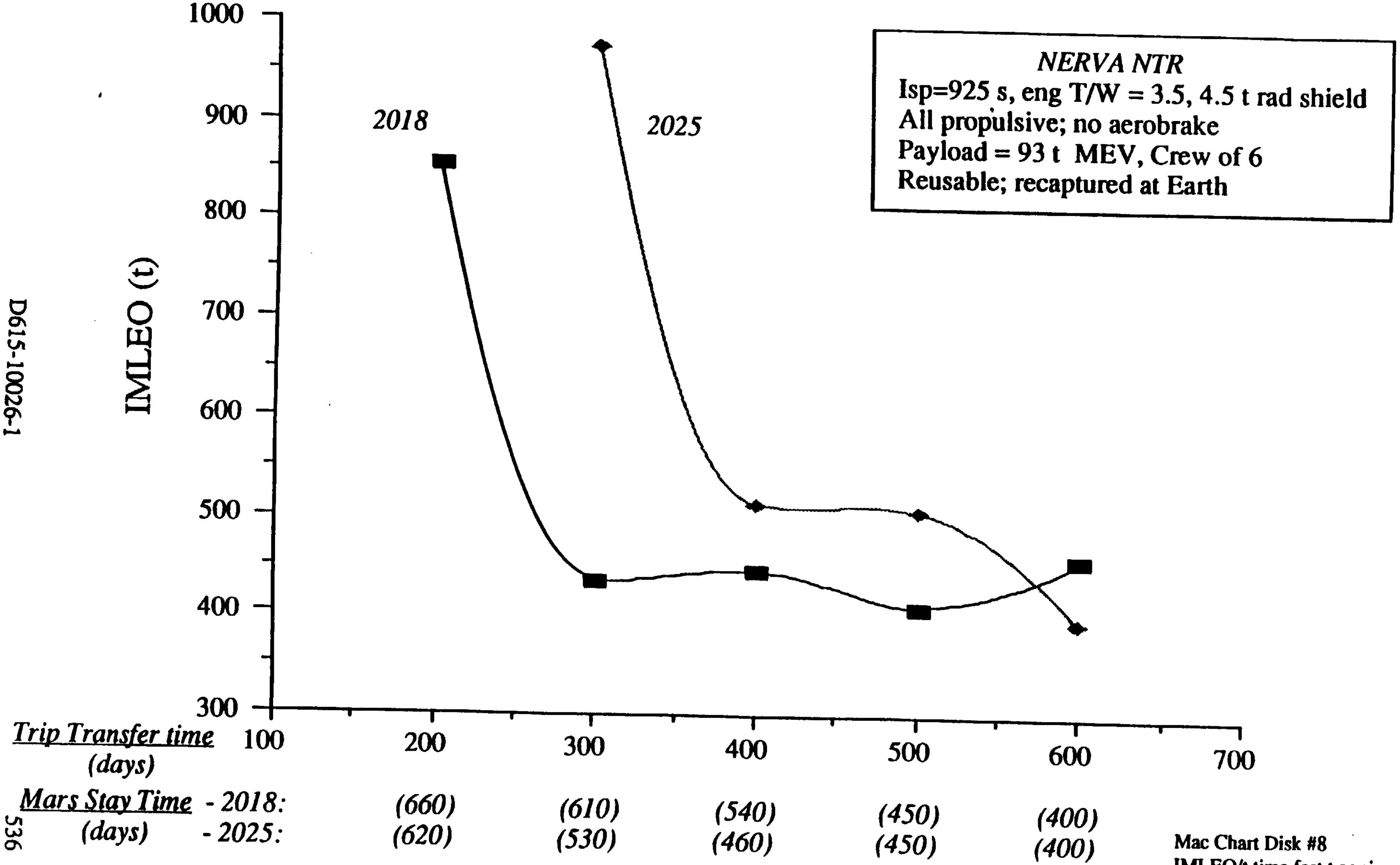
* *245xxxx -, ** 246xxxx Julian dates*

Stay time at Mars for all Oppositions missions = 30 days

disk #8/2010-2023 swby dVs

NERVA NTR vehicle IMLEO vs Transfer time

Fast Transfer Conjunction Missions



NERVA NTR
Isp=925 s, eng T/W = 3.5, 4.5 t rad shield
All propulsive; no aerobrake
Payload = 93 t MEV, Crew of 6
Reusable; recaptured at Earth

Mac Chart Disk #8
IMLEO/t time fast t conj
Synthesis model run # ;377-386

Mission Delta V's and Departure Dates-
Fast Transfer Conjunction Missions

For 2018 & 2025 Conj Missions

dep & arr dates

Vhp limit at Mars capture=7 km/s, ECCV Earth entry Vhp limit=9.7 km/s
Δ = diff between arr Vhp & required arr Vhp limit

