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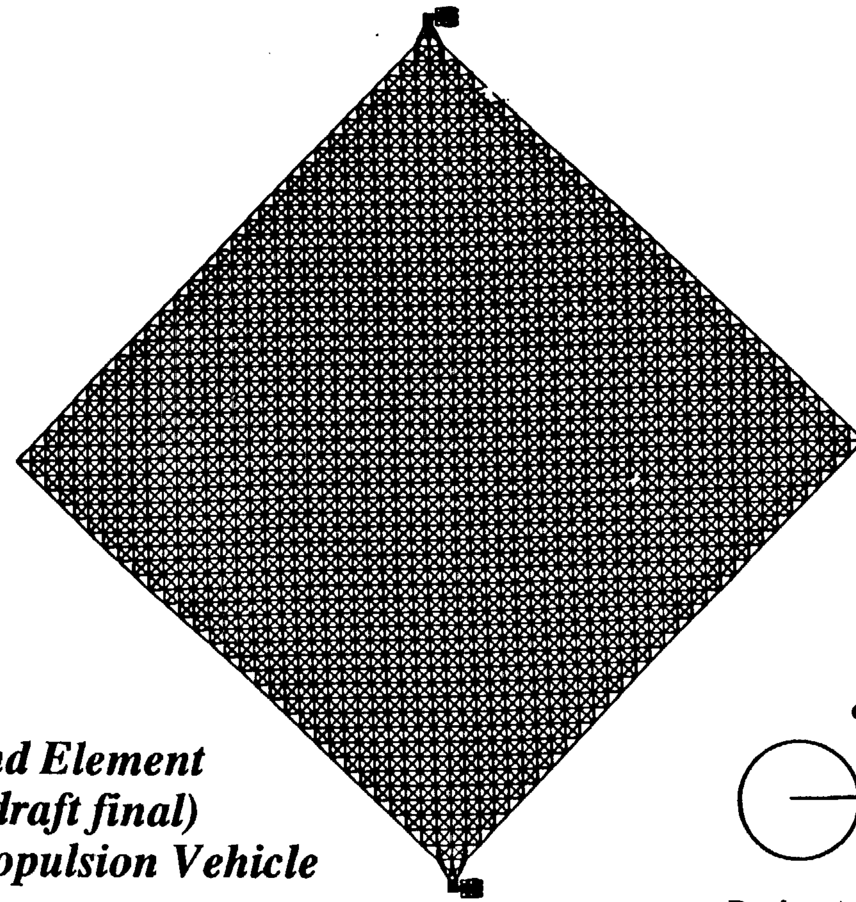

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ADVANCED CIVIL SPACE SYSTEMS

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

Solar Electric Propulsion Implementation Plan and Element Description Document

Boeing Aerospace and Electronics
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Space Transfer Concepts and Analyses for Exploration Missions

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Solar Electric Propulsion Implementation Plan and Element Description Document

Boeing Aerospace and Electronics
Huntsville, Alabama

Documentation Set:
D615-10026-1 IP and ED Volume 1: Major Trades, Books 1and 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
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Implementation Plan and Element Description Document
Solar Electric Propulsion (SEP)
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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
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)
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
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
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
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
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	≤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
σ	Standard deviation
μg	Microgravity

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I. Evolution of Concept

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Concept Development

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EVOLUTION OF THE SOLAR ELECTRIC PROPULSION (SEP) VEHICLE

TECHNICAL ARCHITECTURE PRESUMED LEVEL I REQUIREMENTS - During the course of the STCAEM study, and particularly during the *90 Day Study*, many SEI (then HEI) transportation requirements were generated by Office of Exploration Level II. These are reported as appropriate and necessary in various sections of this report, as well as in the *STCAEM Implementation Plan & Element Description Document* technical volumes. Here, space only permits a summary discussion of the Level I requirements adopted by STCAEM as they evolved during the course of the study. The concepts developed and analyzed ultimately were to accommodate the in-space transportation functions required to support the buildup of a permanent presence on the Moon and initial human exploration of Mars. Thus, our Level I requirement was simply to **deliver cargo reliably to the surfaces of the Moon and Mars, and to get people to those places and back safely.** Vehicles in support of missions to *other* destinations are not part of SEI *per se*, and were not addressed by STCAEM. Planet surface system characteristics and Earth-to-orbit (ETO) launch vehicle characteristics were adopted as needed for manifesting purposes, largely intact from other sources. No design work was performed for these two categories. In addition, the mission planning horizon was limited to the year 2025, about 35 years from now.

The chief Level II requirement governing the dimensions of the vehicle concepts we developed came to us during the *90 Day Study*, and was a crew size of 4 for Mars missions. Subsequently, STCAEM performed a simple skill mix analysis or these long-duration missions. Our result was that doubling up on critical skills (for redundancy), given reasonable expectations of how many skills each crew member could become expert in, requires in fact a minimum of 6 - 7 crew members for Mars missions. For the sake of consistency, our vehicle concepts are shown comparable to the *90 Day Study* results, sized for four crew. Impacts accruing from larger crew sizes are discussed in Section x.3.

CONCEPT DEVELOPMENT METHODOLOGY - A vehicle concept emerges gradually through the iterative combination of requirements analysis, subsystems analysis, mass synthesis, performance analysis and configuration design. Because of the cascading, cause-and-effect nature of specific technical decisions in this cyclic process, the ability for a particular concept to remain fully parametric is incrementally lost, sacrificed for depth of detailing. The need to penetrate deeply even at the conceptual stage is twofold: (1) to uncover subtle integration interactions whose ramifications fundamentally revise the concept as they reflect back up the information hierarchy; and (2) to enable the production of graphical images of the concepts capable of being communicated widely *but grounded firmly in engineering detail.* If circumstances allow the concept development process to engage many cycles of reflexive adjustment, from requirements all the way down through subsystem detailing, the design oscillations subside eventually and the product that emerges is a robust and defensible concept. Basic differences in problems posed and solutions engineered lead concept developments in different directions. "Like" problems and solutions gravitate together; their recombination and resolution results in distinct, identifiable vehicle concepts which constitute *vehicle archetypes*. A concept is archetypal if it spawns concept progeny whose ancestry is clear, and if in so doing its salient features recognizably survive subsequent refinement, development and scaling. The ultimate purpose of the STCAEM Concepts and Evolution tasks was to generate, analyze, evaluate and describe such vehicle archetypes, and the role they could play in human space exploration missions.

The STCAEM architecture analysis identified seven major classes of transportation architecture for SEI lunar and Mars missions. Some are derived from different propulsion technology candidates; some are derived from distinct mission philosophies independent of propulsion method; most have many sub-options. Vehicle archetypes are keyed more closely to propulsion method than to mission mode, however, so we found that all seven SEI transportation architectures can be accomplished by derivative combinations of just five archetypal Mars transfer vehicle (MTV) concepts, two archetypal Mars excursion vehicle (MEV) concepts, and one archetypal lunar transportation family (LTF) concept. The concept evolution of these archetypes is outlined in the Major Trades IP&ED book.

DESIGN AND NECKDOWN CRITERIA - STCAEM concept development was punctuated by four "neckdowns", which winnowed down the option candidates generated at each successive level of detail throughout the study. The four neckdowns were intended to result in: (1) **feasible options**, based on promising propulsion technologies capable of performing SEI-class missions; (2) **preferred options**, representing the handful of candidates whose performance and technological readiness were judged to warrant detailed study; (3) **integrated concepts**, vehicle archetypes developed sufficiently to uncover their major integration concerns and architectural context ; and (4) **detailed concepts**, based on the reconciled integration of traded subsystems. The *90 Day Study* occurred such that the first two neckdowns were effectively reversed; cryogenically propelled, aerobraking technology was necessarily preferred at that time, due to depth of understanding. However, STCAEM later rounded out the picture by completing all four neckdown activities, in an ongoing manner throughout the study.

Studying the program architecture implications of various technology options for SEI missions led to the conclusion that the most generally accessible discriminators, *cost* and *risk*, are driven by more subtle technical discriminators than, for instance, initial mass in low Earth orbit (IMLEO). These can be grouped into three broad categories: *feasibility*, *flexibility*, and *multi-use design*. As indicated above, feasibility was the first filter for all concepts considered by STCAEM. **Flexibility** has three components: (1) *robustness*, which is the ability to perform nominally despite variable or unanticipated conditions; (2) *resiliency*, which is the ability to recover from accidental delays or mishaps; and (3) *evolution*, which is an adaptation over time to changing requirements. Flexibility is thus a measure of a program's technical strength and safety in the face of variable extrinsic factors. **Multi-use design** has two components: (1) *re-usability*, which means using the same hardware item more than once; and (2) *commonality*, which means using the same hardware design in more than one setting. Multi-use design is thus a measure of a program's cost-effectiveness and intrinsic longevity. These two key architecture drivers were paramount in interpreting the results of STCAEM's technical trade studies, and figured prominently in the development of element concepts.

MARS TRANSPORTATION - Four Mars transfer propulsion candidates survived all STCAEM neckdowns: cryogenic chemical, nuclear thermal, nuclear electric, and solar electric. Analysis of aerobraking resulted in two performance ranges of interest for Mars entry (hypersonic L/D = 0.5, and L/D = 1.0), as well as the use of high-energy aerobraking (HEAB) for capture at Mars. Consequently, the five archetypal MTV concepts are based respectively on: cryogenic/aerobraking (CAB), cryogenic all-propulsive (CAP), nuclear thermal rocket (NTR), nuclear electric (NEP), and solar electric (SEP) propulsion technologies. The two archetypal MEV concepts are based on the "low" and "high" L/D performance ranges analyzed.

SEP - Solar electric propulsion represents a non-nuclear, "decentralized" and extremely redundant STCAEM approach to advanced propulsion for SEI missions. It is not, however, a "low-tech" solution to Mars transportation as is commonly held. First, the technology associated with large electric engines is the same for SEP as it is for NEP, because in each case individual 1 MWe engines are ganged together to achieve the appropriate power level. Second, SEP challenges our lightweight, large space structures (LSS) technology more than any other SEI concept. The reference SEP is 203 m on a side, covering an area equivalent to 9.24 football fields; yet its supporting structure must have a mass on the order of only 15 t. Surface accuracy requirements are orders of magnitude less stringent for the SEP photovoltaic (PV) arrays than for high-precision, large space antennas studied in the LSS literature, but the design, fabrication, deployment and maintenance of LSS of SEP-scale remains unvalidated empirically. Third, the size, fragility and unit-repetition appropriate for SEP concepts absolutely requires robotic-mediated maintenance. The positive side of this is that addressing robotic requirements for SEP may help us face up to the necessity and utility of state-of-the-art automation for other vehicle concepts as well. Finally, high-performance, robust, sufficiently lightweight and low-cost PV assemblies have yet to be demonstrated either. The usefulness of SEP hinges critically on our ability to fabricate acres of advanced PV assemblies economically.

Early STCAEM versions of SEP vehicle concepts presumed motorized unfurling of diaphanous, flexible PV blankets across a skeleton of ribs diagonalized by cables. Engine plume impingement of the arrays and structure was avoided by locating two engine assemblies at opposite ends of a long, truss outrigger. The thermal rejection system for the electrical power management and distribution (PMAD) system was centralized in two areas with dedicated radiators. Further investigation of current thinking on practical concepts for LSS led us to adopt the area-truss approach as the only way to get requisite stiffness and remain lightweight. The bay size selected was 7 m, as this limits parts count while not exceeding a reasonable span for projected, strengthened PV blanket technology. The blankets themselves consist of an iso-stress mesh of kevlar fibers to which are bonded stiffened, 4 cm advanced tandem solar cells. The need for engine outriggers was avoided by locating the twin engine assemblies at opposite corners of the square vehicle structure. The vehicle thus sweeps back at 45° angles from the nominal thrust line, and our presumed impingement envelope (a combination of ±20° for plume spreading and ± 20° for engine gimballing) was only 40°. To first order, the thrust line is in-plane with the vehicle because the solar arrays must be sun-facing, while the thrust must average tangential to the transfer orbit.

The STCAEM SEP reference vehicle has an extremely large number of identical parts, and was developed along with a matching robotic assembly, deployment and maintenance scenario. Two kinds of robots are envisioned: (1) a dextrous truss-builder with the ability to move about the vehicle, top or bottom, inspect critical systems and change out defective components; and (2) an array-paver, capable of accepting cassettes consisting of pre-integrated, rolled PV blankets. The paver would attach the blanket to the vehicle structure, removing and rolling up the blanket's protective packaging sheet as it progressed in one-bay-wide strips. On the SEP's first flight, the paver would deploy the sacrificial transfer array, undeploy it once beyond the van Allen belts, and subsequently deploy the full, main array for interplanetary flight.

ARTIFICIAL GRAVITY (SEP) - The need for artificial gravity on long-duration interplanetary transfers has not been established. Neither has the *lack* of such a need, however, so STCAEM was obligated to examine the penalties incurred by requiring continuous artificial gravity *en route* between Earth and Mars. Various approaches to rotating artificial gravity have been proposed; STCAEM assessed all of them, and invented some new ones. The fundamental *design* problems associated with artificial gravity derive from: (1) the need for a countermass for rotation; and (2) the high mass cost of precessing the angular momentum vector of a system having large rotational energy. Elegant solutions to both are elusive, and vary widely with propulsion option. Secondary complications are communications and navigation pointing, flight structures sized to hang heavy vehicles, and possibly material fatigue. The fundamental *operations* problems associated with artificial gravity involve crew EVAs during rotation, robotic maintenance in the vehicle's gravity field, crew physiological and psychological responses to a rotating environment, performing minor course-correction propulsive maneuvers and testing the capability prior to departure. Our work has verified that artificial gravity appears feasible for Mars-class missions, for all propulsion options, at fairly modest mass penalties.

Vehicles based on electric propulsion pose the toughest integration challenge of all for artificial gravity. Being low-thrust systems, they must burn for a substantial fraction of the transfer time. One simple approach is to rotate the vehicle only during the mid-transfer coast period (1 - 2 months) and upon arrival at Mars (if a conjunction profile is used to allow long stay times in Mars space). In case intermittent artificial gravity is an insufficient solution, however, it is important to develop full-blown alternatives. STCAEM examined several configuration options. Required thrust vector histories for low-thrust transfers are not completely understood at this time. Another simple approach would be to keep the thrust vector attitude constant in space, avoiding a need for spin-vector precession. To first order, however, it appears that such repointing would be required, and it is expensive propulsively. We examined using a "cross-product" electric engine located on a long outrigger; even with generous configuration assumptions, the mass penalty is about 10 % of IMLEO. If the spin vector is normal to the transfer plane, little repointing would be required, and we selected this option for both NEP and SEP. We solved the problem of what to use for countermass (particularly acute for the SEP) by baselining a new invention called the "eccentric rotator". With this approach, everything on the vehicle except the habitable and payload systems *is* the countermass. This leads to the despun electric engines themselves tracing out small circles rather than lying along the spin axis. However, their attitude (all that counts for low-thrust propulsion) can remain constant, and the CM excursion is typically small (of order a few meters for NEP and a few tens of meters for SEP) so the gravity loads on the propulsion system are small. The dynamics of such rotating vehicles are not yet fully studied. Mass penalties as well as trip-time penalties appear small, of order 5 % of IMLEO for NEP including a spinup/spindown propellant budget presuming efficient electric thrusting for that purpose. SEP suffers more complications because its distributed structure is so fragile. Effects of the 4 rpm cyclic loading, and the bending moment introduced into the fragile structure by the unbalanced rotor, remain unstudied. Gravity loading of the main truss structure in the eccentric rotator configuration is as high as 0.46 g, and preliminary estimates of the vehicle's structure mass were increased 20 % over the microgravity version to accommodate this (because the SEP structure amounts to only 14 % of the vehicle inerts, however, this results in an inerts increase of 2.6 %).

Low-L/D Mars Excursion Vehicle (MEV) - The MEV archetype development began during, and was resolved just following, the NASA *90 Day Study*. It was originally conceived as a means of delivering 25 t of undefined payload to the surface of Mars. However, the specification of crew cab provisions, the analysis of vehicle mass balance, and consequently the configuration design of the vehicle all depend on specifics of the payload manifest. We assumed a 20 t reference surface module as an integral part of the MEV. This led to a "Mars campsite" design intended to support a crew of four for 30 - 60 d and became or standard lander design. Chief departures from the lunar campsite mode of operation were:

1) The MEV arrives with the crew already onboard, and so is capable of a really self-contained mission.

2) The MEV also brings with it an ascent vehicle (MAV) with a separate propulsion system, configured optimally for the ascent phase (or ascent after breakaway from the descent stage during a descent abort). The crew cab for the MAV is the operations bridge for the MEV during all its mission phases.

3) The MEV is configured for packaging within an L/D = 0.5 aerobrake. For CAB missions, this brake captures the as-yet unmanned MEV into Mars orbit autonomously, before rendezvous with the MTV, and is used again for the descent. For CAP and other types of missions with propulsive Mars orbit capture, this brake is used only for descent. In all design cases, terminal descent engines are extended through ports in the windward surface of the brake at low Mach number, and the brake is jettisoned subsequently, prior to touchdown.

The MEV configuration was developed to permit later removal and relocation of the surface habitat module, with the aid of surface construction equipment. A variant of the MEV, without either surface module or MAV, was analyzed for delivery of heavy cargo on unmanned missions. A quick assessment was made of the feasibility of re-using an MEV, presuming *in situ* production of oxygen and retention of the aerobrake until touchdown. The outcome was positive, although: (1) additional brake hatches appeared necessary for landing gear deployment, crew egress, and cargo offloading; and (2) a lightweight top-shroud appeared advisable due to aerodynamic drag on ascent, and to permit the crew bridge to protrude beyond the presumed wake-protection limit for direct surface viewing during terminal approach. Configuration options for a "split-stage" MEV, in which the same, or a portion of the same, propulsion system is used for ascent as for terminal descent, were also investigated, and shown to be simple variations of the archetype.

Our baseline aerobrake assembly concept presumed robotic-mediated final assembly of pre-finished, rigid aerobrake segments at *Freedom*. Packaging such segments efficiently by nesting them in an ETO launch shroud is made challenging because of: (1) the aerobrake's asymmetrical, deep-bowl shape, in which the maximum depth of a typical "slice" is comparable to reasonable shroud diameters; and (2) the aerobrake's lip, required for both aerodynamic performance and structural stiffening around the free brake edge. Subsequent manifesting analysis, in which segments were configured according to an initial rib-and-spar structure concept, indicated that two ETO flights would be required to launch a single aerobrake in several pieces. Such extremely volume-limited and volume-inefficient manifesting is an unacceptably poor use of the expensively developed capability that a heavy-lift ETO system represents.

In response to this manifesting problem, STCAEM proposed the "integral launch" concept, in which a fully assembled, integrated aerobrake is launched externally, mounted on the side of the launch vehicle exactly analogous to current STS operations. The low-L/D brake is comparable to the STS orbiter in linear dimensions, and is light enough to launch two at once, with capacity to spare for other, shrouded payload as well. Ascent performance of such a flight configuration requires study; the critical question is whether ascent loads would size the aerobrake structure out of the competitive mass range for the mission itself.

Our structural analysis indicates that since the deep bowl-shaped aerobrake loads like a doubly-curved shell, it may be possible to construct an actual "aeroshell" without resorting to ribs and spars or some other articulated skeletal structure system. The shell would be made of a relatively thin honeycomb-type material system with integral TPS. However, lip buckling would still require a stiff rim, probably facilitated by a closed-tube-section structure. Such a brake may be lighter, and certainly simpler, but the thickened rim would still cause packaging problems due to nesting interference.

High-L/D Reusable Mars Excursion Vehicle (RMEV) - The RMEV archetype development occurred in response to three drivers:

(1) Analysis so far indicates that L/D = 0.5 is sufficient at Mars for controlling an aero-vehicle at Mars. However, the existence of some mission design studies in the literature which advocate L/D > 1.5 for Mars, combined with our preliminary understanding of controllability under Mars conditions, make it important to know in detail how different the configuration constraints imposed by higher L/D would be from those imposed by the lower L/D (which by 1989 had come to be regarded generally as appropriate).

2) As the *90 Day Study* stimulated thinking about what the purpose of SEI Mars surface missions should be, concern developed that global, or at least wide, access to the surface of Mars was potentially important. High-thrust Mars transfer propulsion systems (chemical or NTR) tend to be mass-constrained by arrival and departure vector geometry to certain parking orbit conditions. Although there is no lack of interesting (scientifically important) landing sites accessible from the periapsis of *any* orbit at Mars, the fact that performance-optimized parking orbits are unique for each high-thrust opportunity causes a site-access problem if returning to the same surface site is required (for base buildup). Thus for high-thrust transfer propulsion options particularly, an ability to achieve cross-range on lander entry may be important. High L/D enables greater cross-range capability.

3) Certain Mars lander issues not imposed as requirements during the *90 Day Study* required analysis and design validation. Developing a new MEV concept, substantially different from the baseline MEV, allowed us to investigate those issues simultaneously and thoroughly. Specifically, we addressed: (1) a deep aerobrake structure concept, of interest for maximum structural efficiency and therefore reduced brake mass; (2) the ability to deliver large-envelope cargo manifests, represented in our design by a long-duration surface habitat module sized for 10 crew; and (3) re-usability of the MEV, based on *in situ* production of cryogenic propellant.

The vehicle shape represented by the RMEV has applications for other interesting mission modes, concepts for which have yet to be investigated in detail. Three examples are: (1) a smaller RMEV, sized commensurately with the MEV to be a modest cargo-delivery vehicle; (2) a direct-landing MTV, whose return propellant would be manufactured *in situ* on Mars; and (3) re-usable aerobraked "taxi" vehicles capable of performing the Earth-Mars cycler embark/debark function.

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Solar Electric Propulsion (SEP)

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 photovoltic 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 (2000 volts DC) power processors. Low voltage (20-32 volts DC) power processors are still needed for heaters, ionizing potential, and other misc. 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.

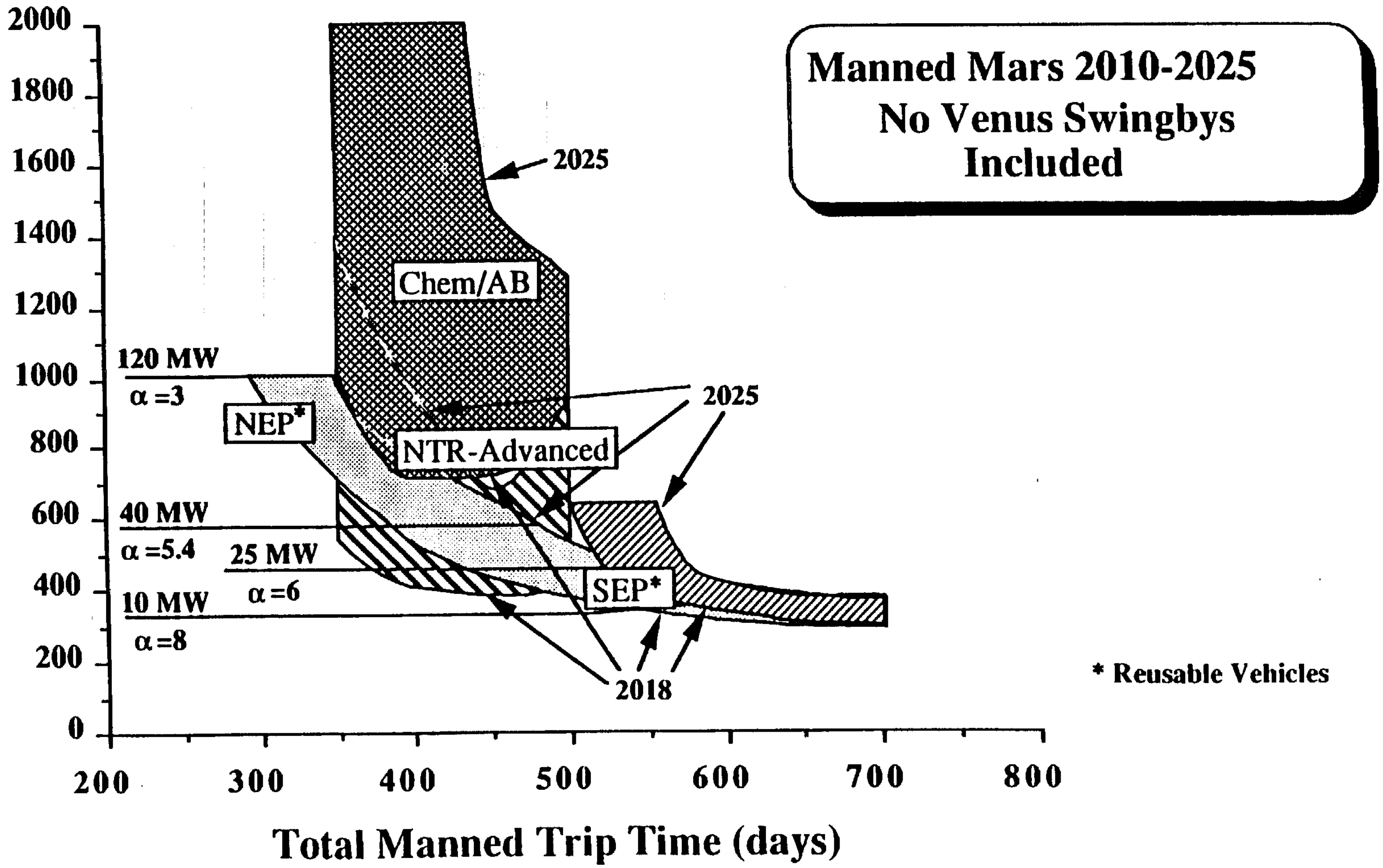
From the mission analysis for various forms of the vehicle indicate that reasonable power levels of 8-15 MW with trip times of 540-620 days , at vehicle specific mass (alpha) from 8-12 kg/kW will yield reasonably low IMLEOs. The use of certain types of gravity assists in flight, around the Moon, Earth and Mars may be employed to reduce trip time or vehicle preservation (flyby and recover). Other techniques, such as an expendable solar array for transfer through the van Allen belts, or staging at L2 may be used to reduce the stress on the vehicle.

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 for aerocapture 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 for aerocapture 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

Propulsion Option Comparison for Opposition Missions

Comparison with a 120 t Payload Opposition Opportunities



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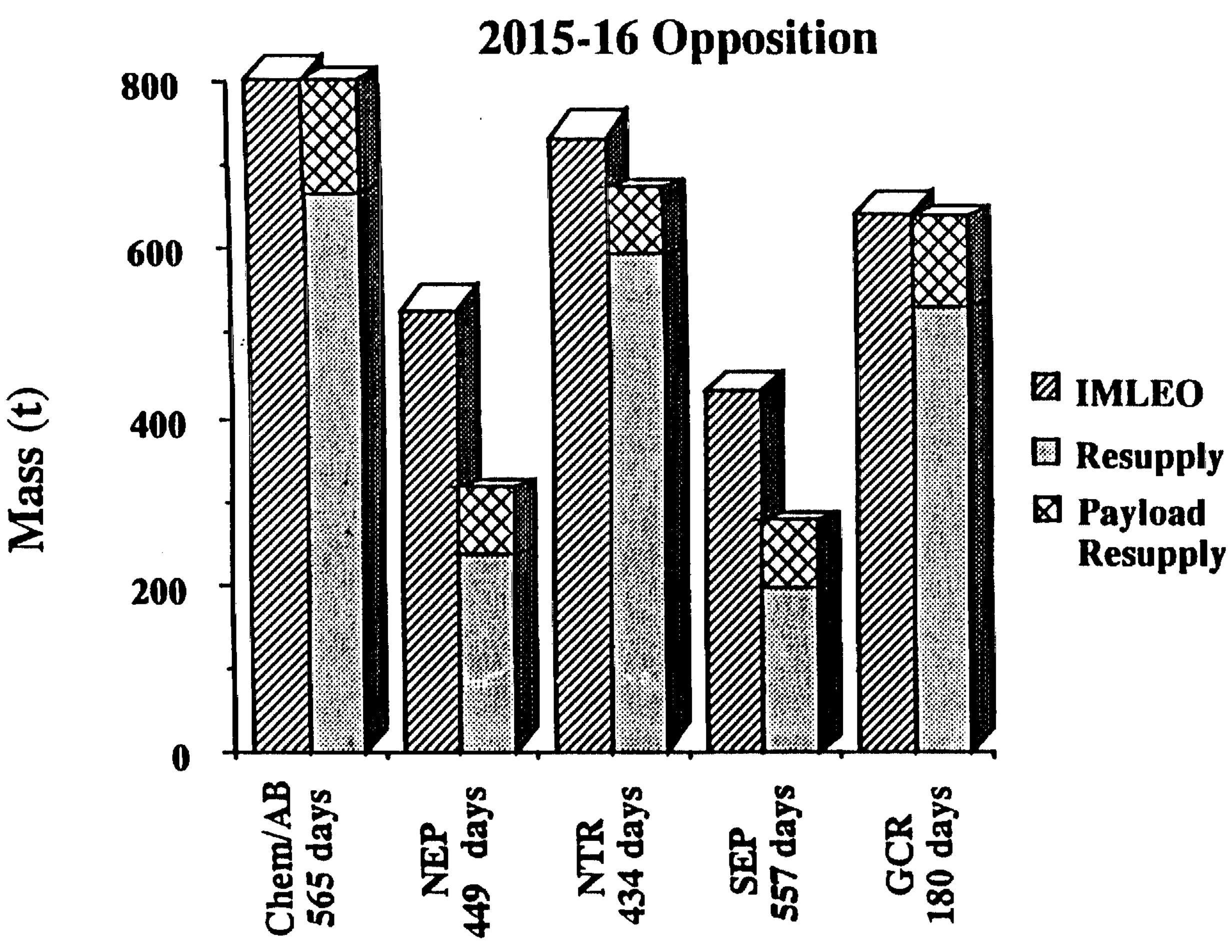
Advanced Propulsion Summary



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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Principal Findings- SEP

- A Lunar flyby can reduce trip times on the order of 10 days.
- A Mars flyby can reduce trip times on the order of 30 days.
- An Earth flyby can reduce trip times on the order of 20 days.
- A transfer array can save ~200 t associated with a chemical boost stage.
- Power levels (for the LEO to HEO transfer) around 5 MWe will be favorable for the vehicles we are transferring.
- Preliminary design was integrated by a CAD model with no incompatibilities noted
- Resupply mass associated with electric propulsion is less than 50% of chemical or NTR resupply mass.
- Isp is the single most important factor in a propulsion system. Electric propulsion offers the highest tested to date.

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SEP

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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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; good 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)

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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 five individual ion thrusters (the total of ten includes two 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 about 35% overall vehicle IMLEO. This relatively low propellant mass is a strong resupply advantage.

Vehicle bus - Thrust loads are extremely low for the electric propulsion (EP) system. Probable maximum loading is from impulses like attatude control system (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 far 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).

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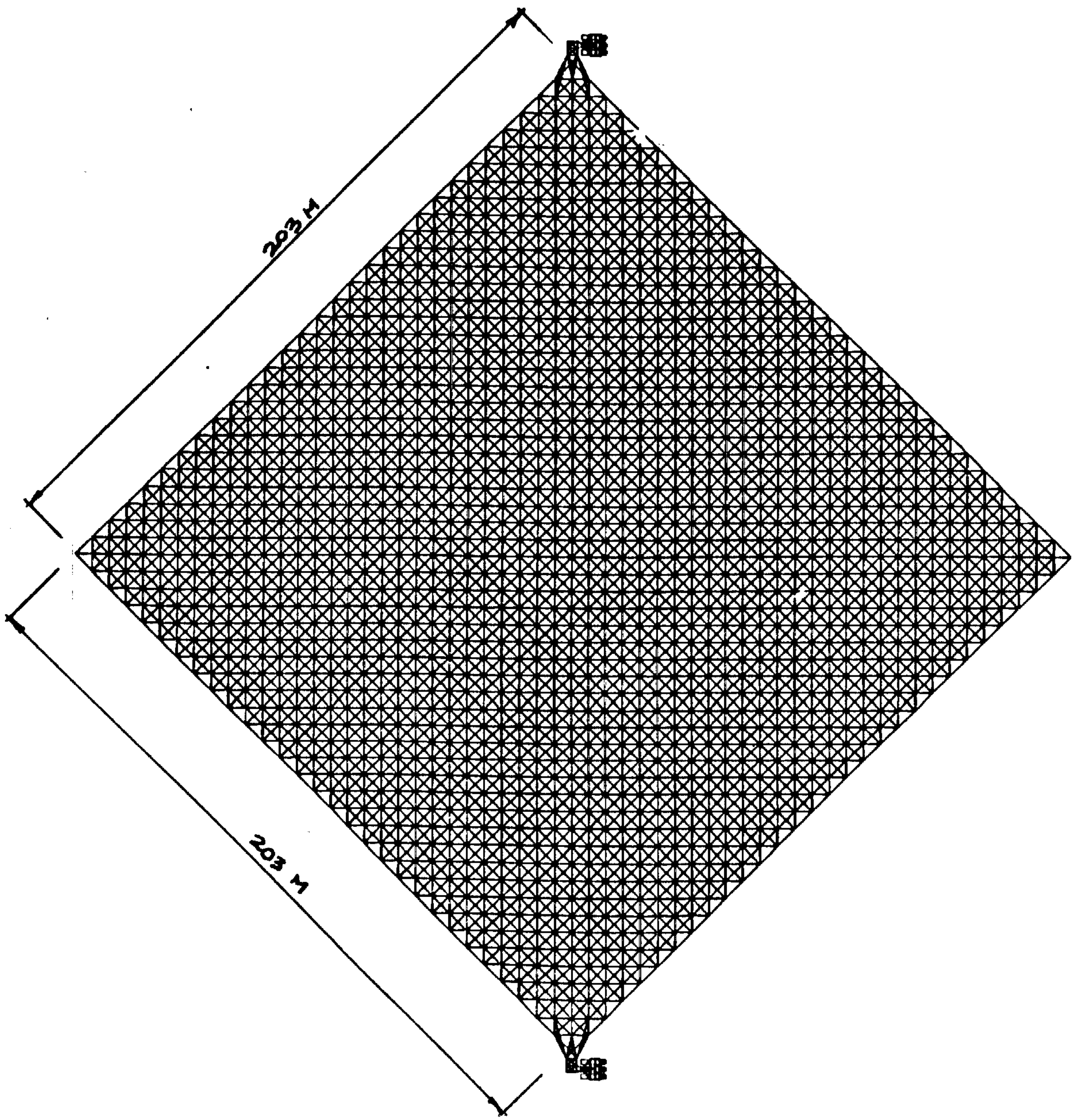
SEP Configuration



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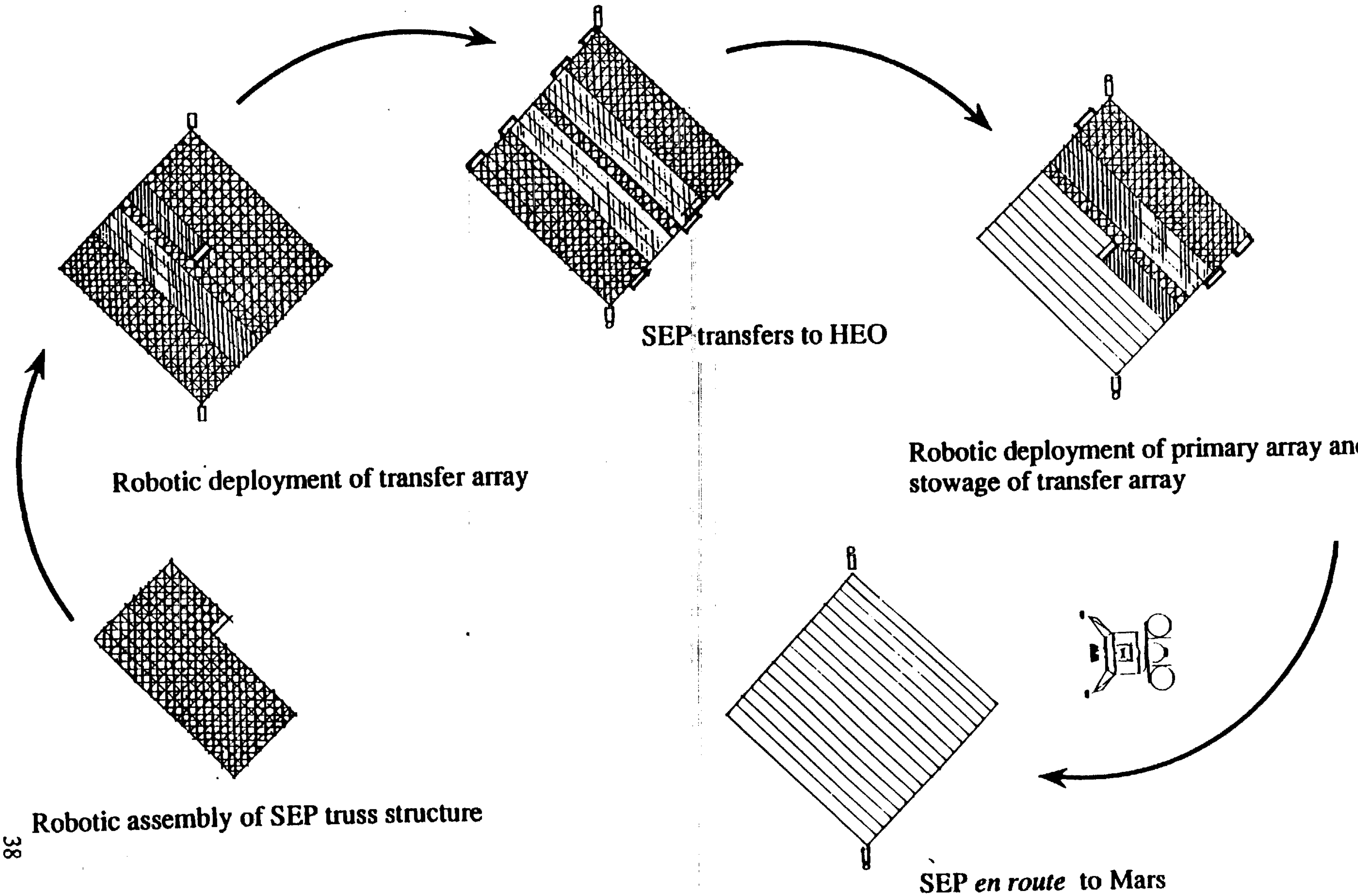
SEP Mission Sequence



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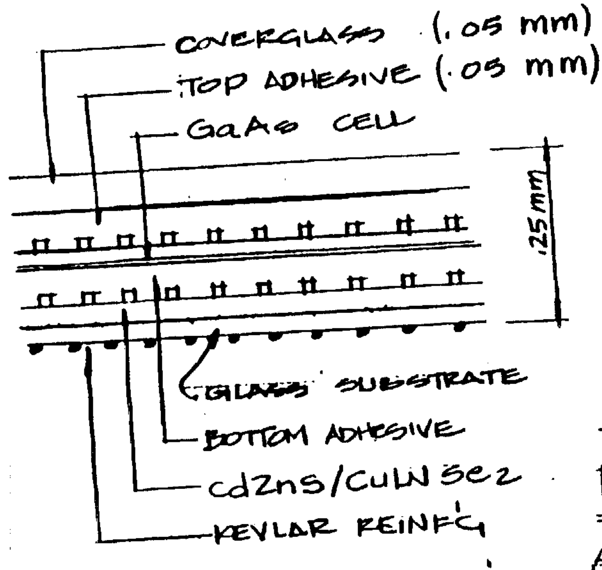
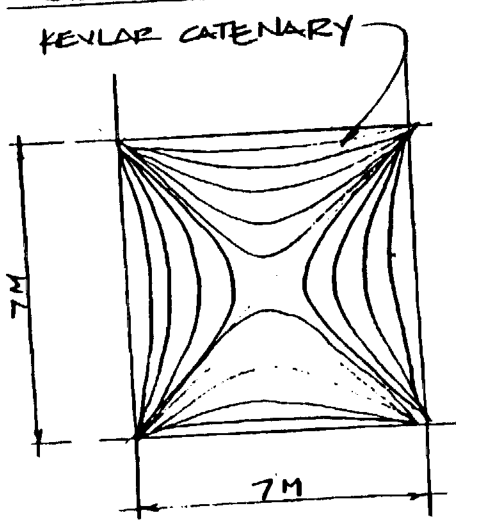
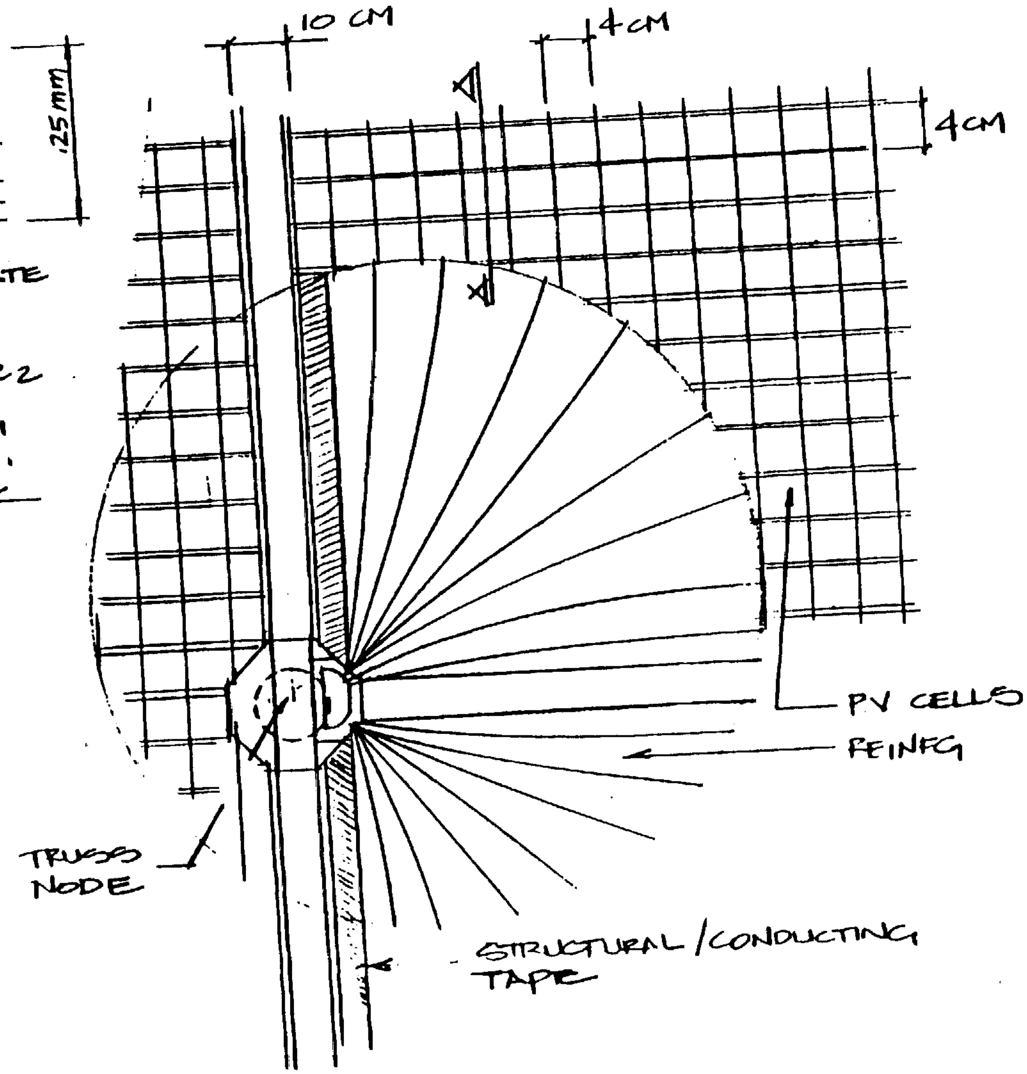
SEP P.V. Array Details



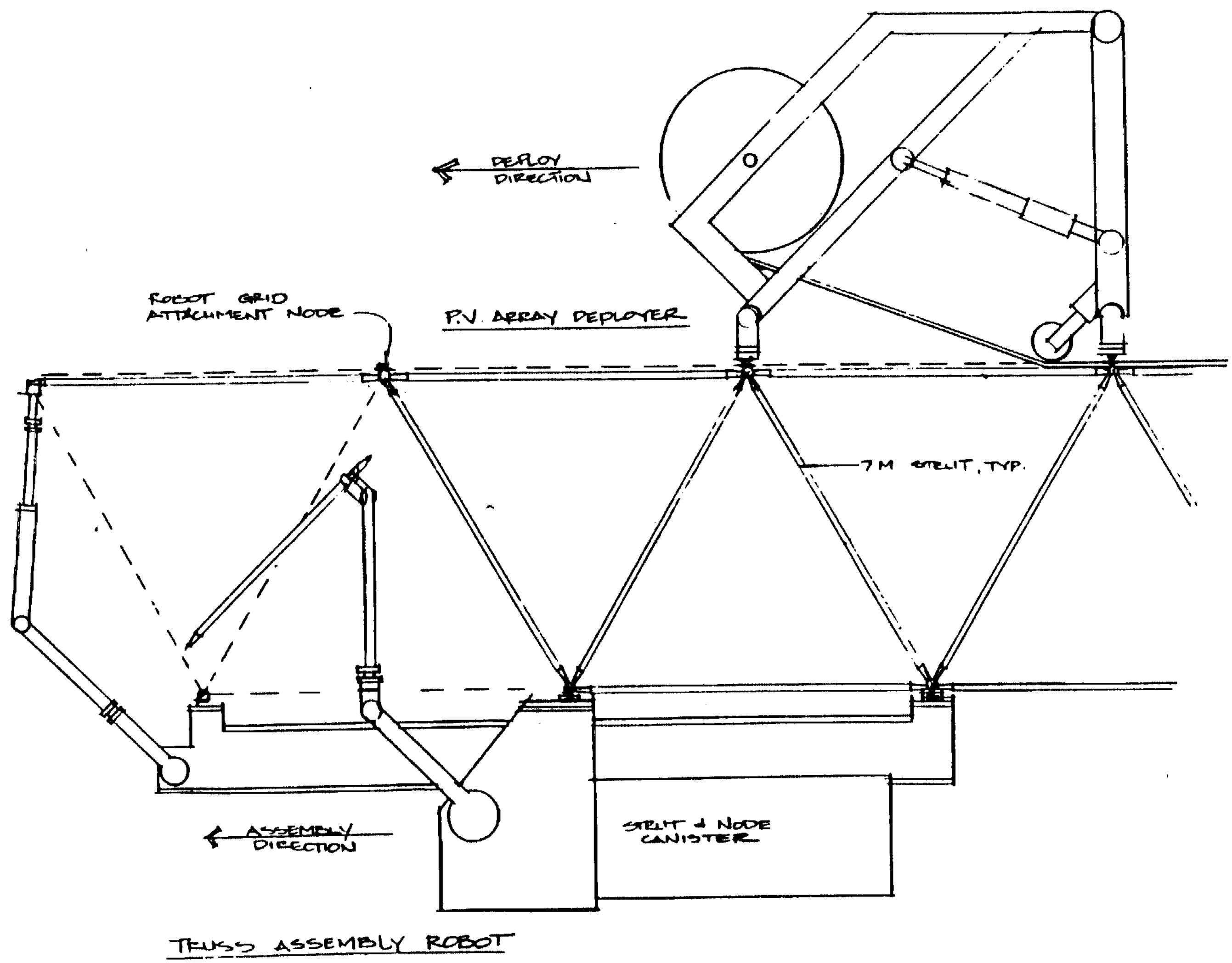
SECTION THRU CELL



ARRAY REINFORCING



Robotic SEP Assembly



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Architecture Matrix

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Reference Matrix to Alternative Architectures

In considering a complex task, it is useful to organize it into a heirarchy of levels. The higher levels are more important or more encompassings, while the lower levels include more detail or are more specific. Constraints (e.g., requirements and schedules) flow down from the higher levels and solutions or implementations build up from the lower levels. The first figure shows a heirarchy of six levels from national goals to performing subsystems. The following section discusses the fourth level, exploration architectures, in terms of the lower levels: element concepts and performing subsystems. Selection of preferred architectures will require the Government (the National Space Council, the President, and the Congress) to first define the top three levels.

Implementation Architectures

Seven architectures have been selected for examination: four different propulsion types (Cryogenic/Aerobrake, NEP, SEP, and NTR); two variations of In-Situ Resource Utilization (ISRU) for propellants with Cryogenic/Aerobrake propulsion (Lagrange point 2 refueling and Mars surface refueling); and a cycling spacecraft concept. Three basic levels of program scope are identified: small, moderate, and ambitious.

Multiple options can be generated within the basic architectures, varying launch vehicle capacity, orbital node type, and mission profile and propulsion type for the various Lunar and Mars vehicles.

Aerobraking is found to be applicable to all seven architectures, placing it as a 'critical' technology. Electric propulsion leads to the lowest reference vehicle mass, and also almost the lowest resupply mass. ISRU/Cryo leads to the lowest estimated resupply mass since most of the propellant is derived locally rather than coming from Earth.

Cost Models

Cost estimation is being performed using "parametric" methods. This technique uses a parameter, usually weight, as an input to empirically derived equations that relate the parameter to cost. It should be recognized that the source data for the cost models is past program experience, while the hardware being estimated will be built one or two decades from now. Therefore these cost estimates should be assumed to have a standard deviation on the order of +-100%. Hardware at technology readiness level 5 may be assumed to have a standard deviation in cost estimate of +-30%. No revenues from sale of products, services, or rights (i.e. patent rights, data rights), or commercial investment, are assumed in the cost estimates. These might appear in a scenario such as the Energy Enterprise.