	Mars Stay time	Transfer trip time	Earth dep	Mars arr	Mars dep	Earth arr	(1) TEI dV	Outb Deep Space burn	MOC Vhp	(2) Δ	MOC dV	TEI dV	EOC Vhp	(3) Δ	EOC dV
2018 'easy year'	660	200	*8300	8400	9060	9160	5.341	0	5.470	0	2.810	3.720	7.540	0	2.800
	610	300	8285	8435	9045	9195	4.132	0	3.480	0	1.390	1.960	3.590	0	1.030
	540	400	8270	8470	9010	9210	3.640	0	3.110	0	1.330	3.440	3.290	0	0.940
	450	500	8245	8465	8915	9195	3.908	0	3.020	0	1.170	2.240	3.140	0	0.900
	400	600	8270	8610	9010	9270	3.980	0	4.420	0	1.980	1.970	3.930	0	1.140
2025 'hard year'	620	200	**0645	0745	1405	1505	8.750	0	10.850	3.850	7.390	8.870	9.690	0.690	4.170
	530	300	0640	0796	1376	1521	5.730	0	5.410	0	2.630	5.300	6.370	0	2.190
	460	400	0621	0831	1331	1521	4.303	0	3.510	0	1.360	3.010	5.990	0	2.000
	450	500	0600	0860	1310	1550	4.783	0	2.900	0	1.110	3.250	3.960	0	1.150
	400	600	0575	0895	1295	1575	3.800	0	2.470	0	0.940	1.960	3.820	0	1.100

(1) *g-losses not accounted for* (2) all aerocapture veh's arriving Mars with Vhp>7 (km/s) use cryo chemical propulsion to slow veh down to Vhp=7 (km/s) for aerocapture
(3) ECCV's arriving Earth with Vhp>9.7 (km/s) use cryo chemical propulsion to slow ECCV capsule down to Vhp=9.7 for entry

** 245xxxx -, ** 246xxxx Julian dates*

disk #8/dVs 2018-2025 fast transf conj

Major Propulsion Element List for 2000-2030 HEI program

Primary Objective: Furnish a top level list of all major propulsion elements necessary to a 3 decade HEI total program entailing Lunar, Mars opposition (short stay) and Mars conjunction (long stay) missions.

Secondary objective: Considering four candidate vehicle combinations (differentiated by propulsion system choice, each of which might satisfy all the space transfer objectives of a comprehensive HEI program) roughly evaluate or 'score' the *total development effort required to bring each propulsion system element/technology up to flight readiness.* Having done so, sum all the element scores for each of the candidate vehicle combinations in order to ascertain which combination meets HEI program objectives with least overall propulsion systems development effort. The 4 candidates are listed below:

(1) Cryogenic Lunar with chemical Mars opposition (zero-g) & conjunction (art-g,tether system)
(2) NTR Lunar, NTR Mars opposition (zero-g) & Mars conj (art-g, vehicle rotation about its Cg, no tether)
(3) Cryogenic Lunar, NEP Mars opposition (zero-g) & NEP conj (art-g, tether system)
(4) Cryogenic Lunar, SEP Mars opposition (zero-g) & SEP conj (art-g, tether system)

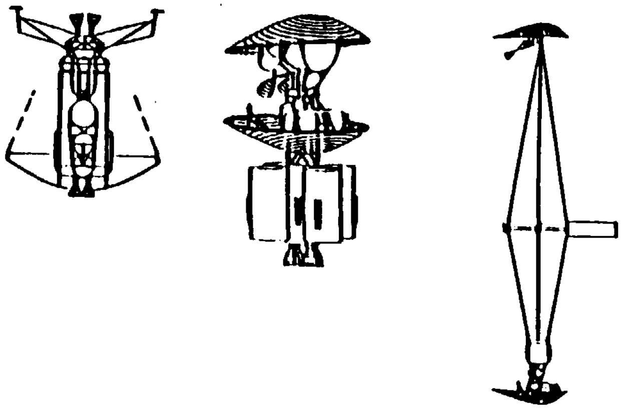
Scores: Primary list: the all NTR set had lowest total propulsion element count of 5, that is, 5 distinct elements were identified. The Cryogenic/SEP combination followed with 8 elements, all Cryogenic with 8, and Cryogenic/NEP with 11. Differences in opinion as to what constitutes 'major' 'and 'distinct' propulsion elements might lead to slight variations in the totals, all depending on who does the counting.