Aa an example, the cost estimate for a NEP architecture shows an average annual funding level of $8 billion per year after initial ramp-up.

The principal cost drivers identified include number of development projects, reuseability, mass in Earth orbit, and mission/operational flexibility.

Analysis Methods

Individual trade studies are performed within each architecture to optimize it against evaluation criteria. The principal evaluation criteria to date has been initial mass in low Earth orbit, as a proxy for cost. The results of this optimization will then be compared to each other in groups. The early Mars group will compare all-propulsive, aerobraking,

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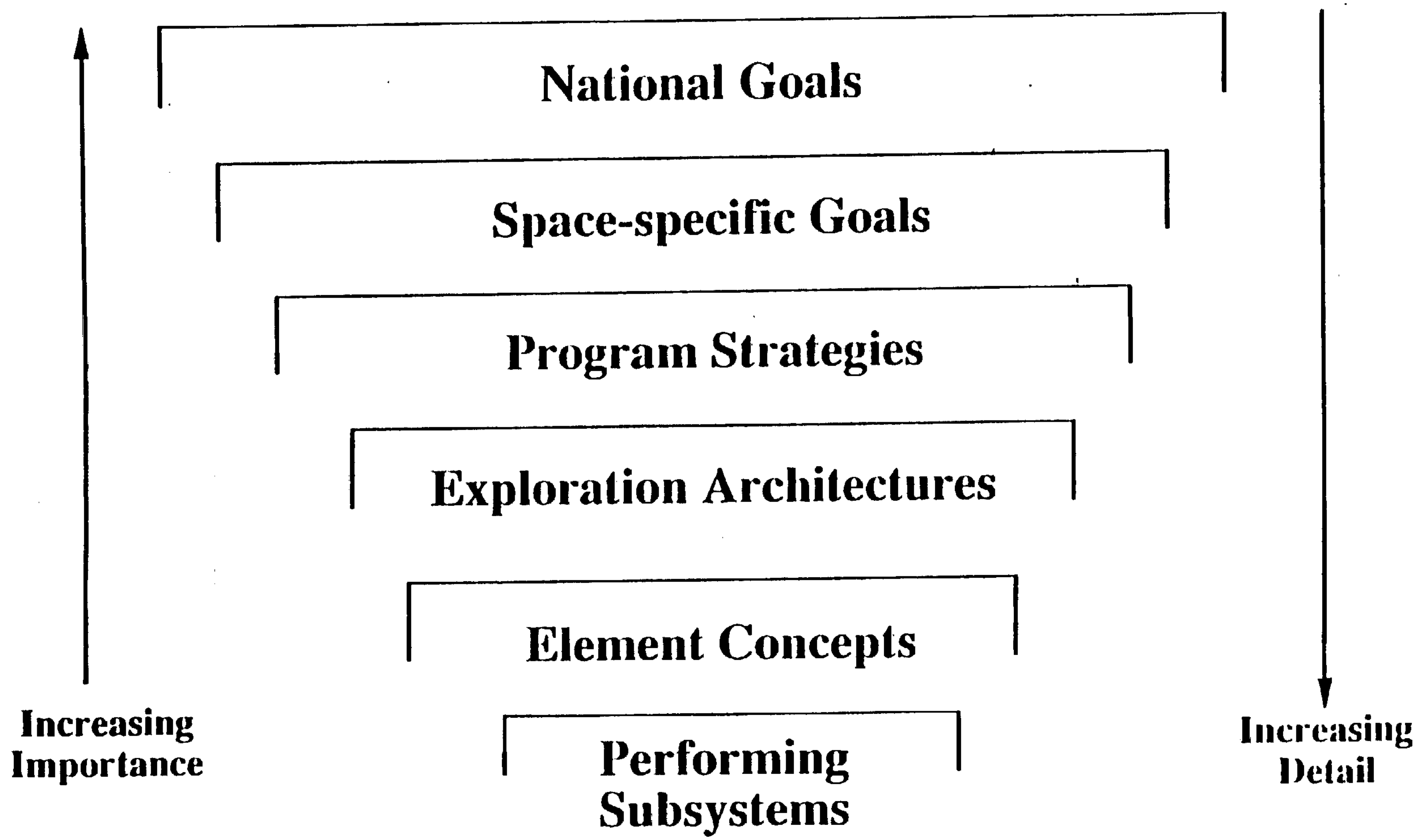
direct travel, and nuclear thermal among themselves. The electric propulsion group will compare SEP and NEP. The innovative group will compare Lunar oxygen to cycler orbits. These concepts may both be retained if it is advantageous to do so. Finally, the choice between early Mars and Late/Evolving Mars will need to be made on the basis of cost, risk, and performance, while combining the best features from each group.

Logical Types for Space Programs

Architectural planning for a space program deals with many levels of information. A major space program like the space exploration initiative must respond directivity to national goals in traceable ways. While we do not determine national goals, it is our business to understand how exploration architectures can be evaluated in terms of national goals.

National goals translate to space specific goals for specific exploration programs such as science emphasis or expanding human presence. These in turn can lead to program strategies for space-specific goals such as low risk, high technology, low cost and so forth. Finally, exploration architectures are integrated assemblages of systems, mission profiles, and operations, necessary to satisfy program goals.

"Logical Types" for a Space Program



Each logical type subsumes all the subordinate types

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Overall Study Flow

The study flow, as required by MSFC's statement of work, began with a set of strawman concepts, introduced others as appropriate, conducted "neckdowns", and concluded with a resulting set of concepts and associated recommendations.

As the study progressed, much discussion among the SEI community centered on "architectures". In this study, architectures were more or less synonymous with concepts, since the statement of work required that each concept be fully developed including operations, support, technology, and so forth.

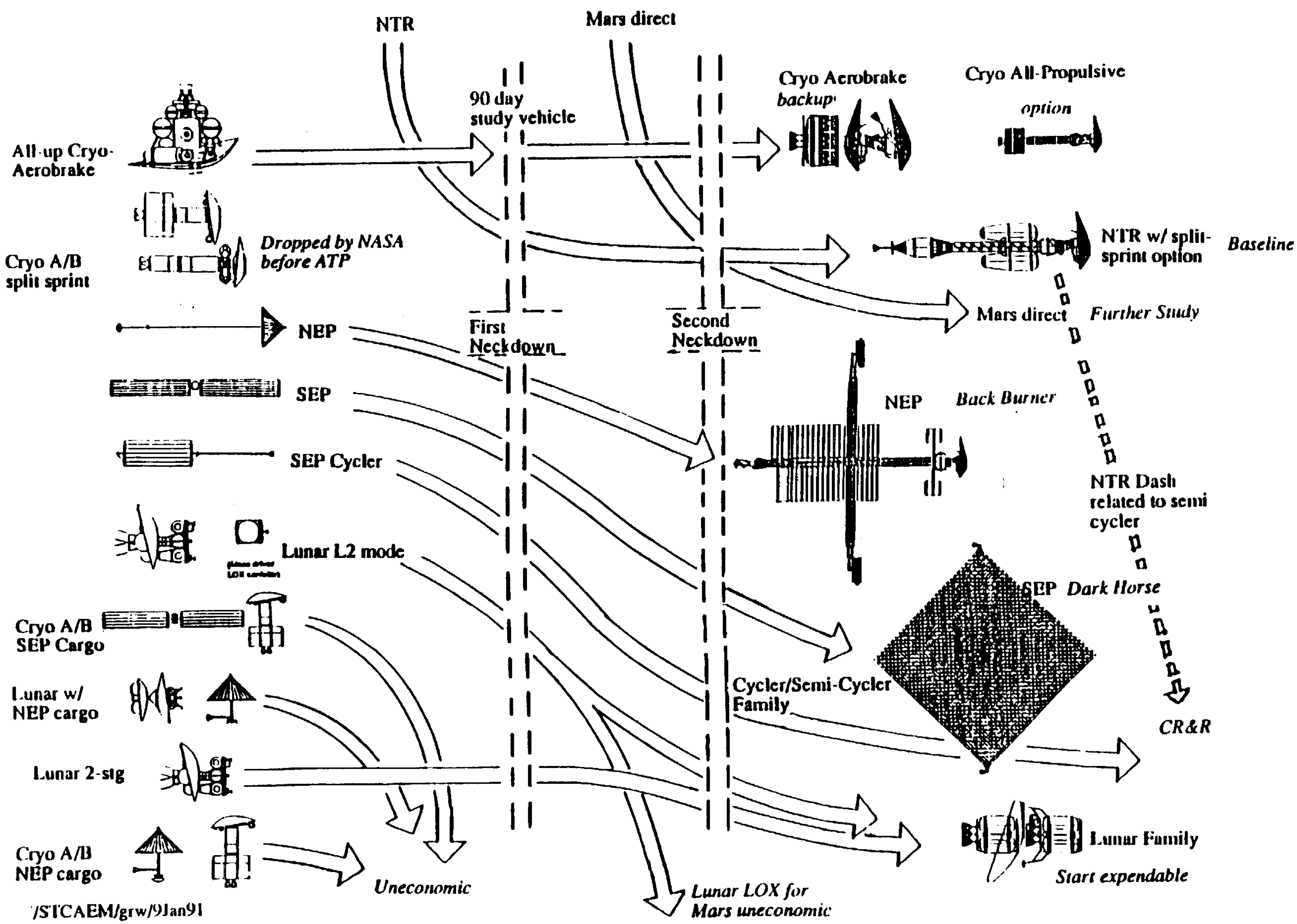
We started with ten concepts as shown on the facing page. Combinations of major technologies, such as electric propulsion and aerocapture, were quickly determined to be uneconomic in view of high development costs. Further, we found that electric propulsion systems could perform both crew and cargo Mars missions if crews are transported to and from the electric system at about lunar distance by a lunar transfer vehicle.

New systems introduced included nuclear thermal rocket (NTR) and Mars direct. NTR was introduced as an option by NASA during the "90-day study". We introduced the Mars direct profile (everything is landed on Mars; the return propulsion system is loaded with oxygen and perhaps fuel as well on Mars) in March 1989. Martin-Marietta subsequently publicized one variant of this concept.

Lunar oxygen for Mars missions was found to be uneconomic because of long payback time for the launch mass required to emplace lunar oxygen production on the Moon. Lunar oxygen has a reasonable return on investment for lunar transportation at two or more lunar trips per year.

The cycler architecture was broadened to include semi-cyclers. Late in the study we introduced an NTR-dash mode (described later in this briefing) closely related to the semi-cyclers.

Overall Study Flow



Program Implementation Architectures

We have selected seven program implementation architectures for architectural analysis. These seven architectures incorporate the advanced propulsion options of principal interest in complete evolutionary architectural scenarios for lunar and Mars exploration. The facing page lists the features of each architecture and the rationale for selection of each.

Some of the architectures include suboptions. For example, the nuclear electric propulsion and solar electric propulsion architectures include optional use of the electric propulsion system for lunar cargo delivery from LEO to lunar orbit. The L2-based cryogenic aerobraking architecture includes use of NTR and NEP vehicles for LEO to L2 cargo delivery as options, and also includes a cryogenic all-propulsive conjunction mission option.

Program Implementation Architectures

Architecture	Features	Rationale
Cryogenic/aerobraking	Cryogenic chemical propulsion and aerobraking at Mars and Earth. LEO-based operations.	NASA 90-day study baseline
NEP	Nuclear-electric propulsion for Mars transfer; optionally for lunar cargo.	High performance of nuclear electric propulsion
SEP	Solar electric propulsion for Mars transfer; optionally for lunar cargo.	High efficiency of solar electric propulsion; find cost crossover for array costs.
NTR (nuclear rocket)	Nuclear rocket propulsion for Lunar and Mars transfer.	High Isp of nuclear rocket enables avoidance of high-energy aerocapture at Mars.
L2 Based cryogenic/ aerobraking	L2-based operations; use of lunar oxygen.	L2 base gets out of LEO debris environment. Lunar oxygen reduces resupply by ~ factor 2.
Direct cryogenic/ aerobraking	Combined MTV/MEV refuels at Mars and LEO. "Fast" conjunction profiles.	Eliminates Mars orbit operations.
Cycler orbits	Cycler orbit stations a la 1986 Space Commission report	Eliminates boosting massive Mars transfer vehicle.

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SEI Program Scopes for Transportation Architecture Analysis

There are many space-specific goals and program strategies. We believe that transportation architectures will respond mainly to program scope. Some architectures are best suited to small program with early goals and others best suited to long range larger programs with ambitious goals. We have selected three representative scopes for small, moderate and large programs as illustrated on the facing page. These scopes permit definition of transportation requirements in terms of numbers of people and amounts of cargo transported to particular locations on particular schedules.

The second important feature of the scopes we intend to investigate is that they cover a scale factor greater than ten. A man tended science station may have few people on the Moon for short periods, or few people on Mars for short periods every other year. Permanent science bases will involve a dozen or so people. Industrial development of lunar resources on a scale of helium-3 scenarios leads to numbers of people presently estimated in the range of thousands by 2050. Beginnings of humans settlement of Mars involves numbers in the range hundreds to thousands. The 20-25 horizon for SEI is expected to permit growth in numbers of people only to dozens or so.

SEI Program Scopes for Transportation Architecture Analysis

Descriptor	Small	Moderate	Ambitious
Lunar Operations	Man-tended science station	Permanent science base 6 - 12 people	Industrial development of lunar resources
Mars Operations	Expeditionary visits ~4 people	Permanent science base 6 - 12 people	Beginnings of human settlement

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Three Activity Levels for Architecture Evaluation

We established three levels of activity to evaluate in-space transportation options. The minimum was 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 median (full science) program aimed at satisfying most of the published science objectives for lunar and Mars exploration. The maximum program aimed for industrialization of the Moon, for return of practical benefits to Earth, and for the beginnings of colonization of Mars. The range of activity levels, as measured by people and materiel delivered to planetary surfaces, was 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.

Activity levels were selected with underlying program objectives in mind:

(1) The minimum lunar program establishes astrophysical observatories on the Moon and provides a man-tending capability to maintain them. To the extent that man-tending lunar visits are not needed for the observatory system, the transportation capability can be used to explore interesting lunar sites for lunar geoscience objectives.

(2) The minimum Mars program is very similar to Apollo, i.e. six sites visited for short periods (two sites per mission and three missions); samples obtained within a few km. of each landing site. If the manned visits are preceded by suitable robotic missions, the scientific payoff for these visits can be high relative to the investment.

(3) The "full science" lunar program adds human permanence at the Moon for extensive scientific and technological exploration. Where the minimum program offers very little opportunity for lunar geoscience, this program offer much. It also permits development of in-situ resource technology for production of surface systems. The reference program also emplaced a lunar oxygen production system to serve the transportation system.

(4) The "full science" Mars program multiplies by several the crew person-days on Mars by including more missions and by more staytime per mission. This program falls short of a permanently-occupied base on Mars, but achieves surface stays greater than a year.

(5) The lunar industrialization program adopts production of helium-3 as a strawman industrial objective and places enough facilities and infrastructure on the Moon by 2025 to return 1 GWe helium-3 fusion fuel to Earth.

(6) The Mars settlement program moves towards Mars settlement. A robust nuclear electric propulsion system is fielded, with convoy flights by 2015. Mars population reaches 24 by 2025, and the transportation system is capable of increasing Mars population by 24 per opportunity by 2025.

Three Activity Levels for Architecture Evaluation

Minimum	Median (full science)	Industrialization /settlement
Just enough to meet President's objectives	*Meet science objectives of lunar/Mars exploration*	*Return of practical benefits to Earth*
• Permanent lunar facilities, not permanent human presence	• Human permanence	• Extensive facilities and infrastructure on the Moon by 2025
• Astrophysics observatories • Man-tending capability • Explore interesting sites	• Opportunity for lunar geoscience • In-situ resource technology	• Lunar population 30 by 2025
• Three missions to Mars	• Order of magnitude more crew time on Mars	• Mars population 24 by 2025
• Similar to Apollo • Two sites per mission • Samples within a few km. of landing sites	• Approaches permanent base (stay time >1 year)	• Capable of increasing Mars population by 24 per opportunity by 2025.

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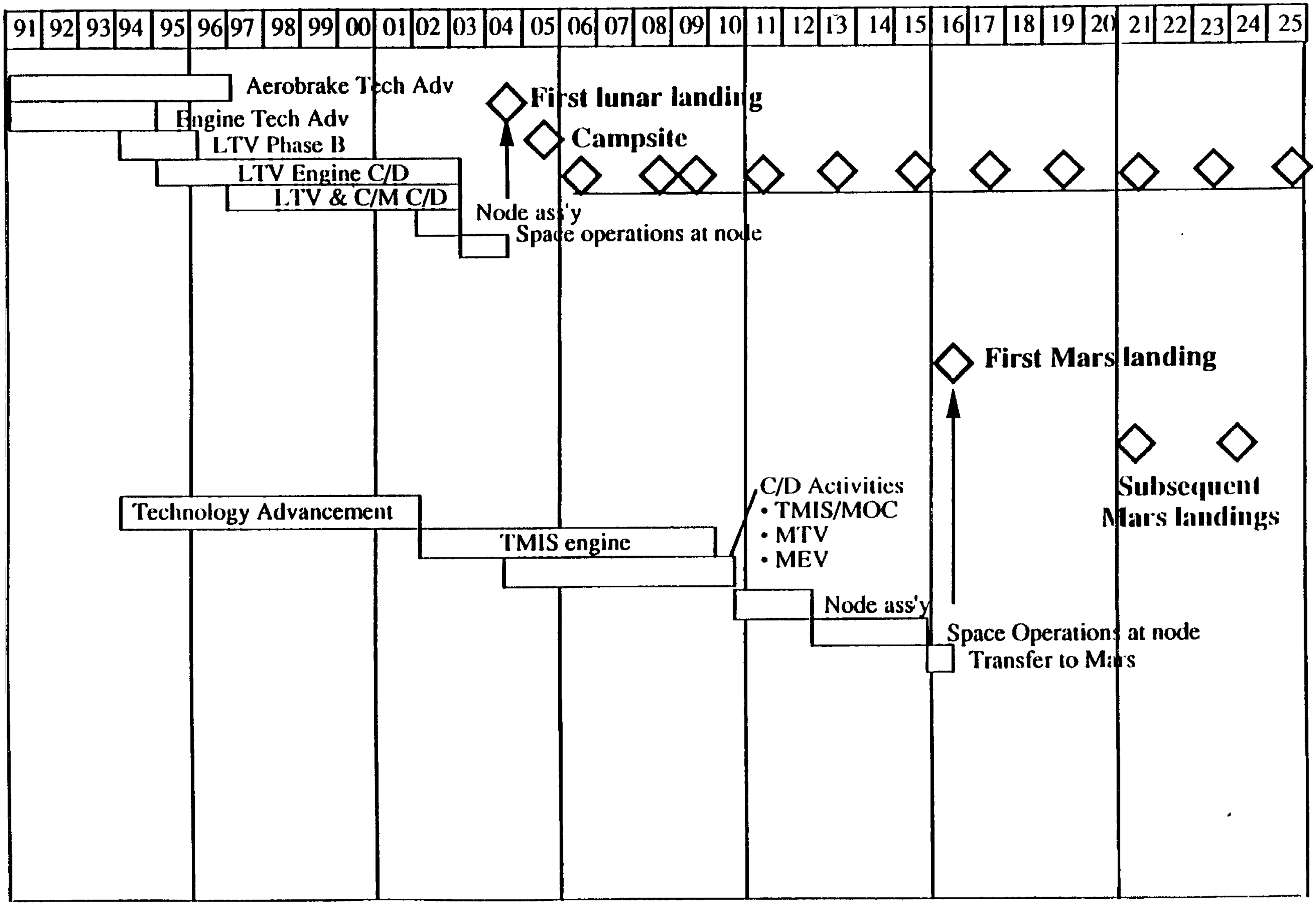
Minimum Program

The minimum program reference averages about 1/2 lunar trip per year and has only three Mars missions. Lunar science facilities are man-tended. Each Mars mission carries two landers (MEVs) for added exploration capability and a measure of rescue capability. Surface stays are about 30 days. Lunar and Mars in-space transportation systems are expendable.

Minimum Program



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Full Science Program

The full science program reference has about 2 lunar missions per year, to establish permanent human presence on the Moon with adequate supplies and equipment for extensive science and exploration. Lunar oxygen for lunar transportation is introduced about mid-way through the lunar program. Six Mars missions are accomplished, with later missions staying on Mars for more than a year. The Mars missions use multiple landers, as many as four late in the program.

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Full Science Program

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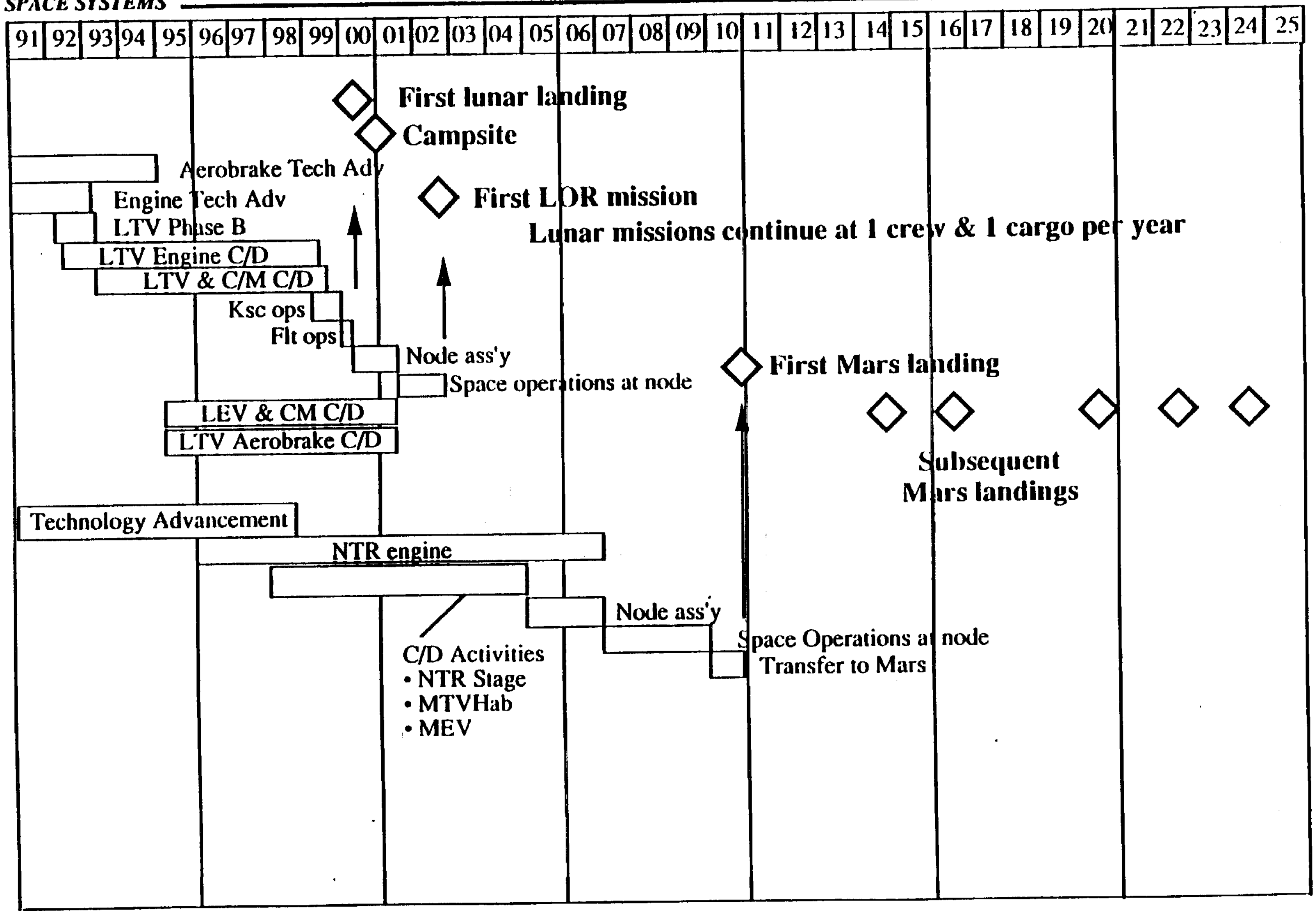

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Industrialization and Settlement Program

The industrialization and settlement program is very aggressive for both the Moon and Mars. Thousands of tons of industrial equipment are delivered to the Moon, driving lunar cargo trips up to five per year. Lunar oxygen is placed in production as early as possible. One crew trip per year leads to a population of 30 because crew stay times on the Moon increase to several years.

Initial Mars missions use a cryogenic/all-propulsive system because the aggressive nature of the scenario merited an initial Mars mission as early as possible, and the reference nuclear electric propulsion system cannot be ready in time. The NEP missions are operated in a crew rotation/resupply mode, opposition profile, with each crew staying one synodic period (about 2.2 years). The reference scenario evolves to reusable MEVs based on Mars, fueled from Mars resources. Heavy cargo capability is provided, up to 250 t. per opportunity by 2020. The Mars population grows to 24, and by the end of the scenario can continue to grow by 24 or more per opportunity.

ADVANCED CIVIL SPACE SYSTEMS

Industrialization and Settlement Program

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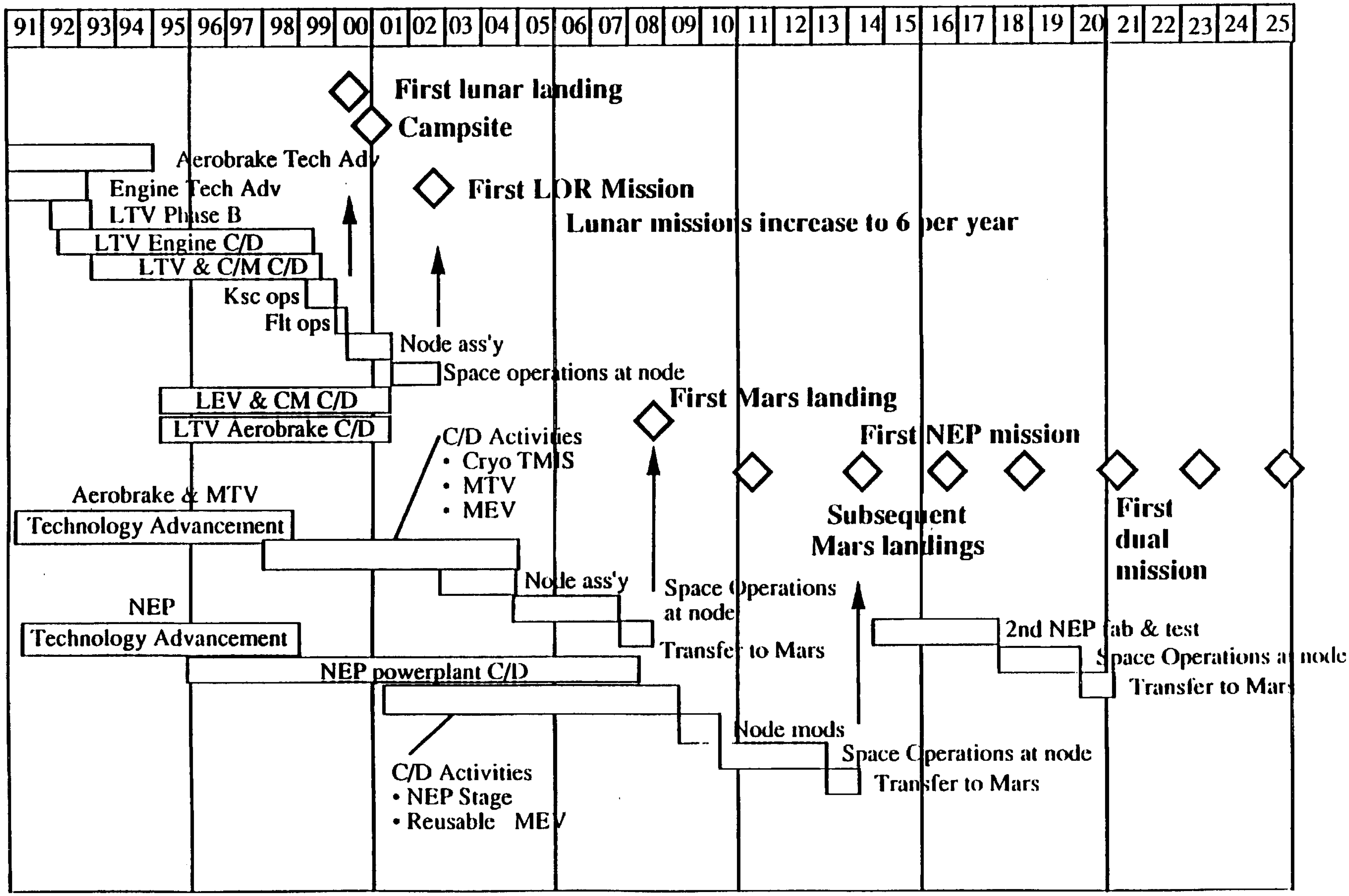

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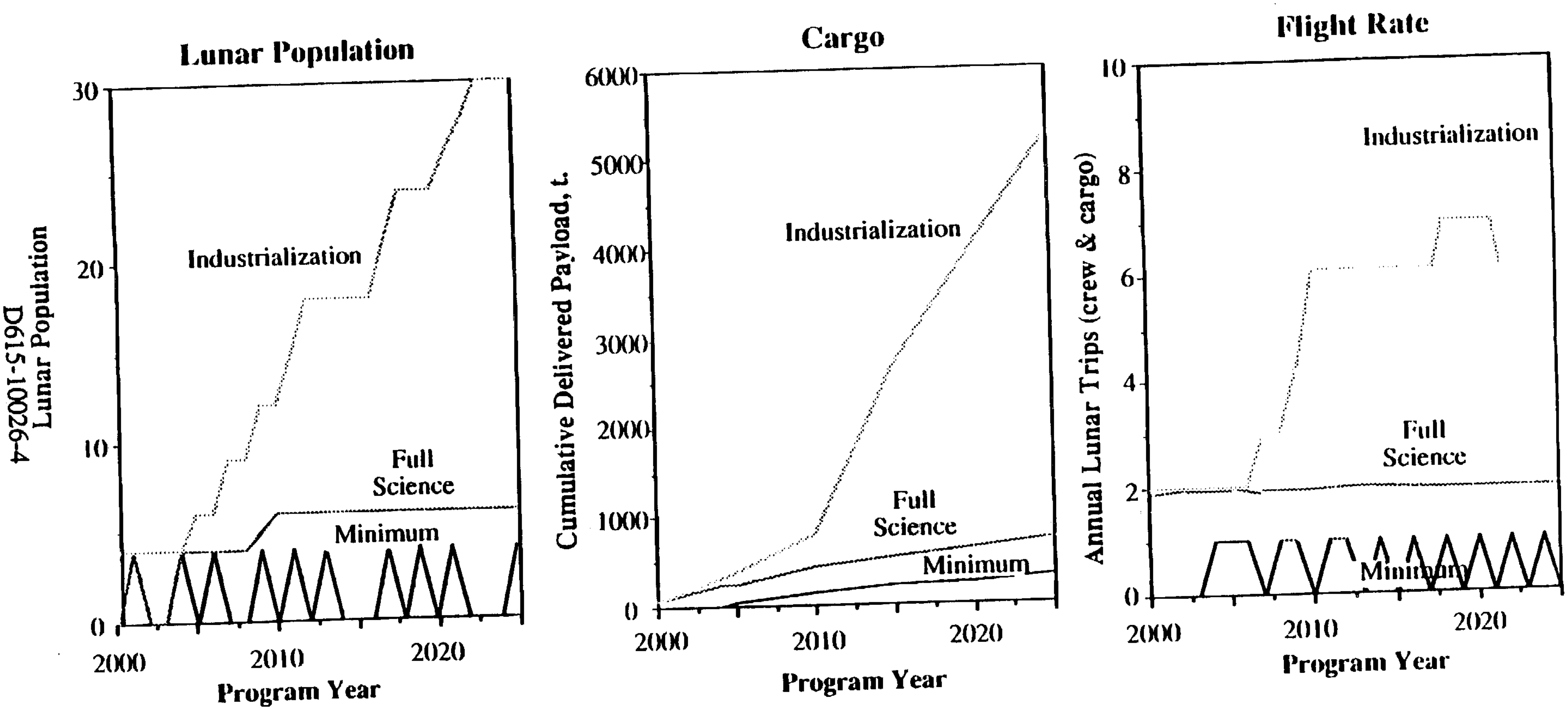
Lunar/Mars Program Comparisons

The next two charts compare the lunar and Mars program scenarios in terms of population, cumulative cargo delivered, and flight rate. The lunar population for the minimum scenario is four people for 30 to 40 days about every other year. The Mars population for the minimum scenario is 6 people on each of 3 conjunction missions, with 30 to 40 day surface stays. The full science menu scenario grows to year-long surface stays on conjunction missions. The lunar industrialization program goes to long stay times with indigenous food growth to build population. The Mars proto-settlement program obtains continuous presence by operating the NEP on an opposition-like profile in crew rotation/resupply mode. Later in this scenario, a second NEP is operated to provide two trips to Mars each opportunity.

These scenarios were the "input" to the manifesting and life cycle cost analyses.

Lunar Program Comparison



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ADVANCED CIVIL SPACE SYSTEMS

Mars Program Comparisons

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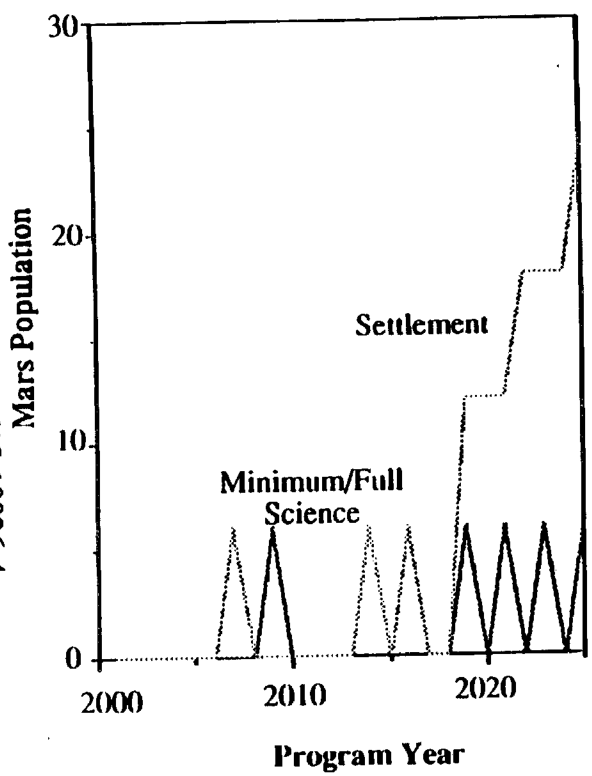
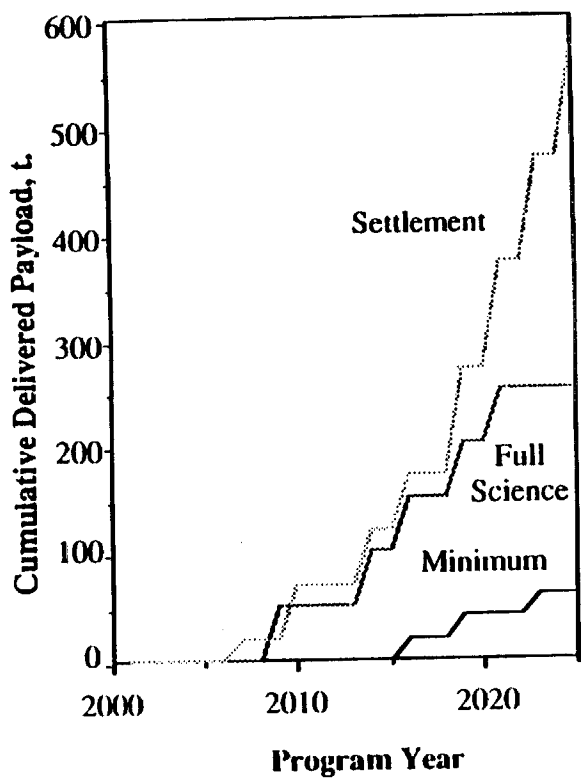
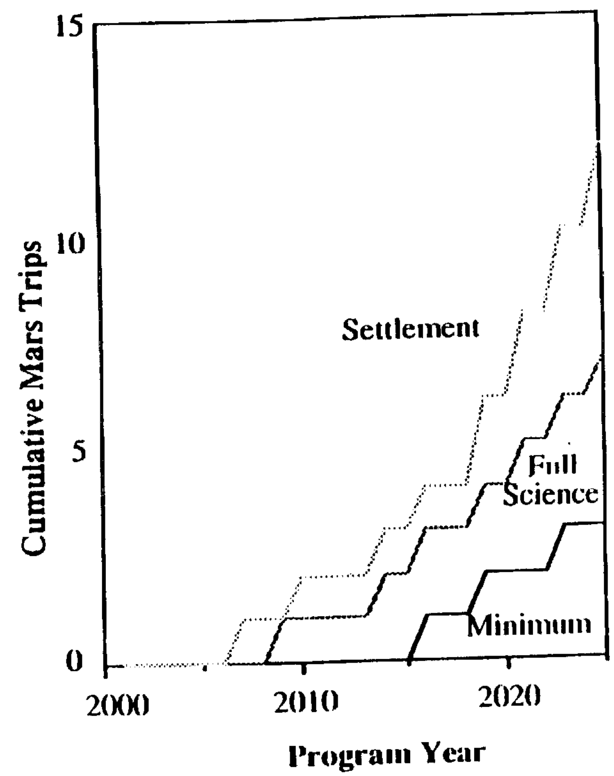
Population



Cargo



Flights



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Architecture/Launch Vehicle/Node Trends

Issues

- Launch vehicle size, shroud size, and lift capacity.
- Node complexity and cost.
- On-orbit assembly complexity
- Number of launches per year
- Development cost
- Per-mission cost

Trends from Architecture Analyses

- Large launch vehicle (up to 300 t. lift) does not eliminate on-orbit assembly.
- Keys to on-orbit assembly are (1) design the vehicle to keep it simple; (2) design for automation and robotics; (3) reusable space vehicles to reduce the frequency of assembly operations.
- Advanced in-space transportation technology reduces launch requirements enough that a 100-t., 10-meter shroud launch vehicle is adequate.
- Ultra-large launch vehicle results in high early program costs and is much more costly than advanced in-space transportation technology.
- Evolution and design for evolutionary transitions are the keys to affordable, efficient programs with long-term growth.

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Available Options

The facing page is a typical listing of the element options making up a total transportation architecture for SEI missions. The options listed are all candidates for incorporation into architectures. Trade studies have not eliminated any of these options. (The list is representative and not necessarily complete.) The number of options on this chart for each row of options is indicated on the far right. In most cases, any option can be combined with any other set of options. Thus, the total possible combinations number in the millions. It is clear that available future effort can not hope to examine all combinations. This drives us to a strategy for architecture sensitivities analysis, to develop key trends and conclusions from relatively few architecture combinations.

Available Options

								No. of options
ETO	100 t.	140 t.	200+ t.	Add prop tanker				3 x 2
Node	SSF	Separate	SSF + separate	Self-assy.	Wet tanks	Refuel vehicles	Propellant depot	4 x 3
Lunar mode	Direct	Direct/ lunar ox.	LOR	LOR/ lunar ox.	L2/lunar oxygen			5
LTV	Cryo all-prop	Cryo aerobrake	NTR	NEP/SEP cargo	Fully reusable	Partially reusable	Expend-able	4 x 3
LEV	Cryo	Storable	Combined with LTV	Fully reusable	Partially reusable	Expendable		3 x 3
Mars mode	2.7 year	1.5 year						2
MTV	Cryo all-prop	Cryo aerobrake	NTR	NEP	SEP	Cycler		6
Mars node	LEO	L2						2
MEV	Cryo	Storable	Combined with LEV					3

Total possible combinations 2,799,360

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Top-Level Trade Table

The facing page considers mission profile, basing at Mars, and propulsion. Four important issues are central to mission profile selection: crew radiation exposure, crew time spent in zero g, the component of mission risk that increases with mission duration, and the added cost of shortening trip time. At one extreme is the notion, frequently expressed, that a Mars round-trip mission should be completed in a year or less. This is possible with certain advanced propulsion technologies, but at considerably higher cost than for longer trips, as described later in this section of the briefing. At the other extreme, trip time is seen as much less important than minimum mass and cost; conjunction profiles should be used. Crew time in zero g can be minimized by arrtificial-g spacecraft design. Increase in risk with duration is difficult to quantify. The mission duration issue presently is concerned mainly with cosmic ray exposure.

Crew radiation exposure comes from solar proton events (flares) and galactic cosmic rays, and from manmade sources if nuclear propulsion or power are used. Unshielded energy deposition from GCRs varies from 50 to 100 milligray (5 to 10 rad) per year. The low end of the unshielded range does not constrain Mars mission architectures, but the high end exceeds the present NCRP astronaut radiation guideline of 500 millisieverts/yr (this guideline is for space shuttle and space station missions; no guidelines have been given for Mars missions). It is possible that guidelines will be reduced in the future.

Five profile options are presented. Conjucntion fast transfer implies transfers much less than one year. Opposition/swingby trajectories vary from about 440 to about 550 days. Opposition/fast profiles imply 450 days or less, without swingby. The split sprint is a variation on the fast opposition profile in which the MEV and propellant for the return from Mars are sent in advance on a low-energy profile.

If galactic cosmic ray exposure must be controlled, we must either provide shielding on the transfer vehicle crew habitat or reduce exposure times. Shielding the transfer vehicle habitat dramatically increases its mass, requiring high performance propulsion such as nuclear, or favoring a cycler concept where massive habitats are emplaced on a suitable repeating trajectory and left there. To reduce exposure time, the applicable profiles are: (a) conjunction missions with fast transfers, i.e. less than 180 days, (b) fast opposition profiles, e.g. less than 1-year round trip, and (c) Mars surface rendezvous (Mars direct). The cycler/semi-cycler architectures offer shielding on the Earth-Mars leg, typically 5 months, and provides a 5-6 month conjunction transfer on the return trip. During the long stay at Mars, the crew must be on the surface most of the time unless a shielded Mars orbit habitat is also provided.

Fast-transfer conjunction missions may require orbit basing. A surface rendezvous mission may not be able to achieve the fast return transfer direct from Mars' surface with reasonable vehicle mass, because of the higher delta V required and because the payload launched from Mars' surface is the entire Earth return habitat rather than a lightweight, short-duration crew cab. Available propulsion options become very limited for fast missions. At one year, the only sensible options are NTR splits, where return propellant is prepositioned at Mars on a low-energy profile, or the use of a nuclear gas-core rocket. Below one year, the gas-core rocket quickly becomes the only option.

Top-Level Trade Table

Mission Profile	Propulsion				Basing	
	Cryo/ All-Prop	Cryo/ Aerobrake	NTR	NEP/ SEP	Orbit	Surface
Conjunction Minimum Energy	√	No advantage over propulsive capture	√	√	√	Later
Conjunction Fast Transfer	Excessive IMLEO	√	√	√	No. Reason for fast transfer is less GCR dose	√
Opposition/ Swingby	Same	√	√	Note 1	√	As a resupply mode
Opposition/ Fast	Same	Excessive IMLEO	√	Not able to make fast trips	√	Same
Opposition/ Split Sprint	Same	Same	√	Cargo only	√	Same

Note 1: NEP flies an opposition/swingby-like-profile but does not benefit from Venus swingby.

Architecture Results for Three Activity Levels

The top-level architecture selection results for the three activity levels are shown on the facing page. For the minimum program, a cryogenic expendable tandem-staged direct mode is the clear economic winner. Its lower development expense causes the operational cost savings for a reusable LOR system to have little payoff. At the median activity level, the reusable system gives about a 5% return on investment (ROI). Our baseline program included lunar oxygen at the median level, but the ROI is estimated only about 3%. At the high lunar activity level, reusable systems and lunar oxygen both have strong payoff, e.g. the lunar oxygen ROI is about 10%

The minimum Mars program is most economic with cryogenic all- propulsive expendable vehicles on conjunction profiles. The NTR has an ROI less than 2% at this level. If natural environment radiation concerns lead to a conjunction fast transfer or opposition profile, the NTR is the preferred solution with cryogenic/aerobraking as a backup. At the median level, the NTR has a 16% ROI versus cryo all-propulsive. Here also, aerobraking is a backup and SEP comes into the picture as a "dark horse", with about 10% ROI if array costs can be reduced to $100/watt, a tenfold reduction from present costs. At $500/watt, the SEP has a negative 10% ROI, showing the great leverage of array cost. At the high level, electric propulsion is indicated as important, but development costs are a problem unless low-cost SEP arrays can be produced. If electric propulsion costs are too high for a settlement-scale Mars program, the NTR/dash and Mars direct modes are viable options.

Architecture Results for Three Activity Levels

Minimum

Lunar:

Expendable

Mars:

- Cryogenic all-propulsive
- Unless radiation environment requires reduced trip times; then nuclear rocket or cryo aerobrake conjunction fast transfer

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Median (full science)

Lunar:

Start expendable, possible growth to LOR reusable, aerobraking

Mars:

- Nuclear rocket, conjunction, multiple landers
- Opposition or conjunction fast transfer options
- Cryo/aerobraking backup
- SEP "dark horse"

Industrialization /settlement

Lunar:

LOR crew and tandem direct cargo, reusable, with lunar oxygen

Mars:

- Early cryo/all-propulsive option
- Electric propulsion for sustained growth (probably SEP)
- Nuclear rocket/dash or Mars direct/Mars propellant, options for crew rotation and resupply.

Seven Architecture Recommendations

The next seven pages contain our main architecture recommendations with data illustrating key points.

Lunar Architecture

- <u>Begin the lunar program with a tandem-direct expendable system.</u>
  - System can be designed to eliminate on-orbit assembly; one docking or berthing required.
  - The number of development projects is minimized. Offers reasonable expectation of return to the Moon by 2004 under likely funding constraints.
  - Flight mechanics constraints for LOR operations are avoided.
  - Tandem-direct LTV is a starting point for evolution to all other identified lunar architectures.
  - Lunar aerobrake can be tested on the unmanned booster stage without risk to the crew. Stage is otherwise expended.

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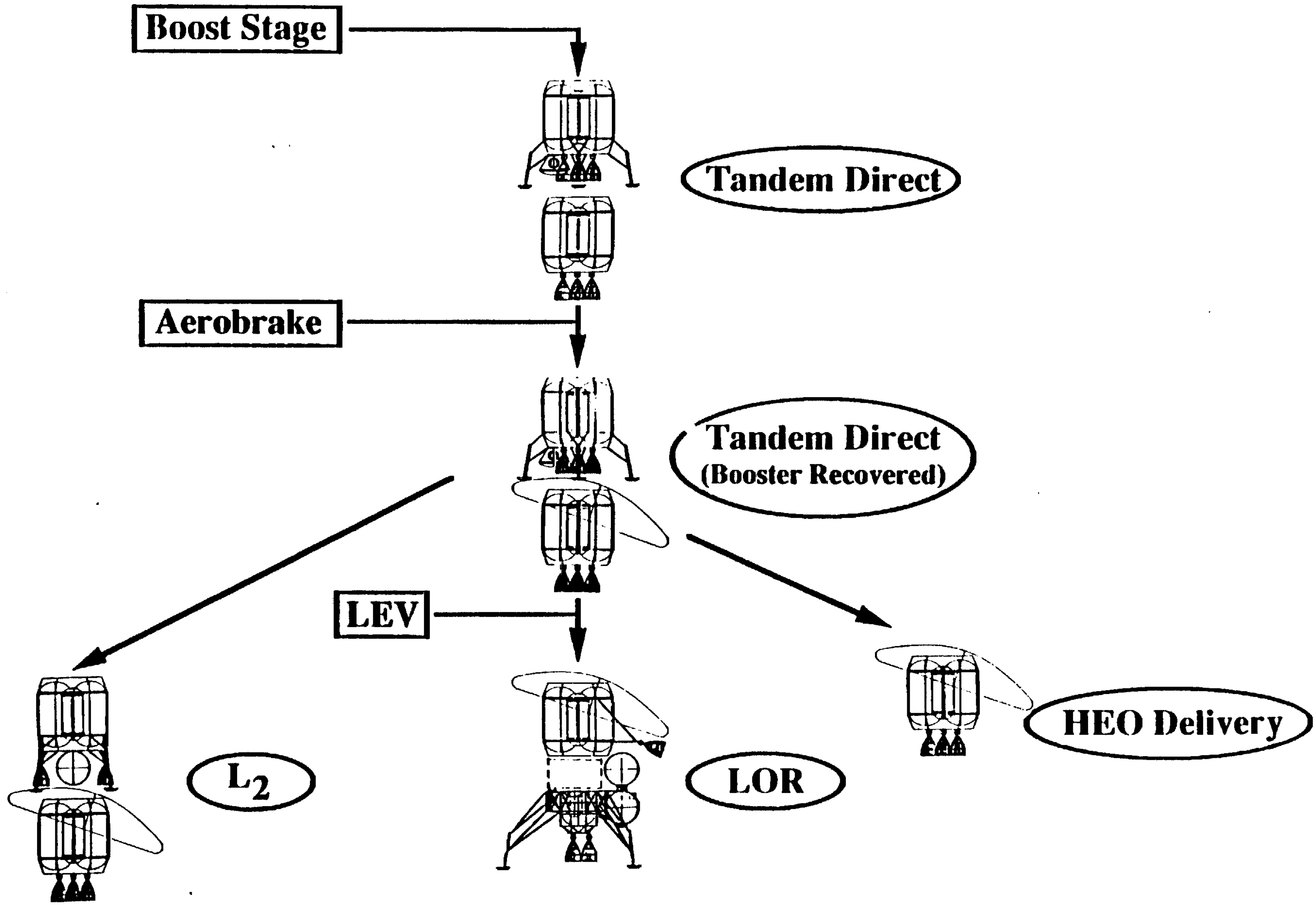
Lunar Transportation Family
Evolution



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Lunar Cryogenic Propulsion

- **Invest in cryogenic storage and management technology.**
  - Without advanced development of a low-boiloff flight-weight cryogenic insulation system, the lunar program may be forced to a storable propulsion system for lunar vicinity operations. Cost impact is billions of dollars.