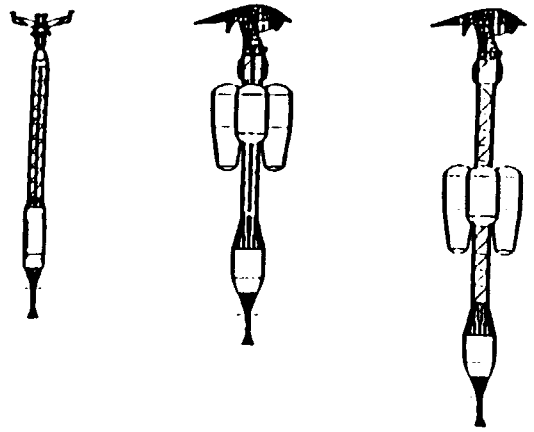
Scores: Secondary list: The all NTR set scored the lowest in total propulsion elements development effort with a score of 13. the chemical/SEP combination and the all chemical set were about even with scores of 18 & 19 followed by chem/NEP at 27. These scores are relative, and only show how the 4 vehicle sets compare to one another; They are also subjective, and the differences in overall scores may be more pronounced, less pronounced or even change in rank depending on who is doing the evaluating. these rankings are not presented herein as the results of a precise technical trade study, but rather the results of a rough comparison 'methodology' with its major emphasis on a top down viewpoint in contrast to an analysis which as its emphasis on optimizing and/or selecting propulsion systems solely for individual missions.

Major Propulsion Element List for Specific Vehicle Sets

to Satisfy Lunar & Mars Objectives of 2000-2030 HEI Program

Moon | *zero-g opposition* **Mars** *artificial-g conjunction*





Propulsion Element	*Development Effort Factor*
*Lunar*Chemical/*Mars*Chemical sys	
Lunar	
1 LTV propul stg	2
2 LTV aerocapt brake	3
3 LEV propul stg	2
Mars zero-g vehicle	
4 MEV propul stg	2
5 MEV/MTV aerocapture brake	3
6 MTV propul stage	2
7 TMI propul stage	3
Mars artificial-g vehicle	
8 Art-g tether system	2
8 distinct propulsion elements with development factor scores:	**19**
*Lunar*NTR/*Mars*NTR sys	
Lunar	
1 LEV propul stage	2
Common Lunar & Mars zero-g	
2 Common LTV/MTV NTR propul stage	6
3 Radiation handling/monitoring/shield	2
4 MEV propulsion stage	2
5 MEV descent heat shield	1
Mars artificial-g	
no necessary additions	
5 distinct propulsion elements with development factors scores:	**13**

***Legend*:** (1) least development effort; (6) most development effort
Expected total resources that must be expended for such a propulsion element to acheive flight readiness

STCAEM/bbb/11June90

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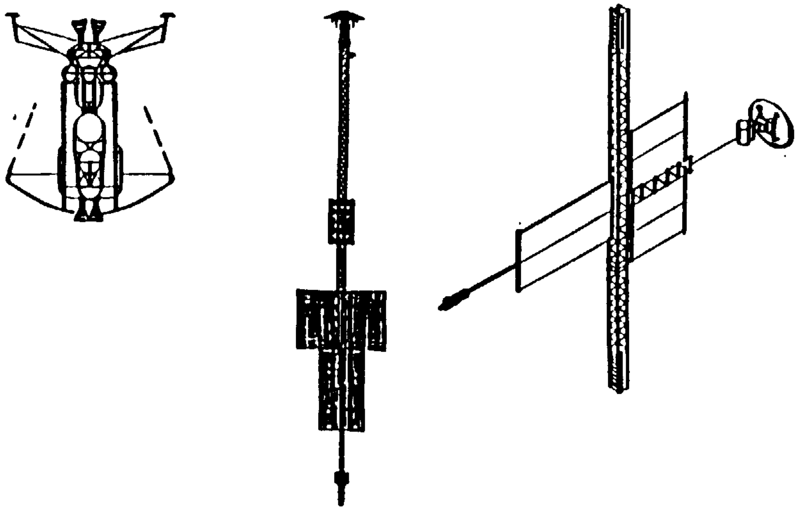
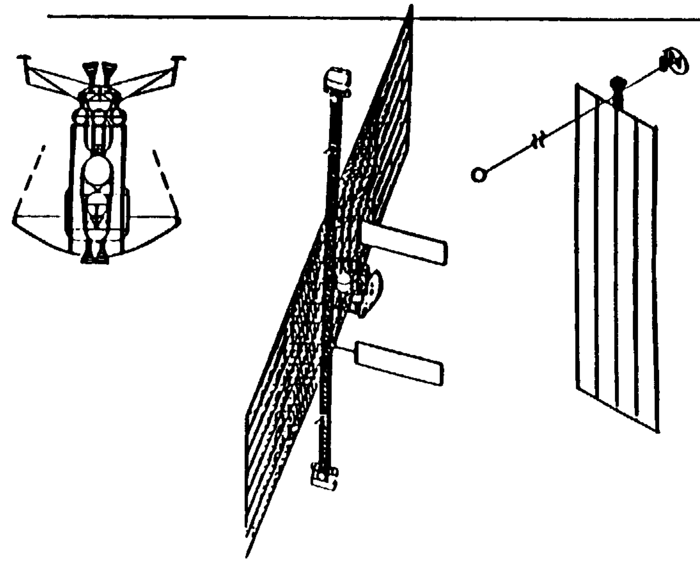
Major Propulsion Element List for Specific Vehicle Sets
to Satisfy Lunar & Mars Objectives of 2000-2030 HEI Program