- **Invest in a 30K-class advanced expander cryogenic engine with 10:1 or better throttling capability.**
  - An advanced expander engine offers about 20 seconds' Isp gain over a modified RL-10; can demonstrate advanced health monitoring and maintainability features essential for Mars missions.

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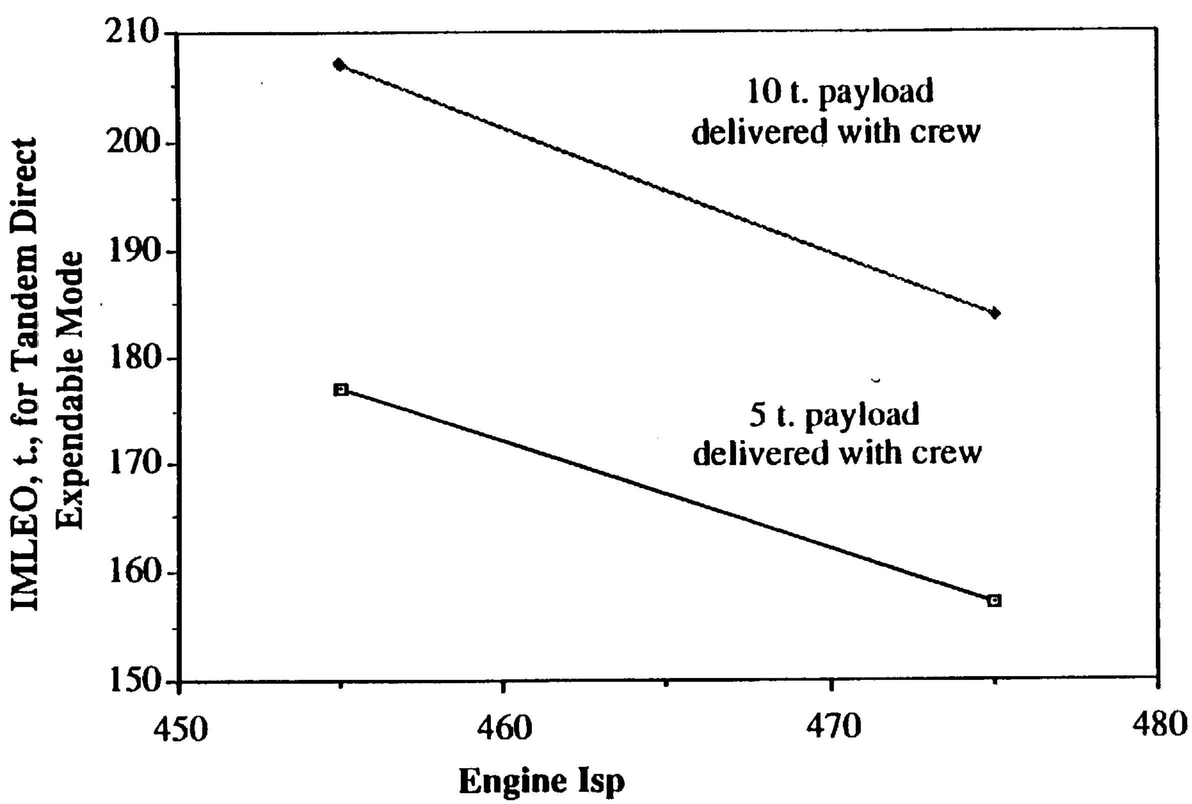
Early Lunar Mode Sensitivity to Isp



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Mars Baseline Architecture

- **Baseline nuclear thermal rocket propulsion for Mars.**
  - **Nuclear thermal rocket indicated as very economic and flexible over wide range of program activity levels.**
  - **Nuclear rocket vehicle mass is sensitive to specific impulse. Isp gain for carbide fuels is well worth the technology investment.**
  - **Development and qualification testing requires proven test facility technology that contains hydrogen effluent and scrubs radioactivity.**
  - **Nuclear rocket performance permits modest lunar program and significant Mars exploration with about six launches per year of 100-tonne class HLLV.**
  - **Nuclear rocket baseline offers reasonable expectation of initial Mars mission by 2010 under likely funding constraints.**

- **Recommended technology advancement program:**
  - **High-performance fuels**
  - **Full-containment ground test facilities.**

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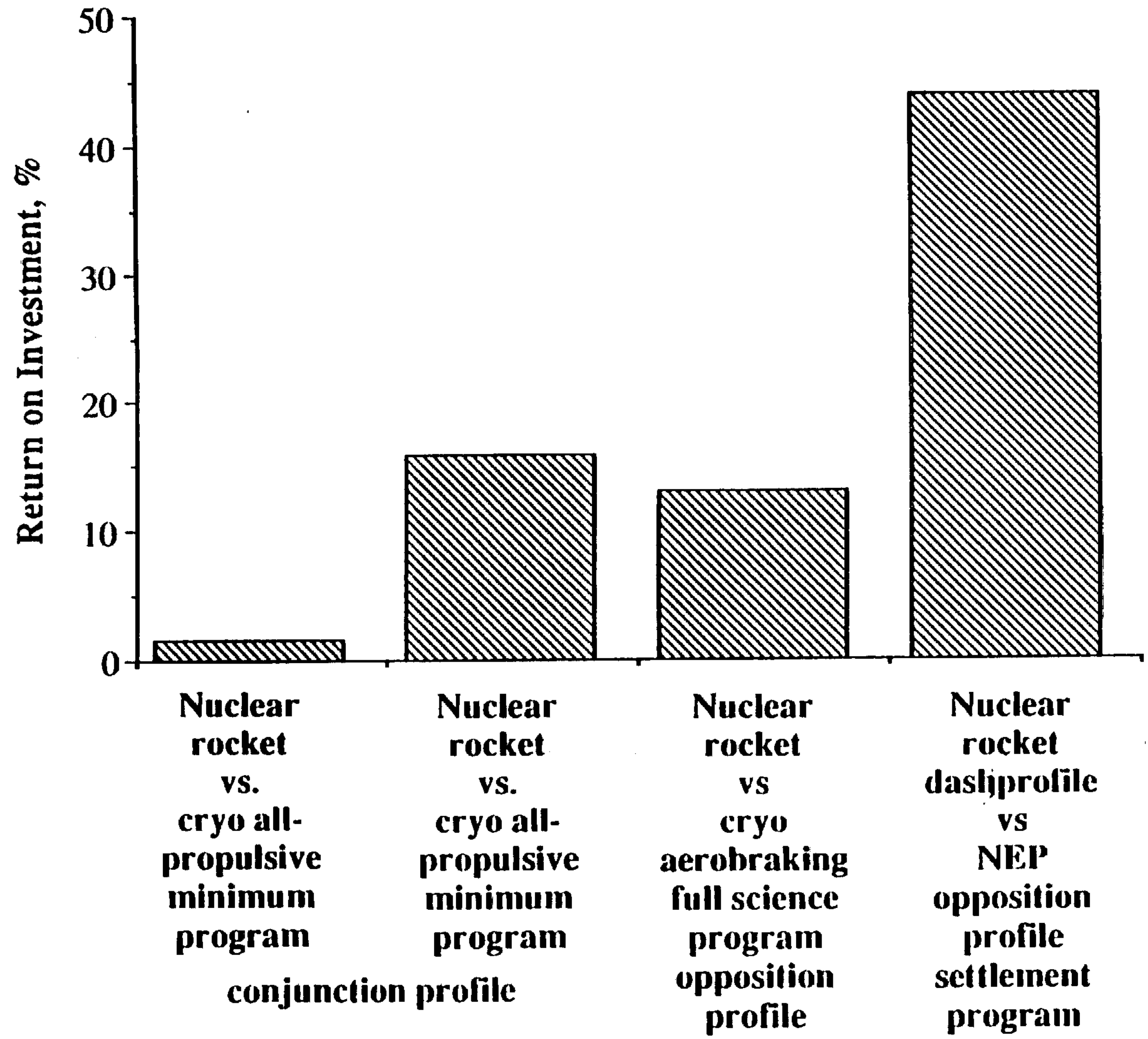
Nuclear Rocket ROI Trades



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Aerobraking Technology

- **<u>Accelerate aerobraking technology for Mars aerocapture as backup to nuclear rocket.</u>**

  - **Target decision between the two in the 1996-2000 time frame.**

  - **NTR performance and cost uncertainties, especially test facilities and testing, merit backup.**

  - **Aerobraking needed for Mars landing. Technology challenges less daunting than aerocapture, but merit technology program.**

  - **Aerobraking technology keeps other options open.**

    - **Conjunction fast transfer**
    - **Mars direct**
    - **Cycler orbits**
    - **NTR-dash profile**

  - **Aerobraking is economic for lunar transportation at >= two flights/year.**

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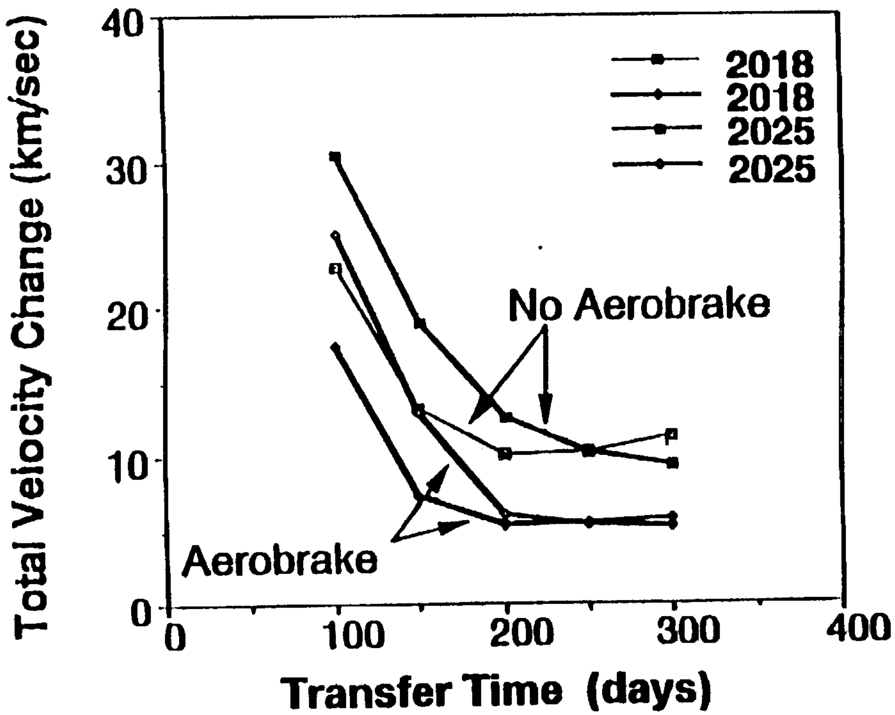
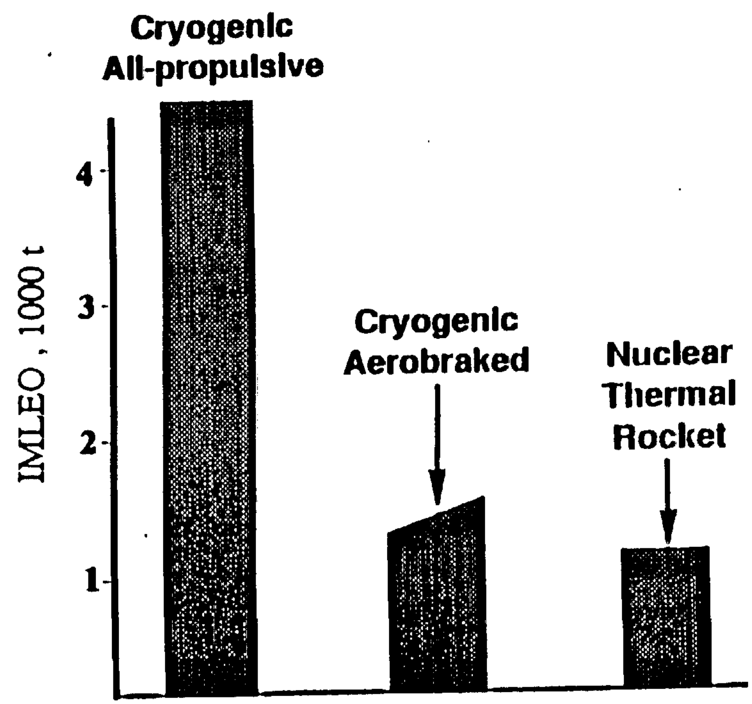
Mars Aerocapture Benefit for Conjunction Fast Transfer





Conjunction Fast Transfer

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Program Implementation Architectures Relation to Aerobraking

The facing page indicates uses of aerobraking for the various architectures. As noted, some form of aerobraking occurs in all of the architectures, in particular for Mars landing and for Earth capture on return from lunar missions. In addition, some of the architectures include an Earth crew capture vehicle (ECCV) for direct return of the crew to Earth in cases where, for example, an NEP or SEP vehicle must spiral back down to LEO or in the case of an NTR where the vehicle captures into a highly elliptic orbit.

Program Implementation Architectures

Architecture	Features	Aerobraking Functio: Mars cap	Mars land	Earth cap/ lunar	Earth cap/ Mars	Earth entry*
Cryogenic/aerobraking	Cryogenic chemical propulsion and aerobraking at Mars and Earth. LEO-based operations.	x	x	x	x	x
NEP	Nuclear-electric propulsion for Mars transfer; optionally for lunar cargo.		x	x		x
SEP	Solar electric propulsion for Mars transfer; optionally for lunar cargo.		x	x		x
NTR (nuclear rocket)	Nuclear rocket propulsion for Lunar and Mars transfer.		x	x		x
L2 Based cryogenic/ aerobraking	L2-based operations; optional use of lunar oxygen.	**	x	x	x	x
Direct cryogenic/ aerobraking	Combined MTV/MEV refuels at Mars and LEO. "Fast" conjunction profiles.	x	x	x	x	
Cycler orbits	Cycler orbit stations a la 1986 Space Commission report	***	x	x	x	x

Notes: * optional/emergency mode **opposition class only *** MEV-class crew taxi (not a large MTV)

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Aerobraking Flight Test Bed

- <u>Perform aerobrake tests on the LTV booster, to put the technology on the shelf for Mars.</u>

  - If the lunar program grows to high activity levels, lunar aerobrake is economically justified.
  - A space-assembled aerobrake is needed for Mars landing.
  - Aerocapture technology is needed as backup to Mars NTR.

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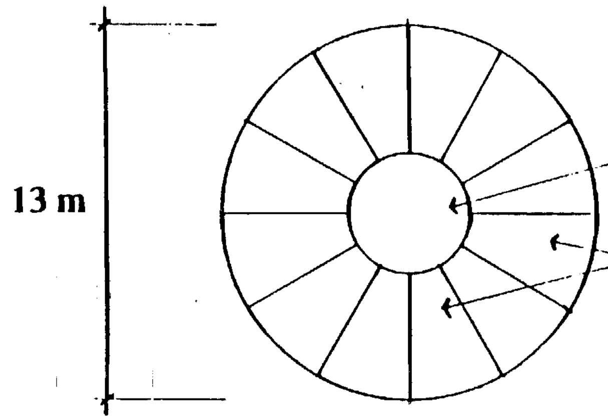
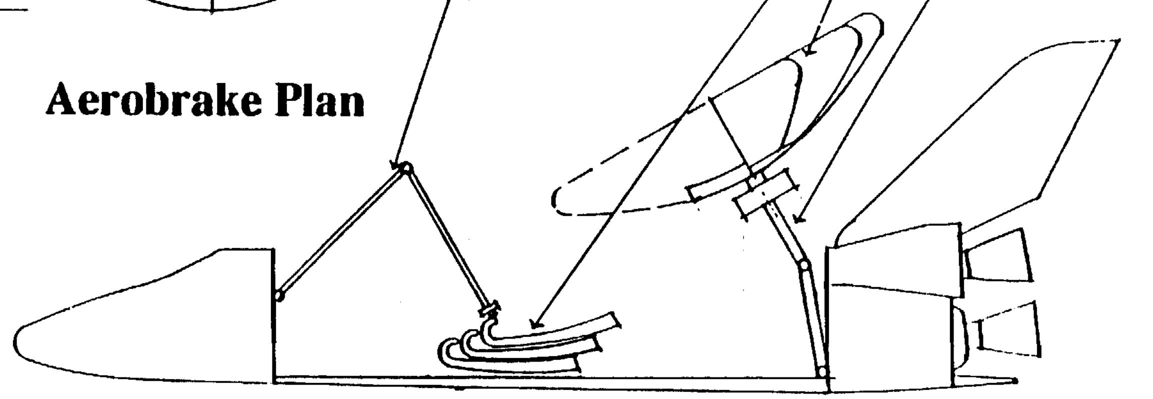
Aerobrake Assembly Test in LEO





Cross Section

Assembly arm rotates brake as outer panels are installed for easy RMS reach and crew visual contact during operations

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SEP as "Dark Horse"

- Designate solar-electric propulsion (SEP) as a "dark horse" for Mars transportation.
  - Technology advancement issues:
    - Light weight, high performance, radiation resistant arrays.
    - Automated production technology, $100/watt
    - Robotics technology for constructing SEP and deploying arrays
    - Long-life, high power density, efficient electric thrusters
  - If safety precludes operation of nuclear propulsion in low Earth orbit, SEP is the only option more economic than cryo-genic/aerobraking.
  - If low-cost array target achieved, SEP is more economic than NEP.
  - SEP is the most likely architecture for eventual private sector use for Mars settlement.
  - SEP technology has derivative benefits, e.g. power beaming to planet surfaces.

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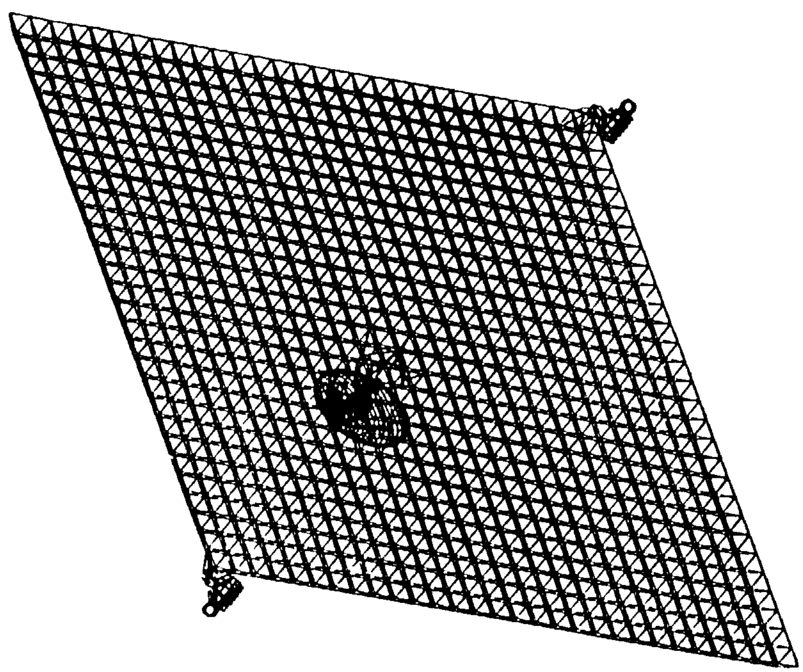
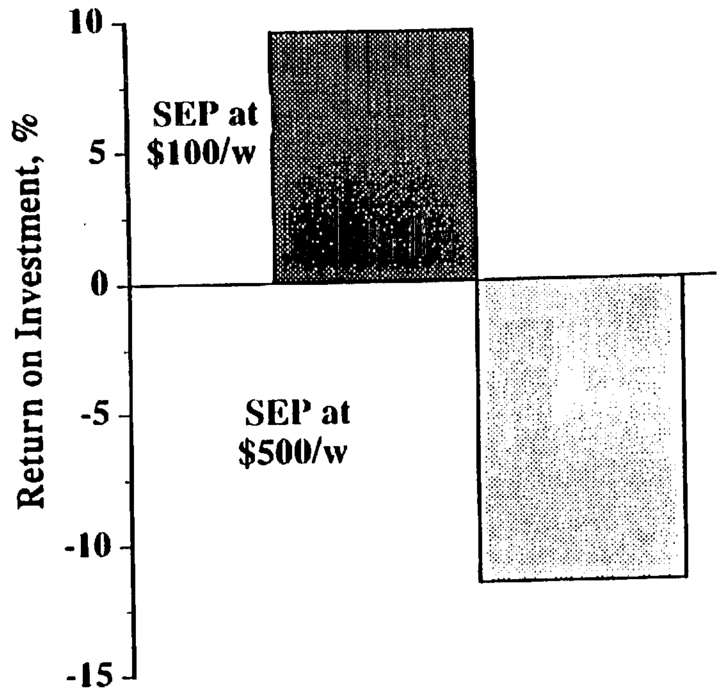
Return on Investment Comparison
SEP versus NTR; Full Science Program





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Nuclear Space Power

- <u>Continue the nuclear space power program towards near-term systems applicable to planet surface power.</u>
  - DDT&E and production cost estimates from this study eliminate nuclear electric propulsion (NEP) as a top contender, but are very preliminary.
  - As NEP systems are better understood, estimates may come down.
  - To keep NEP option open:
    - Further studies to better understand the cost of nuclear power systems suitable for electric propulsion.
    - Modest funding of high-leverage high-performance power conversion technology.

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Mission Risk Comparison

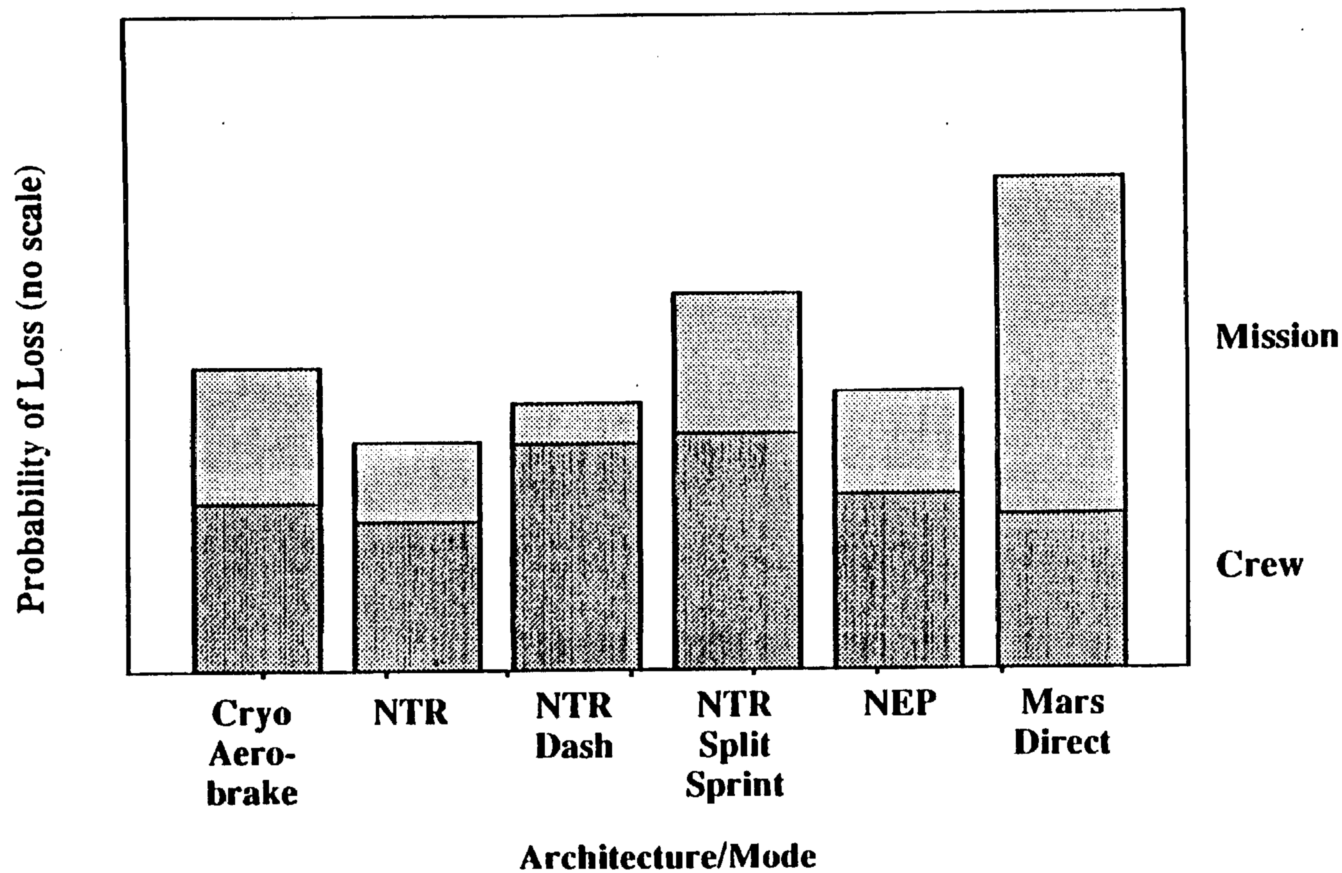
Mission risks were compared in a semi-quantitative way. The methodology is rigorous and quantitative, but reliability and safety estimates for SEI hardware and maneuvers are no more than ballpark guesses today. We made representative estimates with an attempt to be consistent, i.e. the same type of maneuver was given the same number for all cases. Plausible differences were used, e.g. aerocapture was judged higher risk than propulsive capture. Abort modes were included where available.

The facing page shows comparative risks for crew loss and mission loss for several architectures and modes.

NTR shows the least risk because of the 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 is deemed acceptable. 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.

Mission Risk Comparison

②



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Man Rating Requirements

The facing page describes our recommended approach to man-rating and lists the systems/subsystems for which we believe man-rating is required.

Man-Rating Requirements

Approach

- Ground-based testing wherever possible.
- Use flight program activities to bootstrap, e.g. lunar aerobrake program builds confidence in Mars aerobrakes.
- Flight demonstration of critical functions, e.g. Mars cargo landing, before critical manned use.
- Life demo for long-duration systems before critical manned use, e.g. ECLSS on SSF or lunar surface before manned Mars mission.

Subjects

- Aerobrakes
- Cryogenic rocket engines
- Nuclear rocket engines
- Cryogenic propellant systems
- Attitude control propulsion systems
- Nuclear & solar electric propulsion systems
- ECLSS/TCS
- Crew modules/hab systems
- Vehicle power
- Avionics & Communications systems
- Surface transportation systems

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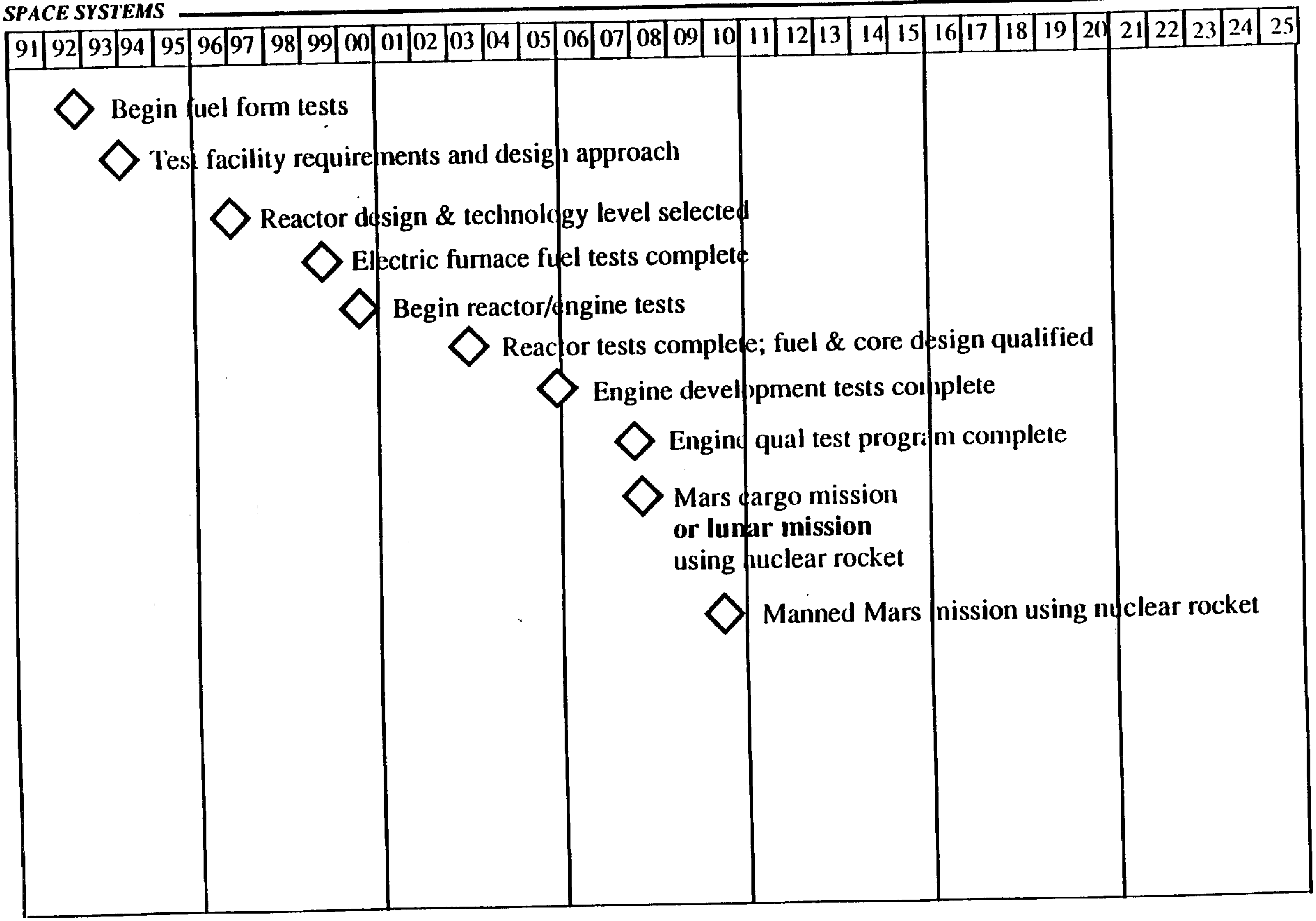
Nuclear Rocket Man-Rating Approach

A sequence of major tests and demonstrations to achieve nuclear rocket man-rating is shown. Note that two flight demonstration options exist. A decision of which to use depends on whether cargo delivery to Mars is needed before the first manned mission, as would be the case if a conjunction fast transfer and long surface stay is required on the first mission to reduce galactic cosmic ray exposure to the crew.

Nuclear Rocket Man-Rating Approach



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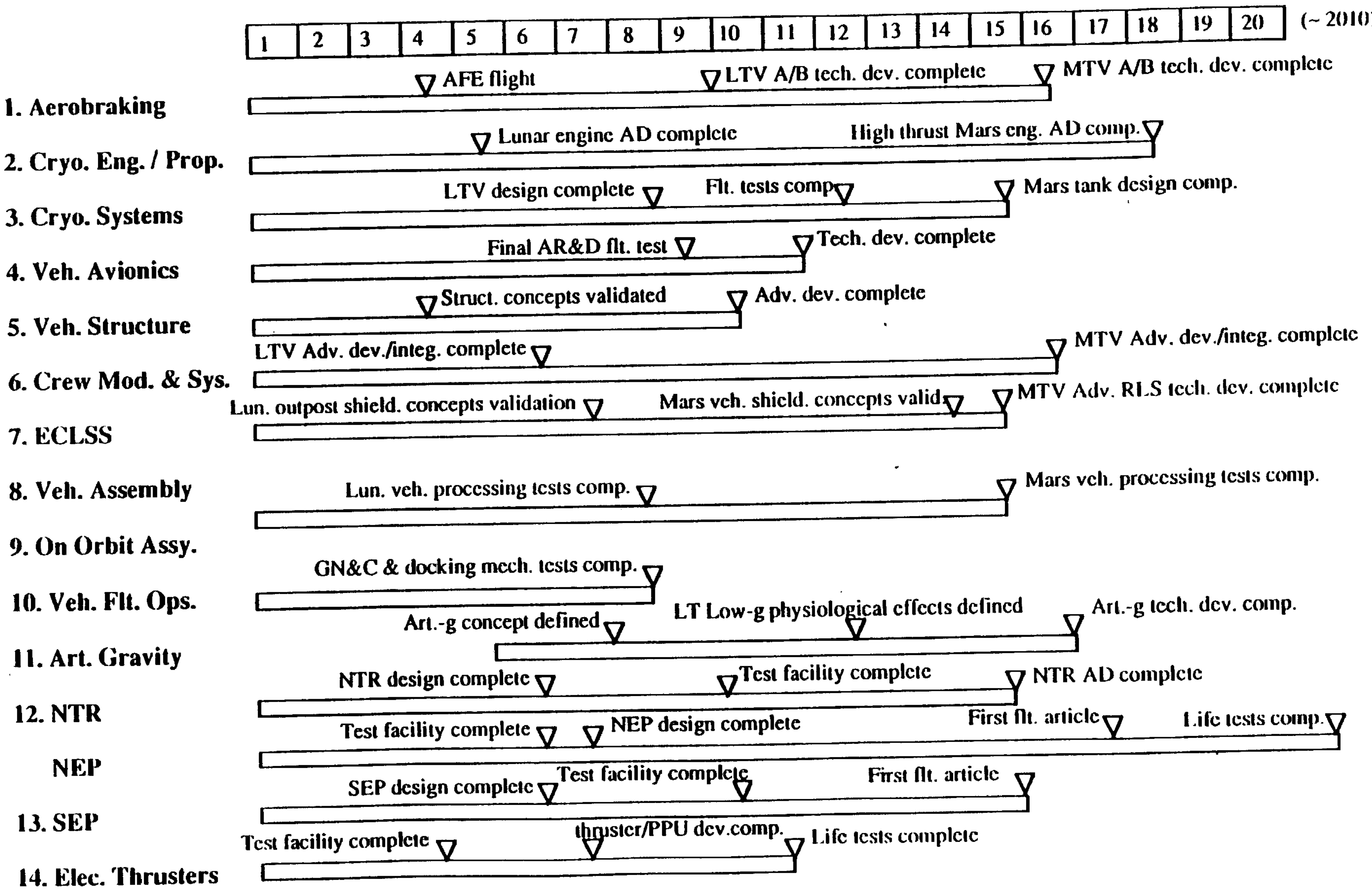
Technology Advancement and Advanced Development

The next three charts present our current recommendations for technology advancement and advanced development, with schedules and funding estimates. The funding level averages about $300 million per year. If we consider the median (full science) program as representative, the technology/advanced development program is about 0.2% of the life cycle cost of the program to 2025, a very modest investment.

Technology Development Schedules
- Overview -



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Technology / Advanced Development Funding Estimates

Technology Category	1	2	3	4	5	6	7	8	9	10	11	Total
1 - Aerobraking* -Technol.	1	6	10	5	5	8	10	10	8	5		68 M
- Adv. Dev.	0	0	30	30	55	20	30	65	65	40	30	400 M
2 - Cryogenic Engines / Prop.	0	30	30	30	20							110 M
- Adv. Dev.	0	0	99	71	65	50	65	65	50			465 M
3 - Cryogenic Systems - Tech.	5	5	5	5								20 M
- Adv. Dev.	0	10	10	20	50	50	50	110				300 M
4 - Vehicle Avionics/Software	2	5	5	5								17 M
- Adv. Dev.	0	0	0	25	45	40	40	40	40	40		270 M
5 - Vehicle Structures - Tech.	3	7	5	5	7	7	5					39 M
- Adv. Dev.	0	0	15	17	11	10	15	15	15	10		108 M
6 - Crew Modules & Systems	0	0	3	3	3	5	5	5	3			27 M
- Adv. Dev.	0	0	15	20	10	10	15	20	20	10		120 M
7 - Environ. Ctrl. & Life Supp.	0	0	3	5	5	10	10	5	5			43 M
- Adv. Dev.	0	0	6	10	10	15	30	30	30	20		151 M

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Technology / Advanced Development Funding Estimates

Technology Category	1	2	3	4	5	6	7	8	9	10	11	Total
8 - Vehicle Assembly - Tech.	5	5	5	5								20 M
- Adv. Dev.	0	5	40	40	40	40	40	40	10			255 M
9 - Orbit Launch & Checkout	5	5	5	5								20 M
- Adv. Dev.	0	4	15	16	5	10	10	10	10	5		85 M
10 - Vehicle Flight Operations - Adv. Dev.	0	0	9	15	10	15	15	15	10	5		94 M
11 - Artificial Gravity - Tech.	0	0	0	2	5	10	10	10	10	3		50 M
- Adv. Dev.												
12 - Nuclear Propulsion												
NTP -	0	10	15	20	20	20						85 M
NEP -	0	15	20	30	30	30	20	20				165 M
13 - Solar Electric Ion Prop.	2	8	10	15	15	10						60 M
Array manufac. Tech. -	0	0	0	30	30	30						90 M
14 - Electric Thrusters	0	5	10	20	20	20	10					85 M
Tech. Development Total	23	120	367	482	461	410	380	460	276	138	30	3147 M

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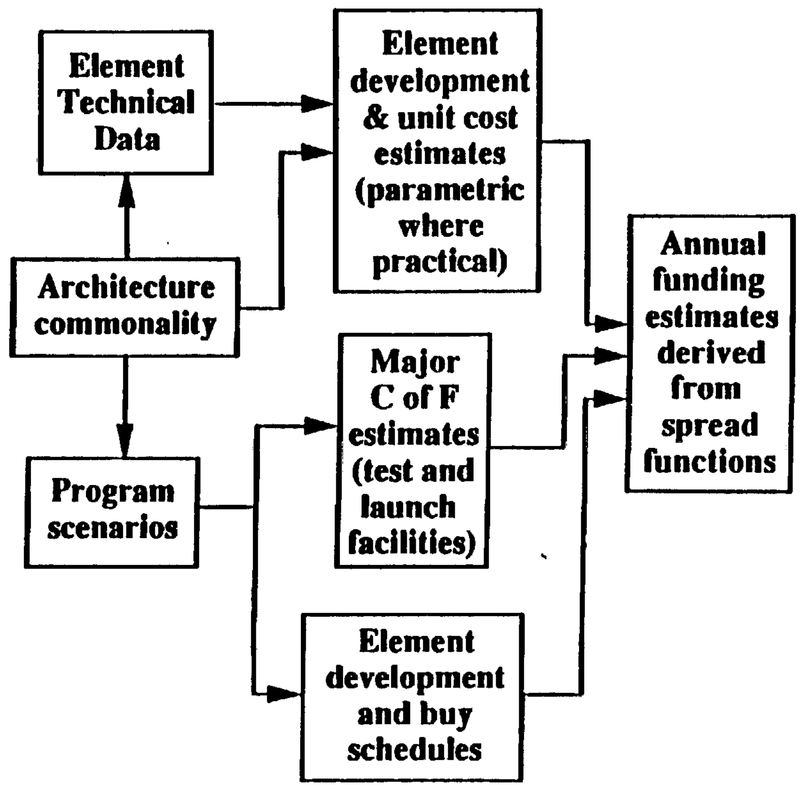
Life Cycle Cost Model Approach

Our basic cost model kernels are parametric cost models. We use the Boeing Parametric Cost Model and the RCA Price models to estimate development and unit cost. The determination of hardware to be costed comes from what architectural elements are needed and from element commonality of the architecture. Program schedules determine requirements and timing for major facilities and for the element development and buy schedules. All of these inputs are used to estimate annual funding for each component of the program, using cost spread functions. The costs are integrated into a spread sheet life cycle cost model to obtain annual funding for complete programs.

The ground rules used in this analysis are indicated on the chart.

The ground rule for use of closed ecological life support (CELSS) and lunar oxygen comes from economics trade studies conducted several years ago through last year.

Life Cycle Cost Model Approach



Ground Rules

- No precursor missions costed.
- NASA contingency not added
- Common element in new application gets 25% delta DDT&E cost.
- No production learning unless production rate > 1 per year.
- Production rates maintained minimum of 1 per 5 years to keep lines open.
- Mission definitions flexible to enable transportation systems to operate at high efficiency.
- All scenarios include closed ecological life support and ISRU for efficiency.

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Architectural Cost Drivers

Our investigations of architectures, while preliminary, indicate the importance of cost drivers, in the order listed on the chart. The number of development projects should be minimized through commonality and phased by evolution so that development costs are reduced and are spread over the life cycle of the program, rather than lumped early in the program.

Space hardware for SEI missions is expensive and should be reused if possible. As an example, our unit cost estimate for the Mars transfer crew module is more than a billion dollars. Reuse of this equipment motivates investment in the advanced transportation technology needed to make it reusable.

The third point is that Earth launch mass drives Earth launch cost. Even if Earth launch cost is reduced by ALS-class vehicles, the Earth launch cost is the largest single part of program cost.

The final point is that design and development of systems with mission and operation flexibility enhances commonality and minimizes the risk that changes in mission requirements force new developments or major changes.

Architecture Cost Drivers

- Number of development projects (minimize through commonality)
- System reuse (maximize)
- Earth launch mass (minimiz
- Mission and operational flexibility (maxi..ize)

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ADVANCED CIVIL SPACE SYSTEMS

In-Space Transportation DDT&E Comparison

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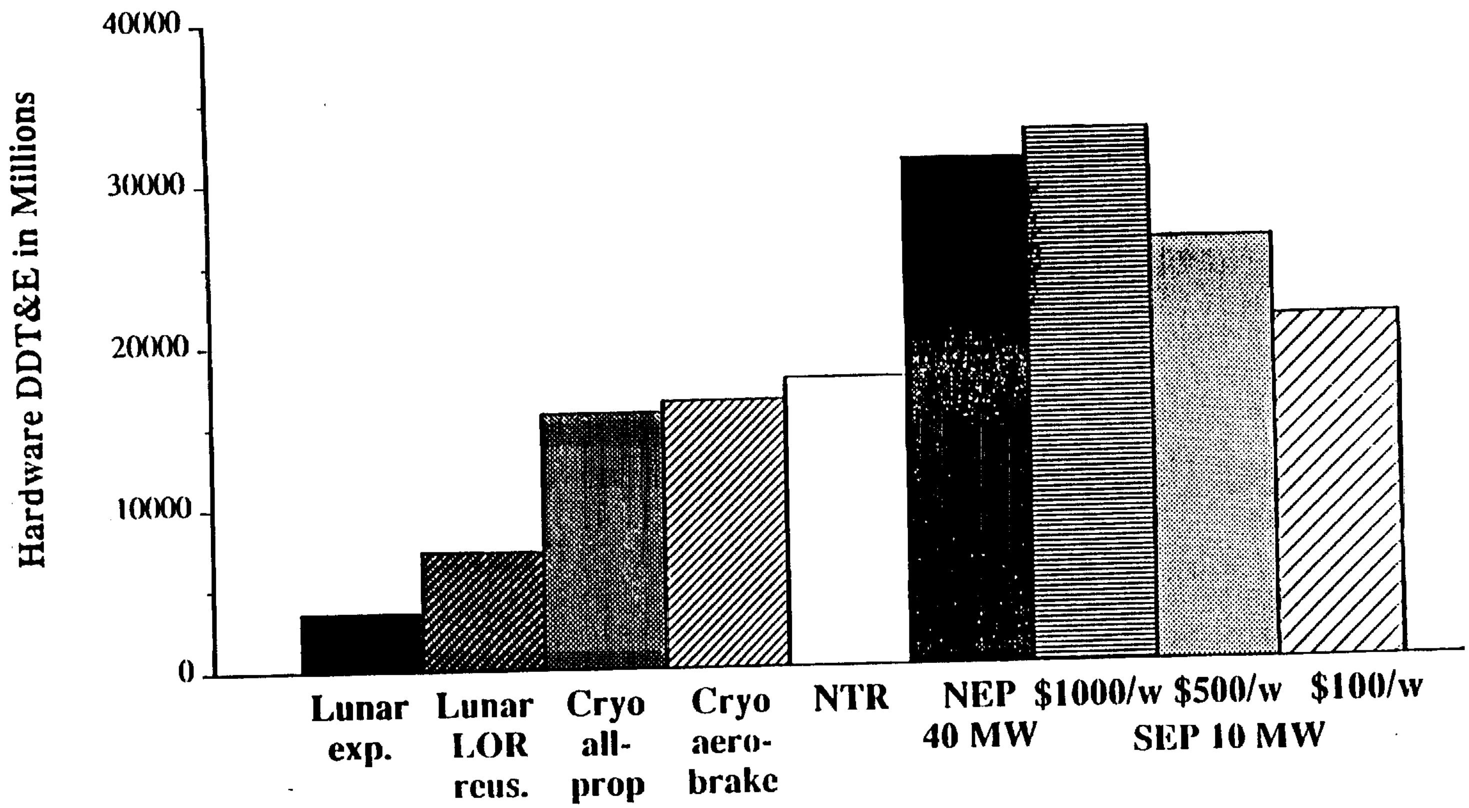

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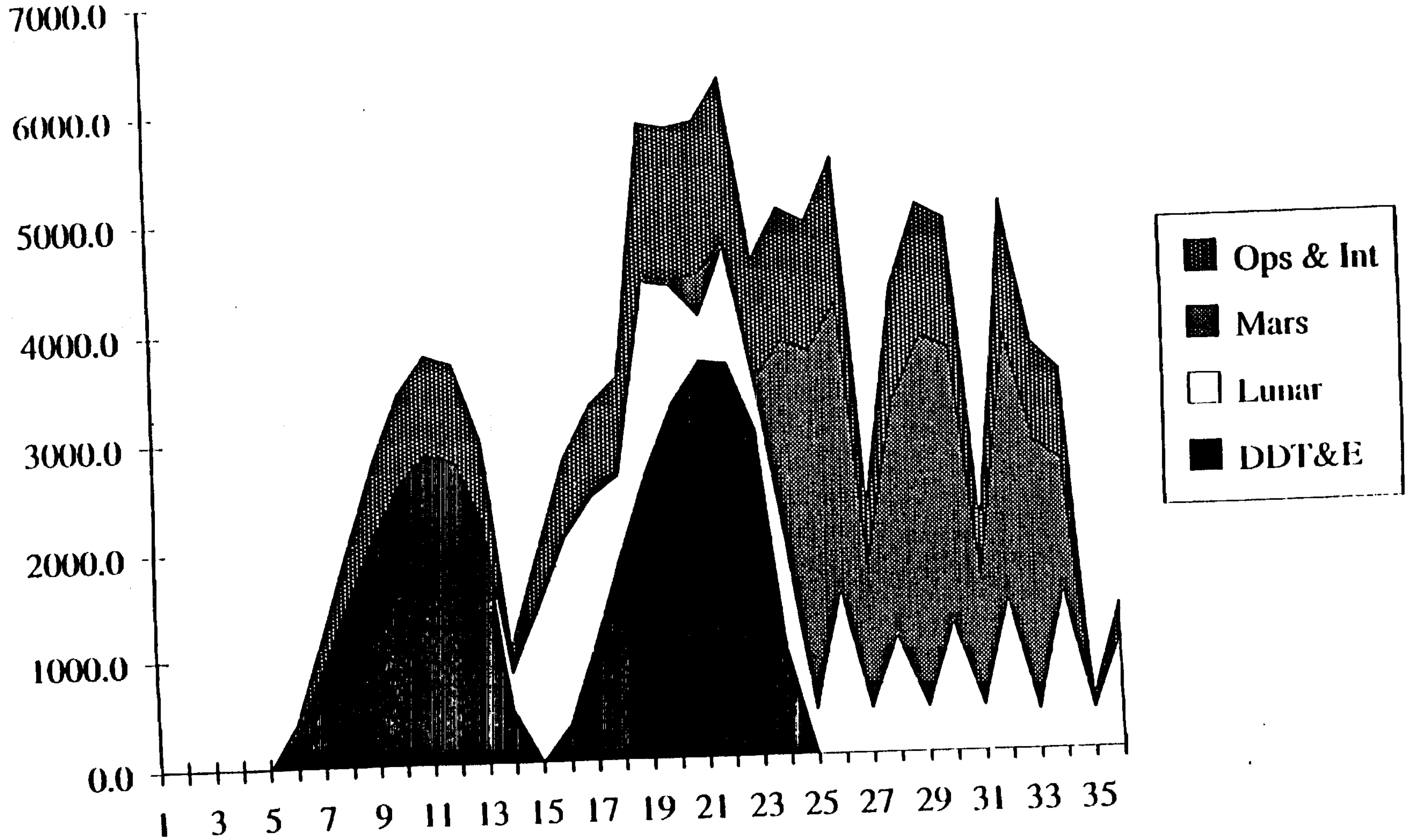
Minimum Program Life Cycle Cost Spread

The minimum program life cycle cost spread peaks between five and six billions per year. The deep valley between lunar and Mars peaks indicates that the Mars program should occur earlier in this program. The minimum program involves relatively modest investments in surface systems and falls well below the SEI funding wedge implied by the Augustine Committee recommendations.