ADVANCED CIVIL SPACE SYSTEMS **BOEING**

Lunar — *zero-g opposition* Mars *artificial-g conjunction*





STCAEM/bbb/11June90

Propulsion Element	Development Effort Factor
Lunar	*Lunar*Chemical *Mars*NEP
1 LTV propulsion stg	2
2 LTV aerocapture brake	3
3 LEV propulsion stg	2
Mars zero-g vehicle	
4 MEV propulsion stg	2
5 MEV descent heat shield	1
6 NEP reactor	6
7 Radiation handling/monitoring/shield	2
8 Dynamic conversion equip	2
9 Radiatiors	2
10 Electric thrustors	3
- Separate crew carrier to NEP 'spirial up' altitude	0 (use LTV)
Mars Artificial-g vehicle	
11 Artificial-g tether system	2
11 elements w sum of devel factors scoring:	**27**
Lunar	*Lunar*Chemical *Mars*SEP
1 LTV propulsion stg	2
2 LTV aerocapture brake	3
3 LEV propulsion stg	2
Mars zero-g vehicle	
4 MEV propulsion stg	2
5 MEV descent heat shield	1
6 SEP Solar array	3
7 Electric thrustors	3
- Separate crew carrier to SEP 'spirial up' altitude	0 (use LTV)
Mars Artifical-g vehicle	
8 Artificial-g tether system	2
8 elements w sum of devel factors scoring:	**18**

***Legend:**(1) least development effort; (6) most development effort*
Expected total resources that must be expended for such a propulsion element to achieve flight readiness

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Mars Transfer Propulsion Technology Issues

Preliminary

	(Constraint on)
High-Energy Aerobraking	
• Lack of understanding of Mars atmosphere dynamics: need precursor mission measurements.	Phase C/D start date
• Radiative aeroheating in Mars atmosphere, 6 to 9 km/sec	Phase C/D start date
• Aeroheating integrity of space-assembled aerobrake TPS	Crew safety
• Severity of Mars atmosphere dust erosion of TPS	Areobrake performance and reuse
Nuclear Thermal Rocket	
• Producibility of carbide fuel elements.	Performance
• Engine life vs. attainable Isp	Performance
• Full-containment ground test facility; effluent capture and cleanup	Rocket reactor testing start date
Electric Propulsion	
• Efficient, long life lightweight power processing	Performance
• High power density, efficient thrusters	Performance
Nuclear Electric Power for Electric Propulsion	
• Long life, reliable, high power, lightweight power generation	Crew safety & performance
• On-orbit assembly of liquid metal components	Feasibility
Solar Electric Power for Electric Propulsion	
• Assembly and control of multi-acre lightweight space structures	Feasibility
• Multi-megawatt high performance arrays, with production cost factor of 5 less than current space arrays	Affordability

Mars Transfer Propulsion Operational Issues

High-Energy Aerobraking

	(Constraints on)
• Complexity of artificial-g implementation	Affordability
• Pre-descent rendezvous in Mars orbit	Mission success
• Short launch windows	Mission success
• Mainly expendable systems, cost and complexity of in-space assembly and mission readiness verification for every mission	Affordability

Nuclear Thermal Rocket

• Nuclear safety in flight operations; EOL disposal	Affordability, practicality
• Turnaround for next mission after elliptic Earth capture	Operations cost

Electric Propulsion

• Navigation complexity for low-thrust trajectories, especially transition from interplanetary to orbital flight.	Operations cost
• Maintenance of complex, high power space electric power systems	Operational life; cost
• STV crew transport to and from EP vehicle to avoid Earth spiral time	Operational cost
• Complexity of artificial-g implementation	Affordability

Nuclear Electric Power for Electric Propulsion

• Nuclear safety in flight operations; EOL disposal	Affordability, practicality
• Reactor and power generator maintenance	Operational life; cost

Solar Electric Power for Electric Propulsion

• Control complexity for large space structure, especially in gravity gradients	Operations cost