Minimum (Baseline W/Ops & Int)



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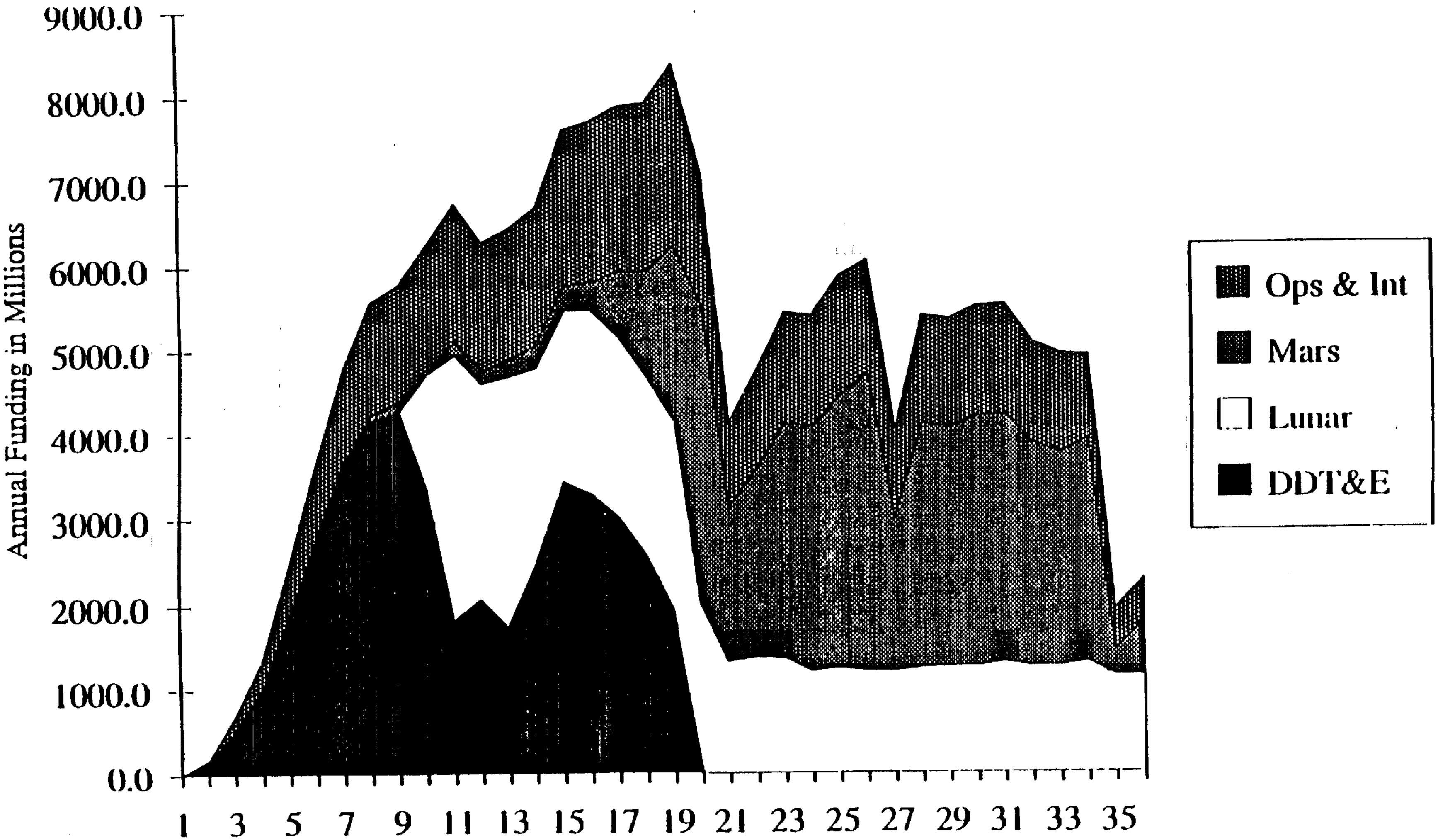
Median (Full Science) Program Life Cycle Cost Spread

The median life cycle cost spread peaks at about eight billions per year. With addition of likely surface systems costs, this program probably exceeds the Augustine guidelines during the peak years.

The median program exceeds by a factor of several the science and exploration potential of the minimum program. Lunar human presence grows from an occasional 45 days to permanent presence of six people, and Mars surface time grows from about four man- years to about 30. In other words, a roughly 50% increase in cost leads to about an order of magnitude increase in exploration and science potential.

C-2

Full Science (Baseline W/Ops Int)



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Median (Full Science) Program Life Cycle Cost Spread
Reduced Early Lunar Program

By deferring major lunar activities, the median program can be brought within the Augustine guidelines. Permanent human lunar presence is delayed until after the Mars DDT&E peak. The early lunar program is like the minimum scenario, i.e. man-tended astrophysics observatories.

Another way to level the funding profile for the median program is to defer Mars by a few years. The reference median program achieves a Mars landing in 2010 (2009 departure). Deferral to about 2016 would probably smooth out the funding profile much as did the reduction of the early lunar program.

Our view was that getting to Mars early was more important than an early buildup to permanent lunar presence. The partially deferred lunar program represented here still achieves astrophysical observatories early, but defers permanent human presence until after the major Mars mission DDT&E is complete.

ADVANCED CIVIL SPACE SYSTEMS

Full Science (Baseline W/Ops Int-Reduced Lunar)

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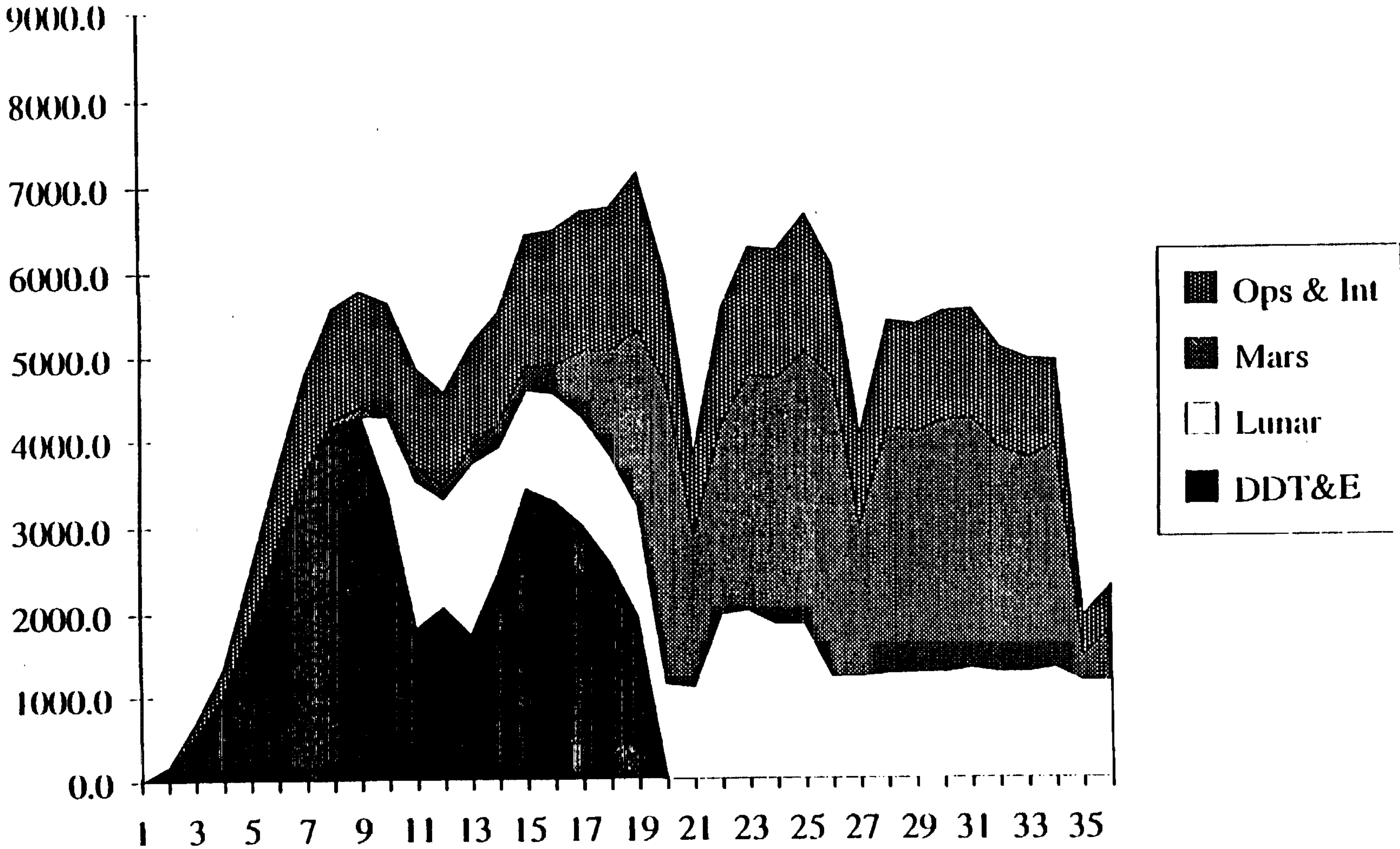

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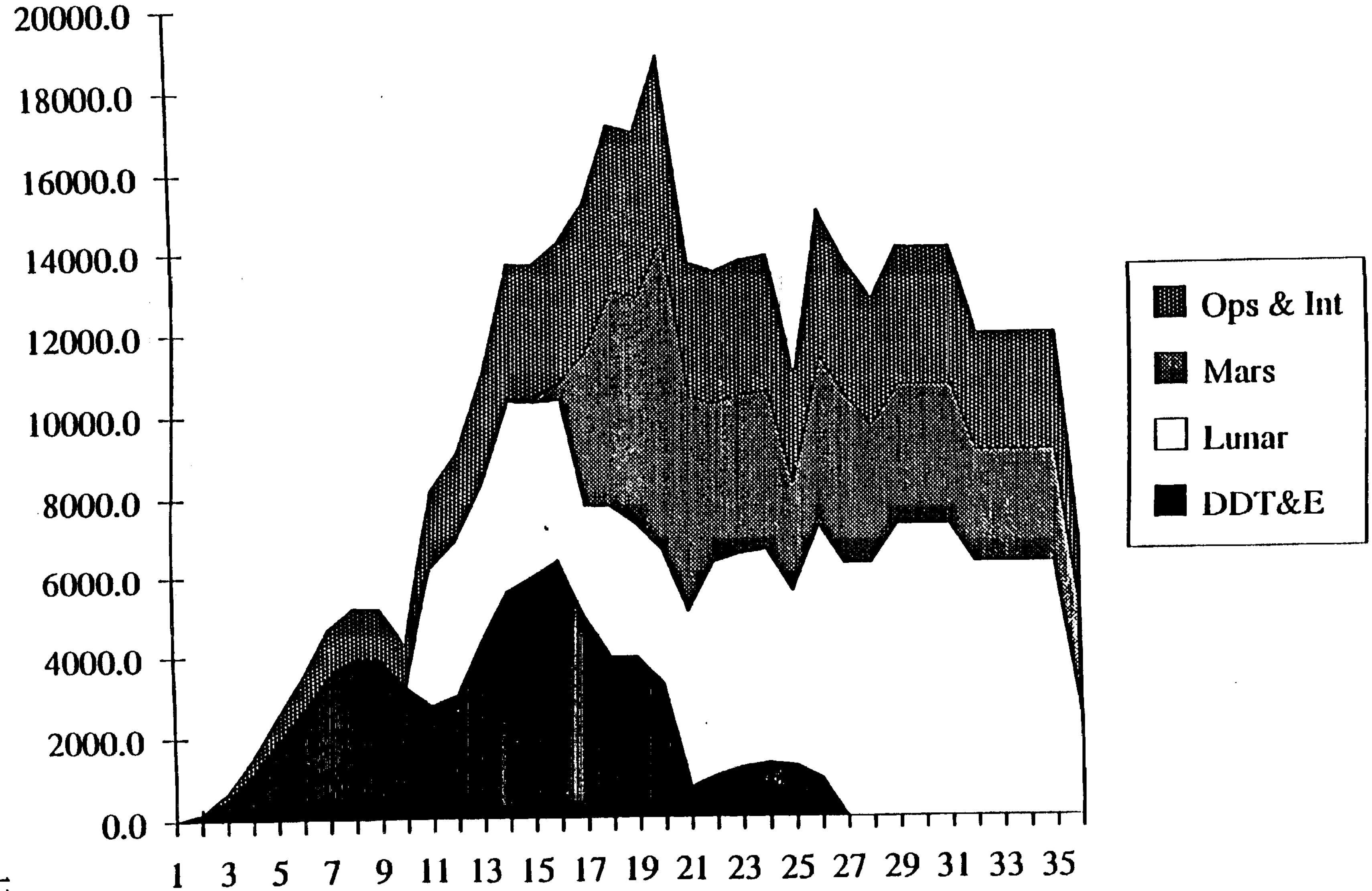
Industrialization and Settlement Cost Spreads

Our maximum scenario involved simultaneous industrialization of the Moon and progress towards settlement of Mars. As the cost spread shows, this is clearly beyond the funding levels recommended by the Augustine Commission. Both of the premises of this scenario, however, suggest significant private sector involvement.

What is significant in the result presented here is that investment on the order of $100 billions over about 20 years stretches from a plausible public-sector program of science and exploration to a program also involving the private sector for industrialization and settlement. This amount of funding is more than the private sector investment in the Alaska oil pipeline by a factor of a few, and probably less than the private investment in oil supertankers since the closure of the Suez Canal.

The economic potentials of lunar and/or Mars industrialization and settlement are presently not at all understood. We have made some stabs at estimating the costs. We have little or no idea as to the eventual payoffs.

Settlement/Industrialization Baseline w/Ops Int



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Results of Return on Investment Analyses

The facing page summarizes results of return on investment analyses. (The ROI methodology is explained in the technology and programmatics section of this briefing book.) Results designated "no ROI" had one case always more expensive than the other. An ROI can be calculated only when funding streams cross.

The storable case has very negative ROI because while less (i.e. no) technology money is spent, more vehicle stages must be developed so that the negative cost impact of not doing the essential cryo management and engine technology is large and early. 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 results were discussed earlier and are included here for completeness.

Return on Investment Analysis Summary

Case	Stor LOR vs cryo direct exp	Reus. LOR vs cryo direct exp	Reus. MEV vs exp MEV	SEP vs NTR $100/w	SEP vs NTR $500/w	SEP vs NEP	NTR vs cryo all-prop	NTR vs cryo all-prop	NTR vs cryo aero brake	NTR dash vs NEP	Lunar oxygen	Lunar oxygen
Program	Min	Full science	Ind/ settl	Full science	Full science	Any	Min	Full science	Full science	Ind/ settl	Full science	Ind/ settl
Result	-85	4.9	No ROI	9.6	-11	No ROI	1.7	15.9	13	44	4	10
Conclusion	Cryo	Reuse case weak	Reus MEV higher LCC	SEP	NTR	SEP better if less cost	CAP	NTR	NTR	NTR	No LLOX	LLOX

Strategy for Architecture Synthesis

The strategy we have adopted is illustrated on the facing page. First, we examined propulsion systems options through trade studies to understand how they work and to define preferred configuration operating modes. Secondly, based on the knowledge gained through these trade studies we chose a set of architectures using combinations of systems and modes, paying attention to integration compatibility, evolutions and commonality. Third, we will compare and trade architectures over a range of scopes and obtain important sensitivities and understand how architectures respond to program scope. We expect this analysis to lead to preferred architectures for various scopes. The final step is to conduct trades within the winning architectures to make further improvements.

All of this is guided by knowledge of the architecture cost drivers described earlier and by the knowledge gained on how systems work together, from the trades conducted within individual propulsion systems.

Strategy for Architecture Synthesis

Conduct trades on systems to find preferred configurations and operating modes → **Synthesize candidate architectures using effective combinations of systems and modes** → **Compare and trade architectures over a range of program scopes to obtain important trends and sensitivities** → **Conduct trades within winning architectures to improve them**

Criteria:	Criteria:	Criteria:	Criteria:
Risk Performance Cost Flexibility	Integration compatibility Evolution Commonality of hardware & technology	Ability to satisfy program goals & schedules Cost	Risk Performance Cost Flexibility

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Architectures Synthesis vs Mission/System Analysis

The facing page compares this approach to the traditional top-down systems engineering approach. The traditional approach shown on the right, starts with program goals, establishes mission requirements through trades, and continues to lower levels. As usually conducted, the traditional approach is faced with the great number of possible combinations noted earlier. The usual outcome is that requirement decisions are made and systems selected without trade studies.

The synthesis technique, on the left, attempts to avoid this problem by a combined top down/bottom up approach. It is similar to a classical optimization problem.

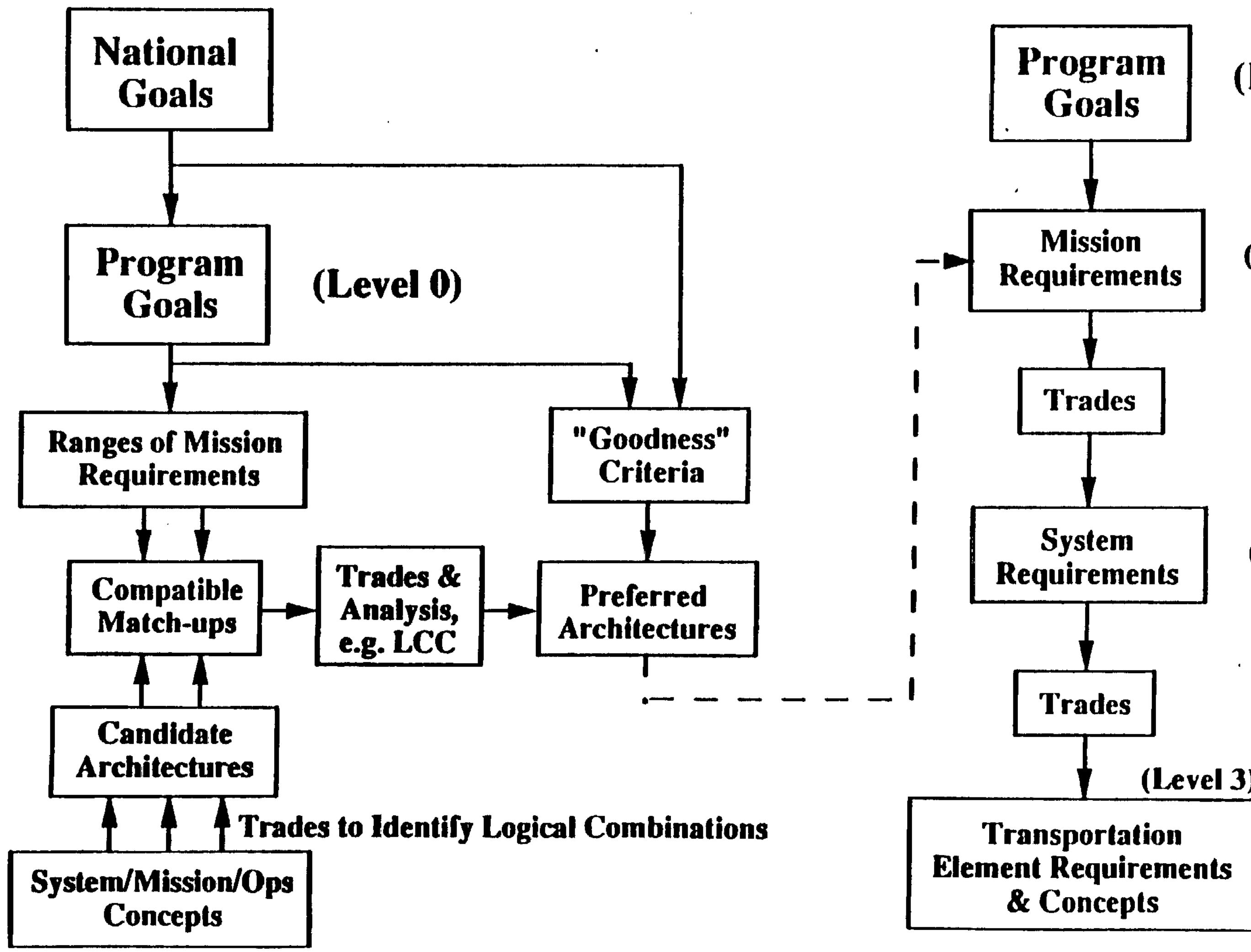
Optimization deals with infinite numbers of paths that satisfy boundary conditions. Optimization is a technique for generating only optimal paths. Any path that satisfies the boundary conditions is the sought optimal path.

Nothing quite as rigorous can be done in architecture synthesis. However, by bottom up trades, assembling systems into "good" candidate architectures, and matching with ranges of program scope, we may come close. The key is knowledge we obtain on what works well what things are compatible and combine well to satisfy mission requirements.

The last step is to conduct trades and analyses such as life cycle cost to identify preferred architectures, apply criteria derived from national goals program goals, to select among preferred architectures.

The dotted line indicates that one could then enter the traditional analysis flow with preferred architectures and their associated requirements and mission profiles, to further refine systems through systems engineering.

Architecture Synthesis vs. Mission/System Analysis



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Architecture Trade Flow

The facing page shows the low level system mission and operations trades that have been conducted or are being conducted for our seven architectures to represent the range of possible architectures for the SEI mission. Most of the trade areas have been presented in this briefing or have been presented in earlier briefings. The knowledge base in this area is fairly complete except that only very preliminary analyses have been done for the cryogenic direct mode and for cycler orbits. When these two options are completed we will be ready to finish up the architecture analysis.

Architecture Trade Flow

Cryo/Aero-braking	Nuclear Electric (NEP)	Solar Electric (SEP)	Nuclear Thermal Rocket (NTR)	L2/Lunar Oxygen	Cryo/Aero braking Direct	Cycler Orbits

• Mission design
• Reuse
• Aerobrake
  • shape
  • heating
  • GN&C
  • structures
  • assembly
• All-propulsive conj. option
• Modularity & commonality
	• Mission design
	  • trip time
	  • gravity assist
	  • node location
	• Power cycle
	• Power level
	• Specific power
	• Redundancy mgmt.
		• Mission design
		  • trip time
		  • gravity assist
		  • node location
		• Solar cell type
		• Power level
		• Specific power
		• Assembly/deployment of large space structure
			• Mission design
			• Isp and T/W sensitivity
			• Reuse
			  • tanks
			  • engines
			  • core stage
				• All-propulsive conj. option
				• Lunar oxygen benefits
				• Integration of lunar & Mars ops.
				• Advanced propulsion for LEO-L2 operations
					• Performance vs. separate MTV/MEV
					• Sensitivity to propellant choice
						• Mission design
						• Feasibility of high Mars encounter velocities
						• Design of "taxis"
						• Operational integration

For all: Overall configuration; key subsystems performance; integration compatibility; operations analyses

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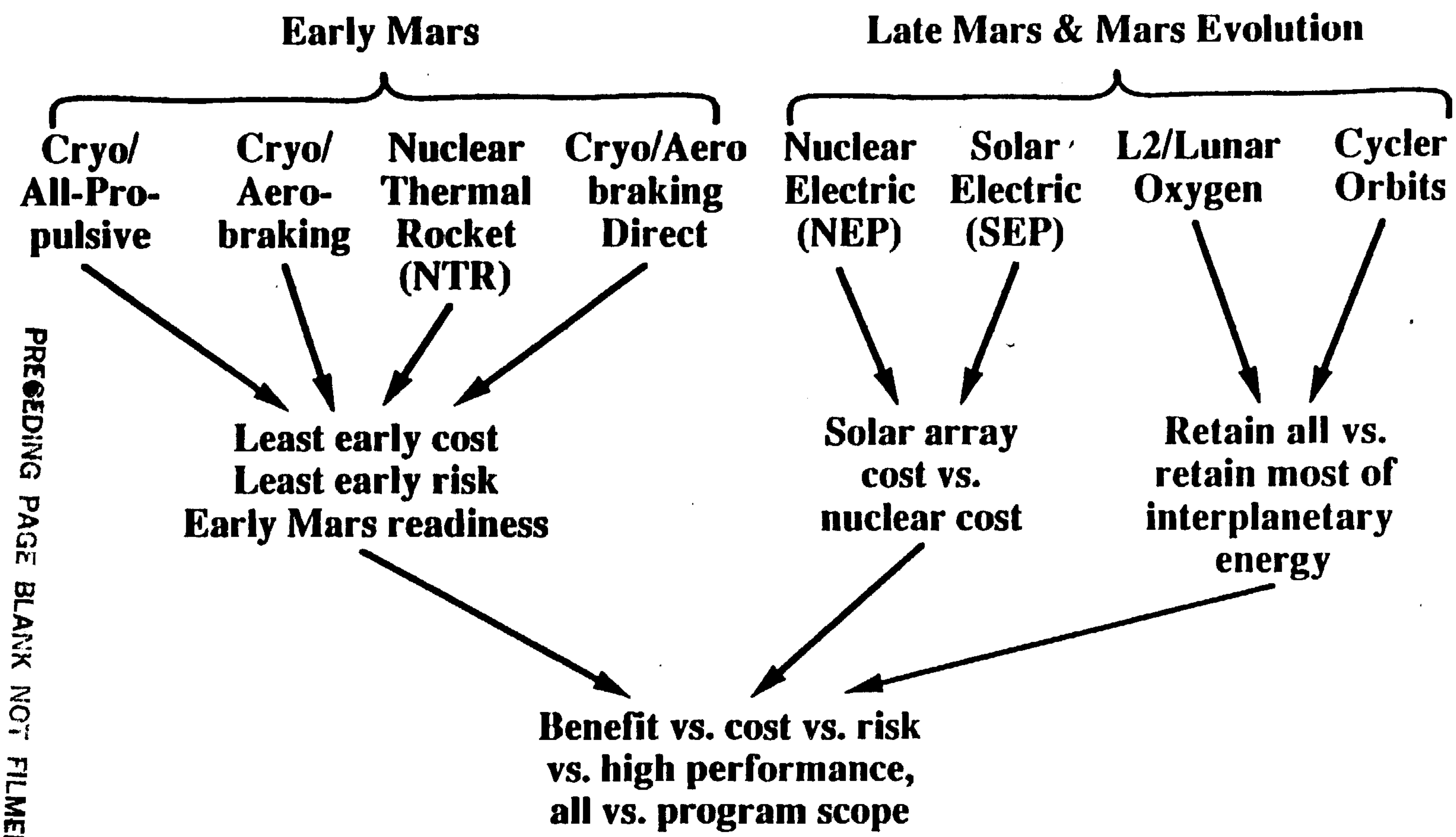
Architecture Evaluation Approach



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Mars Summary

- More than 20 beneficial modes identified.

- Early Mars: Cryo all-propulsive (CAP), ECCV*, conjunction;
  NTR all-propulsive, conjunction or opposition;
  Cryo aerobraking opposition, ECCV;
  (possibly) Direct with Mars oxygen.

- High performance, late Mars or evolution:
  SEP or NEP;
  ISRU, moon or Mars or both;
  Combintations.

- Efficiency range 10:1 measured as RMLEO (resupply mass LEO).

- Reusable MEV/Mars propellant has significant leverage for high-performance options.

* Earth Crew Capture Vehicle, an Apollo-like capsule used for Earth entry and landing or aerocapture to LEO. The rest of the vehicle is expended.

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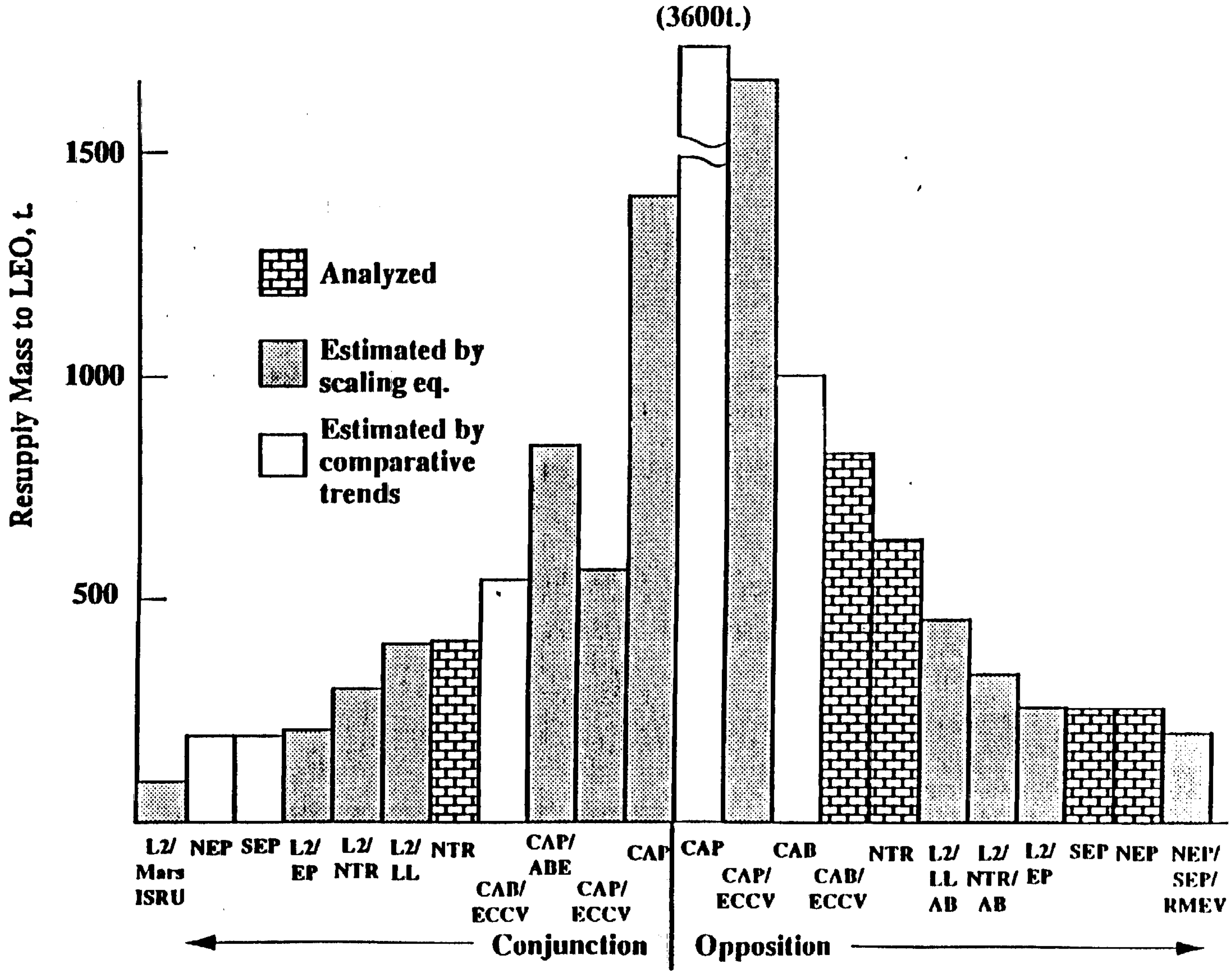
Comparison of Propulsion Options



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Conjunction vs. Opposition Mars Profiles

Opposition Advantages

- Shorter overall trip time, by at least a year.
- Transfer vehicle usually returns in time to be reused on next opportunity.
- Enables crew rotation/resupply mode with synodic period stay time.

Conjunction Advantages

- Lower energy; significantly less RMLEO unless very high Isp available.
- Venus swingby complexity not necessary.
- Long stay times at Mars.
- Shorter transfer times.
- Elliptic parking orbits can be optimized.

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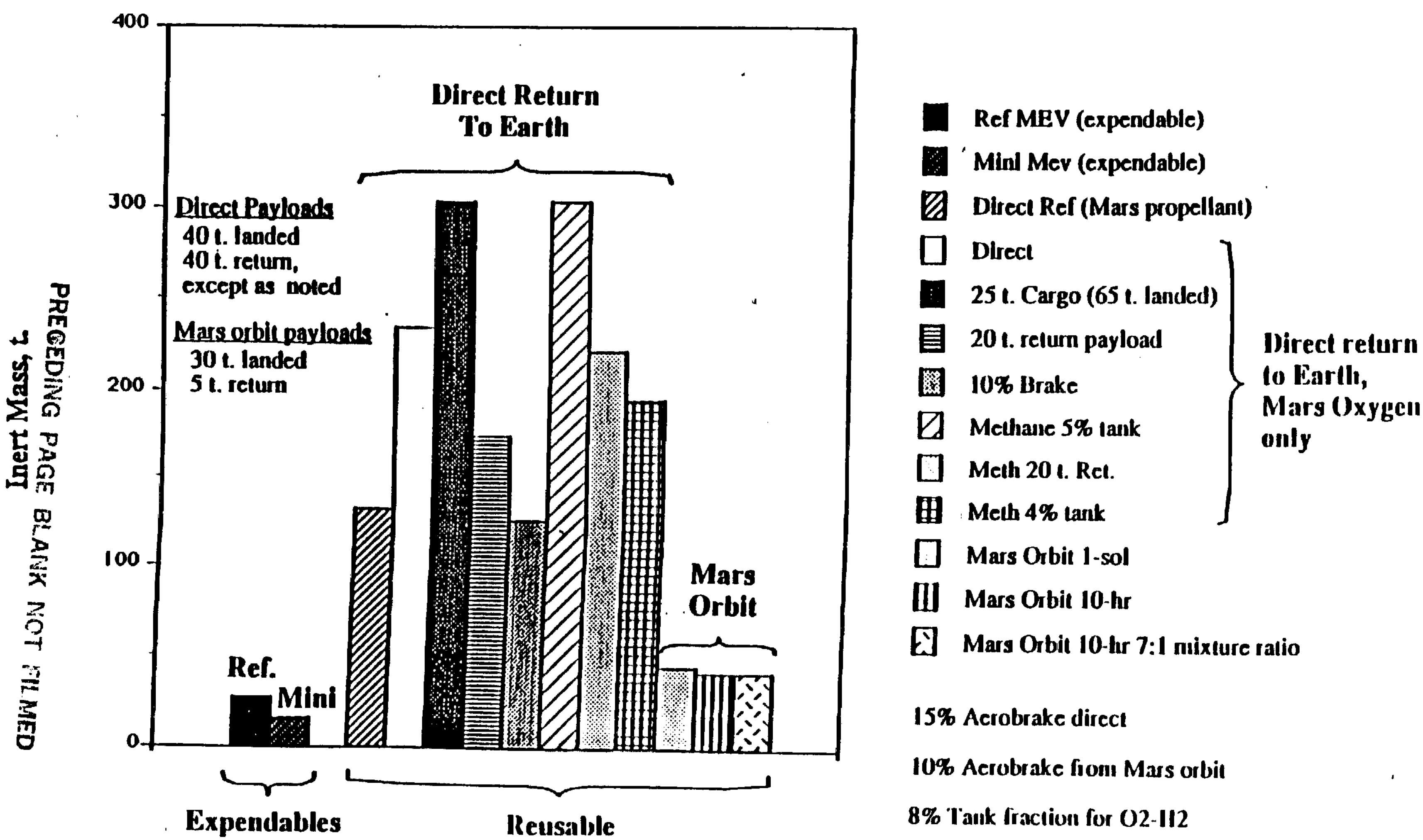
Reusable MEV Sensitivities



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II. Requirements, Guidelines and Assumptions

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Reference and Alternate Missions

Note: Contains material formerly in Mission Analysis

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SEP Mission Analysis

Contained within this section are the following:

- Groundrules for the Mars SEP study
- Propulsion option comparison assumptions
- Propulsion option comparison
- SEP mission profile schematic
- Low thrust mission analysis methodology
- SEP performance parameters
- Trajectory optimization
- Earth gravity well spiral analysis
- Transfer array trade results
- Gravity assist definitions
- Gravity assist results
- Optimum low thrust round trip Earth-Mars mission and system design parameters: Byrd Tucker,SRS,Dec. 27, 1990

Our initial objective for SEP mission analysis was to determine an optimum power level and Isp for a range of projected vehicle alpha's. This information was used to develop a vehicle concept of that class. The results of our initial analysis showed that a vehicle alpha of 10 kg/kW would have an optimum power level in the 10 MWe range. This power level would permit manned trip times that were competitive with chemical propulsion for and assumed 121 t payload. Previous SEP Mars mission studies were primarily aimed at lower power levels because electric propulsion was thought of as a cargo carrier only. Our analysis, in conjunction with the other propulsion option analysis, showed that SEP is a serious contender for manned Mars missions. As time progresses a more detailed vehicle will be developed, allowing more accurate analysis to be performed. Further analysis will still reveal solutions that are in the same class as current analysis. Since vehicle alpha's play such an important role in vehicle performance, this technology area shouldbe given serious attention early in the development program.

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) remains the same, resulting in a higher vehicle mass. 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 to 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 is (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.

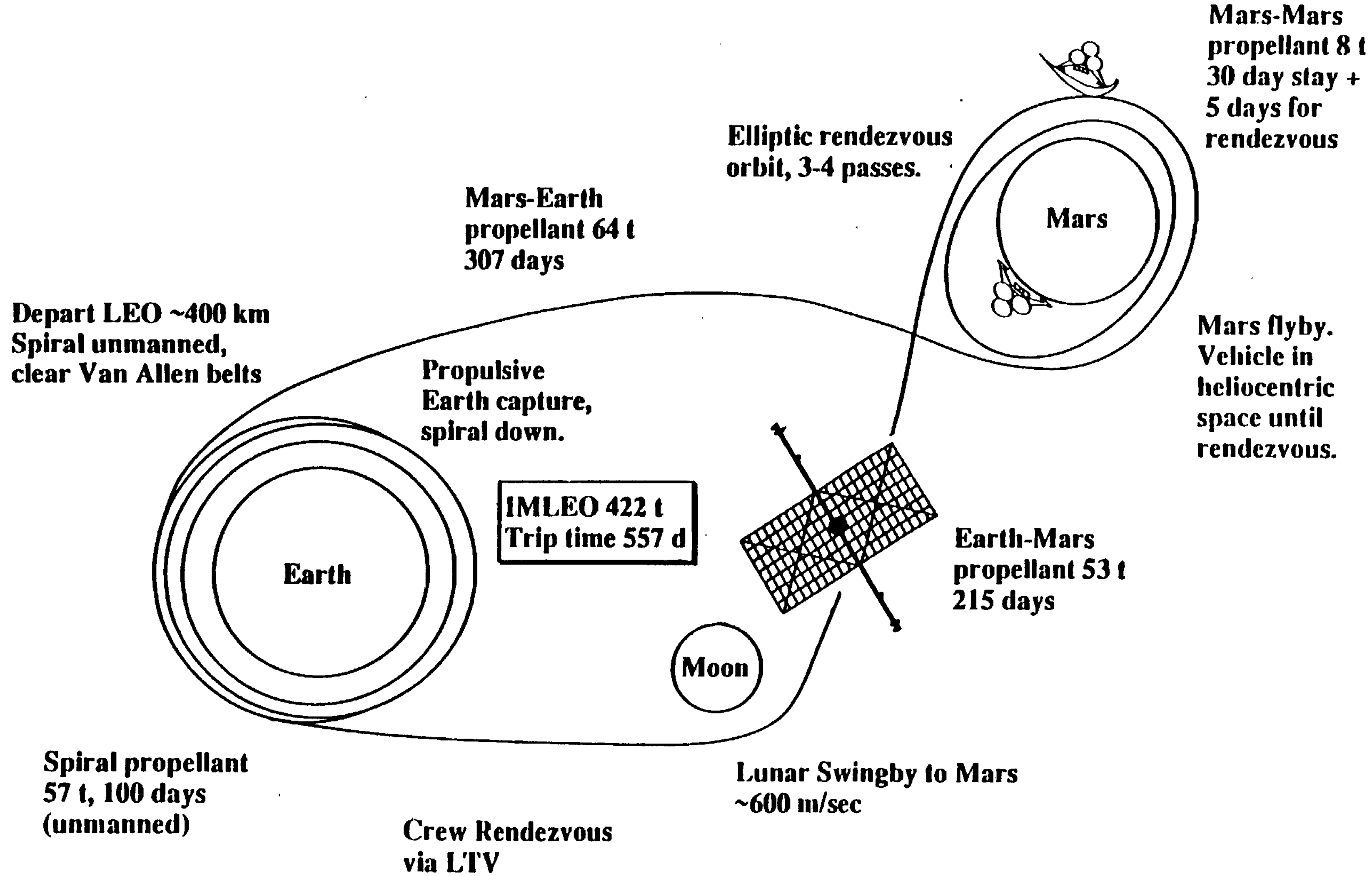
SEP Mission Profile Schematic

We have developed network diagrams for each of the transportation systems under study, to illustrate the main features of their operations. The SEP begins its mission with spiral from low Earth orbit to near the orbit of the Moon's orbit. This spiral takes about one hundred days as noted on the chart. As the SEP reaches the Moon, a crew mission using an LTV meets with the SEP and boards the crew.

A few days thereafter the SEP encounters the Moon with a swing by which gives it a velocity boost on the order of 600m/sec. The SEP encounters Mars dispatches the crew for landing at the first Mars intercept, and uses a Mars fly by with elliptic orbits to settle into a orbit suitable the crew return after their Mars stay. At this time the SEP departs Mars returns to earth. The crew (a) leaves the SEP, by earth crew capture vehicle or (b) is picked up by LTV with SEP at lunar distance.

With this mission profile, the actual crew transfer times exclude spiral time at Earth. Most of the spiral time at Mars occurs while crew preforms the surface mission on Mars.

SEP Mission Profile Schematic



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LOW THRUST ROUND-TRIP EARTH-MARS MISSION SIMULATION

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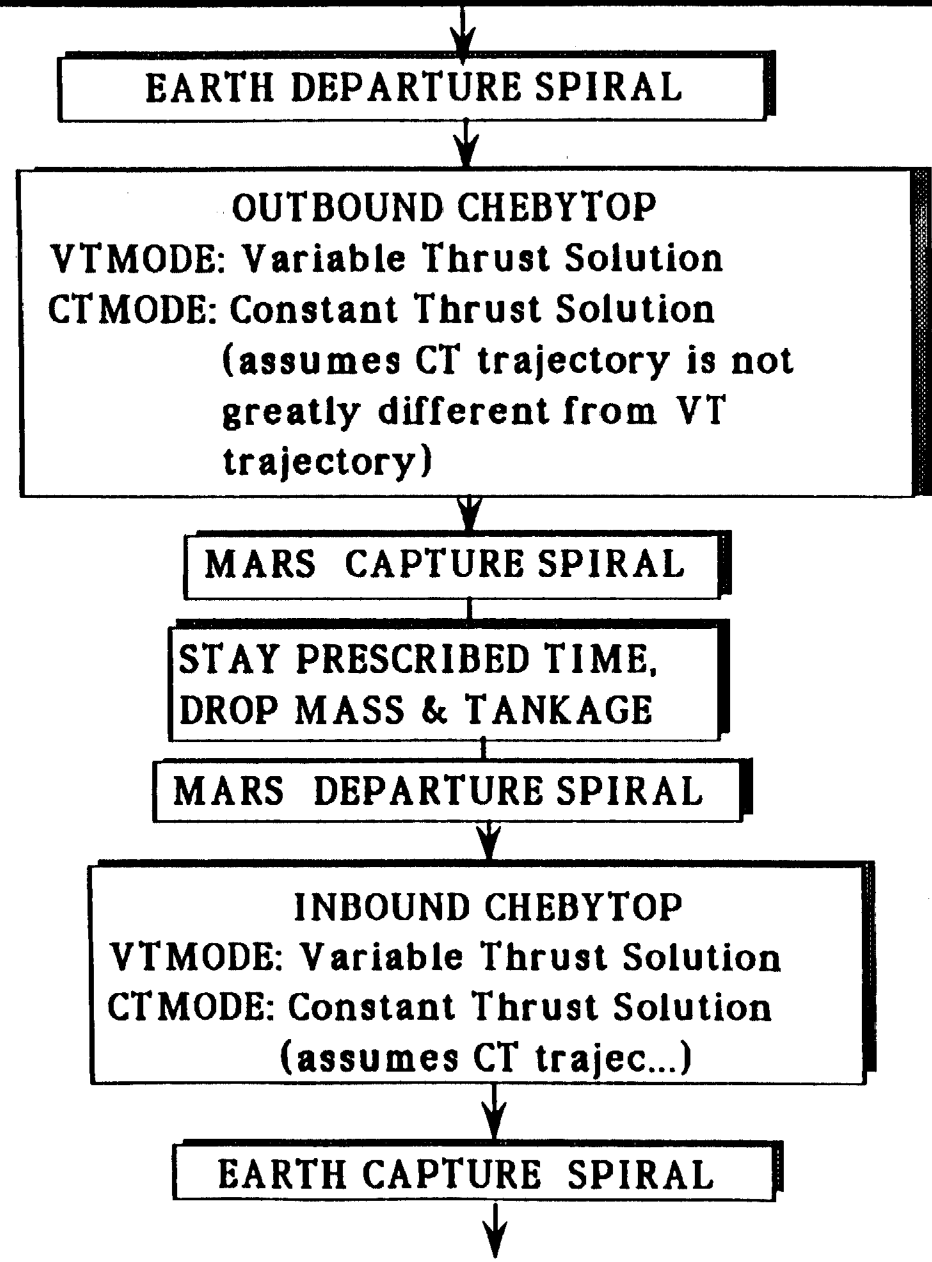

LOW THRUST SYSTEM MODELLING

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NEP: Po = Constant

ETA = -.082668 + 2.6251e-4* Isp - 3.087e-8*Isp**2 + 1.8047e-12*Isp**3 - 4.3169e-17*Isp**4

SEP: P/Po = (1.763 - .8865/R + .0592/R**2)/[R**2(1 - .1171*R + .0528*R**2)]

ETA = 80.193*Isp**2/(96.04*Isp**2 + 5.067e8)

$J = \int_0^T a^2 \, dt$ (trajectory optimization parameter)

$\frac{1}{m_f} = \frac{1}{m_o} + \frac{J}{2 \eta P_o}$ (mass related to trajectory parameters)

$m_{ps} = a P_o$ (power system mass)

$c = g_o I_{sp}$ (exhaust velocity)

$\eta = \eta(I_{sp})$ (thruster efficiency)

$a_o = \frac{2\eta P_o}{c m_o}$ (initial acceleration)

$m_p = m_o - m_f$ (propellant mass)

$m_{tr} = k m_p$ (tankage and reserves)

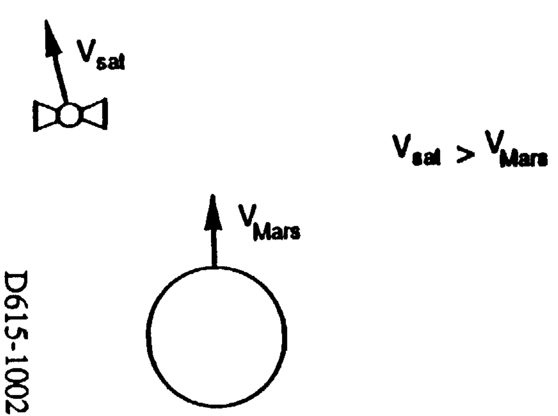
$m_{pl} = m_o - (1+k)m_p - m_{ps}$ (payload mass)

Mars Flyby: Inertial Reference Frame

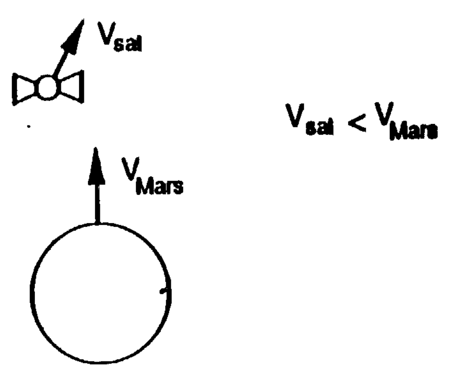
The Mars flyby is shown for a SEP vehicle (Vsat) in the inertial reference frame. Before the flyby, the SEP vehicle is travelling slower than Mars. During the flyby, the vehicle flies in front of the planet, then past the planet allowing the planet to pass the vehicle. When the planet passes the vehicle (approximately 30 days), the vehicle flies past the planet, picking up a gravity boost, therefore reducing the trip time. During this scenario the vehicle does not spiral about the planet as in the reference case.