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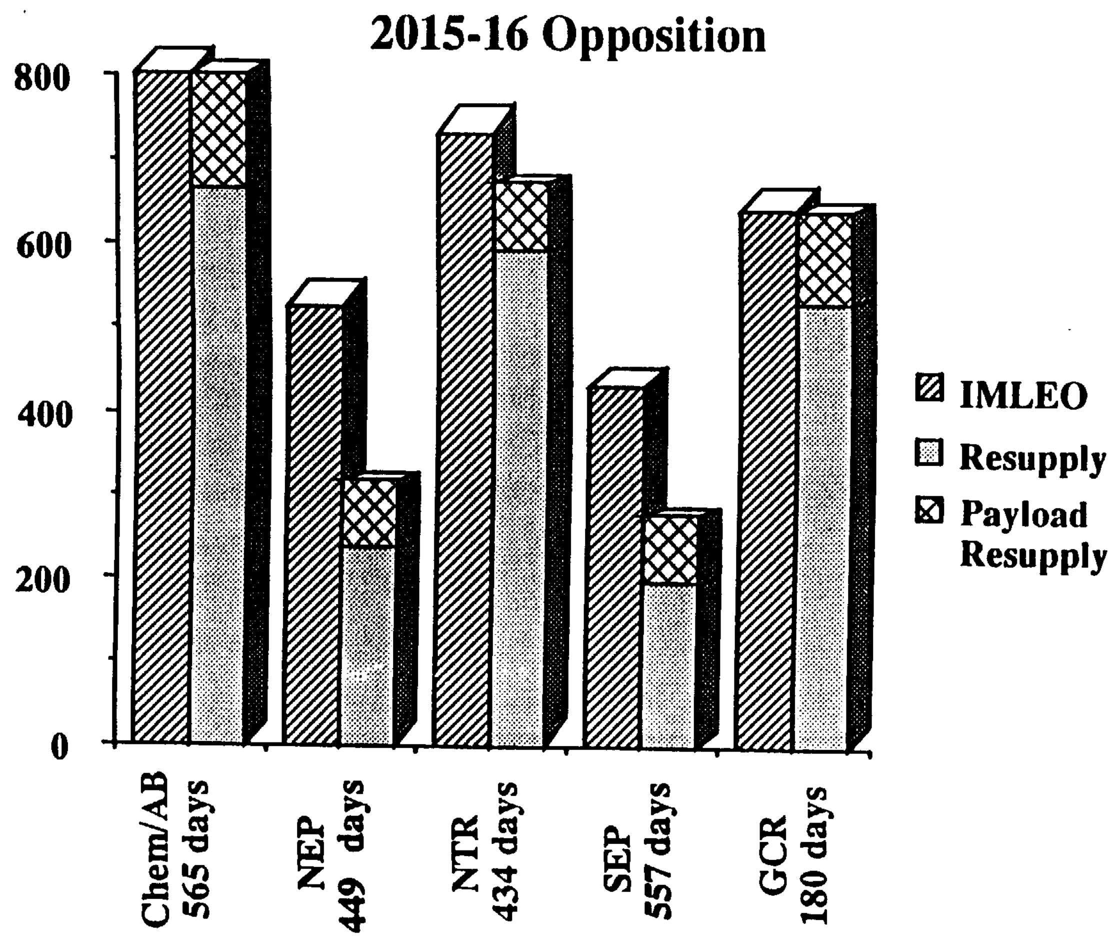
Advanced Propulsion Summary



Assumptions

1. The weights shown for Chem/AB & GCR are for expendable vehicles.
2. Isp's (sec) for options are: Chem-475, NEP-10,000 NTR-925, SEP-5,500 GCR-5,000.
3. NEP departs from LEO, uses Lunar and Mars swingbys.
4. SEP departs from LEO with transfer array, uses Lunar & Mars swingbys.
5. NTR uses Venus swingby.

Key Propulsion Performance drivers:

1. Isp
2. Thrust-to-Weight

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Advantages & Disadvantages

Propulsion Options

	Advantages	Disadvantages
Cryo/AB	Lower development cost Adequate redundancy Good reusability potential if operated from L2 node A large low-energy aerobrake is required for Mars landing with any propulsion option.	High IMEO Sensitive to variations in mission profile requirements Orbital assembly of large aerobrake, with rigorous verification requirements Needs accurate terminal navigation at Mars for successful aerocapture
SEP	Fully reusable Lower IMEO after first trip Trip times competitive with Cryo/Aerobrake Eliminates development and risk of large high energy aerobrakes Less sensitive to launch dates, windows Potential development synergy with existing Pathfinder, and CSTI programs Power supply at destination May offer low development cost	Operated from high Earth orbit or L2 for competitive trip time Susceptible to radiation damage in van Allen belts High power levels (10 MW) required for reasonable trip times Large area required for solar array (1 football field per 2 MW) Array production cost may be too high. Variable power over trajectory
NEP	Fully reusable Lower IMEO after first trip Faster trip times at high power, <200 days each way Power supply at destination Eliminates development and risk of large Aerobrakes Less sensitive to launch dates, windows Power source independent of solar distance Potential development synergy with existing SP-100, Pathfinder, and CSTI programs	Operated from High Earth Orbit for competitive trip time Limited redundancy and long operating times Dynamic power conversion required Nuclear Power requires further technology development High power levels required for reasonable trip times (10 MW) Nuclear systems in ETO launch

Advantages & Disadvantages

Propulsion Options (cont.)