Mars Flyby: Inertial Reference Frame

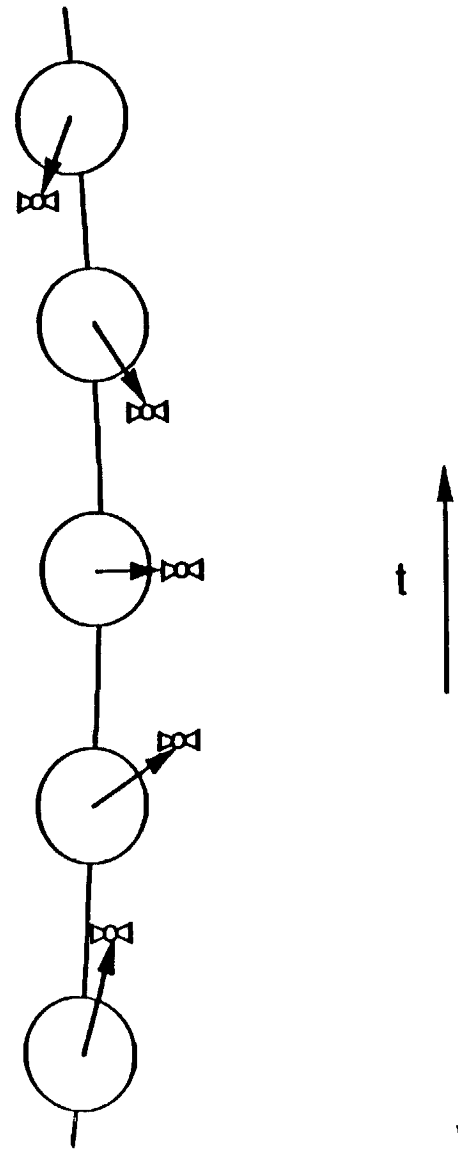
After Flyby



Before Flyby



During Flyby

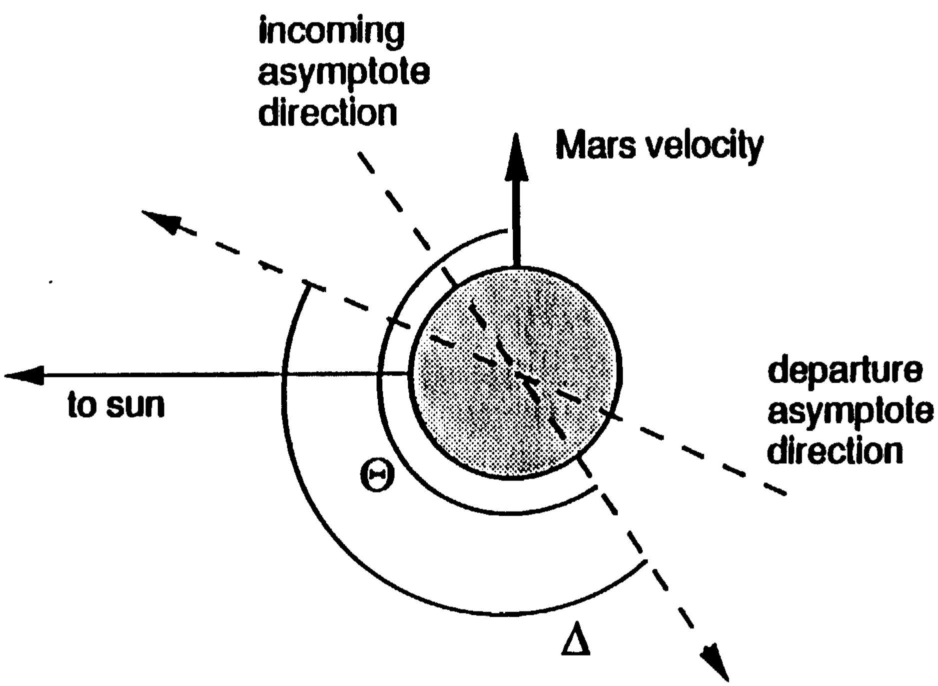


Mars Flyby: Mars Reference Frame

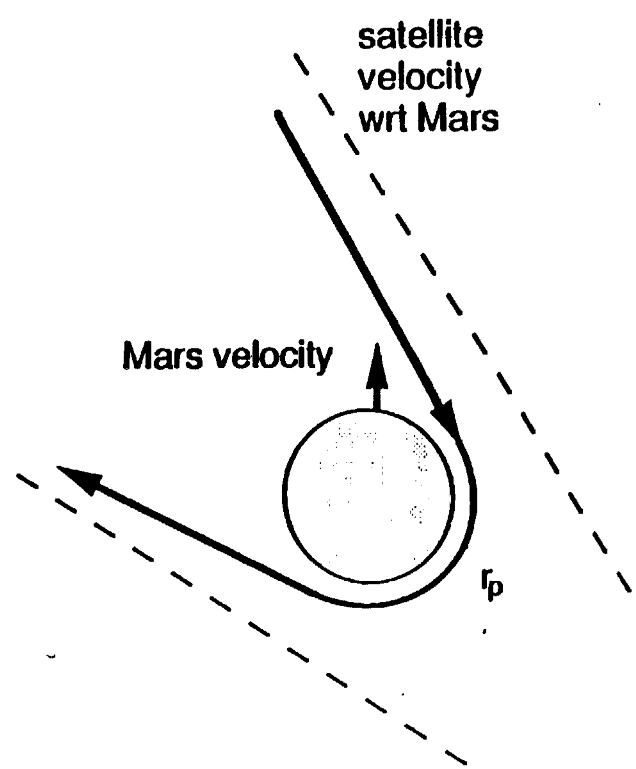
The following figures depict the Mars flyby from the Mars reference frame. The views show what the SEP vehicle would appear to be doing from the surface of Mars. During the 30 day stay time, the SEP vehicle would not enter an orbit about Mars, but would appear to be almost stationary.

Mars Flyby: Mars Reference Frame

Flyby Parameters



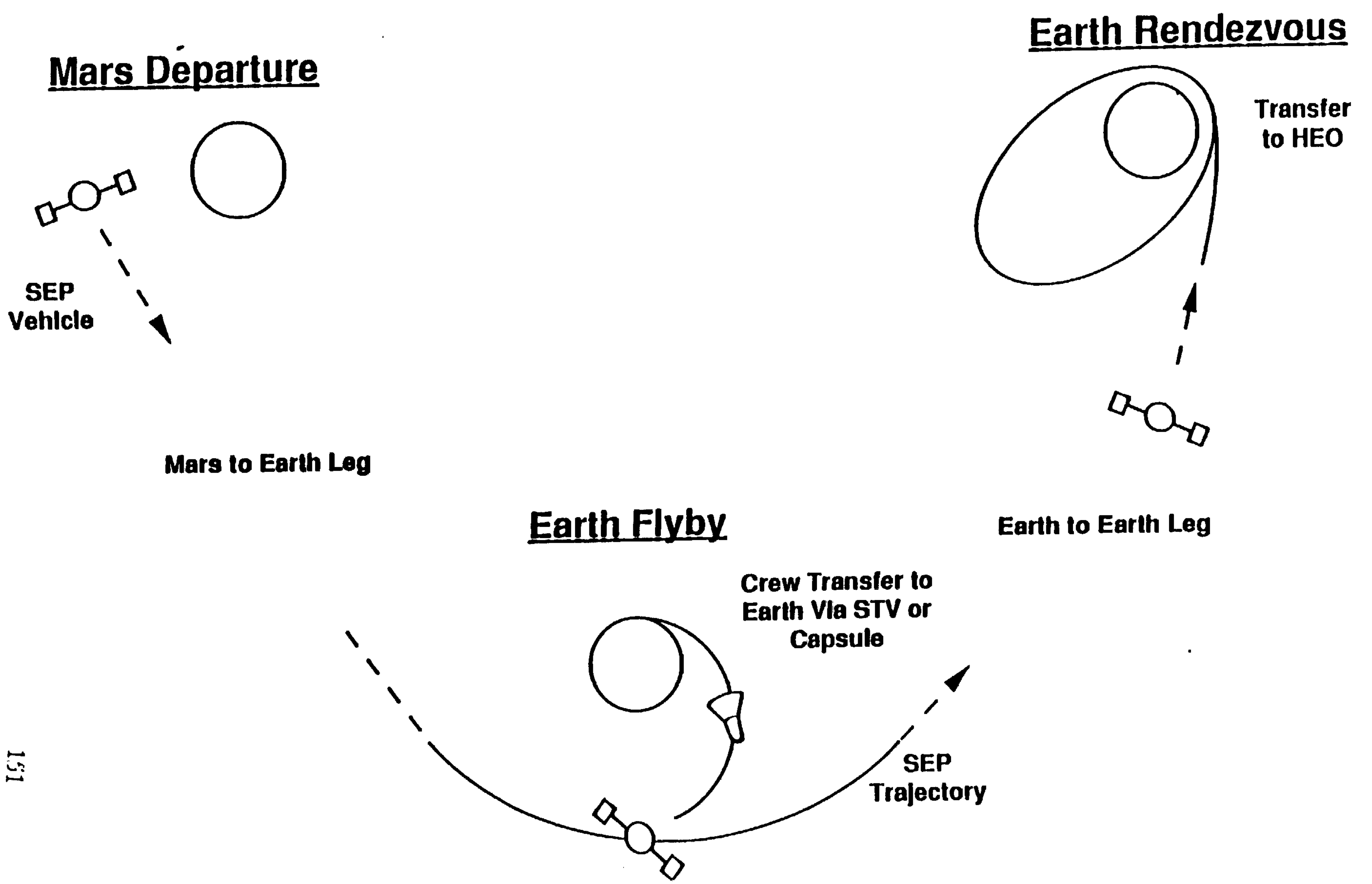
View from Mars Reference Frame



Earth Flyby

The advantage gained by an Earth flyby is due to the vehicle being able to accelerate for a longer period of time, before it has to decelerate. The vehicle will approach Earth with an excess speed limited to 5 km/sec. The vehicle will drop the crew off at earth via a STV or ECCV. The vehicle will spend up to 200 days "catching back up with" the Earth. The amount of return time can be traded against time saved during manned flight. One issue is thruster lifetime, which might limit the time the vehicle can spend trying to rendezvous with the earth. The Earth flyby will allow for a reusable SEP.

Earth Flyby

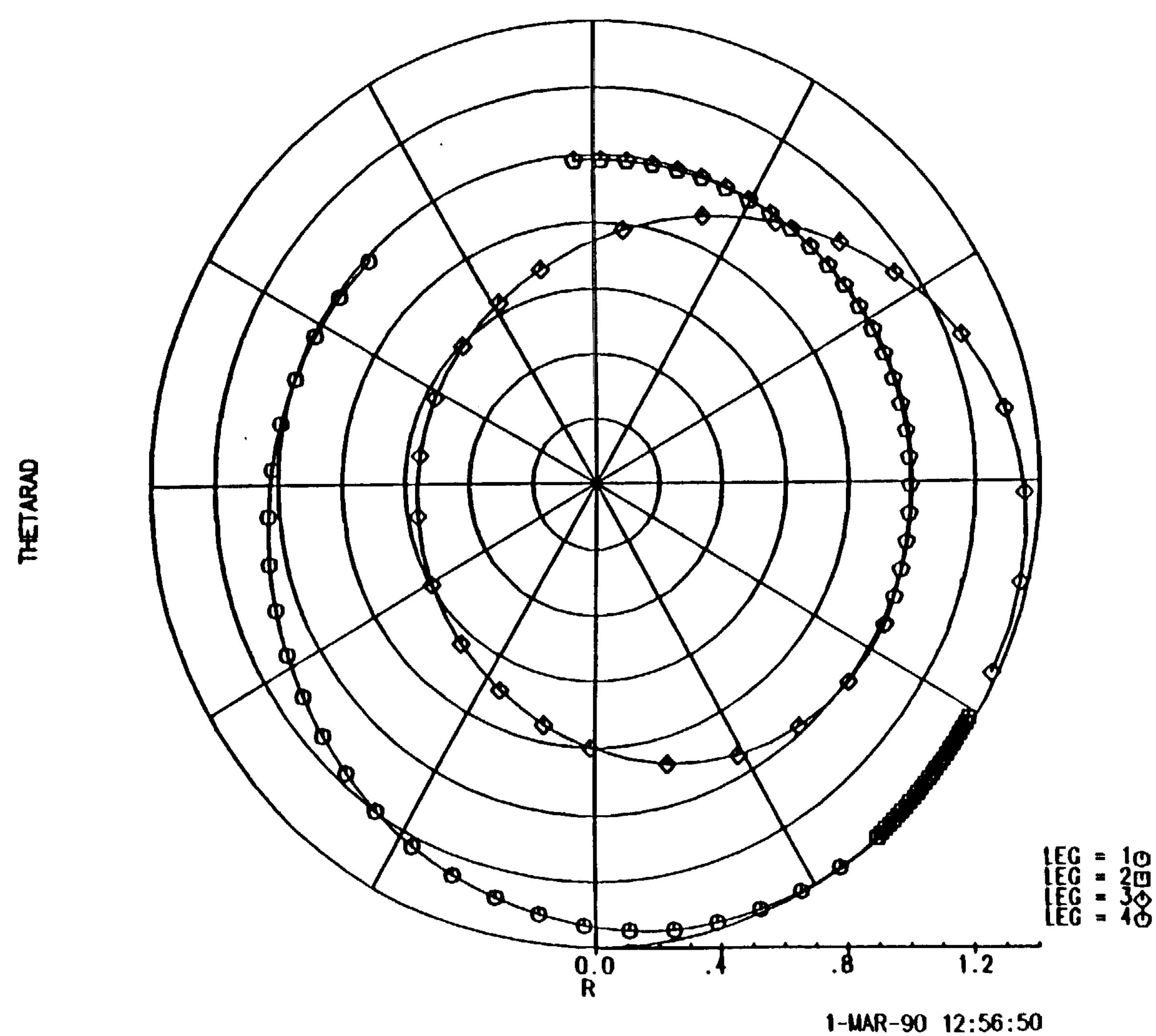

Mars Departure
SEP Vehicle
Mars to Earth Leg
Earth Flyby
Crew Transfer to Earth Via STV or Capsule
SEP Trajectory
Earth to Earth Leg
Earth Rendezvous
Transfer to HEO

SEP Trajectory with Flybys

The trajectory plot combines the planetary flybys referenced previously and plots an actual trajectory generated by CHEBYTOP. The Mars stay and the Earth flyby and rendezvous are included.

SEP Trajectory with Flybys

THETA VERSUS R



2-5354-WGV90-078
29 November, 1990

To:	Brad Cothran	M/S JX-23
c:	John Hardtla	M/S 82-26
	Dana Andrews	M/S 8K-02
	Vince Weldon	M/S 82-48

Subject: Conjunction-Class Missions to Mars Using Solar Electric Propulsion

References:

1) Johnson, F. T., "Improvement of the QUICKTOP Digital Computer Program (CHEBYTOP III)," D180-15371-1, April, 1973.

Discussion:

Trade studies were performed for the proposed conjunction class manned Mars mission using solar electric propulsion. A nominal case for the 2016 opportunity was generated with rendezvous conditions at all encounters, and the result of adding a Mars flyby leg was studied. Trades were then performed in which power level and vehicle specific mass were varied, and specific impulse was varied for each power level at a fixed specific mass. Using a baseline case of 10 MW, specific impulse (Isp) of 6500 seconds and a vehicle specific mass (alpha) of 10 kg/kW, trajectories were generated for each of the opportunities between the years 2010 and 2026. Four different power degradation (with distance from the Sun) curves were then compared for a given vehicle.

Assumptions for the study were as follows:

- Variables included flyby leg duration, Isp, initial mass in orbit, power level, alpha, launch date and power degradation curve.
- Trip time was defined as Earth escape to Mars and return to Earth. Mars residence was not included.
- The equation used to calculate thruster efficiency was:

$$\eta = \frac{BB}{1+\left[\frac{DDT}{Isp*g_o}\right]^2}$$

where DDT=22.96 and BB=0.835, constants from CHEBYTOP (Reference 1).

- Outbound payload of 116 MT
- Inbound payload of 43 MT
- Tankage mass equal to 10% of propellant mass
- Earth spiral to escape delta V of 4000 meters/second (GEO to escape)
- High elliptical Earth capture orbit
- Mars capture orbit of 24.5 hour (one Martian day) period with perigee altitude of 360 km.
- Vehicle alpha was defined as the ratio of the total inert weight to the power delivered to the thrusters.

The following tables 1, 2, and 3 show the flyby benefits, vehicle masses, alphas, specific impulses and trip durations for 600 day Mars residence missions, 2016 opportunity:

Hyperbolic Excess Velocity (km/sec)	Flyby Leg Duration (days)	Delivered Payload (MT)
1.93	600	50.2
2.16	500	44.4
1.97	400	44.0
0	300	42.8
1.40	200	48.2
1.48	100	48.2
0	0	43.0

Table 1 Nominal Case and Flyby Benefits

The nominal case in Table 1 consisted of an outbound trip to Mars, 600 day Mars residence time and a return trip to Earth. A 10 MW vehicle operating at an Isp of 5000 seconds was chosen as the starting point for the solar electric propulsion (SEP) conjunction class mission study. The vehicle alpha was 10 kg/kW and its initial mass in geosynchronous Earth orbit (GEO) was 355 MT, enabling it to deliver the desired payloads to Mars (116 MT) and back to Earth (43 MT). The vehicle was then allowed to fly by Mars at a finite (optimized) speed and return a specified duration later, reducing the delta V required for the first leg and thus reducing the required propellant mass. As a result, the payloads delivered to Earth upon return are higher for the flyby cases, and are shown in Table 1. Note that the payload for the 300 day flyby leg is nearly identical to the no flyby case, as the flyby velocity optimizes (?) to be zero. The zero value could be a result of the optimizer stopping on a local minimum. In any case, the flyby leg generally reduces the delta V requirement of the Earth to Mars leg. However, from a mission practicality/safety standpoint, the benefit may not be worth the inconvenience of not having a return vehicle nearby.

Initial Mass in Low Earth Orbit (MT)	Power (MW)	Total Vehicle Alpha (kg/kW)	Trip Time (days)
473	15	12	460
408	15	10	438
363	15	8	420
390	10	12	457
355	10	10	430
326	10	8	410
295	5	12	555
256	5	10	525
253	5	8	500

Table 2 SEP Power Level Trades

Trades of initial mass versus trip time for three different power levels are shown in Table 2. The vehicle alpha was varied for each power level, and a Mars residence of 600 days was used. Due to lower initial power levels and additional power degradation near Mars (power decreases roughly with the square of the distance from the sun), the SEP vehicles do not have nearly the flexibility in initial mass versus trip time that the NEP vehicles do. The combination of high Isps and low power levels limits the total available delta V, forcing the SEP vehicles to fly on low energy trajectories. As a result, increasing the propellant weight does not necessarily decrease the trip time, since the total available delta V is thrust-constrained rather than propellant mass-constrained. Each power level and vehicle combination therefore flies best within a relatively narrow range of propellant mass fractions. A representative sample is shown above.

Initial Mass in Low Earth Orbit (MT)	Specific Impulse (sec)	Power (MW)	Trip Time (days)
309	10,000	15	445
319	7500	15	445
363	5000	15	420
262	10,000	10	490
265	7500	10	470
326	5000	10	410
218	7500	5	568
253	5000	5	500

Table 3 SEP Isp Trades

Table 3 shows the effect of varying specific impulse for a given power level. All vehicles used an alpha of 8 kg/kW, since at higher Isps, a low alpha was the only way the 5 MW vehicle could get to Mars. For the 5 MW vehicle, an Isp of 5000 is the practical upper limit. Higher Isps result in such low thrust that the vehicle must lengthen its trip time to well beyond the Hohmann transfer trip time simply to allow the thrusters enough time to generate the required delta V to complete the transfer. At 15 MW, an Isp of 10,000 seconds reduces the initial mass in orbit substantially while maintaining a reasonable trip time. The 10 MW vehicle operated best at an Isp between 5000 and 10,000 seconds, and for the remainder of the study a vehicle with an Isp of 6500 seconds was chosen as a good compromise between low initial mass and reasonable trip times. Factors that could affect this choice are cost of delivering mass to orbit, feasibility of extremely large structures for higher power levels, and human tolerance to extended time in space.

Opportunity (year)	Launch Date (Julian Date-2440000)	Trip Time (days)	Stay Time (days)
2010	15110	535	450
2012	15875	520	470
2014	16657	463	500
2016	17452	401	550
2018	18233	372	600
2020	19032	418	550
2022	19800	505	500
2024	20570	530	450
2026	21340	530	450

Table 4 Trajectory Summaries for 10 MW SEP Vehicle, Various Opportunities

Using the baseline vehicle (10 MW, alpha=10 kg/kW, Isp=6500 seconds) and trajectory, a trade was performed in which the year of opportunity was varied through the entire Earth-Mars opportunity cycle. Results are summarized in Table 4. When arrival and departure from Mars occurs near the apoapsis of the Martian orbit, Mars is further away from and Earth and is traveling slower. Both of these factors require a corresponding increase in necessary total delta V for the same trajectory geometry. As a result, longer trip times and higher initial masses in LEO are required for some of the oppositions than others. The SEP vehicles cannot make up for higher energy requirements by increasing the delta V (the propellant mass available), so shortening the stay time is used as a way of maintaining relatively efficient paths on the "more difficult" opportunities.

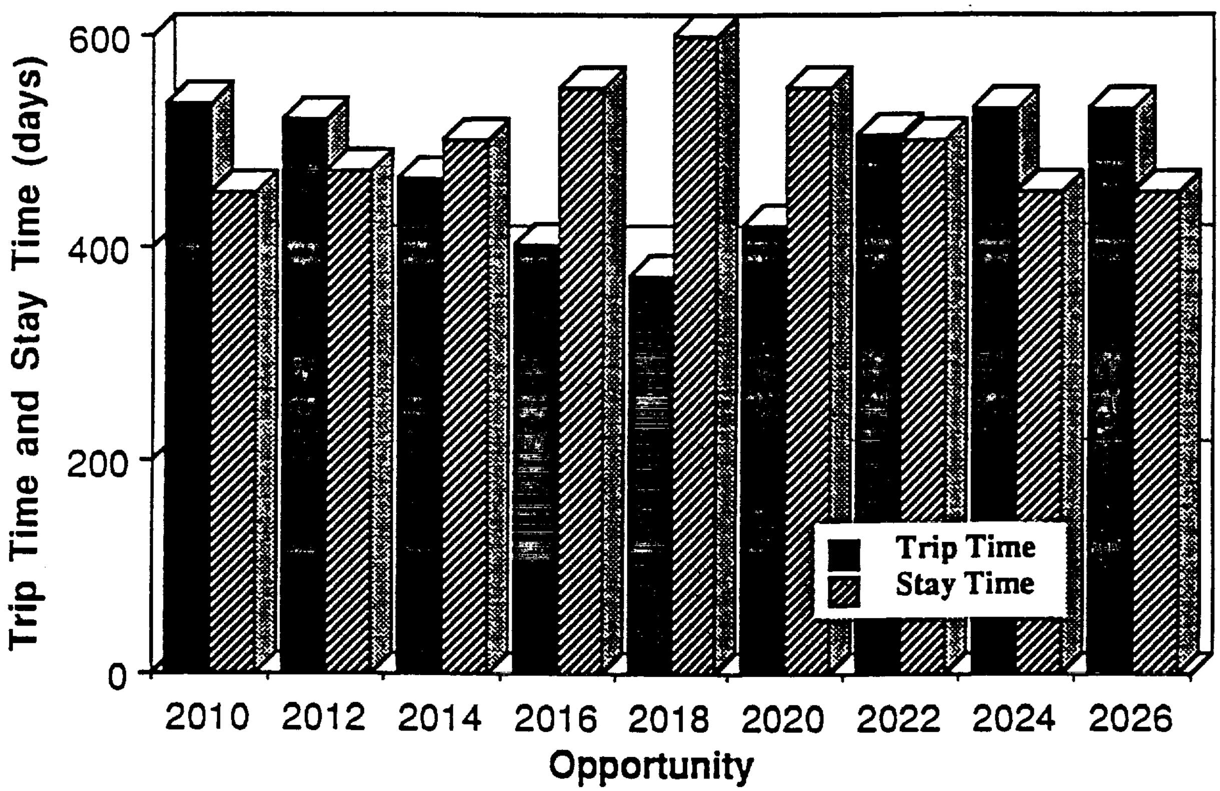


Figure 1 Trip Time and Stay Time for Various Earth-Mars Opportunities

The Mars residence time and trip time as a function of opportunity is illustrated in Figure 1. The vehicle initial mass in GEO was 322 MT for all cases, and stay time and trip time were varied in order to produce the required payloads. The 2018 launch opportunity represented in the previous data is one of the "easier" opportunities in that Mars is near perigee when the SEP vehicle arrives and departs. The total distance traveled is shorter and the required delta V is lower. Correspondingly, trip time is low and stay time can be increased to at least 600 days while still maintaining relatively efficient paths to and from Mars. For the difficult opportunities, the delta V limitations require that the vehicles travel along longer, more efficient paths. To maintain efficient geometry, stay time is reduced. For an opportunity that requires significantly more delta V, such as the 2010 opportunity, a higher power level may be beneficial due to thrusting limitations on lower-powered vehicles.

Opportunity	Power Curve (see below)	Trip Time (days)	Stay Time (days)
2018	1	372	600
2018	2	383	600
2018	3	388	600
2018	4	380	600
2026	1	530	450
2026	2	538	440
2026	3	538	435
2026	4	531	442

Table 5 Trip Time Variations with Power Degradation Curves

Four power degradation curves were used in this study:

Power Curve 1: Reference JPL-50, used in previous studies
Power Curve 2: $P/P_0 = 5.5989-8.5331*R+5.0004*R**2-1.0463*R**3$
Power Curve 3: $P/P_0 = 5.2461-7.8198*R+4.5087*R**2-0.9352*R**3$
Power Curve 4: $P/P_0 = 4.4917-6.1930*R+3.3679*R**2-0.6667*R**3$

where R is distance from the Sun in A. U.'s.

The effect of the different power degradation curves on trip time is shown in Table 5. The initial mass was held fixed at 322 MT and the stay time was allowed to vary for the 2026 opportunity. The power curves affected the trip and stay time to some extent, but in no case did they force a different power level or vehicle alpha to be used.

Conclusions:

For the more efficient opportunities (e. g. 2016, 2018), a 10 MW vehicle provides a good compromise between low initial mass in Earth orbit and short travel times to and from Mars. For the opportunities which require substantially more energy, a higher power vehicle may improve the overall performance for the mission.

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OPTIMUM LOW THRUST ROUND TRIP EARTH-MARS MISSION AND SYSTEM DESIGN PARAMETERS

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1.0 INTRODUCTION

The objective of this task is to determine optimum mission and system design parameters for both Nuclear Electric Propulsion (NEP) and Solar Electric Propulsion (SEP) systems performing round trip Earth-Mars missions in the 2011 to 2028 time frame, subject to a variety of both equality and inequality constraints. The following constraints are enforced throughout the entire study:

- Payload at Mars arrival is 124,300 (kgs).
- Propellant reserves and tankage is 10% of the propellant loading.
- Mass dropped at Mars is 84000 (kgs), plus the propellant reserves and tankage for the Earth-to-Mars leg of the mission (including the Earth escape and Mars capture spirals).
- Payload at Earth return is 40300 (kgs).
- Stay time at Mars is 30 days. It is assumed that the crew will exit the low thrust vehicle and descend to the Mars surface (using a high thrust system) in a relatively short time . The crew will also ascend using a high thrust system, and will rendezvous with the low thrust vehicle for the Mars-to-Earth return leg of the trip. However, the low thrust descent and ascent spiral propellants are included as part of the low thrust system being optimized. At Earth departure, it is also assumed that the crew will use a high thrust system to rendezvous with the low thrust vehicle just before Earth escape. At Earth return, the crew will leave the low thrust vehicle before spiralling down into Earth orbit. Thus, the Earth escape and capture spiral propellants are charged to the low thrust system mass, but the spiral times are not counted as part of the mission.
- Minimum acceptable distance of the spacecraft from the sun is 0.3 AU, on either the outbound or inbound leg of the mission. This constraint never becomes a factor in this study because the minimum distance on all missions examined is about 0.5 AU.

2.0 SIMULATION AND OPTIMIZATION PROCEDURES

A parameter optimization program, ~~referred to~~ as **POP**, is used to drive the optimization process. **POP** is an acronym for "Parameter Optimization Program." It can be interfaced with any system model and, when the parameters are communicated properly between the system model and POP, it will drive the simulation to find the set of parameter values that satisfies all of the defined constraints and minimizes a cost functional. Both equality and inequality type constraints are acceptable. System parameters may be designated as fixed (in which case POP ignores them in its optimization search) or variable (in which case POP allows them to vary in its optimization search). The theoretical foundation for POP is given in Reference 1.

It is well known that SIMPLEX only solves linear systems of equations; thus, an obvious question is "How is SIMPLEX used to solve nonlinear problems?" The answer is that all the required partiial derivatives are supplied to SIMPLEX as the coefficients in its system of linear equations, and the search is constrained to a "linear neighborhood" of the current system states. In this way, on any one call to SIMPLEX a linear system of equations is solved and the answers are returned to POP, which then reevaluates all relevant relationships, with all their nonlinearities, and sets up to take another step with SIMPLEX. This procedure of sequentially feeding SIMPLEX small linear chunks of a large nonlinear problem ultimately results in a solution of the large nonlinear problem. It is quite surprising how robust POP is in this role. Reference 1 exhibits some results for a difficult and highy nonlinear problem, but over the years since POP was first developed, it has been used to solve a host of difficult nonlinear problems.

One advantage of using POP over several other optimization techniques is the ease with which the cost functional, the constraints (both equality and inequality types), and the parameters to be fixed or variable during the optimization can be changed. Any variable in the system model can be used as a parameter by equivalencing it to a member of the parameter set. Any parameter in the set can be fixed by simply setting an input flag properly for that parameter. The cost functional or constraints can be changed by changing the proper equations in the constraint subroutine and recompiling.

Performing system optimization is somewhat like walking through a mine field, "You never know what might happen after the next step!" Optimization with POP is no different. The user must be wary of several potential problem areas.

Estimating the partial derivatives is one potential problem area. The partials are estimated empirically, as indicated in the following equation:

$$\left(\frac{\partial C_i}{\partial p_j}\right)_o = \frac{C_i\,(p_{jo}+\delta p_j) - C_i\,(p_{jo})}{\delta p_j}$$

where C_i (as $i = 1,\ldots,N$) represent the cost functional and all the constraints, and p_j (as $j = 1,\ldots,M$) represent all variable system parameters. The user must input values for δp_j , and the value for each "δp_j" must be chosen such that the resulting matrix of partial derivatives adequately approximates the matrix of true but unknown partial derivatives. This is not a trivial exercise for problems that you are not familiar with. POP allows you to set a DEBUG flag in the input so that you can see the results of $C_i\,(p_{jo}+\delta p_j)$ and $C_i\,(p_{jo})$ and interactively change the δp_j to find values that result in credible approximations for the partials. You should input values for δp_j such that the differences in the numerator in the equation for the partials retains 4 or 5 significant digits. Failure to do this properly can result in much wasted manhours and computer time.

Determining a linear neighborhood of the current system states can also be difficult. POP uses input variables called BFAC to control the search region for POP. BFAC is a multiple of δp_j , which defines the region within which POP is allowed to vary each p_j on one iteration. POP then dynamically adjusts BFAC based upon the linearity of the cost functional during each search. When the cost functional increases with respect to BFAC, POP reduces BFAC by (0.75*BFAC).

A maximum (BFMAX) value and a minimum (BFMIN) value are also input. These values restrict the range of values within which BFAC can vary. BFMIN should be 1.0 if the δp_j values have been chosen reasonably. BFMAX is not so easy to specify, and can have a great influence on the optimization process. If BFMAX is too large it is possible for the process to bounce around from one local "valley" to another, and perhaps never really converge. If BFMAX is too small the process may move very slowly toward the minimum of a local valley, which may not be the best valley anyway. POP has no facility for assuring that the local minimum it finds is the global minimum. The user is responsible for analysing the results and the problem to decide whether the results are in fact the desired optimum.

Figure 1 shows a macroflow diagram of the POP optimization procedure. After input and initialization, it calls the system simulation routine with "nominal" values for all of the parameters to determine nominal system performance. It then varies each "free" parameter by a prescribed "delta" amount and uses divided fifferences to empirically estimate the partial derivative of each constraint (i.e. the cost functional, all equality constraints, and all inequality constraints) with respect to each free parameter.

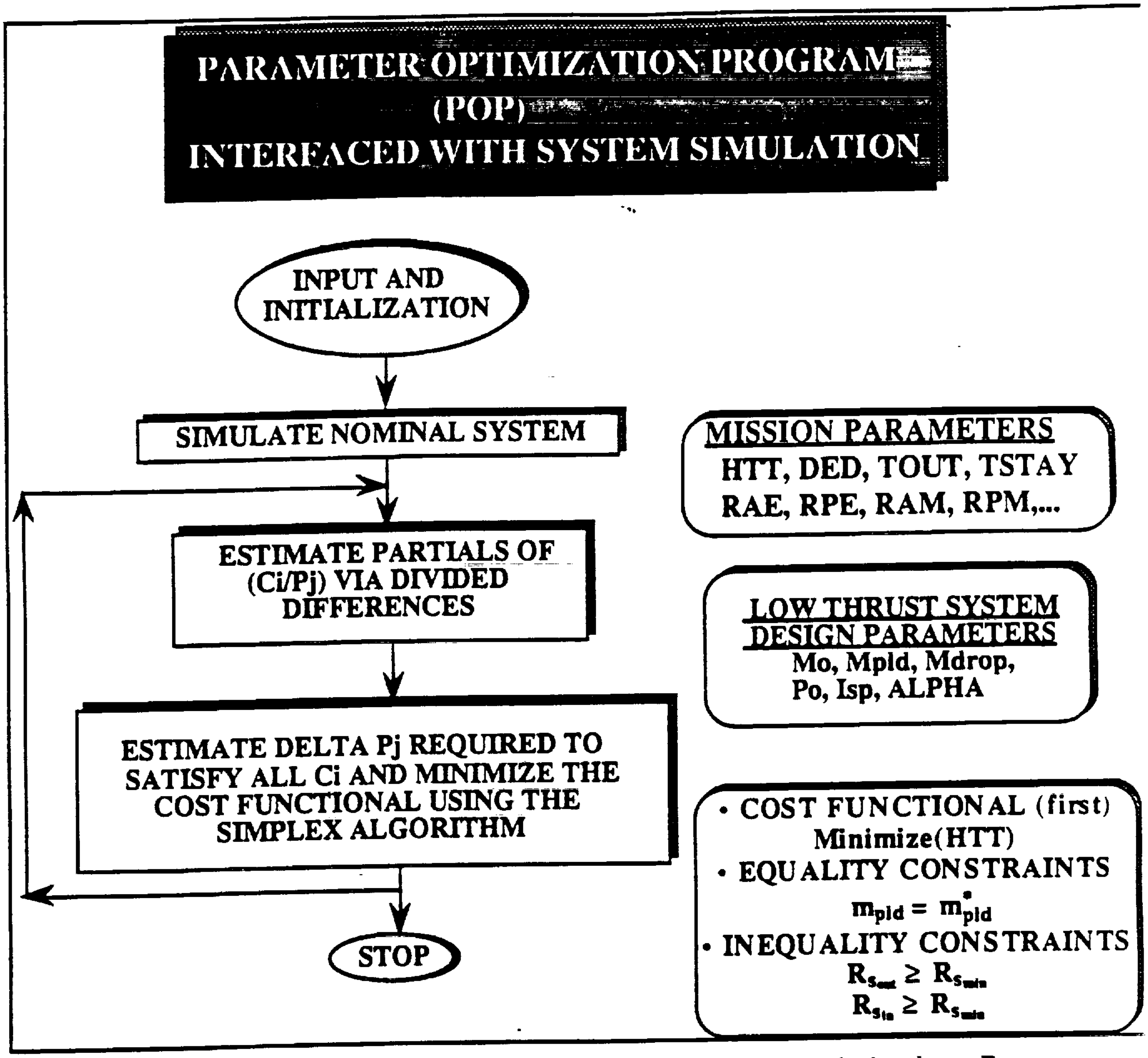


Figure 1. Macroflow Diagram of The Parameter Optimization Program (POP)

The SYSTEM subroutine used in this study is structured using low thrust escape and capture spiral subroutines based on the results of Reference 2, and low thrust Earth-Mars and Mars-Earth trajectory subroutines based on the CHEBYTOP development by The Boeing Company in the late 1960s and early 1970s, as documented in Reference 3.

Figure 2 presents a macroflow diagram of the system subroutine used for this study. Departure is always from a circular Earth orbit, and the spiral is simulated out to escape (C3E = 0). CHEBYTOP routines are then called to simulate the trajectory to Mars capture (C3M = 0). The arrival spiral subroutine simulates the trajectory from C3M = 0 to the specified circular Mars orbit. If the departure or arrival orbit is

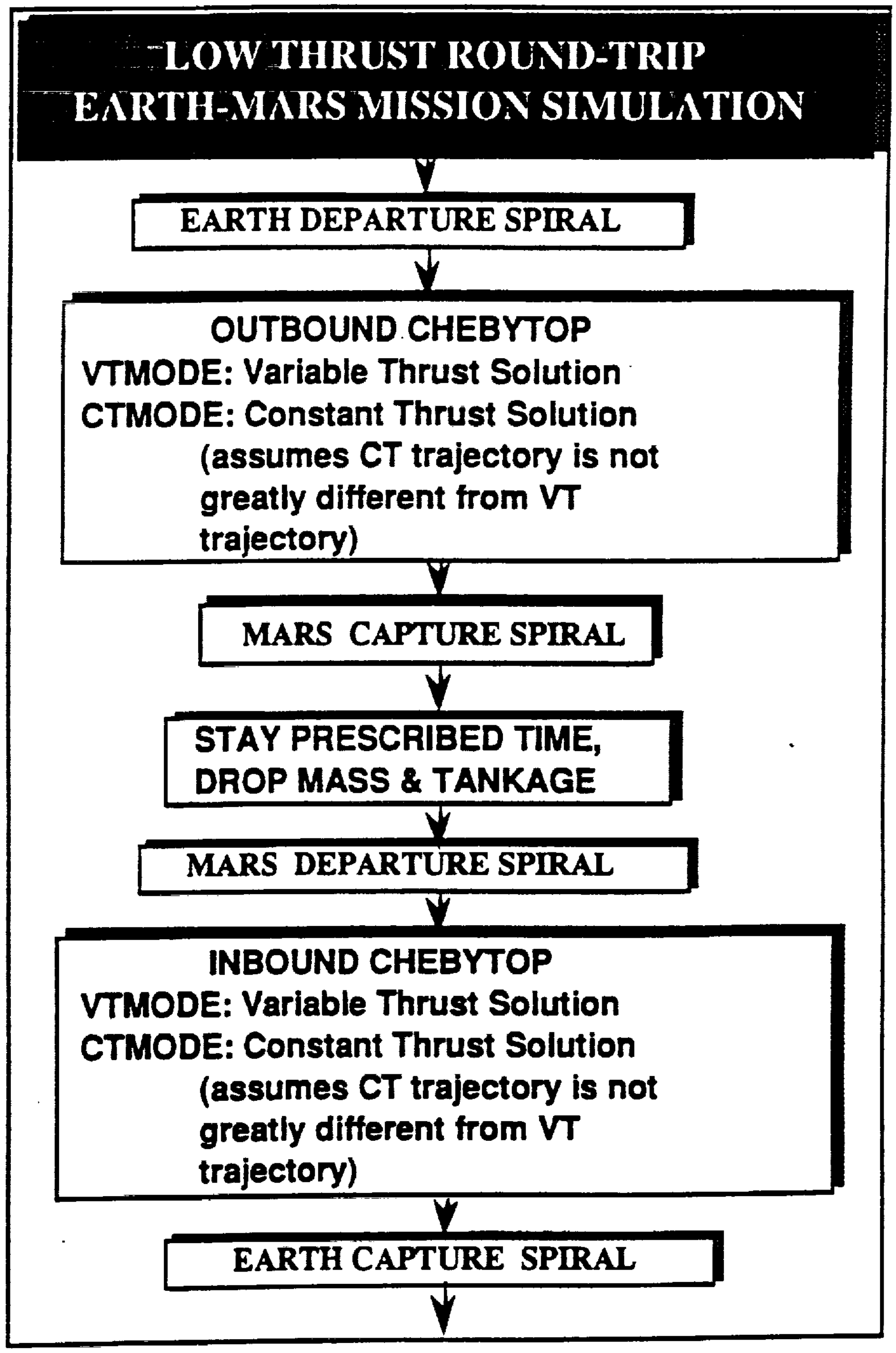


Figure 2. Macroflow of the Low Thrust Round-Trip Earth-Mars Mission Simulation

elliptical, the spiral subroutine uses the semi-major axis as if it were the radius of a circular orbit. This approximation is made because the spiral subroutines are developed for departure from and arrival at circular orbits.

CHEBYTOP is used in this analysis primarily as a trajectory generator. It optimizes the thrust attitude angles and coast arcs when it generates a trajectory, but nothing else. POP is used to optimize all of the other mission and system parameters. A significant problem surfaced during this analysis as POP kept stressing the system to minimize the cost functional. Since CHEBYTOP assumes that the VTMODE trajectory is not greatly different from the CTMODE trajectory, and POP keeps pushing the system to its limits, even for the VTMODE, it gets to a point where the CTMODE approximation does not converge, and in this analysis we are primarily interested in CTMODE performance results. Thus, the question arose: "How can the optimization search volume be constrained to a region such that the CTMODE always converges?" This was accomplished by constraining both the outbound and inbound CTMODE payload mass fractions to desired values.

To be more specific, suppose that POP is minimizing the total heliocentric travel time, and a particular iteration results in a CTMODE payload mass of 30,000 (kgs). Since the desired payload value of 40,300 (kgs) is different from that achieved on that iteration, the desired payload mass fraction is computed using the desired payload mass with all the mission and trajectory data from the iteration. The difference in the desired mass fraction and the mass fraction achieved on the iteration is entered as an error in the constraint subroutine. This is done on both the outbound and inbound legs of the mission. It is evident that the desired mass fraction value changes from one iteration to the next because the mission and trajectory data change, but this "floating" of the desired value has caused no discernable difficulty. This "floating end condition" concept was used successfully on an Apollo lunar targetting problem (see Reference 4).

This scheme accomplished the desired results, i.e. it kept the iteration constrained to a region in which the CTMODE was close enough to the VTMODE results to converge. However, the user should be aware that this reduced the search volume to accomodate the CTMODE approximations, and it may be possible to achieve better results with an unconstrained trajectory generator. It is not likely, however, that such improvement would be sufficiently large to change the trends or trades resulting from this analysis.

3.0 EARTH-MARS ROUND TRIP MISSION PARAMETERS

The mission begins with the Earth departure spiral out from an Earth orbit to C3E = 0. The orbit is specified by input of its apogee and perigee radii, RAED and RPED. As was mentioned earlier, the spiral algorithm assumes departure from circular orbit. If apogee radius is different from perigee radius, the algorithm uses the semimajor axis as the radius of the circular orbit. The spiral out time is ignored, but the propellant required is included as a part of the low thrust system mass.

At escape (C3E = 0) CHEBYTOP computes the outbound leg of the heliocentric portion of the flight. Beginning time of this outbound leg is called the "date of Earth departure, DED," and is an input. The "heliocentric travel time, HTT," is input and is the sum of the outbound Earth-to-Mars trip time (from C3E = 0 to C3M = 0) and the inbound Mars-to-Earth trip time (from C3M = 0 to C3E = 0). Note that HTT does not include stay time at Mars or any of the spiral times.

The "outbound trip time, TOUT," is also input, and the inbound trip time is computed as TIN = HTT - TOUT. The Mars arrival date is DMA = DED + TOUT. The arrival spiral is from C3M = 0 to a Mars orbit specified by its apoapsis and periapsis radii, RAMA and RPMA. If they have different values the algorithm uses the semimajor axis. Again, the spiral down time is ignored, but the spiral down propellant is considered part of the outbound propellant requirement. At Mars, the input value for drop mass [84,000 (kgs)] is dropped, along with the outbound tankage and reserves, which is 10% of the sum of propellants used in the Earth escape spiral, the outbound heliocentric leg, and the Mars capture spiral.

The Mars departure date is DMD = DMA + TSTAY, where TSTAY is input. The Mars departure orbit is specified by input of RPMD and RAMD, periapsis and apoapsis radii of the departure orbit. The Mars departure spiral is out to C3M = 0 and the propellant used is a part of the inbound propellant for the system.

Earth arrival date is DEA = DMD + TIN. CHEBYTOP computes the inbound heliocentric leg of the mission from C3M = 0 to C3E = 0 in time TIN. The Earth capture spiral is from C3E = 0 down to an Earth orbit specified by input of RPEA and RAEA. The spiral down time is ignored, but the propellant used is included in the inbound propellant requirements for the system.

Two versions of POP were used: one minimizes HTT; the other minimizes the initial mass in Earth orbit, IMEO, with HTT fixed at a desired value. Mission parameters that are available for POP to use in its optimization are:

- DED: Date of Earth departure
- TOUT: Heliocentric outbound travel time (from C3E = 0 to C3M = 0)

- HTT: Sum of outbound and inbound heliocentric travel time
- TSTAY: Stay time at Mars (from C3M= 0 at arrival to C3M = 0 at departure)

4.0 LOW THRUST SYSTEM PARAMETERS

The fundamental relationships for modelling the low thrust system are listed below:

$$J = \int_0^T a^2\, dt \quad \text{, (trajectory optimization parameter)}$$

$$\frac{1}{m_f} = \frac{1}{m_o} + \frac{J}{2\eta P_o} \quad \text{, (mass related to trajectory parameters)}$$

$$m_{ps} = \alpha P_o \quad \text{, (power system mass; } \alpha = \text{specific mass; } P_o = \text{initial power)}$$

$$c = g_e I_{sp} \quad \text{, (exhaust velocity)}$$

$$\eta = \eta\ (I_{sp}) \quad \text{, (Thruster efficiency)}$$

$$a_o = \frac{2\eta P_o}{c m_o} \quad \text{, (initial acceleration)}$$

$$m_p = m_o - m_f \quad \text{, (propellant mass)}$$

$$m_{tr} = k m_p \quad \text{, (tankage \& reserves)}$$

$$m_{pl} = m_o - (1+k) m_p - m_{ps} \quad \text{, (payload mass)}$$

The system design parameters available to POP for use in its optimization are listed below:

- IMEO: Initial mass in Earth orbit
- HISP: Specific impulse of the low thrust system
- PO: Initial power of the low thrust system

Note that the "specific mass, ALPHAW or α," is an input but is never varied in the optimization.

5.0 NUCLEAR ELECTRIC PROPULSION (NEP) RESULTS

Design parameters for the NEP system are its (1) initial power, Po, (2) specific mass, α, and (3) specific impulse, Isp. In some of the following NEP results Isp is optimized, but specific mass and Po are held constant.

Thruster efficiency, η, was specified as a tabulated function of Isp. Thus, when Isp is optimized it is neccessary that the η(Isp) be represented functionally so that the partial derivative can be evaluated. The tabulated data was fit with the following fourth order polynomial for that purpose:

η = -0.082668 + 2.6251e-4*Isp - 3.087e-8*Isp**2 + 1.8047e-12*Isp**3 -4.3169e-17*Isp**4

The tabulated η(Isp) data only extends to an Isp value of about 12500 (sec). Thus, any time the NEP Isp value is optimized, it is constrained such that its value is less than or equal to 12500 (sec).

All these NEP results assume Earth departure and return at a "nuclear safe orbit" of radius 7070 (km), i.e. about 700 (km) altitude; Mars arrival and departure is at a circular orbit of radius 23000 (km).

5.1 NEP SYSTEM DESIGN PARAMETRICS FOR THE 2016 OPPOSITION

This section presents parametric data for the 3/2016 launch opportunity for various NEP system design options. Detailed optimization results for this section are presented in the following tables:

For the Po/α = 120/3 System

HTT	*302.042	325	400	500	600
DED	17470.46	17470.80	17459.48	17428.49	17404.25
TOUT	126.834	129.300	155.195	202.428	245.647
IMEO	997.689	865.390	737.102	676.761	652.971
HISP	10000	10000	10000	10000	10000
ETA	.83	.83	.83	.83	.83

For the Po/α = 80/4 System

HTT	*342.049	400	500	600	
DED	17462.80	17459.74	17427.82	17403.00	
TOUT	142.822	156.637	205.568	249.653	
IMEO	854.930	694.094	627.554	602.483	
HISP	10000	10000	10000	10000	
ETA	.83	.83	.83	.83	

For the Po/α = 40/4 System

HTT	*359.262	400	500	600	700
DED	17458.42	17458.07	17437.6	17401.00	17365.96
TOUT	156.242	161.844	203.7	256.093	302.327
IMEO	548.281	443.885	396.197	379.753	375.463
HISP	10000	10000	10000	10000	10000
ETA	.83	.83	.83	.83	.83

For the Po/α = 24/6 System

HTT	*439.964	500	600	700	
DED	17456.85	17440.79	17401.42	17354.13	
TOUT	189.924	203.105	261.178	321.480	
IMEO	448.792	384.341	363.858	358.385	
HISP	10000	10000	10000	10000	
ETA	.83	.83	.83	.83	

For the Po/α = 10/12 System

HTT	*610.319	650	700	800	
DED	17431.76	17404.75	17390.34	17346.97	
TOUT	270.478	272.068	297.456	349.266	
IMEO	377.595	345.701	342.290	342.310	
HISP	10000	10000	10000	10000	
ETA	.83	.83	.83	.83	

The first value in each table (with the asterisk, *) is the minimum HTT value achievable with that NEP system design and launch opportunity. The other HTT values are fixed and the IMEO values are the minima for those HTT values.