	Advantages	Disadvantages
NTR	Lower IMEO Trip times about 400-450 days possible all opportunities Broader launch windows than chemical systems Technology demonstrated through ground testing phase Eliminates development and risk of large high energy aerobrakes Reusable and restartable Simpler trajectory design	Nuclear systems in ETO launch High cost development
GCR	Lower IMEO Very quick trip times (< 100 days each way) possible with masses comparable to chemical/aerobrake opposition class missions Broader launch windows than chemical systems Reusable and restartable Eliminates development and risk of large high energy aerobrakes Simpler trajectory design	Nuclear systems in ETO launch Technical feasibility in doubt: requires significant basic research and development Open cycle system introduces radioactive effluent concerns

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ADVANCED
CIVIL
SPACE
SYSTEMS

Advanced Propulsion WBS

BOEING

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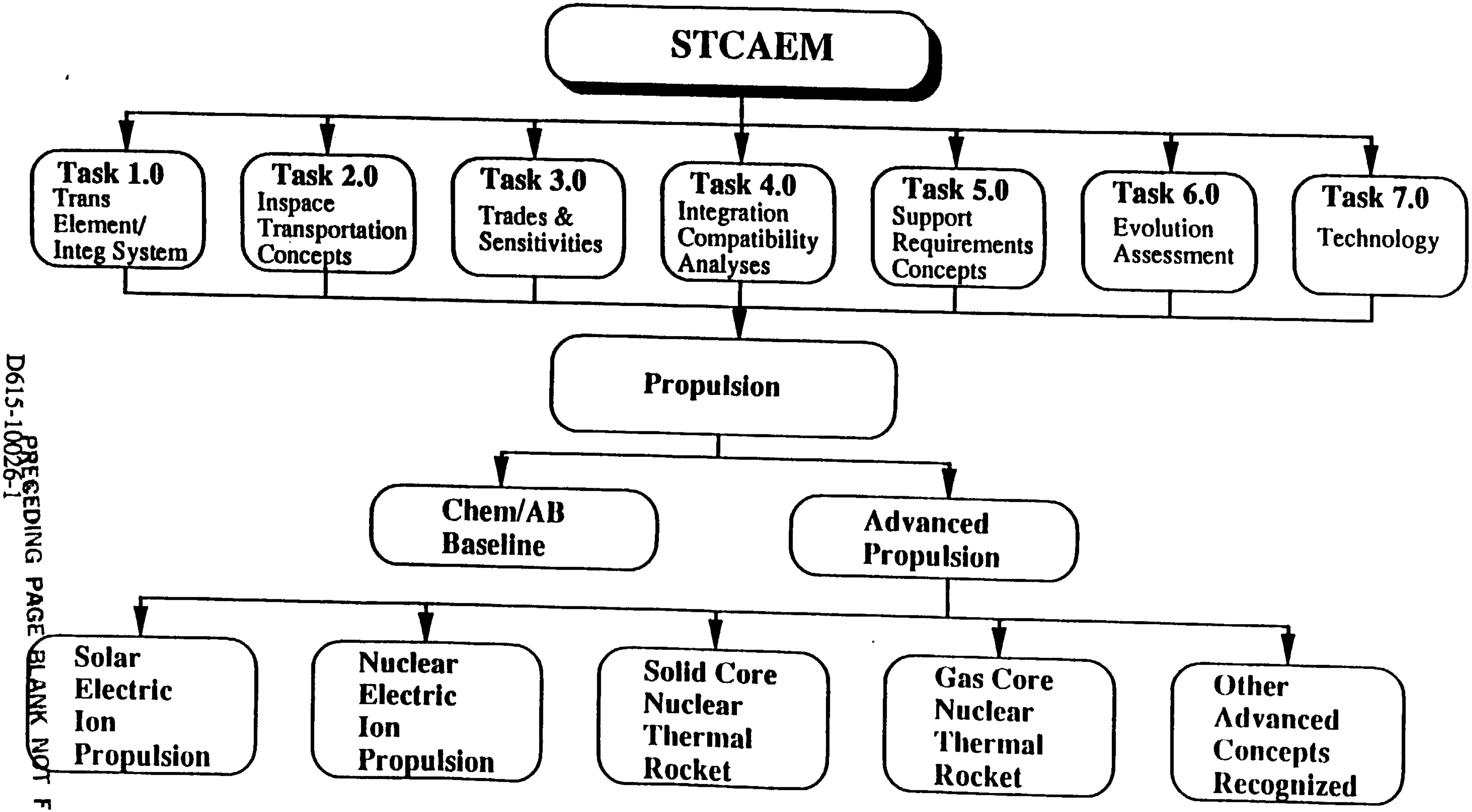

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Task 1.0 Trans Element/ Integ System

Task 2.0 Inspace Transportation Concepts

Task 3.0 Trades & Sensitivities

Task 4.0 Integration Compatibility Analyses

Task 5.0 Support Requirements Concepts

Task 6.0 Evolution Assessment

Task 7.0 Technology

Propulsion

Chem/AB Baseline

Advanced Propulsion

Solar Electric Ion Propulsion

Nuclear Electric Ion Propulsion

Solid Core Nuclear Thermal Rocket

Gas Core Nuclear Thermal Rocket

Other Advanced Concepts Recognized

Advanced Propulsion WBS (cont.)