Figure 3 shows the minimum IMEO required for various NEP design options to perform missions of various durations (various HTT values). Keep in mind that all these NEP designs are assumed to have Isp = 10000 (sec) with an efficiency of about 0.83. The minimum value of HTT shown in Figure 3 is the minimum HTT value achievable with that NEP design, characterized by its Po, Isp, and ALPHA. Suppose that a mission of HTT = 302 days is required. Figure 3 shows that the only one of these NEP designs that has that capability is the **Po = 120** with **α = 3** . It is also evident from the figure that the NEP system having the lowest Po value will perform any HTT mission with the minimum IMEO, if it can achieve the desired HTT value. For example, if an HTT of 600 days is required, it is cheaper in terms of IMEO to perform the mission with the (24,6)

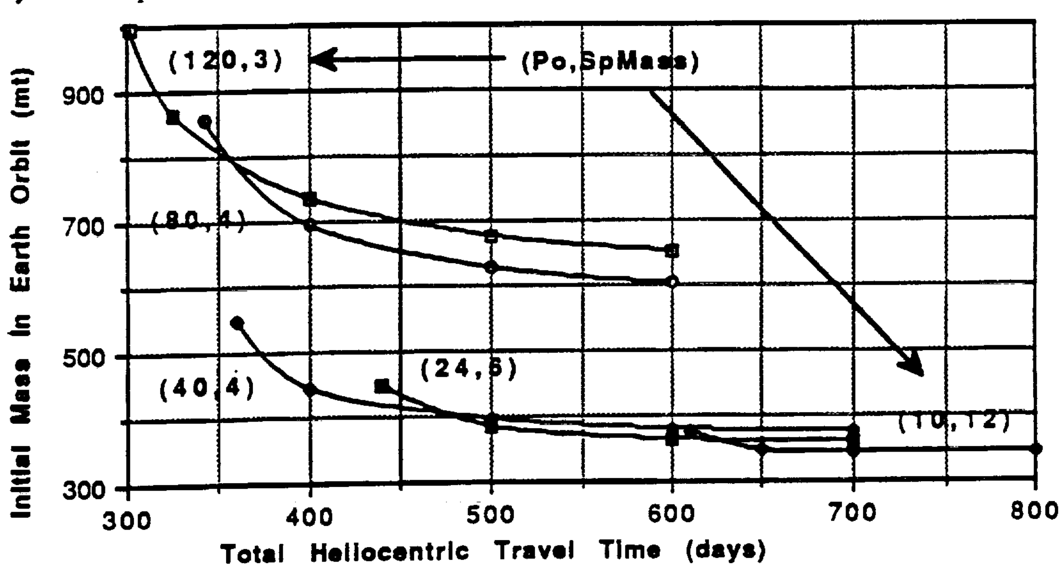


Figure 3. Initial Mass Required in Earth Orbit for Various Missions and Nep System Designs

system than with any other system examined. That mission can't be done with the (10,12) system; the figure shows that the minimum HTT achievable with the (10.12) system is about 610 days.

Figures 4 and 5 are companions of Figure 3, showing the optimum Date of Earth Departure (DED), and duration of the outbound leg of the mission (TOUT), for the same set of mission and NEP system design options.

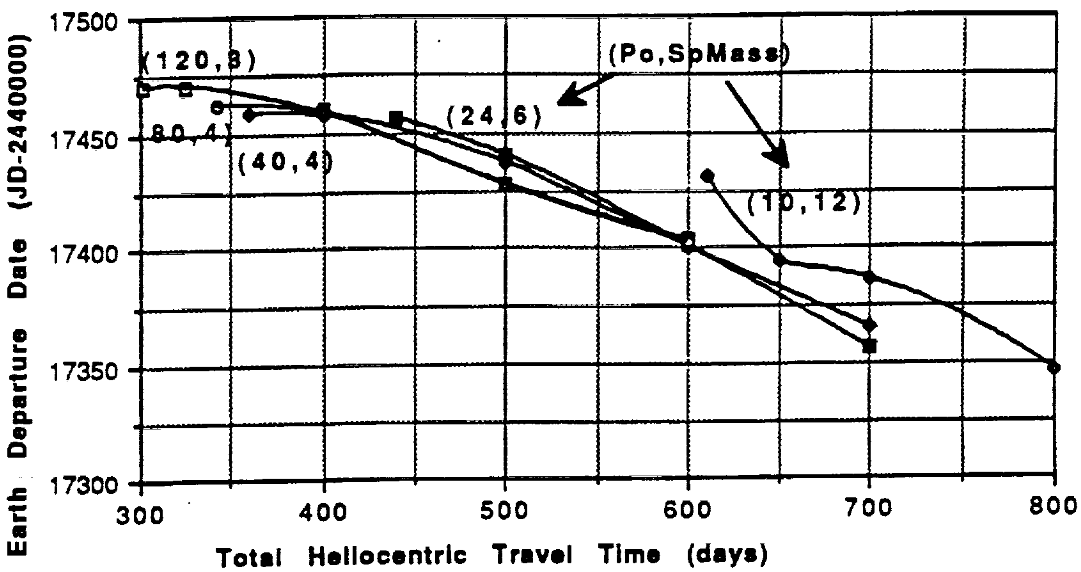


Figure 4. Date of Earth Departure for Various Mission and NEP System Design Options

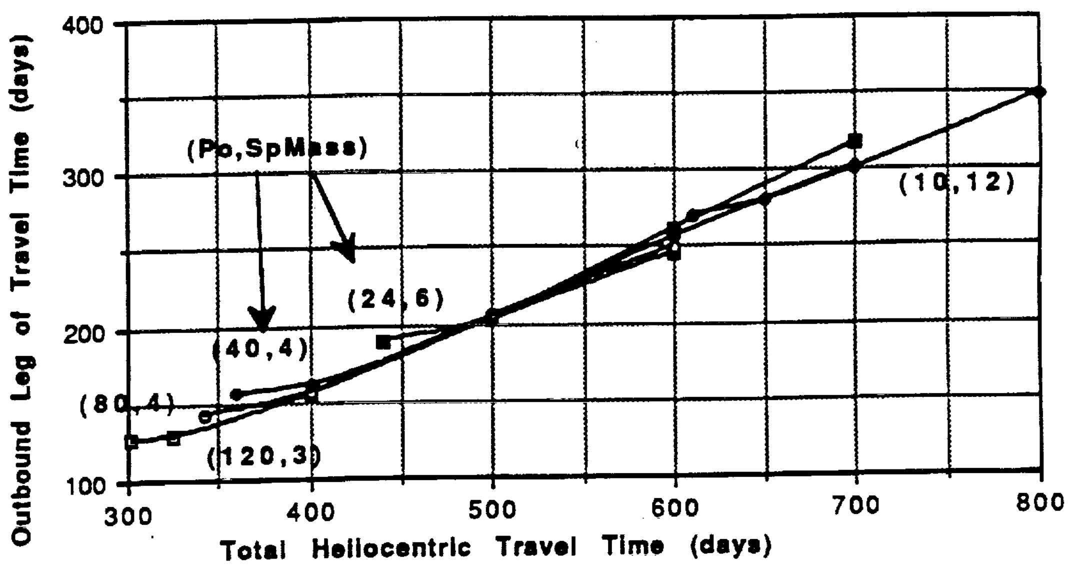


Figure 5. Duration of the Earth-to-Mars Leg of Various Missions Using Various NEP System Design Options

Figures 4 and 5 show that the HTT value primarily controls the value of DED and TOUT, with the (Po, α) combination of the NEP system having a second order effect.

5.2 OPTIMUM PARAMETERS FOR A (40,4) NEP SYSTEM OVER AN EARTH-MARS SYNODICAL CYCLE

This section of NEP results shows the capability of the (40,4) NEP system design to perform various HTT duration missions at every opposition opportunity throughout an entire Earth-Mars synodical cycle (about 17 years). Another difference in this section is that here POP is required to optimize the Isp value instead of using a fixed input value. A detailed tabulation of the optimization results is presented in the following tables, one for each opportunity in the cycle.

For the 12/2011 Opportunity

HTT	393.284	415	450		
DED	15911.07	15909.21	15917.80		
TOUT	177.700	186.952	191.998		
IMEO	608.13	487.949	424.472		
HISP	9239.51	11845.98	12500.0		
Po/α	40/4	40/4	40/4		

For the 1/2014 Opportunity

HTT	377.693	400	450		
DED	16677.06	16682.42	16662.38		
TOUT	172.036	178.599	195.820		
IMEO	576.664	473.663	408.957		
HISP	9087.88	11755.01	11704.17		
Po/α	40/4	40/4	40/4		

For the 3/2016 Opportunity

HTT	351.920	375	450	500	600
DED	17461.44	17463.78	17445.94	17442.81	17436.37
TOUT	150.521	159.026	192.566	209.839	262.106
IMEO	576.191	479.979	389.350	373.980	365.636
HISP	8712.68	11562.10	12485.21	12500.0	12337.92
Po/α	40/4	40/4	40/4	40/4	40/4

For the 5/2018 Opportunity

HTT	337.232	360	450	500	600
DED	18256.64	18256.78	18244.99	18232.53	18219.99
TOUT	132.650	139.945	168.746	183.692	234.245
IMEO	596.977	488.938	383.935	371.391	361.997
HISP	8161.83	10814.83	12481.89	12438.91	12500.0
Po/α	40/4	40/4	40/4	40/4	40/4

For the 7/2020 Opportunity

HTT	379.002	400	450		
DED	19054.95	19061.12	19057.82		
TOUT	145.916	152.106	174.737		
IMEO	542.929	467.359	405.551		
HISP	9992.22	12456.85	12500.0		
Po/α	40/4	40/4	40/4		

For the 9/2022 Opportunity

HTT	394.025	415	450		
DED	19839.79	19837.23	19845.02		
TOUT	162.840	170.467	180.965		
IMEO	641.691	505.436	430.601		
HISP	8519.58	11167.38	12500.0		
Po/α	40/4	40/4	40/4		

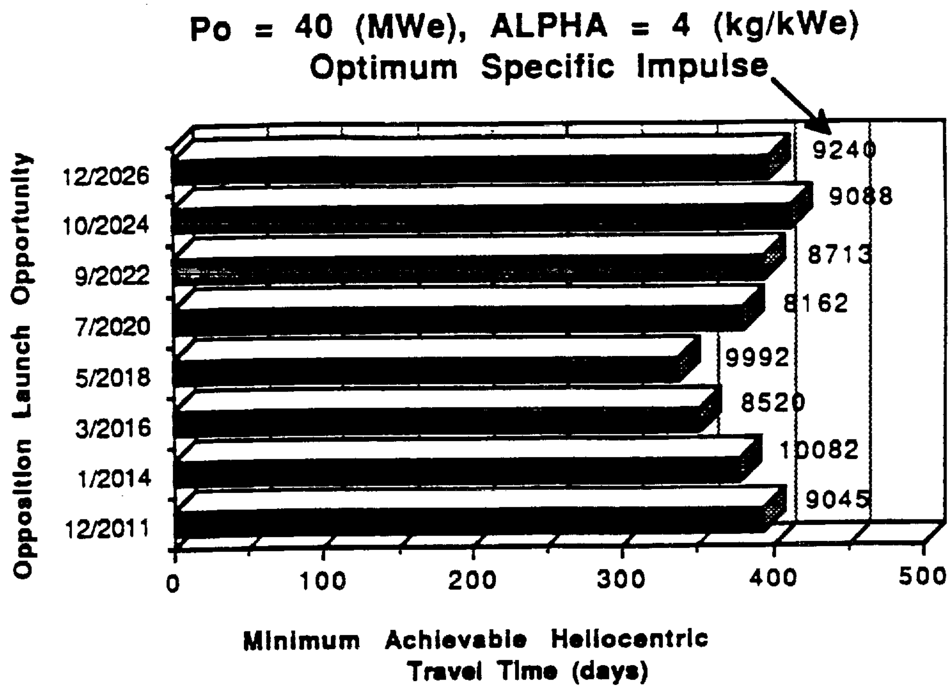
For the 10/2024 Opportunity

HTT	410.990	430	450		
DED	20608.18	20608.66	20603.76		
TOUT	179.339	187.531	200.715		
IMEO	568.192	480.936	440.611		
HISP	10082.04	12424.57	12493.19		
Po/α	40/4	40/4	40/4		

For the 12/2026 Opportunity

HTT	397.610	415	450		
DED	21376.39	21374.78	21376.02		
TOUT	178.784	188.840	206.339		
IMEO	615.834	511.763	432.825		
HISP	9045.30	11097.85	12452.05		
Po/α	40/4	40/4	40/4		

This database of optimum NEP parameters for an entire Earth-Mars synodical period can be used to generate a multitude of interesting plots. The following plot is just one example of the kind of plots that might be of interest. It is clear from the plot that optimum specific impulse values do not form a consistent pattern with minimum achievable HTT. There is most likely a dependence on Earth-Mars distance that is not shown in the plot. (Earth-Mars distance is not included in the database).



5.3 CONTINGENCY OPTIONS FOR A NEP REACTOR FAILURE AT MARS

The Boeing Company raised the question: "How can a mission be planned so that the mission can still be accomplished if one of the reactors goes out at Mars (assuming a dual reactor NEP system)?"

The first option considered was the possibility of carrying enough extra propellant to allow the return leg to be completed with only half of the outbound power, Po. The second option considered was to change the stay time at Mars from 30 days to a different value that would allow the return leg to be completed with the nominal propellant loading. It was somewhat surprising that both options handle the problem with minor changes from the nominal. The following table lists the propellant required and the masses to be dropped for the various trajectory segments.

Using IMEO to handle the problem requires that an extra 1777.8 (kgs) of propellant be carried out to Mars. If the reactor does not fail, then the extra propellant would be offloaded and the nominal return trajectory would be flown. If one of the ractors does fail at Mars, then the extra propellant would be utilized as shown in Column 3 of the table to successfully execute the return trajectory.

Using stay time at Mars, TSTAY, to handle the problem results in the values shown in Column 4 of the table. All of the propellant loadings are at their nominal values, but the stay time is reduced to 28.852 days (instead of 30) which distributes the propellant usage as shown in Column 4. Differences between the two contingency plans and the nominal are shown in Columns 5 and 6.

	NOMINAL VALUES	REACTOR OUT/IMEO	REACTOR OUT/TSTA	DIFF. FOR IMEO	DIFF. FOR TSTAY
INITIAL MASS IN EARTH ORBIT	479898.5	481676.3	479898.5	1777.8	0
EARTH ESCAPE SPIRAL PROP	28027.669	28134.028	28027.669	106.359	0
OUTBOUND HELIO PROPELLANT	65545.948	66349.384	65545.948	803.436	0
MARS CAPTURE SPIRAL PROP	3245.529	3253.583	3245.529	8.054	0
MASS DROPPED AT MARS	84000	84000	84000	0	0
TOTAL OUTBOUND PROPELLANT	96819.146	97736.995	96819.146	917.849	0
OUTBOUND TANKS AND RESERVES	9681.9146	9773.6995	9681.9146	91.7849	0
MARS ESCAPE SPIRAL PROP	2351.625	2535.767	2528.435	184.142	176.81
INBOUND HELIO PROPELLANT	65981.5	66236.571	65550.059	255.071	-431.441
EARTH CAPTURE SPIRAL PROP	12664.554	12923.688	12919.243	259.134	254.689
TOTAL INBOUND PROPELLANT	80997.679	81696.026	80997.737	698.347	0.058
INBOUND TANKS AND RESERVES	8099.7679	8169.6026	8099.7737	69.8347	0.0058
PAYLOAD AT EARTH RETURN	40299.992	40299.977	40299.929	-0.0156	-0.0638

6.0 SOLAR ELECTRIC PROPULSION (SEP) RESULTS

The solar electric propulsion (SEP) system in this analysis differs from the NEP system only in the η(Isp) function, and in the power profile as a function of distance from the sun (power is constant for the NEP system). Both of these are specified for the SEP system by the following equations:

η(Isp) = 80.193*Isp**2/(96.04*Isp**2 + 5.067e8)

P/Po = (1.763 - 0.8865/R + 0.0592/R**2)/[R**2 (1 - 0.1171 R + 0.0528 R**2)]

ALPHA, or α , i.e. specific mass, is assumed to be 10 (kg/kwe) for all these SEP results.

For SEP missions Earth departure and return is assumed to be at a geosynchronous orbit of radius 42241(km); Mars arrival and departure is at a circular orbit radius of 23000 (km).

6.1 OPTIMUM SEP SYSTEMS FOR 2016 OPPORTUNITY MISSIONS

This section presents optimum SEP system designs for performing various HTT duration missions **at the 2016 launch opportunity**. Specific mass is always fixed at 10 (kgs/kwe) for these SEP systems. Detailed optimization results are presented in the following tables (the value with the asterisk, * , is the minimum achievable HTT with that SEP design):

For the Po/α = 10/10 SEP System

HTT	*549.011	600	650	700	
DED	17429.39	17426.76	17410.93	17391.33	
TOUT	237.493	249.244	272.514	300.179	
IMEO	489.382	354.204	352.331	335.492	
HISP	4569.95	5521.95	5023.71	5527.80	
Po	10000	10000	10000	10000	

For α = 10, With Optimum Po and Isp SEP System

HTT	520	549	570	600	650
DED	17442.44	17434.35	17430.22	17425.44	17410.72
TOUT	214.211	232.164	240.661	255.401	280.790
IMEO	578.197	492.843	372.044	319.656	297.859
HISP	5597.12	4191.12	5931.11	6328.13	4883.08
Po	18212.79	9919.88	9611.80	7644.50	4424.60

Figures 6 throught 10 are for these SEP systems performing missions for the 2016 launch opportunity. Figure 6 shows the minimum IMEO required for the SEP

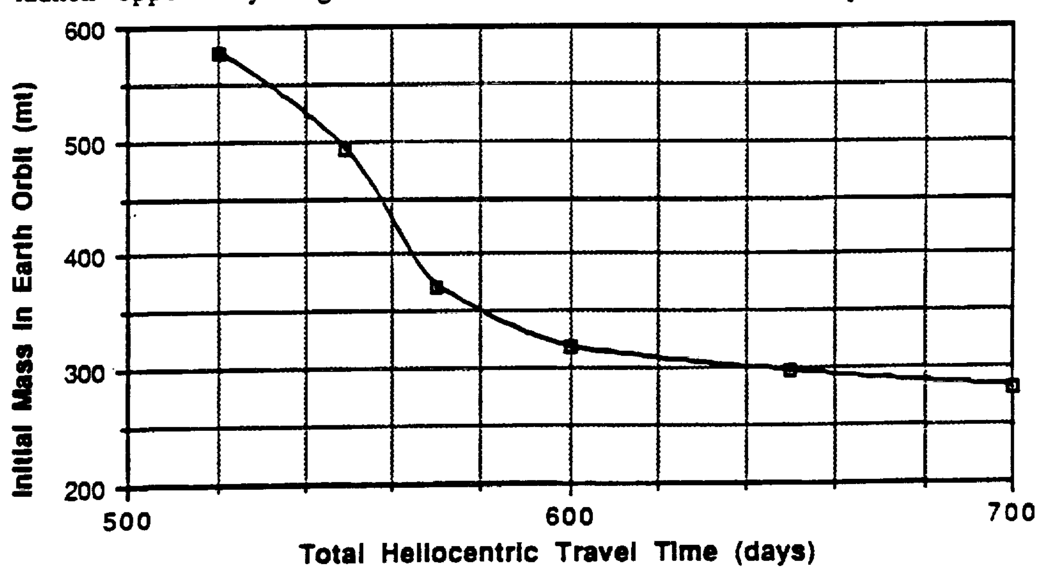


Figure 6. Minimum Initial Mass in Earth Orbit for SEP System to Perform Various HTT Missions With Optimum Po and Isp

system to fly various HTT duration missions, with both the initial power level, Po, and Isp values optimized.

Figures 7 and 8 are companion charts that show optimum Po and Isp values associated with the HTT missions shown in Figure 6.

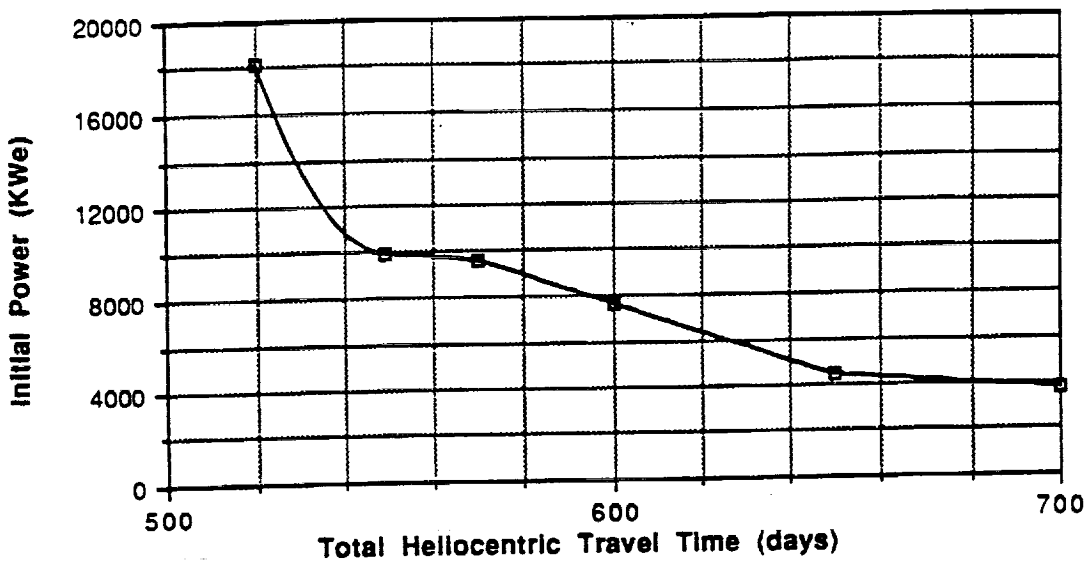


Figure 7. Optimum Initial Power Values for Missions Having Various Heliocentric Travel Times (HTT)

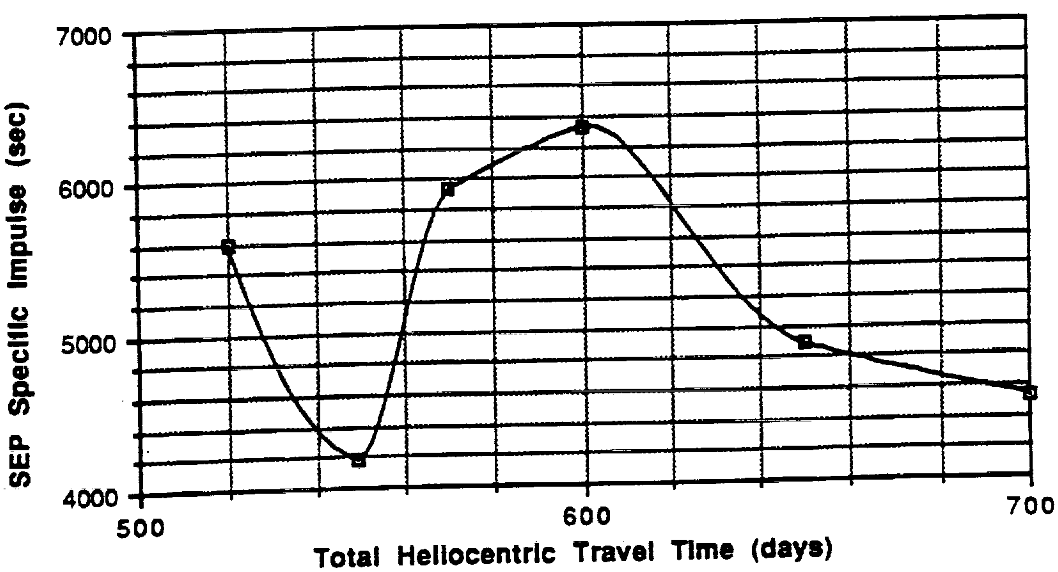


Figure 8. Optimum Specific Impulse Values for Missions Having Various Heliocentric Travel Times (HTT)

Figure 8 exhibits an optimum Isp value for HTT = 549 days that appears to be inconsistent with all of the other values. This problem has not been analysed further to determine what causes the inconsistency.

Similarly, Figures 9 and 10 are companion charts that show optimum Earth departure date (DED) and optimum outbound heliocentric trip time (TOUT) for the same missions shown in Figures 6, 7, and 8.

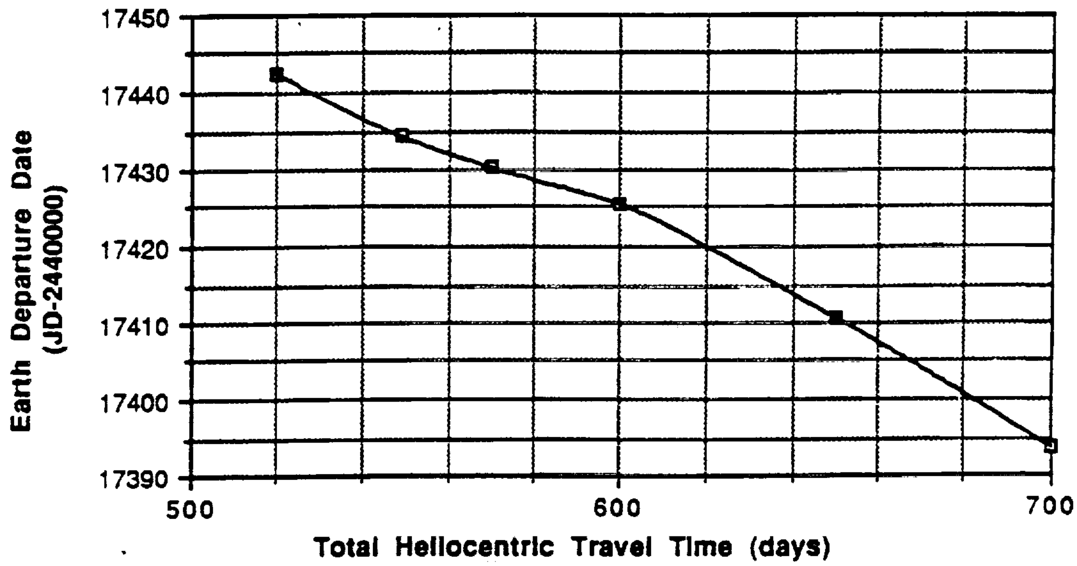


Figure 9. Optimum Earth Departure Dates for Missions Having Various Heliocentric Travel Times (HTT)

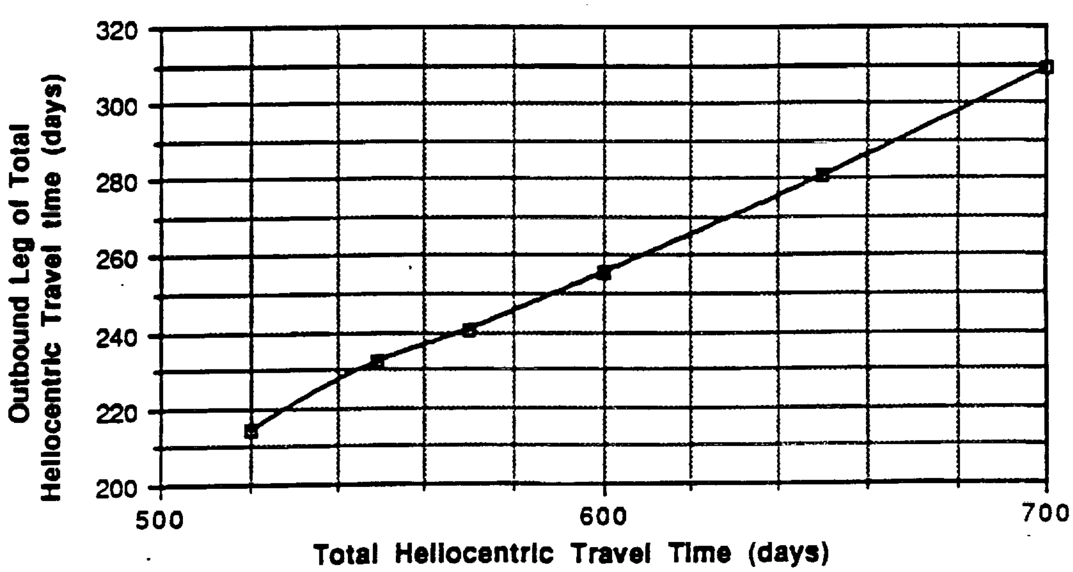


Figure 10. Optimum Outbound Trip Time for Missions Having Various Heliocentric Travel Times (HTT)

6.2 LOW EARTH ORBIT (LEO) TO GEOCENTRIC EARTH ORBIT (GEO) TRANSFERS

The Boeing Company suggested the possibility of making the LEO to GEO transfer with a disposable solar array. This would allow the array to be discarded at GEO due to expected damage caused by passage through the Van Allen radiation belt. Boeing estimated the mass of the disposable array to be about 28000 (kgs).

Relationships developed in Reference 5 are used to (1) estimate the mass required in LEO to transfer a specified mass to GEO, and (2) the time required to accomplish that transfer. Thus, the IMEO requirements presented earlier in this survey for the SEP system to perform various missions of HTT duration would become the specified mass to be transferred to GEO. The computational procedure for this LEO to GEO transfer estimation is as follows:

$$m_w = P_o\,\alpha \text{ (power plant mass)}$$

$$m_{pld} = m_{geo} - m_w \text{ (payload mass for the transfer)}$$

$$m_{st} = 28000\text{(kgs) (structural mass for the ...)}$$

$$m_f = m_{pld} + m_{st} \text{ (final mass for the ...)}$$

$$R = \frac{m_w}{m_f}$$

$$\gamma = \frac{R}{1+R} = \frac{m_{pr}}{m_{leo}} \text{ (ratio of propellant mass to mass in LEO)}$$

$$\Delta V = V_{c_{leo}} - V_{c_{geo}} \text{ (transfer velocity required)}$$

$$V_c = \frac{\Delta V}{\gamma} \text{ (characteristic velocity)}$$

$$m_{leo} = \frac{m_w}{(\gamma - \gamma^2)} \text{ (mass required in LEO)}$$

$$T = \frac{V_c^2\,\alpha}{2000\,(86400)} \text{ (time required ..days)}$$

The following tables list detailed results of a parametric survey showing the mass required in LEO to transfer desired quantities of mass to GEO, and the time (in days) required to accomplish that transfer, using various power levels.

Mass Required in LEO to Transfer Desired Mass (mgo) to GEO

Po/mgo	250	300	350	375	400	425	450	500	550
1	288.37	338.31	388.27	413.25	438.24	463.23	488.21	538.19	588.18
2	299.55	349.30	399.12	424.04	448.98	473.92	498.87	548.79	598.72
3	311.63	361.02	410.59	435.41	460.26	485.13	510.01	559.81	609.64
4	324.72	373.55	422.73	447.41	472.12	496.87	521.65	571.28	620.97
5	338.97	386.99	435.62	460.08	484.61	509.20	533.84	583.23	632.73
6	354.51	401.43	449.32	473.50	497.78	522.16	546.61	595.69	644.95
7	371.56	416.99	463.91	487.71	511.69	535.79	560.01	608.70	657.65
8	390.32	433.81	479.48	502.81	526.39	550.16	574.08	622.29	670.85
9	411.09	452.03	496.13	518.88	541.96	565.31	588.88	636.49	684.60
10	434.18	471.86	513.97	536.00	558.49	581.33	604.45	651.36	698.92

Days Required to Transfer Desired Mass (mgo) to GEO

Po/mgo	250	300	350	375	400	425	450	500	550
1	946.78	1318.0	1750.4	1989.6	2244.1	2514.0	2799.1	3415.3	4092.8
2	236.70	329.49	437.61	497.41	561.03	628.49	699.77	853.82	1023.2
3	105.20	146.44	194.49	221.07	249.35	279.33	311.01	379.48	454.75
4	59.17	82.37	109.40	124.35	140.26	157.12	174.94	213.46	255.80
5	37.87	52.72	70.02	79.59	89.77	100.56	111.96	136.61	163.71
6	26.30	36.61	48.62	55.27	62.34	69.83	77.75	94.87	113.69
7	19.32	26.90	35.72	40.61	45.80	51.31	57.12	69.70	83.53
8	14.79	20.59	27.35	31.09	35.06	39.28	43.74	53.36	63.95
9	11.69	16.27	21.61	24.56	27.71	31.04	34.56	42.16	50.53
10	9.47	13.18	17.50	19.90	22.44	25.14	27.99	34.15	40.93

Figures 11 and 12 show plots of the parametric survey tabulated above.

Figure 11 shows the mass required to transfer various desired mass values from a geocentric circular orbit of radius 6770(km) to a geosynchronous orbit of radius 42241(km), using various power levels, and Figure 12 shows the time required to accomplish the same transfers.

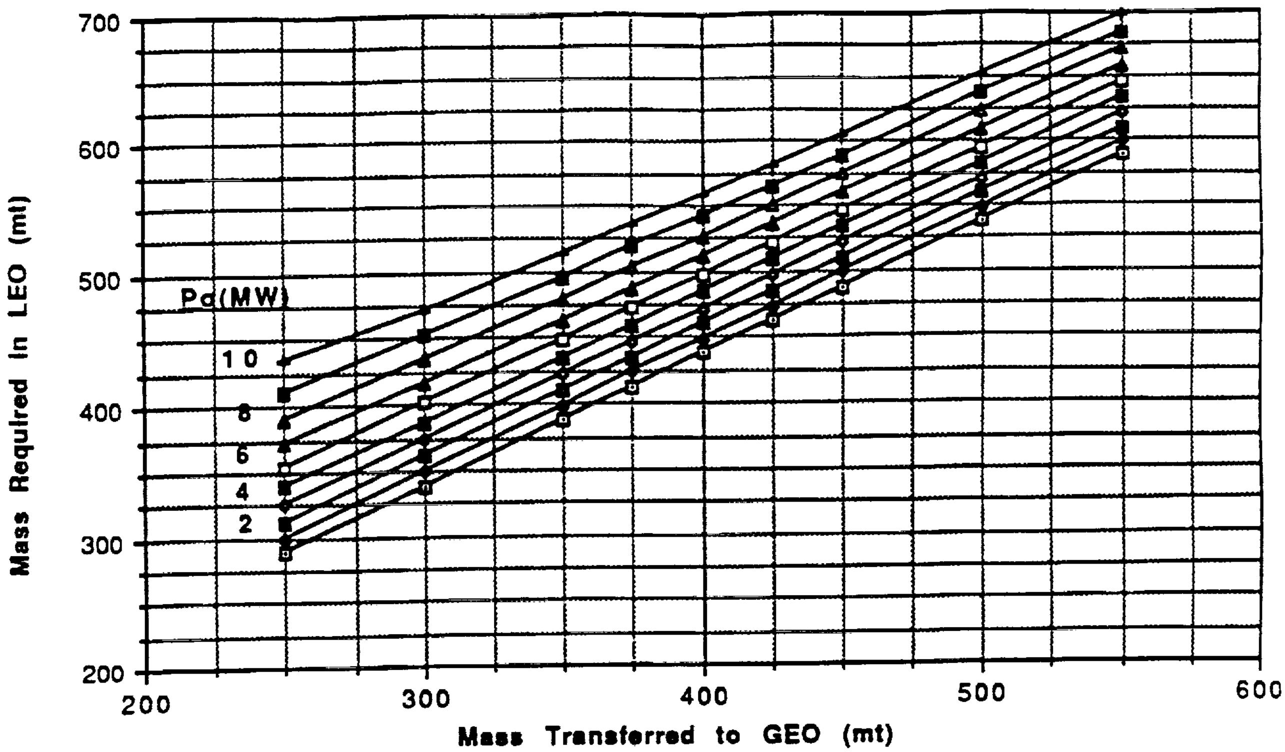


Figure 11. Orbit Transfer Mass Requirements for SEP System Using a Disposable Solar Array

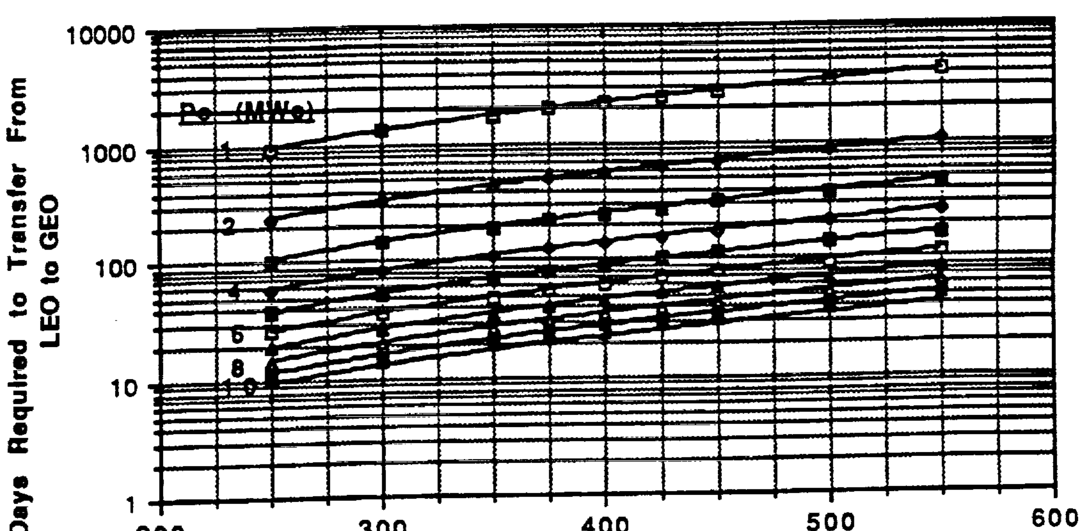


Figure 12. Orbit Transfer Time Requirements for SEP System Using a Disposable Solar Array

Figures 11 and 12 provide the user with a means of trading the time required to transfer various mass values from LEO to GEO" with the initial mass required in LEO to accomplish the transfer, using various SEP power levels. Reference 5 assumes a constant acceleration in deriving the estimating ralationships.

As a specific example, assume that a total manned trip of 600 days is desired. This implies HTT = 570 days (HTT = 600 - TSTAY). Figure 6 shows that the minimum IMEO required at GEO is about 375(mt), Figure 7 shows the optimum Isp value is about 5925 (sec), and Figure 8 shows the optimum Po value is about 9.6(MW). Now, the LEO to GEO transfer is not required to use the same Po value as the interplanetary phase. Thus, we can still trade Po values to get required IMLEO and time to make the transfer. Suppose that it is desired that the IMLEO be no more than about 450 (mt). Figure 11 shows that a Po value of about 4(MW) requires about 450(mt) in LEO to transfer 375(mt) to GEO, and Figure 12 shows that it takes about 125(days) to make the transfer.

8.0 REFERENCES

1. Williams, D.F., and W.B. Tucker, "*Computation of Quasi-Optimal Reentry Trajectories Using The SIMPLEX Algorithm of Linear Programming,*" M-240-1208, Northrop Services Inc. Huntsville, Alabama, April 1973 (UNCL)

2. Ragsac, R.V., "*Study of Trajectories and Upper Stage Propulsion Requirements for Exploration of the Solar System,*" F-910352-13, Volume II: Technical Report, Final Report, United Aircraft Corp., East Hartford, Connecticut, September 1967 (UNCL)

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4. Tucker, W.B., "*Some Efficient Computational Techniques Including Their Application to Time Optimal Trajectories From Parking Orbit,*" NASA TN D-2691, George C. Marshall Space Flight Center, Huntsville, Al., March 1965 (UNCL)

5. Keaton, Paul W., "*Low-Thrust Rocket Trajectories,*" LA-10625-MS, Rev., Los Alamos National Laboratory, Los Alamos, New Mexico 87545 (UNCL)

Performance Parametrics

Note: Contains material formerly in Mission Analysis

[redacted]

Propulsion Option Comparison Assumptions

Chemical/AB

- Expendable - ECCV return
- Isp = 475 sec
- AB weight = 10 % for comparison

NTR-NERVA

- Expendable - ECCV return
- Isp = 925 sec, Tc=2700 K, Composite , Pc = 1000, nozzle AR = 500:1
- Engine T/W = 3.5
- No shield (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)
- No shield (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

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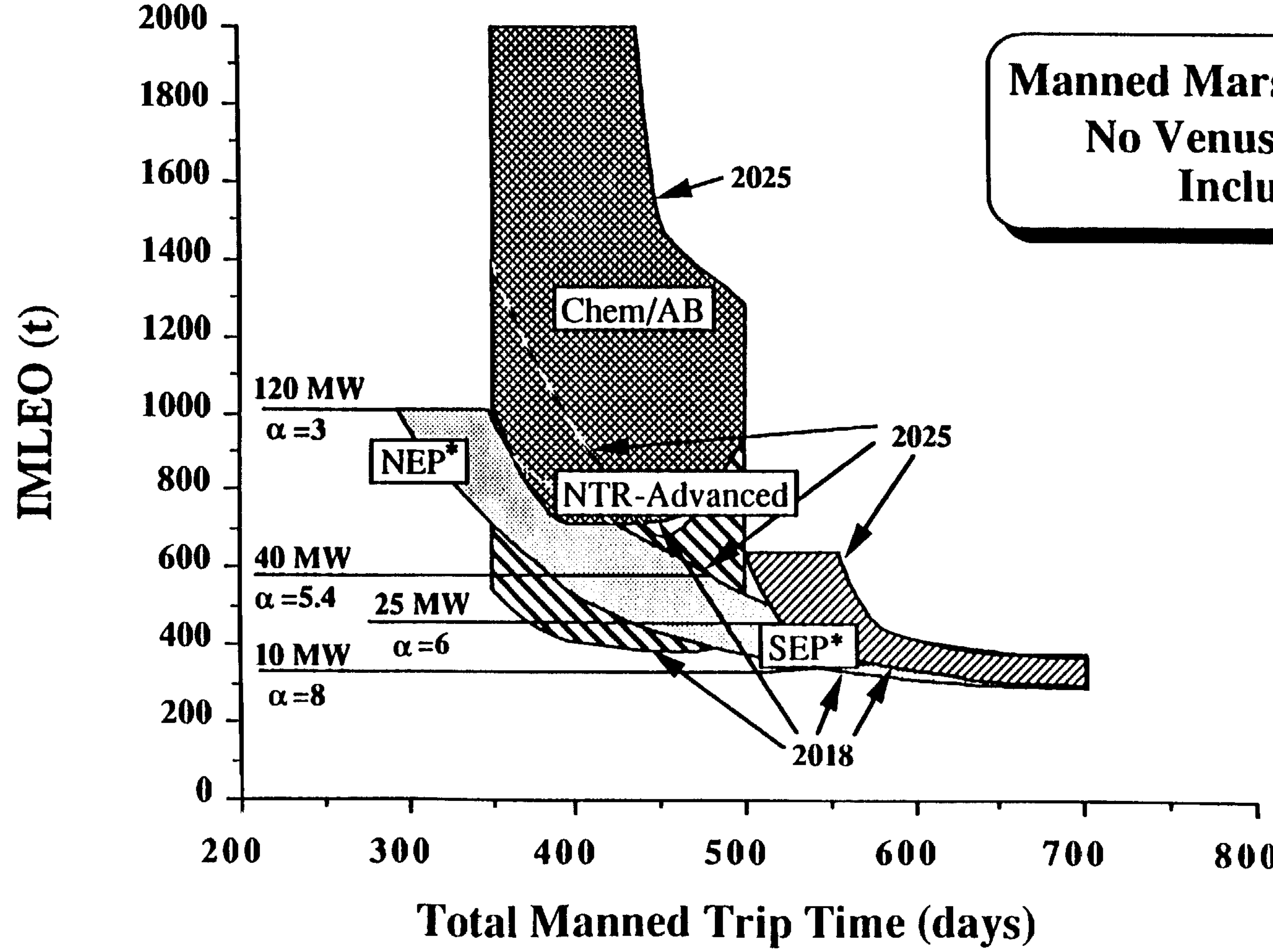
Propulsion Option Comparison for Opposition Missions

Comparison with a 120 t Payload Opposition Opportunities



* Reusable Vehicles

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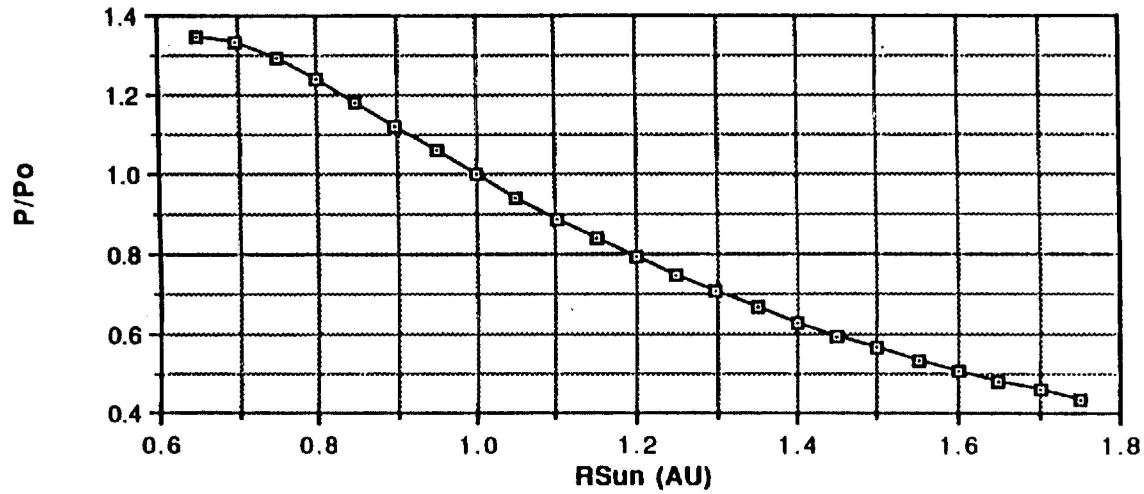
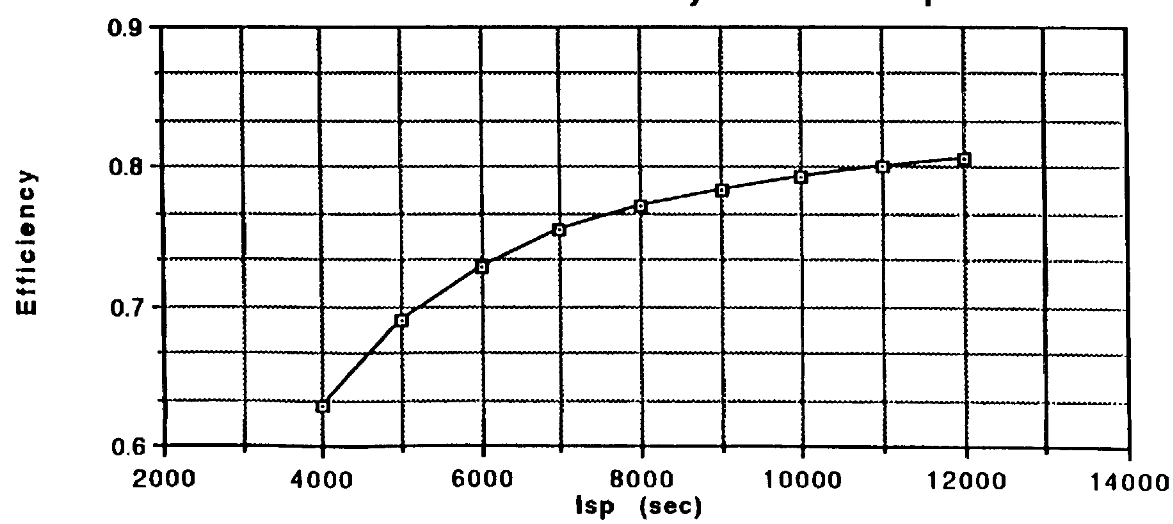
SEPS Performance Parameters

SEPS Power Ratio as Function of Distance From the Sun



Ion Thruster Efficiency for Various Isp



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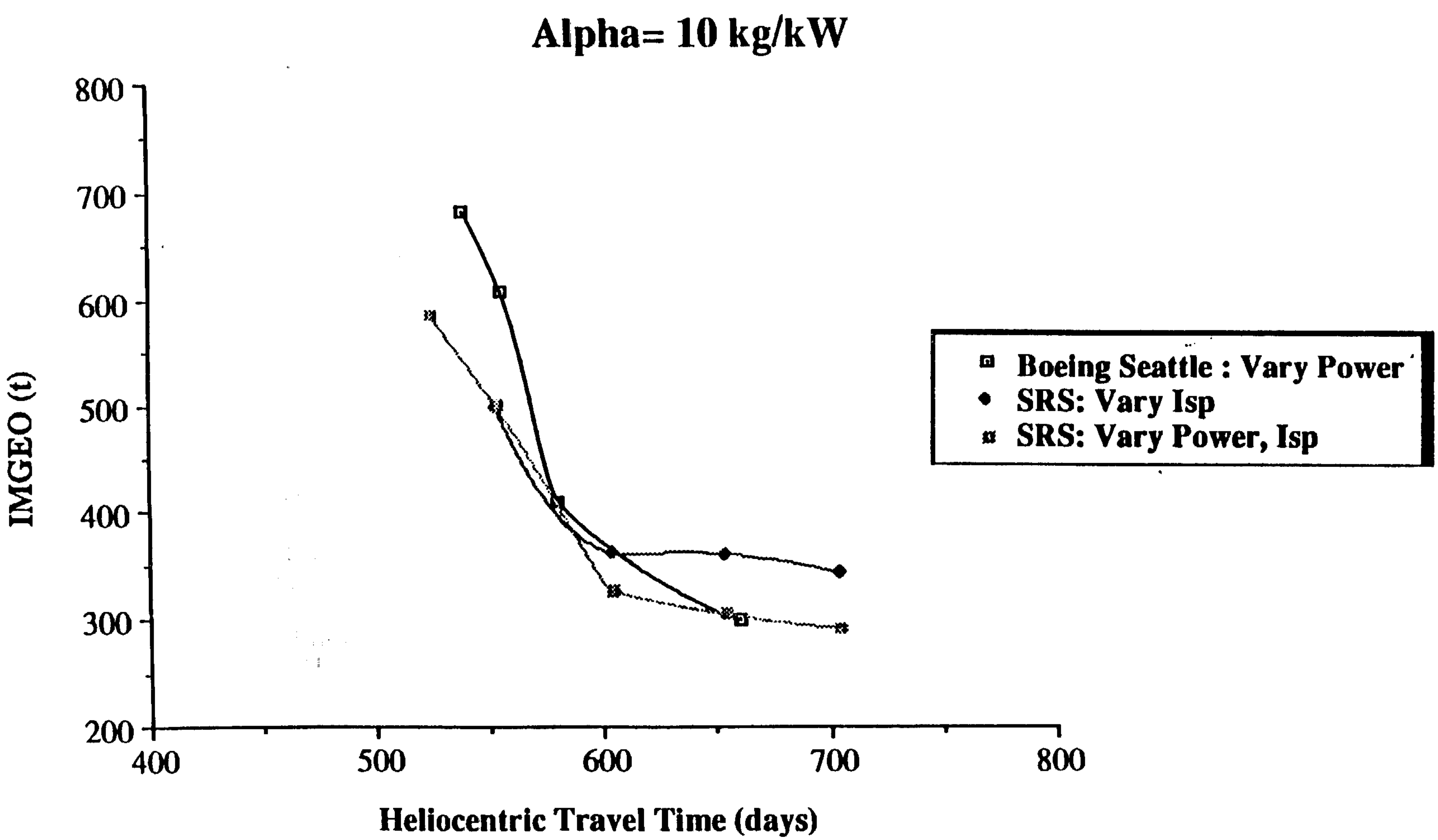
Mission Parameters For Various SEP Vehicles

The SEP mission analysis plot shows initial mass at GEO vs heliocentric travel time (HTT). Heliocentric travel time for electric propulsion is defined as total manned trip time minus Mars stay time. Our results show that for a HTT of 570 days, a power level of 10 MW is optimum. Increasing the HTT will allow for a lower power level which also decreases the initial mass. Decreasing the HTT results in increasing the power level. Converting IMGEO to IMLEO, requires weight for a chemical booster stage.

Mission Parameters For Various SEP Vehicles

BOEING

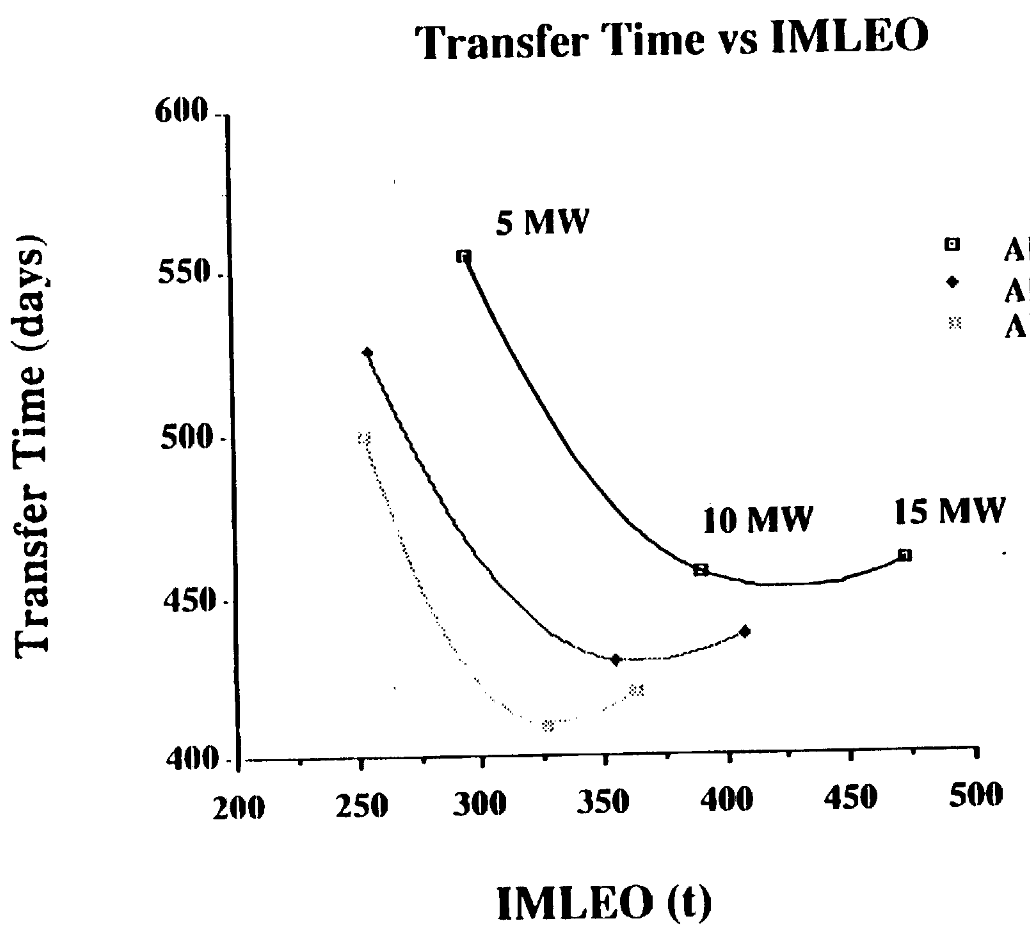


* Does not include planetary swing-bys

SEP Conjunction Power Trades

The following chart depicts crew transfer time versus IMLEO for different vehicle alphas. The power was varied from 5 to 15 MWe for each of the vehicles. An interesting finding is that the transfer time actually increases from 10 MWe to 15 MWe for a constant vehicle alpha. The reference vehicle design yields a vehicle alpha of 10 kg/kW. The major assumptions for this analysis can be found on the chart. The key point of this chart is that a power level of 10 MW is the upper limit for a low thrust SEP vehicle design. The 10 MW power level is in agreement with previous opposition class analyses. This common power level between the conjunction and opposition missions, yields a more flexible propulsion option.

SEP Conjunction Power Trades



Assumptions:

- Mars stay time of 600 days
- Isp= 5,000 sec
- Outbound payload of 116 t
- Inbound payload of 43 t
- 2016 Opportunity
- Power varied from 5 MW to 15 MW

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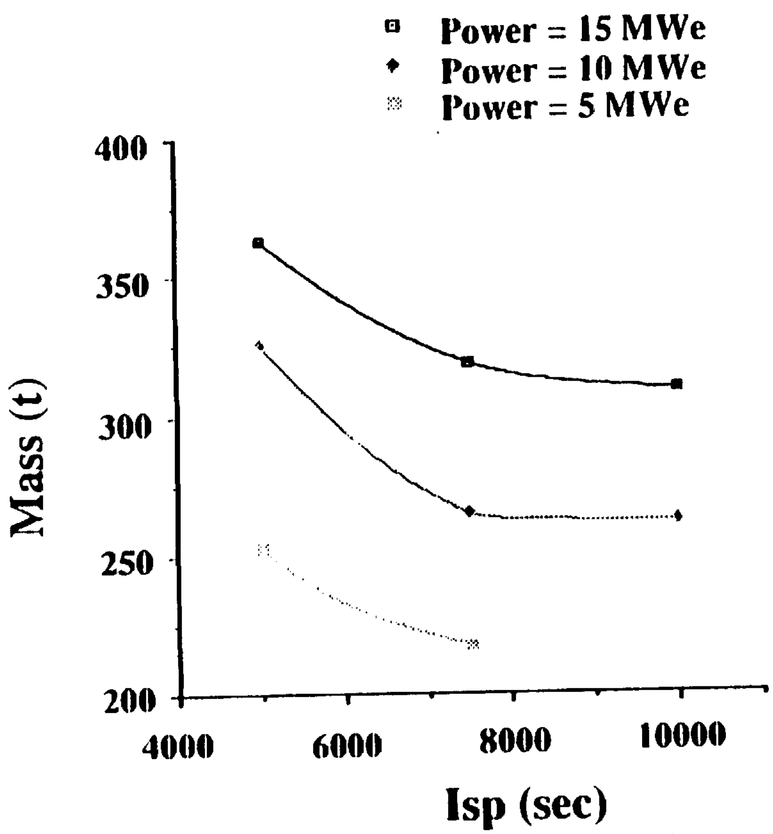
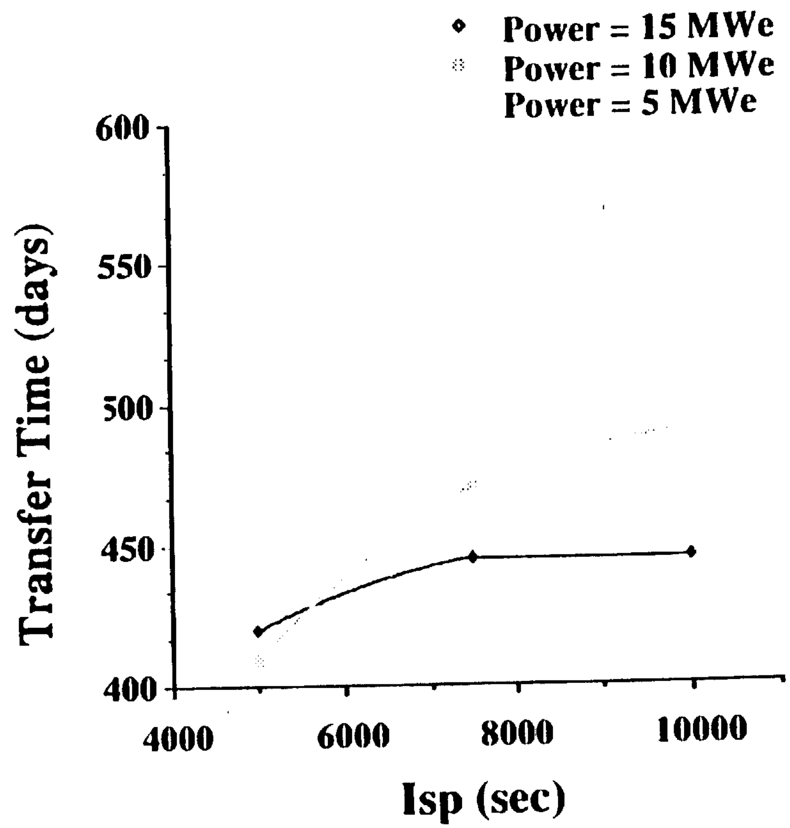
SEP Conjunction Isp Trades

The following chart shows the trade-off between initial mass and transfer time over the Isp variation. Three different power levels are shown with 10 MW being the baseline power level. It can be observed that increasing the Isp above 7,500 sec does not decrease initial mass, but does increase transfer time. The optimum power level resides in the 5,000 to 7,500 sec range. For the remainder of the study an Isp of 6,500 was chosen as a good compromise between low initial mass and reasonable trip times. Factors that could affect this choice are cost of delivering the mass to orbit, feasibility of large structures for higher power levels, and human tolerance to extended time in space.

SEP Conjunction Isp Trades





Assumptions:

- Mars stay time of 600 days
- Outbound payload of 116 t
- Inbound payload of 43 t
- 2016 Opportunity
- Power varied from 5 MW to 15 MW

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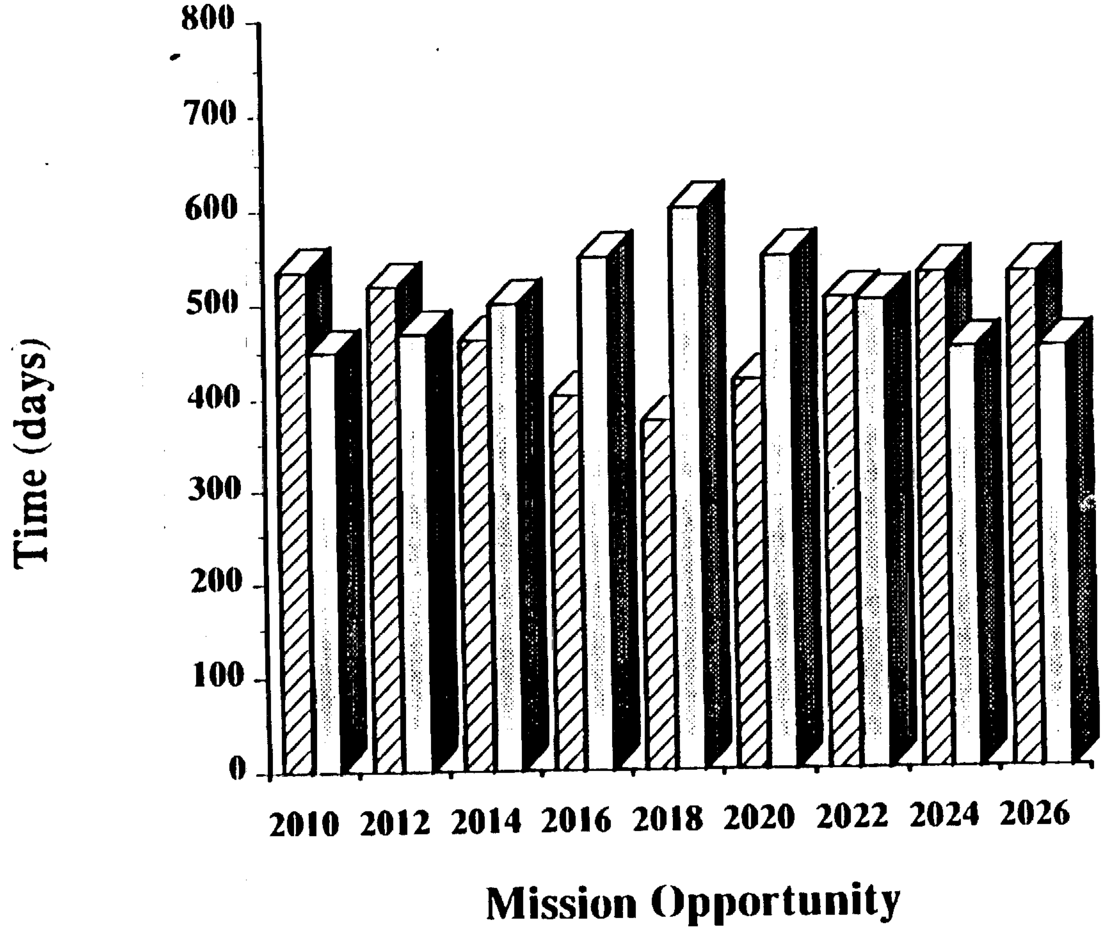
SEP Conjunction Various Opportunities

Using the baseline vehicle and trajectory, a trade was performed in which the year of the opportunity was varied through the entire Earth-Mars opportunity cycle. When arrival and departure from Mars occurs near the apoapsis of the Martian orbit, Mars is further away from Earth and is traveling slower. Both of these factors require a corresponding increase in necessary total ΔV for low thrust vehicles for the same trajectory geometry. As a result, for a low thrust mission, longer trip times and higher IMLEO are required for some of the conjunction missions. The SEP vehicles cannot make up for higher energy requirements by increasing the propellant mass available, so shortening the stay time is used as a way of maintaining relatively efficient paths on the "more difficult" opportunities.

SEP Conjunction Various Opportunities



Assumptions:

- Outbound payload of 116 t
- Inbound payload of 43 t
- Power = 10 MWe
- Alpha = 10 kg/kW
- Isp = 6500 sec

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SEP Missions Selection Criteria

Criteria	Rationale
1. Total IMLEO	• Missions with less IMLEO are favored.
2. Orbital Debris Shield Mass	• Missions operating in LEO will require greater debris shielding
3. Mass @ Spiral Initiation	• Missions where payload and main vehicle are integrated separately may impact this parameter.
4. Mass @ Departure	• Total mass at departure.
5. Isp (transfer stage)	• Isp directly correlates with propellant mass.
6. Spiral Time	• The less the spiral time, the less the array degradation, the lower the probability of debris hits, the greater the mission efficiency.
7. DeltaV	• Mission with less total deltaV is favored.
8. Days Exposure to Radiation	•The greater the time the SEP spends in the Van Allen belts, the greater the degree of solar array degradation.
9. Days Exposure to Orbital Debris	•The greater the time the SEP spends in LEO, the damage to the solar array due to orbital debris.
10. Total Mission Time	• Driven by assembly time and spiral time.
11. Resiliency	• Time required to recover from failure mode. Higher nodes require greater time to recover from a failure.
12. Infrastructure Cost	• Assembly/departure node locations may drive mission costs
13. Infrastructure Complexity	• Missions requiring extensive space based support hardware, are more complex and are subject to increased chances of failure.
14. % Degradation of Solar Array	• Each mission will result in varying degrees of array degradation.
15. Total HLLV Flights	• Certain mission options may result in a less number of total HLLV assembly missions.
16. Reusability of Used Hardware	• Transfer array on certain mission options may be reused for power beaming applications.
17. GCR Exposure to crew	• Operating in higher nodes increases amount of GCR received by crew during assembly and checkout.
18. Flight Proven Technology	• An EOTV provides a proving ground for SEP technology prior to mission.

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SEP Mission Options Trade

Criteria	SEP Mission Options												
	Option 1		Option 2		Option 3		Option 4		Option 5		Option 6		Weights
1. Total IMLEO	5	4.15	5	4.15	1	.83	4	3.32	3	2.49	3	2.49	.83
2. Orbital Debris Shield Mass	1	.56	1	.56	5	2.80	5	2.80	1	.56	2	1.12	.56
3. Mass @ Spiral Initiation	2	.66	2	.66	3	.99	3	.99	3	.99	2	.66	.33
4. Mass @ Departure	3	.84	1	.28	3	.84	3	.84	3	.84	3	.84	.28
5. Isp (transfer stage)	5	3.60	5	3.60	1	.72	5	3.60	5	3.60	3	2.16	.72
6. Spiral Time	1	.50	2	1.00	5	2.50	5	2.50	5	2.50	4	2.00	.50
7. DeltaV	5	3.90	5	3.90	2	1.56	1	.78	5	3.90	4	3.12	.78
8. Days Exposure to Radiation	1	.39	2	.78	5	1.95	5	1.95	5	1.95	4	1.56	.39
9. Days Exposure to Orbital Debris	1	.44	1	.44	5	2.20	5	2.20	1	.44	2	.88	.44
10. Total Mission Time	3	1.83	1	.61	5	3.05	2	1.22	1	.61	4	2.44	.61
11. Resiliency Time	5	4.10	5	4.10	1	.82	1	.82	5	4.10	4	3.28	.82
12. Infrastructure Cost	4	3.80	5	4.75	3	2.85	2	1.90	3	2.85	4	3.80	.95
13. Infrastructure Complexity	4	.68	5	.85	2	.34	1	.17	2	.34	3	.51	.17
14. % Degradation of Solar Array	5	3.35	4	2.68	5	3.35	5	3.35	5	3.35	2	1.34	.67
15. Total HLLV Flights	5	4.45	5	4.45	1	.89	4	3.56	3	2.67	3	2.67	.89
16. Reusability of Used Hardware	3	.03	1	.01	1	.01	5	.05	5	.05	1	.01	.01
17. GCR Exposure to Crew	3	.33	5	.55	1	.11	1	.11	3	.33	4	.44	.11
18. Flight Proven Technology	1	.06	1	.06	1	.06	5	.30	5	.30	1	.06	.06
Total Scores	33.67		33.43		25.87		30.46		31.87		29.38		

1 unfavorable 2 3 average 4 favorable 5

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SEP Conjunction Key Findings

- The baseline vehicle for this study was determined to be a 10 MW vehicle at 10 kg/kW
- The baseline vehicle has a transfer time of 430 days and an IMLEO of 355 t for the reference 2016 opportunity
- An Isp of 6,500 seconds was chosen as a good compromise between low initial mass and reasonable trip times
- The Mars staytimes optimized between 450 and 600 days for the entire Earth-Mars synodical cycle
- The transfer times for the reference vehicle ranged from 372 to 535 days over the entire cycle
- Different power curves (P/Po) were analyzed for concentration ratios of 1-4. This analysis resulted in small deviations from the reference case.

Spiral Time Analysis for Earth Spiral

Transfer time required in days is shown for corresponding initial masses at GEO for the different power levels. The referenced transfer time is the time it takes the SEP vehicle to spiral from LEO to GEO with its own propulsion system. This analysis was performed to determine the time penalties associated with the transfer array scenario (TAS). The transfer array scenario was developed to eliminate the heavy chemical boost stage necessary to transfer the vehicle from LEO to GEO. The TAS will provide a LEO mass savings benefit on the order of 200t. Once the SEP vehicle reaches GEO or a higher orbit, it can drop the arrays of to be used for a lunar power beaming platform or other uses. The arrays will experience roughly a 35% degradation due to the spiral through the van Allen belts. Some issues to be resolved are the amount of debris and radiation damage the vehicle will experience while traversing the belts. A power level of 5 MWe will transfer the vehicle in less than 100 days.

ADVANCED CIVIL SPACE SYSTEMS

Spiral Time Analysis for Earth Spiral

BOEING

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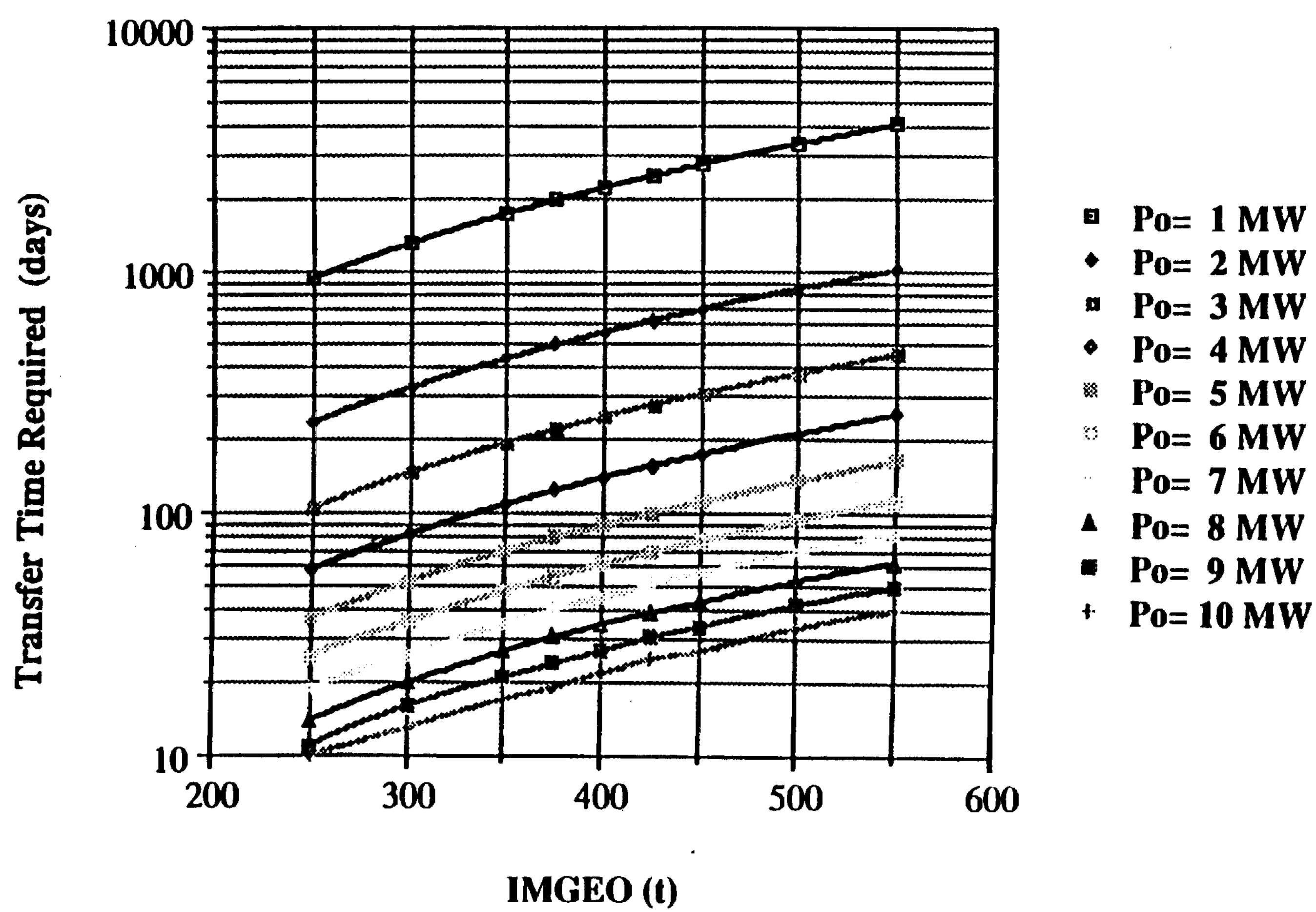
LEO to GEO Electric Spiral Analysis



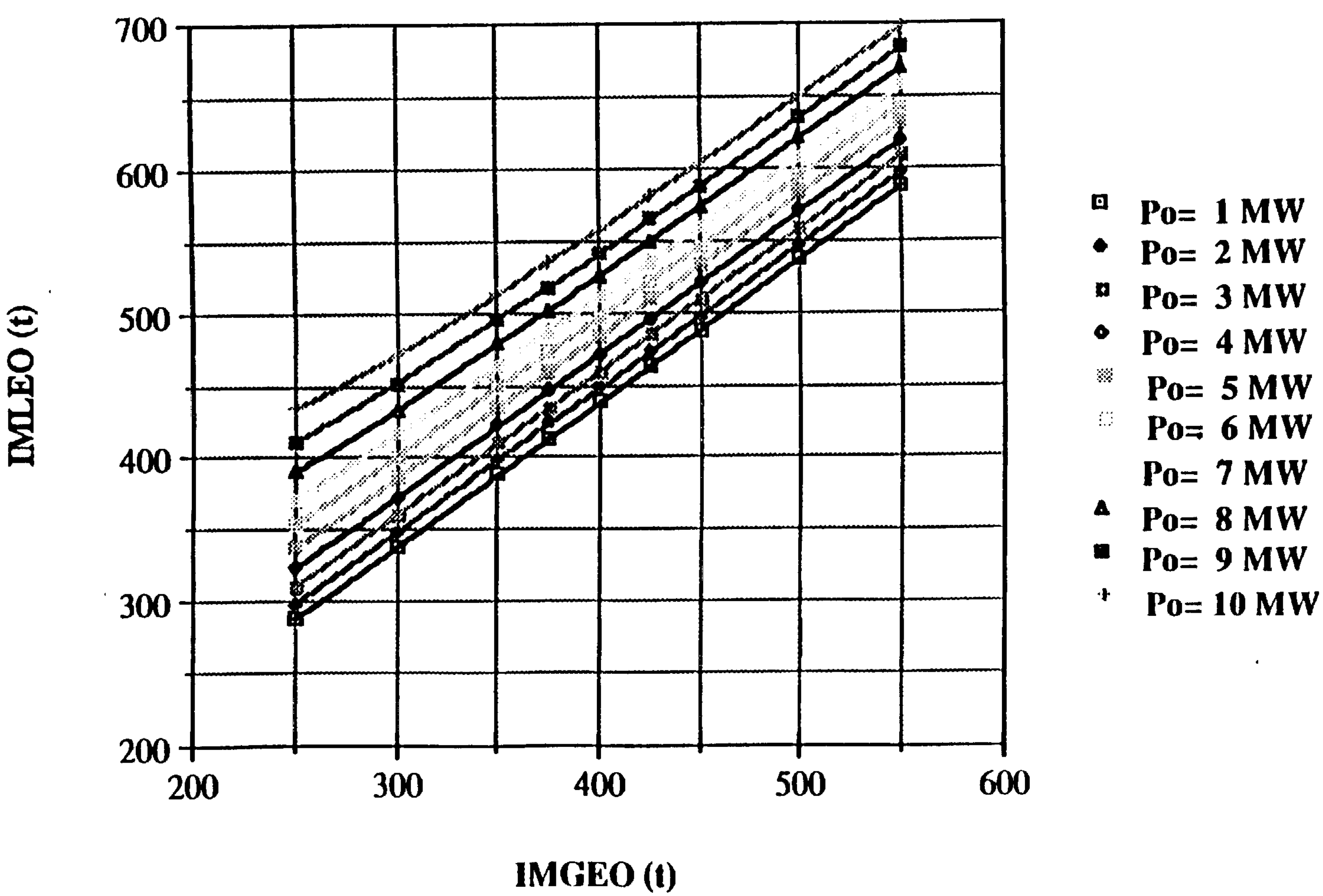
Solar Array Mass Trade for Earth Spiral

The following graph shows the equivalent mass in LEO for the transfer array scenario vs. what the vehicle weighs at GEO. This weight in GEO would be the same if a chemical stage had boosted the vehicle from GEO. The trade was performed for different power levels to determine the most advantageous solution. To determine the correct power level, one must also take into account the time associated with the power level.

Solar Array Mass Trade for Earth Spiral

STCAEM/brc/9Feb90

LEO to GEO Electric Spiral Analysis



Mass of Expendable Array Scenarios vs. Reference Cases

The bar graph is a summary of the two preceding charts containing data on the spiral times and LEO mass for a given mass at GEO. From these charts it seems that a power level between 3 and 6 MWe would provide the best combination of weight savings and transfer times. For a reference point, 5 MWe has been chosen.

Mass of Expendable Array Scenarios Vs. Reference Cases

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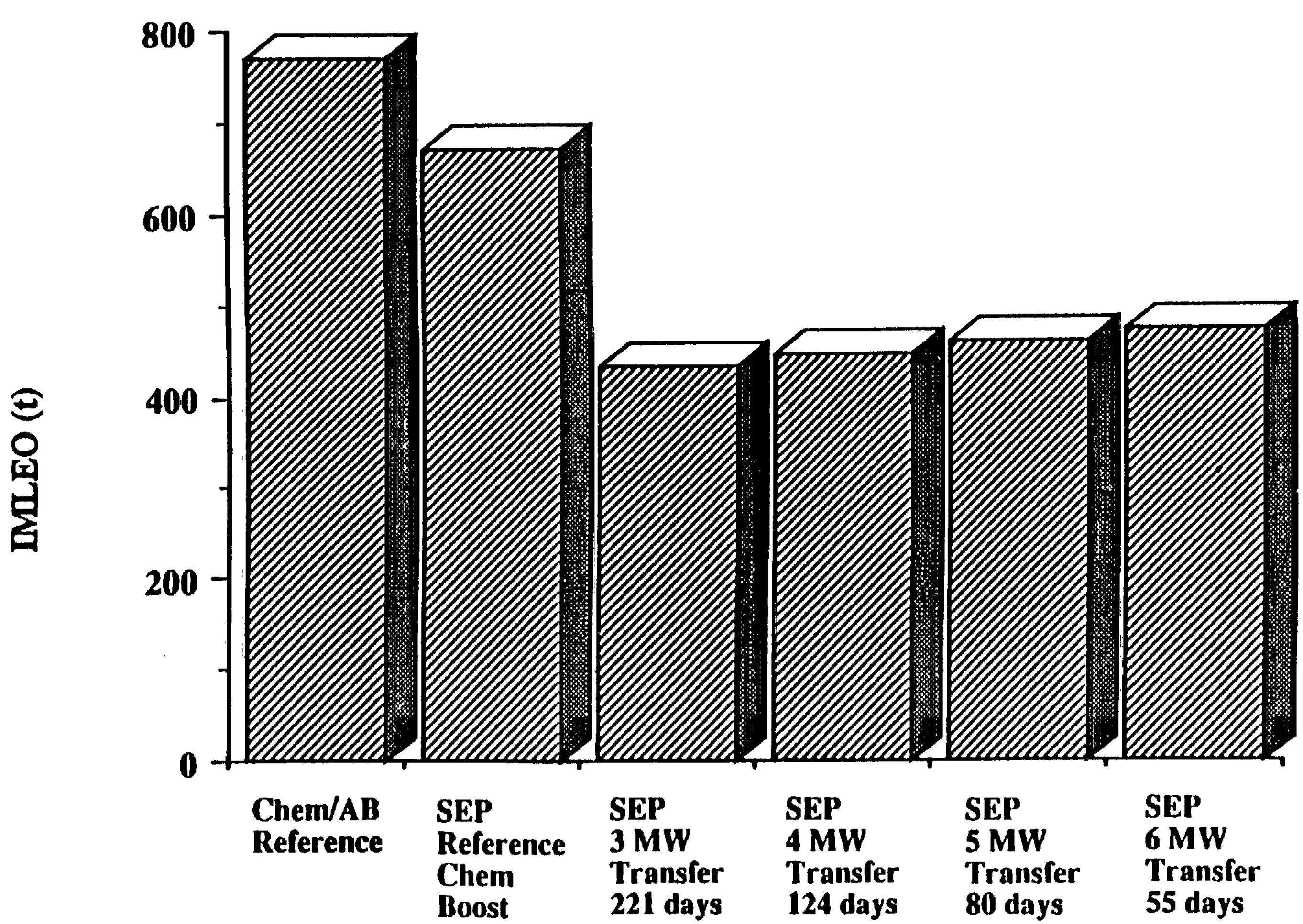
Assumes a 10 MW, 375 t (@ GEO) SEP Vehicle



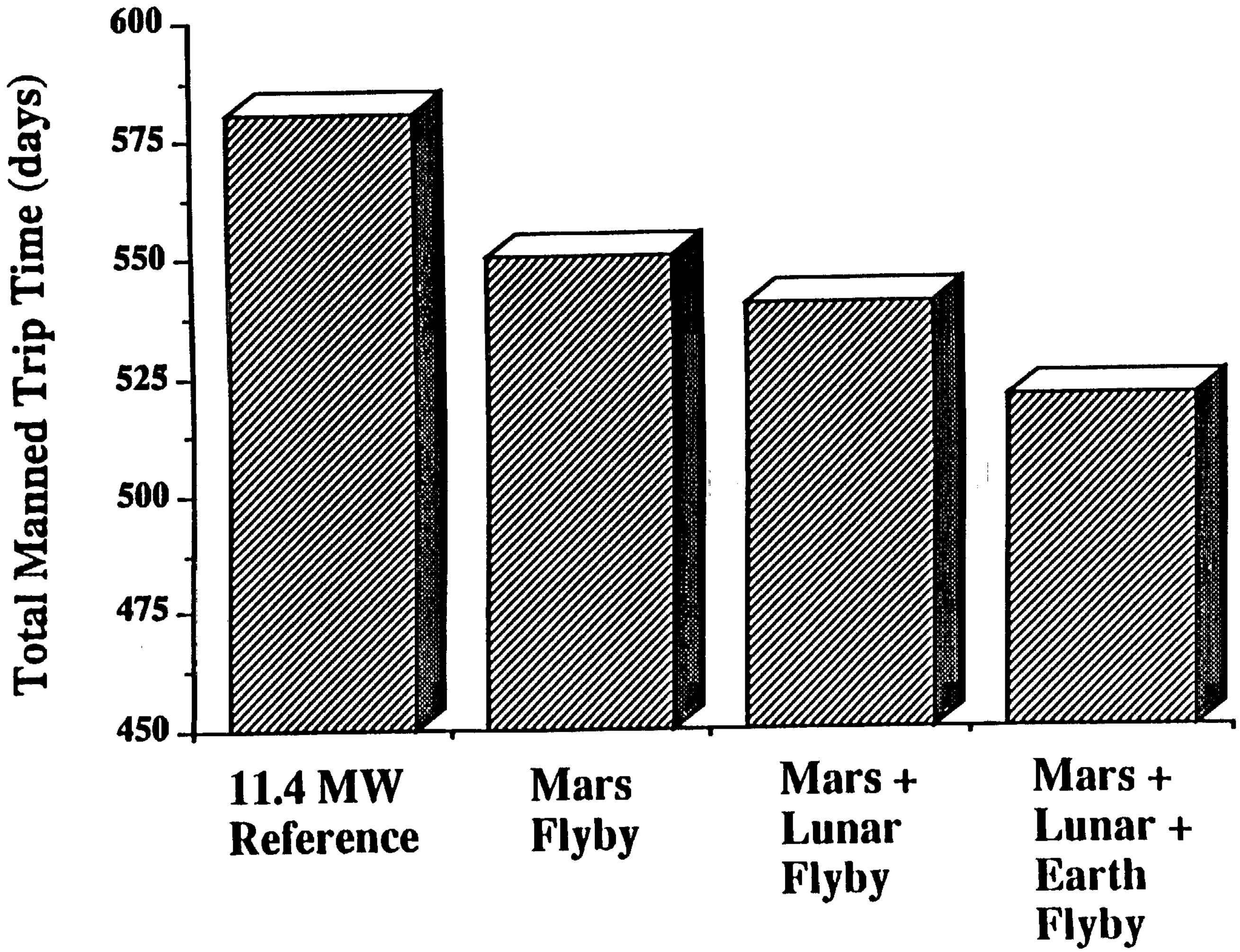
SEP Travel Time vs. Mission Type

The purpose of the swingby mission analysis was to decrease manned trip time for electric propulsion vehicles. Three different swingbys were analyzed that showed favorable results. A lunar, Earth, and Mars swingby showed preliminary benefits of trip time savings. A Venus swingby opportunity has not been found that would provide benefits for a low-thrust vehicle at this time.

The following graph shows a reference SEP vehicle and corresponding trip time for comparison purposes. The advantages or gains of the three different swingbys can be seen. A total manned trip time of 520 days (for the given vehicle) can be obtained, when all swingbys are employed.

SEP Travel Time vs. Mission Type

Time Comparison for SEP Missions Using Swingbys



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Levied Requirements

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Solar Electric Propulsion (SEP) - System Requirements

During the course of the Space Transfer Concepts and Analyses for Exploration Missions contract (STCAEM), Boeing's Advanced Civil Space Systems group (ACSS) has conducted regular review meetings in order to define and derive requirements, conditions and assumptions for system currently being developed.

As system definition and development progresses, technical experts provide documentation and rationale for requirements that have been derived. This real-time capturing prevents requirements and their associated rationale from being lost or neglected. For example, a vehicle configurator may see the need for providing a minimum passage dimension for vehicle egress or ingress. This requirement would then be captured at an early development stage and would provide a history for the decision. This seemingly simple requirement may have large impacts on the design down the road and its traceability is important.

Derived requirements and rationale are later transfered to the Madison Research Corporation (MRC) where they are then entered into the system data base which has been developed for ACSS using ACIUS's 4th Dimension® software. The data base allows for easy access and traceability of requirements.

The charts that are contained within this document represent two collated copies of principal requirements and assumptions for February 2, and May 30, 1990. The systems defined include: (1) the Mars Transfer Vehicle (MTV), (2) Mars Excursion Vehicle (MEV), (3) Trans-Mars Injection Stage (TMIS), and the Earth Crew Capture Vehicle (ECCV). Each system is then broken down into subsystem headings of: (1) design integration, (2) guidance, navigation and control (GN&C), (3) electrical power, (4) man systems, (5) structure and mechanisms, (6) propulsion, (7) ECLSS, (8) and command and data handling (C&DH). The initials of each of the technical experts responsible for developing the supporting rationale for each of the requirements is indicated parenthetically next to each entry.

Although the majority of the derived requirements listed are directly applicable to all vehicles such as those powered by Nuclear Electric propulsion (NEP), Nuclear Thermal Rockets (NTR), Solar Electric propulsion (SEP) and reference Cryo, there are some that are not. Those requirements that are only directly applicable to a specific vehicle type are indicated within the entry. The italicized entries indicate a modification to an original requirement prior to the second revision of May 30, 1990.

Defining and re-examination of derived requirements will continue through the current contract.

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Derived Requirements

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SEP Node Resupply Requirements

STCAEM/brc/25Apr90

Component	Design Life	# of Missions	Replace	Refurb.	Comments
Ion Thrusters	15,000 hrs	1	X		ORU or Refurb per mission
Power Processors	10 yrs	3		X	Refurb necessary components
Power Distribution	10 yrs	3		X	Refurb necessary components
Solar Array - Main	10 yrs	3	X		Refurb when necessary
Solar Array - E. Orbit Xfer	1 year	1			Leave at HEO node for further use
Propellant Tanks	10 yrs	3		X	Refurb when necessary
Propellant - Argon	NA	1	X		Good for one mission
Structure	10 yrs	3		X	Refurb if necessary
Payload					
Habitat					
Consumables	NA	1	X		
ECLSS	10 yrs	3		X	Refurb necessary components
Structure	10 yrs	3		X	Refurb necessary components
Avionics	10 yrs	3		X	Refurb necessary components
Power Subsystem	10 yrs	3		X	Refurb necessary components
Radiators	10 yrs	3		X	Refurb necessary components
Aerobrake					
Structure	10 yrs	3		X	Refurb necessary components
TPS	1 Mission	1	X		Replace per mission
MEV	1 Mission	1	X		

MTV Derived Requirements

- **Design Integration**
  - Two (2) communications satellites deployed in Mars orbit with total mass = 3000kg (GW)
  - Crew module must accommodate alternative advanced propulsion options (BD)

- **GN&C**
  - Capture trajectory entry interface for aerocapture not to exceed 6'g' limit and to preclude an uncontrolled skip-out (PB)

- **Electrical Power**
  - Solar power to be used for transfer phase, batteries to be utilized for sun occultation time while in Mars orbit (BC)

- **Man Systems**
  - Added protection to crew from Solar Proton Events (SPE) will incorporate use of a "storm shelter". (MA)
  - Consumables stored will suffice for crew residence time from 443-1018 days (includes abort), assumes 100% ECLSS closure of water and oxygen, 0% closure on food and .25 kg leakage per day (PB)
  - Two (2) astronauts able to pass through major circulation paths while wearing EVA suits. (SC)
  - Crew quarters shall provide sufficient volume for casual conversation between at least two (2) crew members (SC)

MTV Derived Requirements

(continued)

- **Man Systems** (continued)
  - Crew visibility during all maneuvers (docking/rendezvous) (SC)
  - There shall be 2 means of egress from each module for emergency escape (SC)
  - Crew module to accommodate 0'g' and induced 'g' environments (SC)

- **Structure and Mechanisms**
  - Airborne support equipment for aerobrake shall be 20% of aerobrake mass (PB)

MEV Derived Requirements

- **Design Integration**
  - Provide 15% of active weight for spares (JM)
  - MAV must be able to abort-to-orbit during descent phase (PB)
  - Twenty-five (25) ton down payload on manned vehicles (BS)
  - Protective covers provided for all mission critical systems (BS)

- **GN&C**
  - L/D range from 0.5 to 1.0 (GW)
  - Deorbit from 1 sol x 250 km periapsis orbit (nominal) (GW)
  - Currently, cross range = ± 500km (GW)
  - Engine start before aerobrake drop (GW)
  - Approach path angle = 15° (GW)
  - Capture trajectory entry interface for MEV aerocapture at Mars not to exceed 6'g' limit on crew members and equipment and to preclude an uncontrolled s[illegible]t of the Mars atmosphere (PB)
  - Landing accuracy after aerobrake jettison will be unaided by landing beacons assuming 1km cep and with beacon assuming 30m cep (PB)
  - Aerobrake jettisoned in controlled manner during powered descent phase (BS)

MEV Derived Requirements
(Continued)

- **Propulsion**
  - Pre-descent checkout of engines to be provided (checkout extent TBD) (BD)
  - One (1) meter clearance established between engine bells and surface (SC)

- **Electrical Power**
  - Solar arrays to supply power to MEV following separation from MTV for fifty (50) day approach to Mars (BC)
  - Power for 50 day approach sequence to Mars shall be provided by solar arrays separate from the full MTV configuration. Arrays to be retracted 12 hours prior to Mars encounter, power shall be provided by batteries or other internal source (BC)

- **ECLSS**
  - Capability of two (2) crew cab represses (BD)

- **Man Systems**
  - Consumable s will suffice for a crew residence time of 30 - 600 days dependent on mission stay time and abort scenarios, assumes 100% ECLSS closure of water and oxygen, 0% closure of the food and .15 kg leakage per day (PB)
  - The maximum surface stay time is 600 days (PH)

MEV Derived Requirements

(Continued)

- **Structure and Mechanisms**
  - Shall be at least two (2) functionally independently pressurized areas for emergency conditions The shall be two (2) EVA suits stored in these areas (PB)
  - Establish 30cm clearance between all elements to allow for movement during high-stress maneuvers (SC)
  - Crew cab to have SSF diameter (4.4m), width (1.4m), and penetrations and attachments occur at rings. (SC)
  - Surface hab system to be: removable later by surface construction transport vehicle and protected from damage by MAV blast during ascent start (BS)

MTV - TMIS Derived Requirements

- **Design Integration**
  - Flexible to support reference missions (interconnect design to support reference mission requirements (GW)
  - Fully modularized to utilize ETO capacity , the amount of modularization shall be a function of the ETO vehicle chosen (PB)
  - Assembly to be accomplished on-orbit, remotely and robotically (BS)
- **Propulsion**
  - Reference vehicle is launched "wet" with top-off (dry/wet issue to be traded) (JM)
- **Structure and Mechanisms**
  - Thrust structure - tanks - intertanks used as primary structure (GW)
  - The airborne support equipment mass for launch to Earth is assumed to be 7% for all hardware sets (PB)

Mars Transfer System Derived Requirements

- **Design Integration**
  - Wake closure cone behind all aerobrakes is 44° wide (BS)
  - Equipment design life must account for mission duration plus one year (BS)
  - All components designed for 5 missions with refurbishment (except aerobrake) (BS)
  - Design for range of crew sizes, from 4 to 12 (BS)
  - L/D range from 0.5 to 1.0 for aerobrake vehicles at Mars (BS)
- **GN&C**
  - 8500 m/s maximum entry velocity at Mars (GW)
  - 100 m/s error-correction (post aerocapture) (GW)
- **Propulsion**
  - Engine out capabilities in all mission phases (BD)
  - Engine must continuously track C.G. of vehicle from beginning to end of all burns (BD)
  - Maximum gimbal angle of engines TBD (BD)
- **Man Systems**
  - Solar Proton Event (SPE) protection to be provided (MA)
  - Allow for direct viewing of all docking, berthing and landing procedures (SC)

Mars Transfer System Derived Requirements
(Continued)

- **Structure and Mechanisms**
  - All critical function lines and redundant systems shall run non-parallel (PB)
  - All systems shall function up to 2 years in a dormant state and having been subjected to the harsh space environment (PB)
  - The airborne support equipment mass for launch to Earth orbit shall be assumed to be 15% for all hardware except the aerobrake (PB)
  - Airborne support equipment mass assumption for the aerobrake shall be 20% of the aerobrake mass (PB)
  - Aerobrake will be launched to Earth orbit in sections for on-orbit assembly as the reference case (PB)
  - MTV and MEV aerobrakes have common layout of attach points (BS)
  - Vehicle elements will have removable debris shield panel cladding for protection during LEO operations. These panels will be removed and saved in LEO to be used for the next mission-opportunity. The panels will not add to the LEO debris environment (BS)
  - Mission vehicles will carry a robotic manipulation capability to inspect and maintain all exterior areas and systems (BS)
  - Structure optimized to minimize weight, operations, complexity and development effort (BS)
  - Greater than 30cm separation between all major vehicle exterior systems (i.e., tanks, modules) (BS)

- **C&DH**
  - Connectability between links maintained 90% of the time. Availability when scheduled - 98% connectability (PH)

MTV - ECCV Derived Requirements

- GN&C
  - Capture trajectory entry interface for ECCV aerocapture or aeroentry into Earth atmosphere not to exceed 6'g' limit on crew and personnel, and to preclude an uncontrolled skip out of Earth atmosphere (PB)
  - L/D = 0.25 (MF)
- Structure and Mechanisms
  - Interior materials must conform to NASA standards for outgassing, fire hazards,etc. (SC)

MTV - TMIS Derived Requirements

- **Design Integration**
  - Assembly to be minimized to extent practical. (KS)
- **Propulsion**
  - Passive thermal control system including zero-'g' thermodynamic vent system coupled to multiple vapor cooled shields. (JM)
  - TMIS insulating system is a continuously purged MLI over foam design optimized for minimum ground-hold, launch, and orbital boiloff. Includes vapor cooled shield (coupled to TVS) outside of foam. (JM)
  - TMIS tanks launched late in assembly sequence to minimize orbital stay time before TMI burn (, 6 months). (JM)
  - MTV tank insulation system is thick (2-4") MLI blankets. Multiple vapor cooled shields placed at optimum points in the MLI. (JM)
- **Structure and Mechanisms**
  - *Thrust structure - tanks - intertanks used as primary structure for cryo/aerobrake only* (GW)

Note: Changes to existing derived requirements dated 02 February 1990 are shown here in italics

Mars Transfer System Derived Requirements

- **Design Integration**
  - Wake closure cone behind all aerobrakes is 44° wide. The total wake closure angle is centered on the velocity vector. (BS)
- **GN&C**
  - *200 m/s error correction (post aerocapture)* (GW)
- **Propulsion**
  - *Engine out capabilities in all mission phases. NTR engine out capabilities TBD* (BD)
  - All passive cryogenic thermal control system.
  - No. MTV-TMIS fluid transfer before Earth departure. (MEV tanks refrigerated or filled after MOI)
- **Structure and Mechanisms**
  - Aerobrake externally mounted to vehicle for launch to Earth orbit ("Ninja Turtle" concept) (PB)

Note: Changes to existing derived requirements dated 02 February 1990 are shown here in italics

MEV Derived Requirements

- **Design Integration**
  - Down payload on manned vehicles
    - ~ 25 mt down payload for reference MEV (includes habitat module) (BD)
    - ~ 0.7 mt down payload for the 'Mini-MEV' (crew habitat is provided by the ascent/descent cab) (BD)

- **GN&C**
  - *Currently, cross range = ± 1000 km for high L/D aerobrake* (GW)
  - *Landing approach path angle = 15°* (GW)
  - *Landing accuracy after aerobrake jettison will be unaided by landing beacons assuming 1 km CEP and with beacon assuming 30 m CEP* (PB)

- **Propulsion**
  - Engine out capabilities for ascent/descent stages (BD)
  - Passive cryogenic storage system: MLI with vapor cooled shields (JM)
  - Gravity field environment eliminates need for zero-'g' acquisition and venting. (JM)
  - Vacuum jacketed ascent tanks for Mars boiloff reduction. (JM)
  - MEV propellant transferred from MTV prior to descent. (JM)

- **Electrical Power**
  - *Solar arrays to supply power following separation from MTV for ~ 50 day approach to Mars. Arrays to be retracted TBD hrs. prior to Mars descent (cryo/aerobrake).* (BC)
  - Batteries or fuel cells to provide power for ascent and descent phases. (BC)

Note: Changes to existing derived requirements dated 02 February 1990 are shown here in italics

MTV Derived Requirements

- **GN&C**
  - *Capture trajectory entry interface for aerocapture options not to exceed 6'g' limit and to preclude an uncontrolled skip-out* (MC)
  - Aerocapture exit errors not to exceed 0.25° inclination, RAAN, and ARCP, and a 0.1 hr period (MC)
  - GN&C requirement for advanced propulsion TBD: (MC)
    - ~ NTR - capture into planned orbit ± TBD
    - ~ EP (electric propulsion options) - TBD
- **Electrical Power**
  - *Solar power to be used for transfer phase, batteries or fuel cells to be utilized for sun occultation time while in Mars orbit except for NEP*. (BC)
  - NEP power derived from existing power system with a backup energy supply via fuel cells (BC)
- **Man Systems**
  - Volume per crew guidelines extrapolated from historical data (SC)
    - ~ Transfer hab = 112 m^3/crew
  - Two independant pressurized volumes for safety (SC)
  - Gravity condition emphasized to accommodate 0-'g' and 1-'g' and for surface commonality (SC)
  - 2.3 m standard ceiling height for psychological and locomotion (SC)
- **Structure and Mechanisms**
  - All penetrations occur in barrel section to minimize mass. (SC)

Note: Changes to existing derived requirements dated 02 February 1990 are shown here in italics

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Guidelines and Assumptions

Groundrules for Mars SEP Study

Task: Investigate trajectories for Mars mission to optimize total trip time

Departure Date: Optimize around year 2015 opportunity

Mars Mission: Depart GEO to 23,500 km Mars orbit and return to GEO

Trip Time: Defined as $C_3 = 0$ to $C_3 = 0$

Mars Stay Time: 30 Days

Trajectory: No trajectory inbound of .3 A.U. should be considered

Vehicle: SEP

Payload: 124.3 t - Earth to Mars
40.3 t - Mars to Earth

Propulsion Stage: 10 kg/kW based on power delivered to thrusters at 1 AU (includes solar array, thrusters, power processor, supporting structure and systems, and 20% contingency). FUEL AND TANKS ARE NOT INCLUDED.