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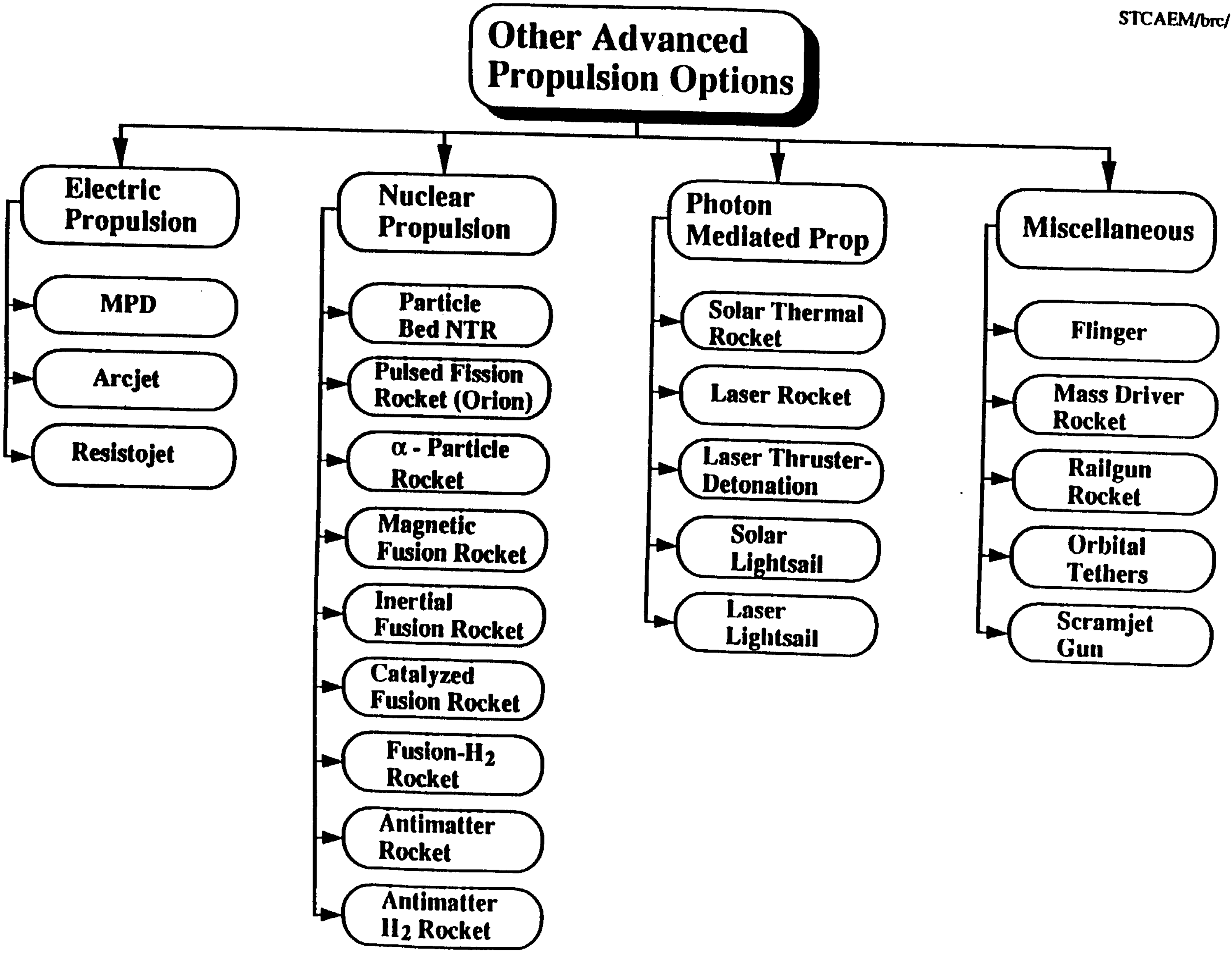

Other Advanced Propulsion Options

- **Electric Propulsion**
  - MPD
  - Arcjet
  - Resistojet
- **Nuclear Propulsion**
  - Particle Bed NTR
  - Pulsed Fission Rocket (Orion)
  - α - Particle Rocket
  - Magnetic Fusion Rocket
  - Inertial Fusion Rocket
  - Catalyzed Fusion Rocket
  - Fusion-H_2 Rocket
  - Antimatter Rocket
  - Antimatter H_2 Rocket
- **Photon Mediated Prop**
  - Solar Thermal Rocket
  - Laser Rocket
  - Laser Thruster-Detonation
  - Solar Lightsail
  - Laser Lightsail
- **Miscellaneous**
  - Flinger
  - Mass Driver Rocket
  - Railgun Rocket
  - Orbital Tethers
  - Scramjet Gun

Advanced Propulsion WBS (cont.)

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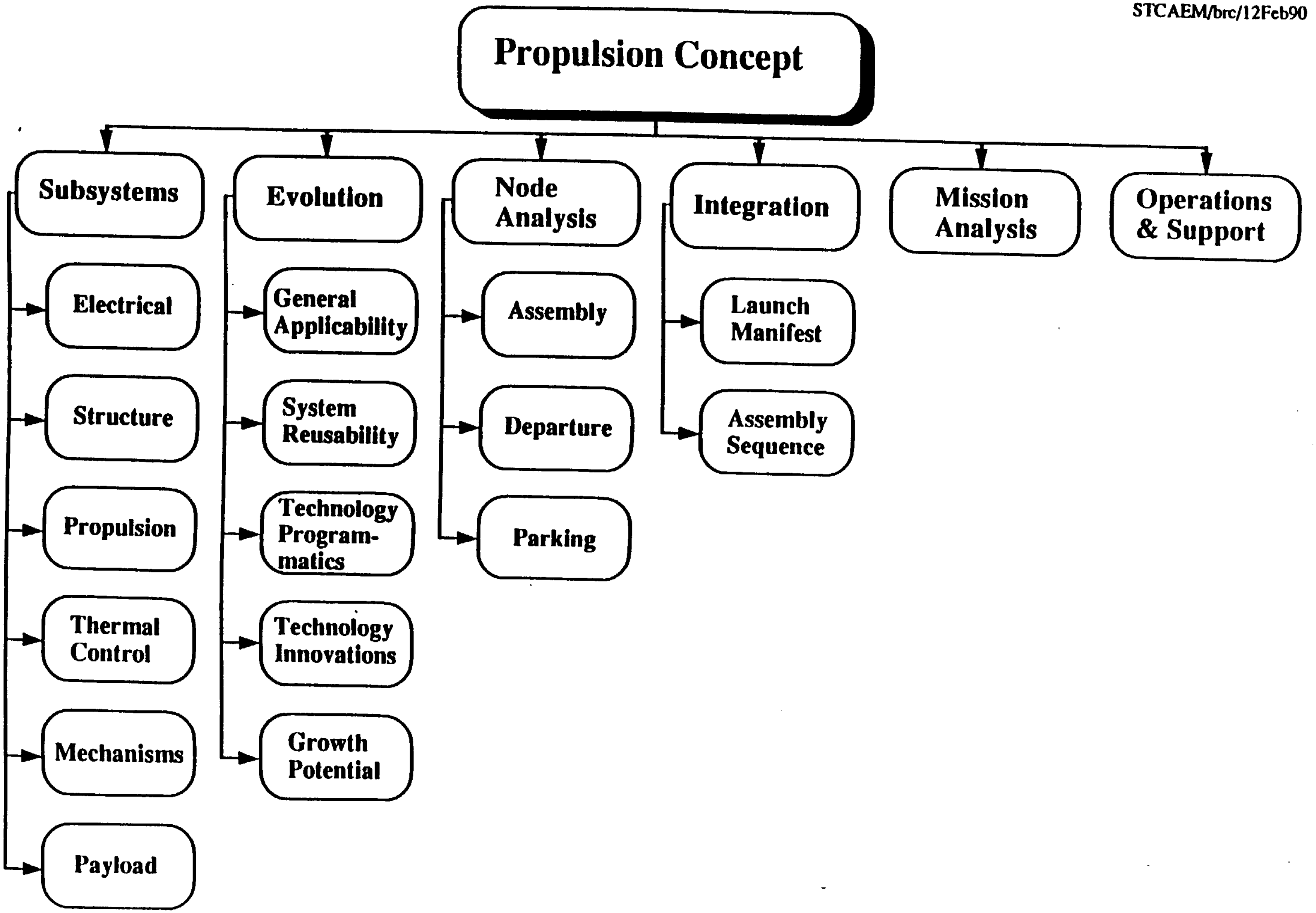

Propulsion Concept

- Subsystems
  - Electrical
  - Structure
  - Propulsion
  - Thermal Control
  - Mechanisms
  - Payload
- Evolution
  - General Applicability
  - System Reusability
  - Technology Programmatics
  - Technology Innovations
  - Growth Potential
- Node Analysis
  - Assembly
  - Departure
  - Parking
- Integration
  - Launch Manifest
  - Assembly Sequence
- Mission Analysis
- Operations & Support

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