Thruster Efficiency: $EFF = 80.193 \times Isp^2 / (96.04 \times Isp^2 + 5.067 \times 10^8)$

System Efficiency: Base of array to thruster = 91%

Solar Array: $P/Po = R^{-2} (1.763 - .8865R^{-1} + .0592R^{-2}) / (1 - .1171R + .0528R^2)$

* Valid outside of .65 AU, inside of .65 AU power is constant at .65 AU level.

The preceding groundrules are those agreed to in meeting at General Dynamics/Huntsville on September 5, 1989.

SEP Groundrules and Assumptions
Ground- Earth Escape

- SEP sub-assemblies will be delivered to SSF orbit by HLLV
- The SEP vehicle will be assembled at LEO. A separate transfer array will be deployed in LEO and powers the vehicle until it reached HEO (out of the van-Allen belts) .
- Once at HEO the primary Arrays will be deployed and the vehicle will begin the escape spiral.
- During the escape spiral, the crew will rendezvous with the SEP vehicle via a Lunar Transfer Vehicle (LTV). The crew will transfer to the SEP vehicle and the LTV will carry the transfer array to LEO.
- Escape altitude will be approximately 150 Earth radii.

SEP Array Groundrules

- The primary solar array blankets will not be deployed until the vehicle is at HEO
- The array support structure is composed of graphite composite struts and titanium nodes. The structure is assembled using robotics.
- The array(s) are deployed by telerobotic equipment.
- The assembly and deployment of the vehicle can be done by supervised telerobotics
- The disc shaped extensions of the structural nodes serve as attachment points and power connections.

SEP Program Guidelines

- Technology assumed for vehicle systems must be at technology readiness level of 6 by year 2005.

- Mission design/technology influenced by weighting many interdependent Figures Of Merit (FOM) such as

    IMLEO
    Trip Time
    Safety/Reliability
    Operational/Mission Flexibility
    Number of Technology Developments

- The solar electric propulsion project is evolutionary; the program will allow for system upgrades as technology progresses.

- Manufacturing technology will play a key role in reducing the cost of large multi-megawatt solar arrays.

- An electric orbital transfer vehicle would provide a testbed and data base for electric propulsion applications for Mars missions.

STCAEM/brc/16Dec90/disk9

SEP Guidelines and Assumptions

The vehicles propulsion system will be composed of an electric ion propulsion system.

The power system will be designed for a 10 year lifetime.

Resupply mass for hardware was amortized over the 10 year lifetime with a 3 mission/10 year assumption.

For mission analysis purposes, the vehicle was assumed to depart from GEO and return to GEO.

Further operations trades will dictate assembly,departure and refurbishment node locations as well as mission operation modes.

The SEP vehicle will perform an unmanned spiral out of the Earth's gravity well with crew rendezvous prior to Earth escape.

A lunar and Mars gravity assist are baselined to decrease IMLEO and trip time requirements.

STCAEM/brc/16Dec90/disk9

SEP Guidelines and Assumptions (cont.)

- Mission analysis assumed a baseline alpha of 10 kg/kW for the 10 MWe vehicle.
- The large truss structure supporting the array will be assembled robotically.
- The solar array and power distribution will be designed for high reliability and multiple power paths.
- The power distribution system will be high voltage to reduce transmission mass.
- Propellant for the ion thrusters will be argon.
- The solar array was assumed to be composed of multi-junction cells at 26% efficiency.

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III. Operating Modes and Options

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Reference

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SEP Operating Modes

As the vehicle is slowly spiraling towards Earth escape, the crew will rendezvous with the SEP by a LTV class vehicle a few days prior to escape. Just prior to escape, the SEP vehicle will perform a Lunar fly-by to gain a delta V boost. After Earth escape the vehicle will continue thrusting just prior to the "halfway" point. After a short coast time (20 - 40 days), the vehicle begins the deceleration portion of the interplanetary leg. The deceleration portion is started a little later than normal, since the vehicle will be performing a Mars fly-by. The vehicle does not capture at Mars upon arrival due to an excess delta V, but does drop the MEV containing the crew at Mars. The excess delta V is low and does not impose any significant impacts to the MEV aerobraking scenario. The vehicle continues in heliocentric space, in close proximity to the planet, until it is able to capture into a loose rendezvous orbit. The amount of time the vehicle continues in heliocentric space will be designed to be synonymous with the crew surface stay time. At the end of the surface stay, the crew will return to orbit in the MEV ascent cab. After crew rendezvous, the SEP vehicle will return to Earth. At Earth capture, the crew will depart the SEP and return to Earth by an ECCV or a LTV. A parking orbit for refurbishment requirements is TBD.

Mars Mission Operational Task Flow
Solar Electric Propulsion

The following charts show the top-level operations that must be performed for the SEP manned Mars missions. These include the ground, near-Earth, outbound transfer Mars vicinity, inbound transfer, and earth capture operations. Several options exist :

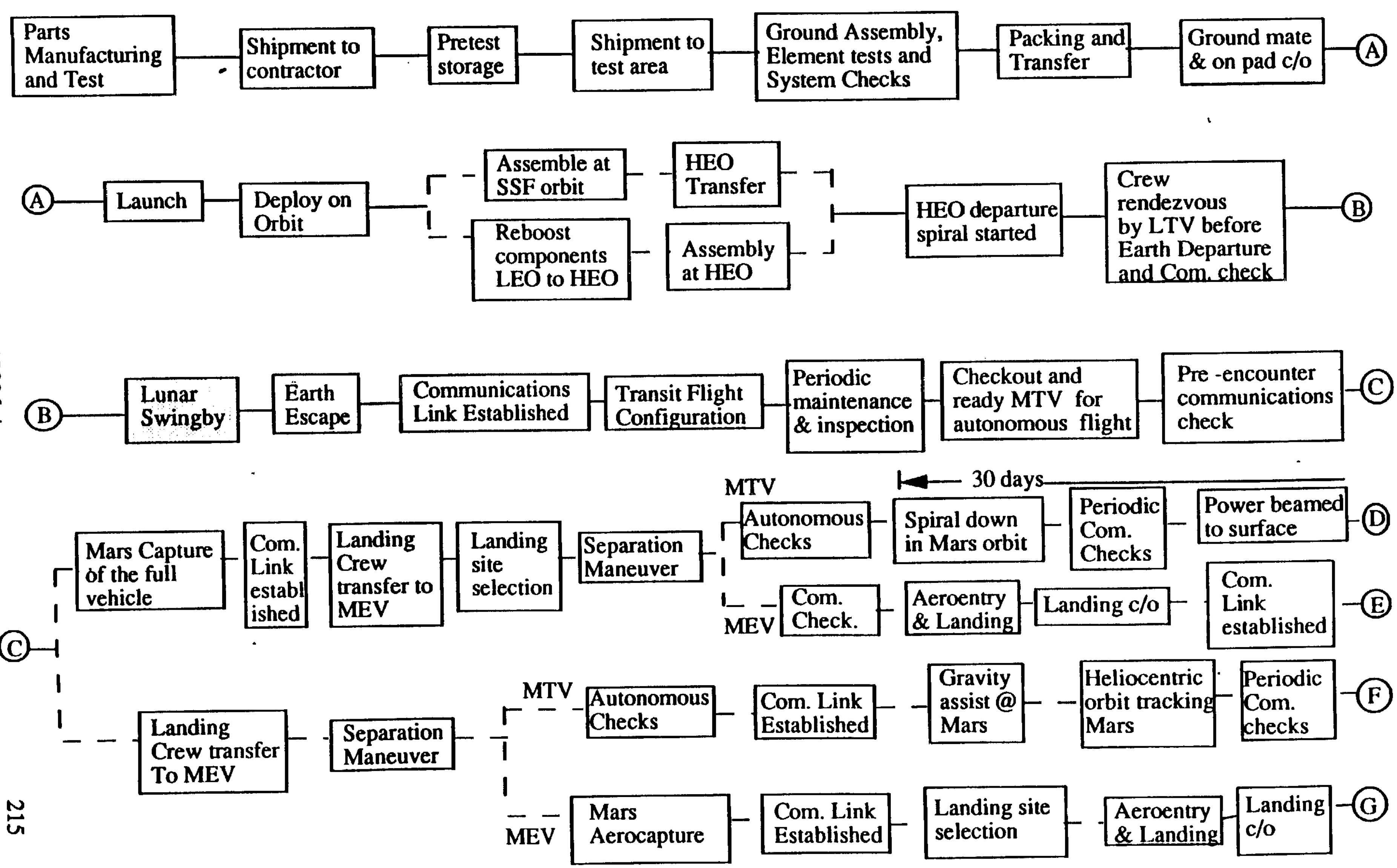
a) the near-Earth buildup node point, may be at SSF orbit or at HEO

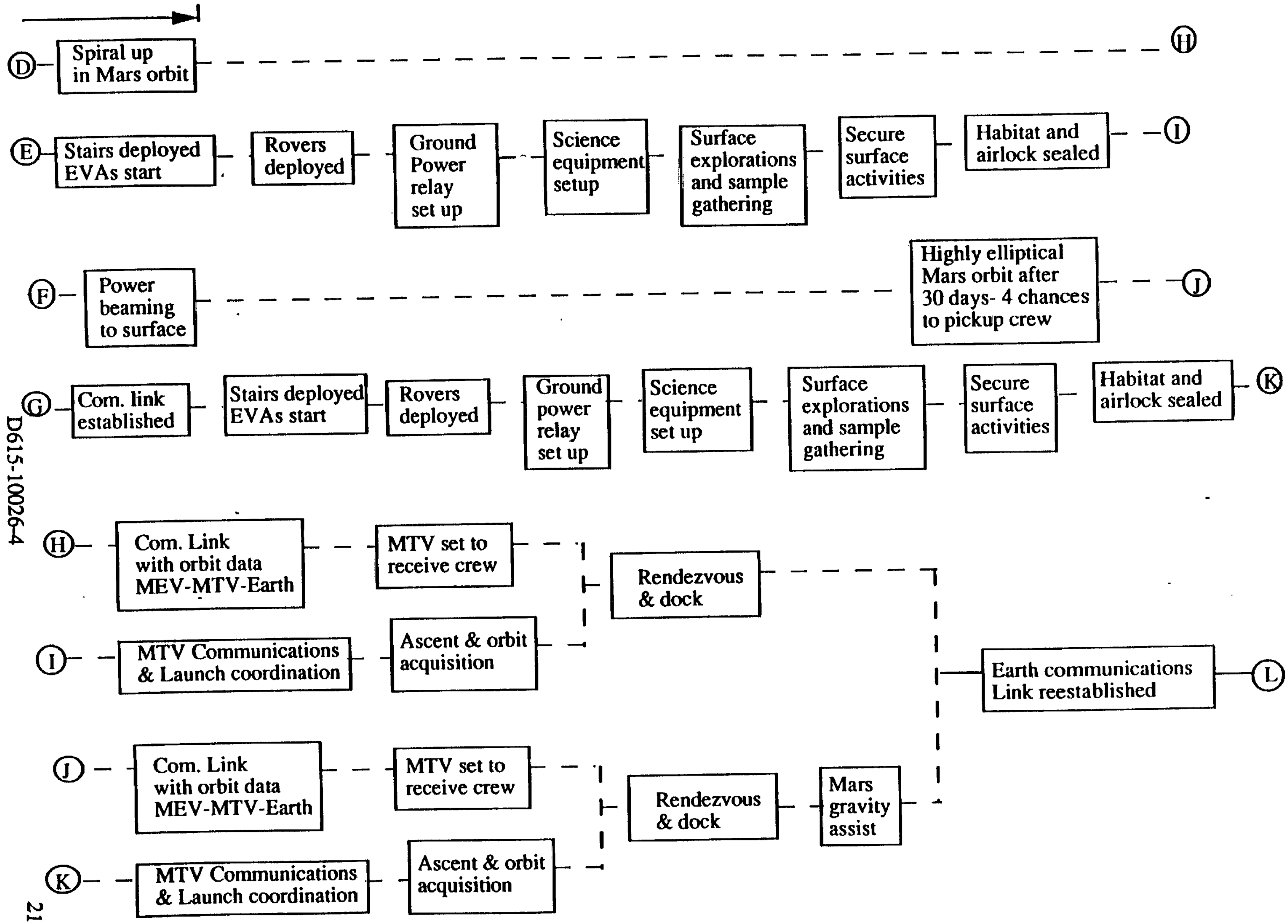
b) the use of a lunar swingby which may or may not be available depending on the mission timing

c) to do a full capture of the SEP transfer vehicle, leaving the MTV in orbit and landing the MEV from near Mars orbit , the MTV would beam power to the MEV on the surface, or swingpast Mars, drop off the MEV to aerocapture and land, "formation fly" with Mars and on the return thirty days latter swing around Mars to allow 4 possibilities to rendezvous and dock the MTV and MEV, then use the final swing around Mars to gravity assist the vehicle on the trajectory back to Earth.

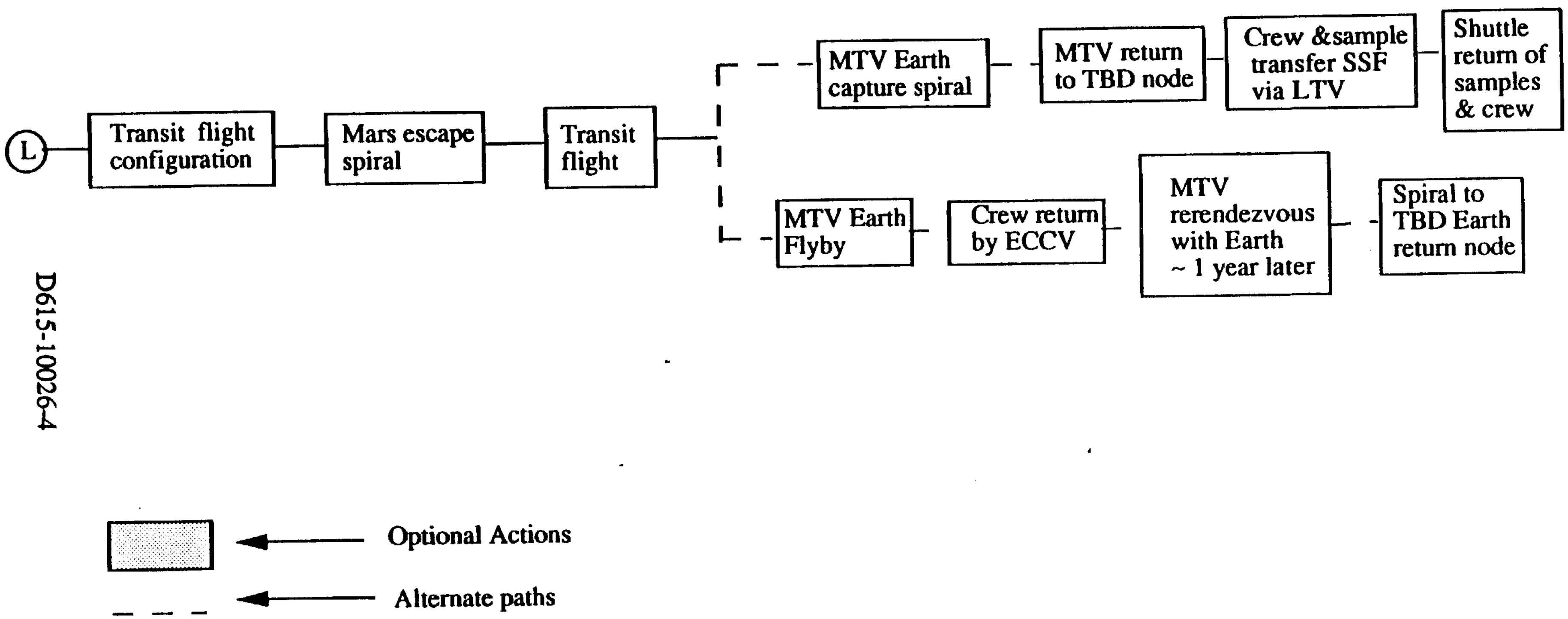
d) either an ECCV direct entry to Earth from the MTV with the MTV doing an Earth flyby to rerendezvous and capture with Earth one year later or the MTV may be captured into LEO and the crew transferred to SSF then to Earth

Mars Mission Operational Task Flow
Solar Electric Propulsion (SEP)





D
Spiral up in Mars orbit
H
E
Stairs deployed EVAs start
Rovers deployed
Ground Power relay set up
Science equipment setup
Surface explorations and sample gathering
Secure surface activities
Habitat and airlock sealed
I
F
Power beaming to surface
Highly elliptical Mars orbit after 30 days- 4 chances to pickup crew
J
G
Com. link established
Stairs deployed EVAs start
Rovers deployed
Ground power relay set up
Science equipment set up
Surface explorations and sample gathering
Secure surface activities
Habitat and airlock sealed
K
H
Com. Link with orbit data MEV-MTV-Earth
MTV set to receive crew
Rendezvous & dock
I
MTV Communications & Launch coordination
Ascent & orbit acquisition
Earth communications Link reestablished
L
J
Com. Link with orbit data MEV-MTV-Earth
MTV set to receive crew
Rendezvous & dock
Mars gravity assist
K
MTV Communications & Launch coordination
Ascent & orbit acquisition


L
Transit flight configuration
Mars escape spiral
Transit flight
MTV Earth capture spiral
MTV return to TBD node
Crew &sample transfer SSF via LTV
Shuttle return of samples & crew
MTV Earth Flyby
Crew return by ECCV
MTV rerendezvous with Earth ~ 1 year later
Spiral to TBD Earth return node
Optional Actions
Alternate paths

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IV. System Description of the Vehicle

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SEP System Description

Contained within this section are the following:

- Block diagram for direct screen drive
- Power path efficiency
- Parts description
- Vehicle and payload mass statements

The vehicle can be broken out into four main subsystems: (1) power (2) propulsion (3) support systems (4) payload. The payload is common to all propulsion systems with minor adjustments for each option such as attachments, trip time implications, and power sources. The vehicle support systems such as communications, avionics, and structure, are not necessarily unique to the SEP and hence no detail will be covered here. For a more detailed description of these systems, refer to the vehicle configuration section.

The power system is unique to the SEP vehicle and is the largest subsystem from a mass and area standpoint. The solar array necessary for power generation is the design driving point of the vehicle. A GaAs/CIS tandem junction cell has been baselined for the array due to its high efficiency and low specific mass. The high efficiency allows for a relatively small array when compared to a silicon array of the same power. The array is a flexible planar array based on 26% efficiency, 460 W/kg, and 10 year life. A concentrator array was not chosen due to the pointing accuracy required and the P/Po performance of the array throughout the trajectory. Several other concepts have been looked at such as a planar array with concentration ratio's of 1-4. MSFC performed analysis of these lower level concentration ratio cells compared to the JPL 50 curve (Silicon array, CR=1) and determined that there were no benefits to using the CR=1,2,3,or 4 arrays. Temperature plays an important role in the P/Po curve and it was found that even though the higher CR arrays performed better around 1.5 AU, they lost power around 1 AU due to temperature restraints and cosine losses. Therefore a planar array with CR=1 was chosen.

The power subsystem is also composed of the Power Management and Distribution (PMAD) circuitry. The PMAD comprise a sizable weight percentage of the whole vehicle. One method to decrease the PMAD mass is to employ direct screen drive (DSD). Designing the vehicle for DSD will save mass, but the vehicle will not operate in the plasma environment about the Earth. The plasma environment will cause arcing if potentials get above ~200 volts. One alternative that would allow the mass savings of DSD would be to design the vehicle for DSD and use a boost vehicle to get through the plasma environment.

The propulsion subsystem is composed of 1m X 5m ion thrusters and the argon propellant subsystem. The thrusters use electrical power generated by the solar array and conditioned by the PMAD to ionize and repel the argon ions at an extremely high velocity.

SEP Configuration Development

Features

- Extremely lightweight skeleton, diaphanous array
- Repetitive geometry
- Distributed, redundant power system
- HEO based, transfer array, LTF ferry
- Robotic-mediated assembly, deployment, maintenance

Remaining Issues

- Structural dynamics analysis & design
- PV blanket manufacturing process / design
- Robotics concept development
- g_a resolution: Eccentric rotator? Countermass?
- Propulsive flight attitude analysis

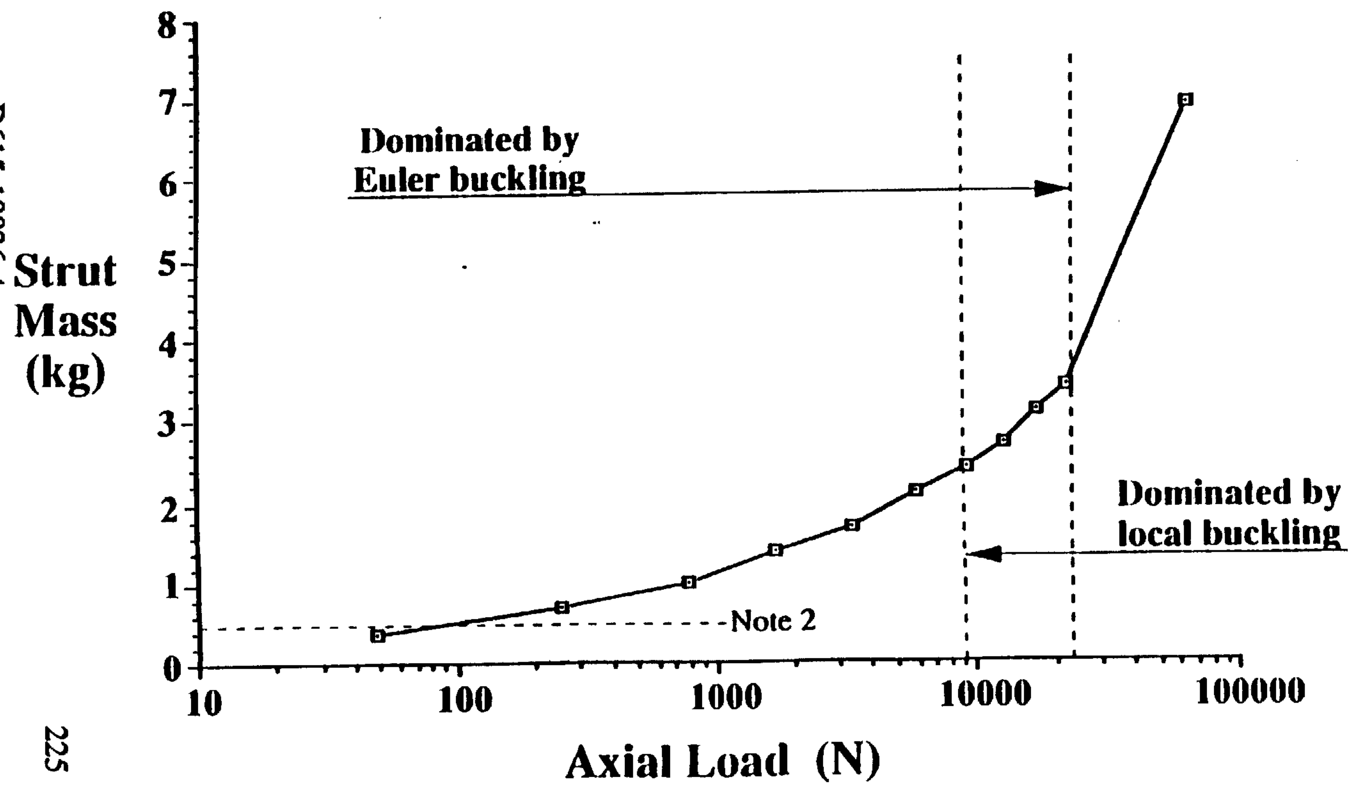
/STCAEM/crf/ 9Jan91

Truss Strut Parametric Mass

The SEP tetrahedral truss strut was sized using the parametric analysis shown on the facing page. The structure is designed to be very stiff, yet as light as possible, with minimum guage wall thickness for handling and assembly.

Truss Strut Parametric Mass

- 7 m strut length, pinned ends
- Shape factor knockdown for imperfections:
  Pinned: 0.9 (< 1.0) & Circular section: 0.36 (< 0.6)
- Gr/Ep, 0° plies with 90° inner and outer wraps
- Minimum-gauge assumed > 0.57 mm



1 Graph data adapted from: Mikulas, M.M.; Bush, H.G.; Card, M.F. (1977) Structural Stiffness, Strength and Dynamic Characteristics of Large Tetrahedral Space Truss Structures. NASA TM X-74001, LaRC.

2 Point design for comparison using 0.35 mm minimum gauge, 7 m long Gr/Ep strut from: Hedgepeth, J.M.; Adams, L.R. (1983) Design Concepts for Large Reflector Antenna Structures. NASA CR-3663, LaRC

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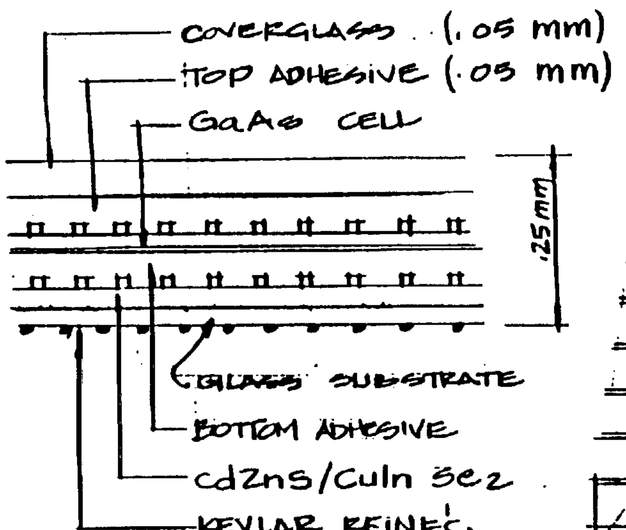
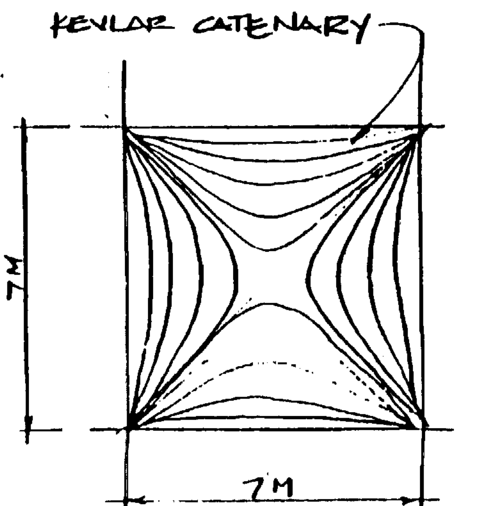
SEP Photovoltaic Array Details

The solar array blanket is constructed of tandem junction solar cells, bonded to a kevlar catenary support structure that transfers forces from the array, into the structural nodes of the tetrahedral truss.

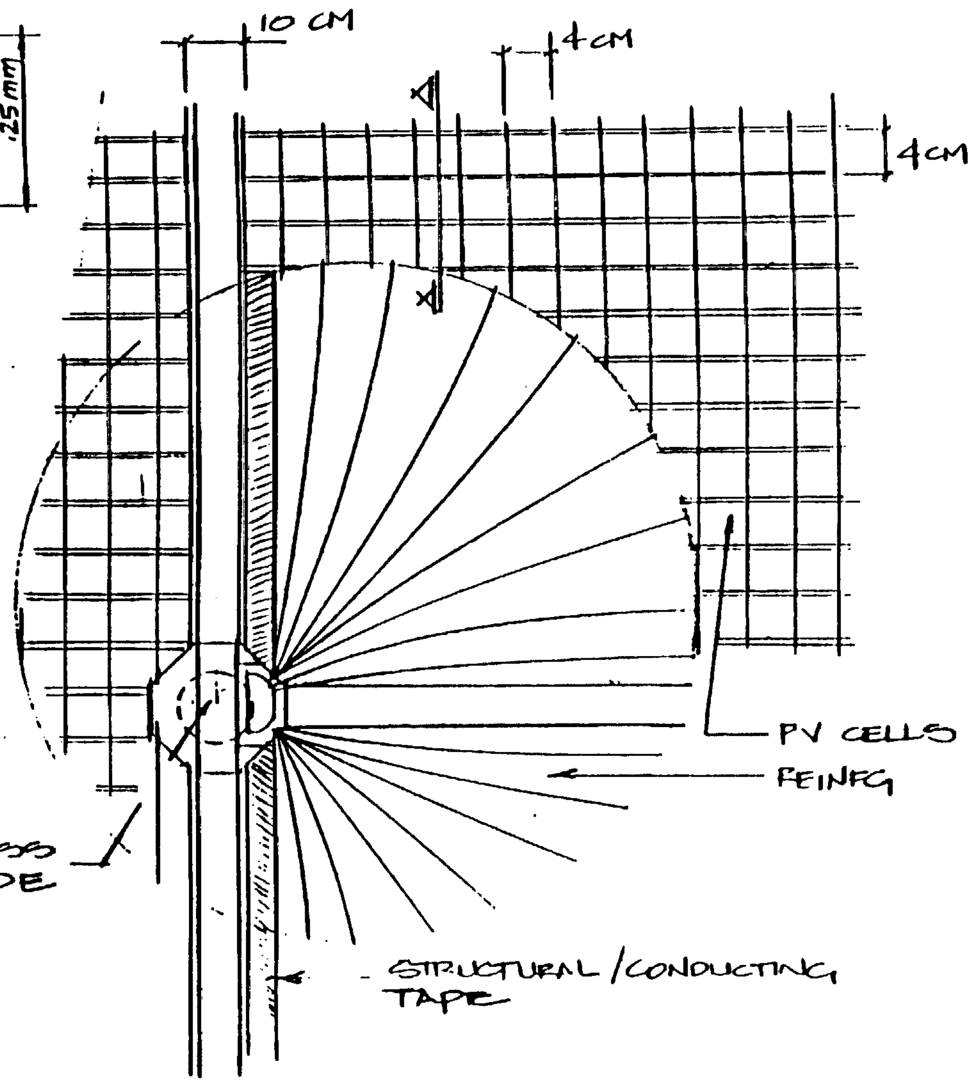
SEP P.V. Array Details



SECTION THRU CELL



ARRAY REINFORCING



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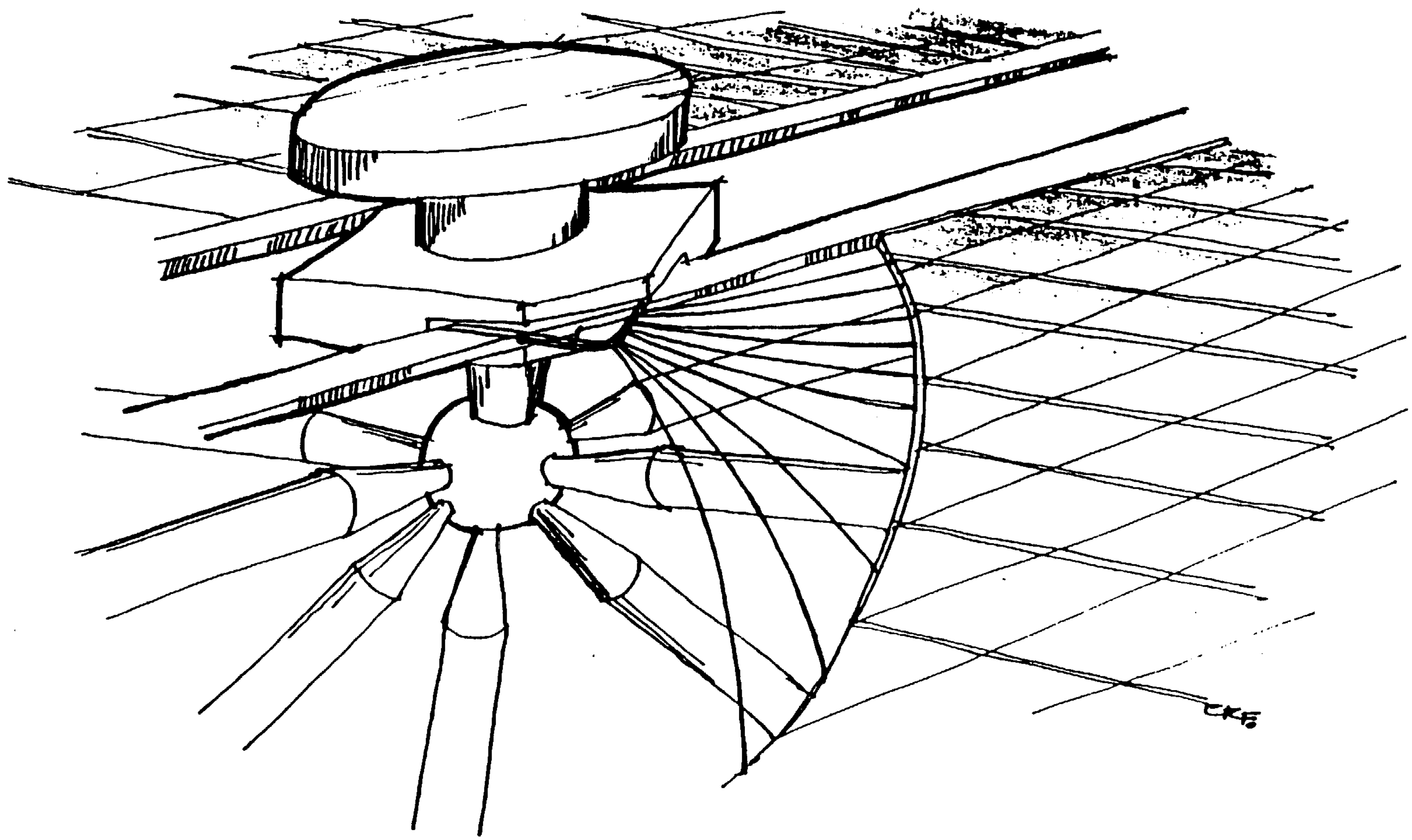
SEP Array / Robot Attachment Node

A sketch of the array blanket / truss structure attachment is shown below. The cutaway view of the array shows the catenary pattern of kevlar reinforcing that transfers stress to a load point at the edge of the blanket, and then to the tetrahedral truss through a structural node.

The telerobotic assembler and blanket deployer attach to the node for structural, power, and data connection.

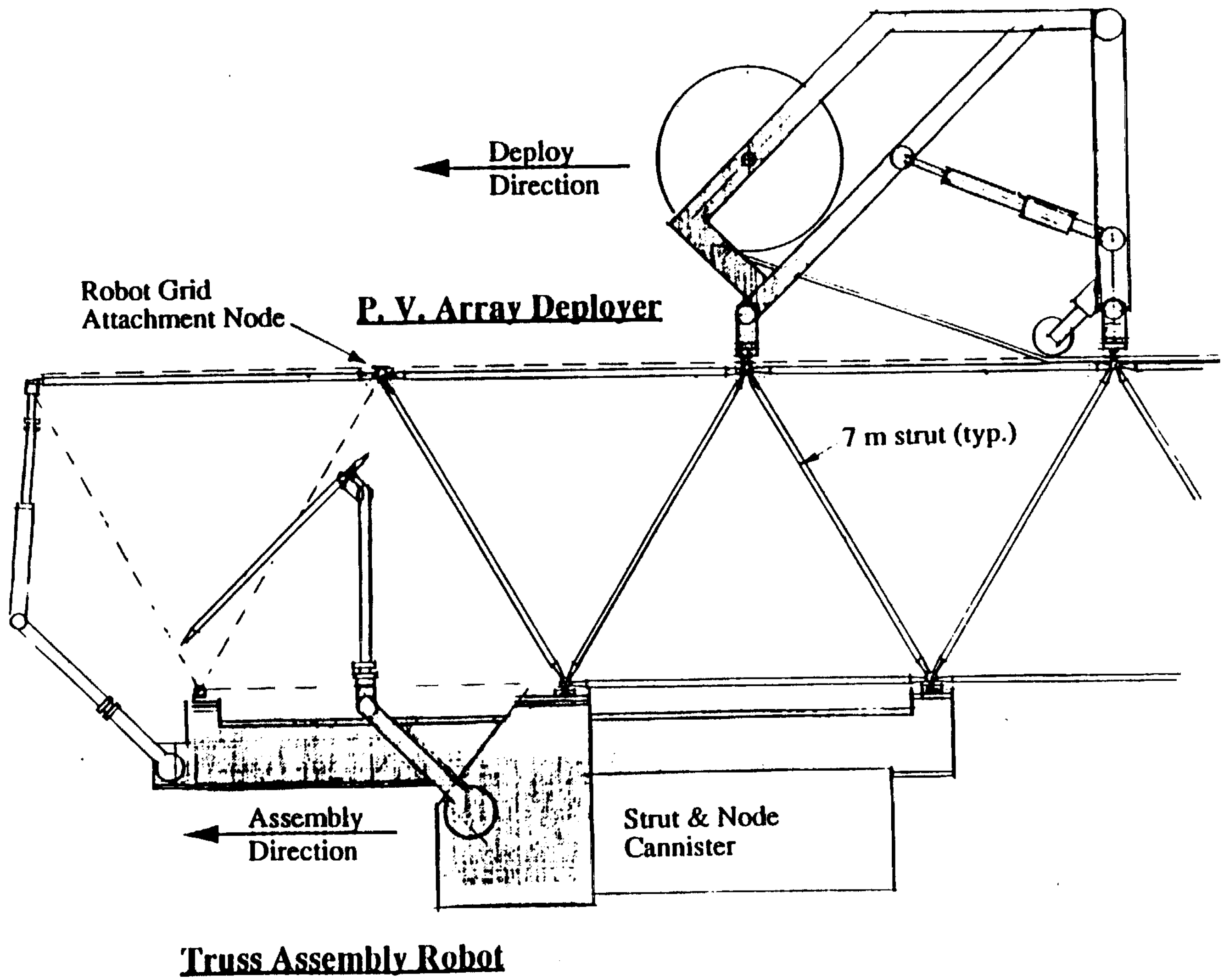
SEP Array / Robot Attachment Node



Robotic SEP Assembly

The sketch shown on the opposite page illustrates a concept for telerobotic assembly and solar array deployment. The assembly and deployment sequence is visualized as being linear in fashion, starting at one corner of the vehicle and proceeding along it's edge. Both the assembler and deployer would progress one 7 meter bay at a time, while spanning two bays in width.

Robotic SEP Assembly

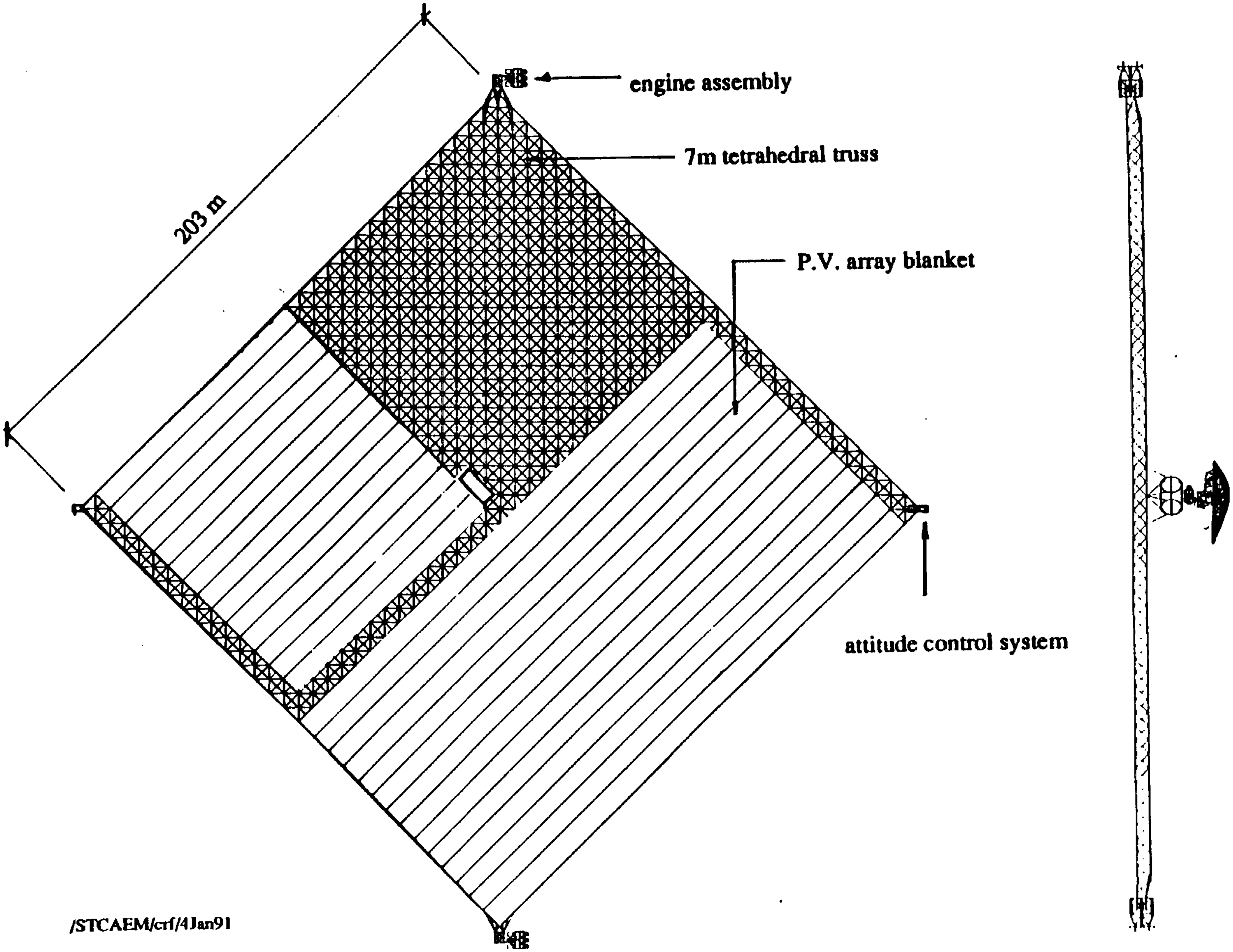


/STCAEM/crf/4Jan91

SEP Configuration

Shown on the facing page is the micro-gravity version of the solar electric vehicle concept. Part of the array blanket is removed to show the location of payload and truss structure configuration.

SEP Configuration

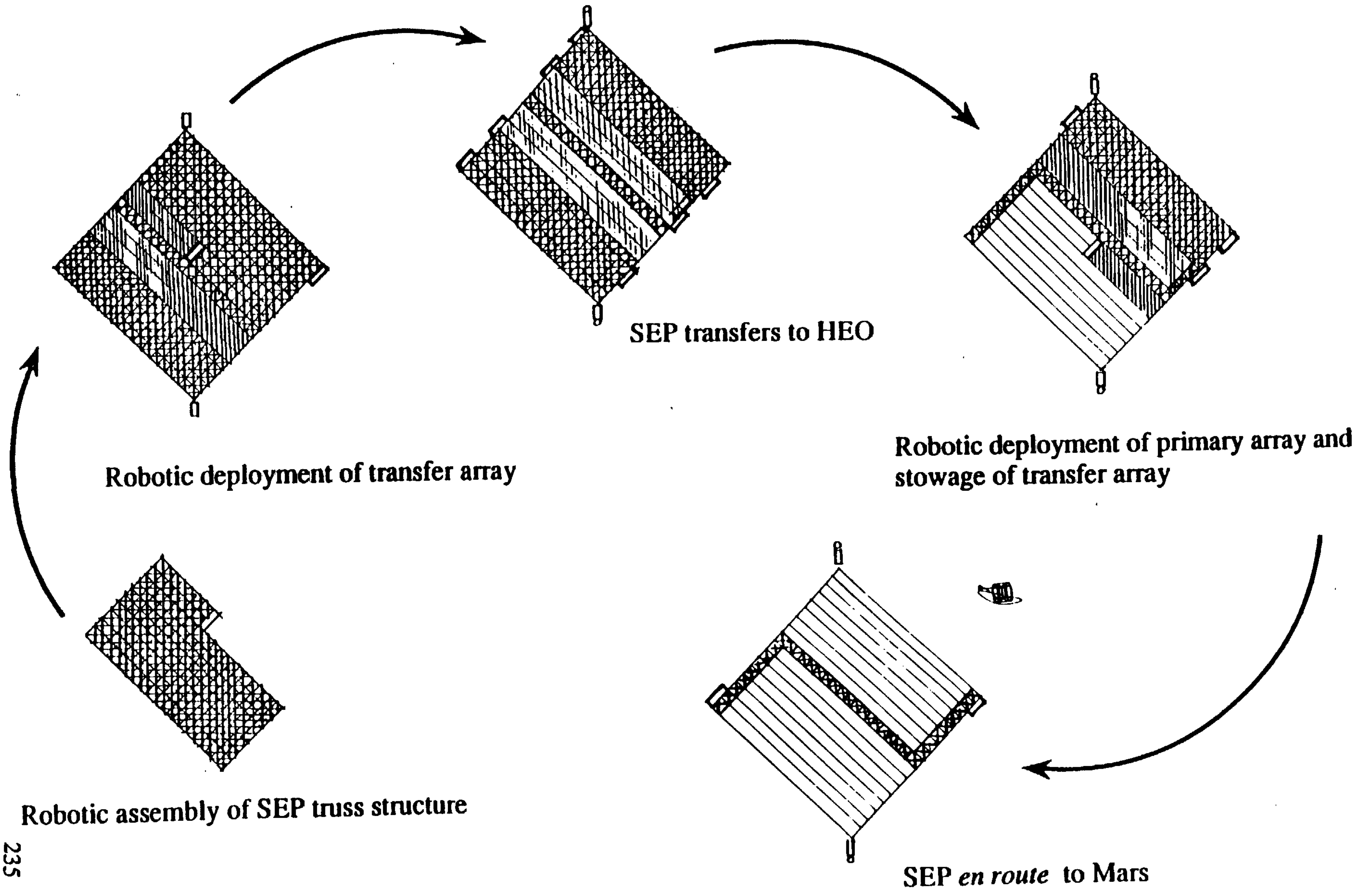


/STCAEM/crf/4Jan91

SEP Operations Sequence

The facing page depicts SEP operations from telerobotic construction in LEO, thru solar array deployment and TMI.

SEP Mission Sequence



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SEP Mass Statement

Shown is a system / subsystem mass statement for the micro-gravity SEP vehicle, with a 4 crew hab module and assembled tetrahedral truss structure.

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 (for 4 crew)	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	x 23,464,336 cells
Reinforced SiO2	211mg/cell	=27.7 t
Adhesive (fiber/cell)	60mg/cell	
Kevlar Support Structure	68mg/cell	

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 kg/kW

/STCAEM/crl/17DEC90

SEP PMAD Design for Mass Reduction

Conditioning: Could configure array for direct-screen drive (avoids conditioning equipment mass)

Distribution

- High voltage system —— risk arcing in ionospheric plasma environments
- Low voltage system —— very large conductor mass

Potential Solutions to the high - V/ low - V trade

- Low voltage PMAD (hundreds of 1" diameter conductors; very high mass)
- Low voltage prosthetic PMAD for LEO operation (left in HEO ; complex connections
- Low voltage EOTV tug (Van Allen Belt - tolerant ; V heavy)
- *High voltage PMAD robust InP; (heavily insulated)*

/STCAEM/crf/ 9Jan91

Block Diagram of a Typical Regulated Solar Array for Direct Screen Drive

Regulated Array

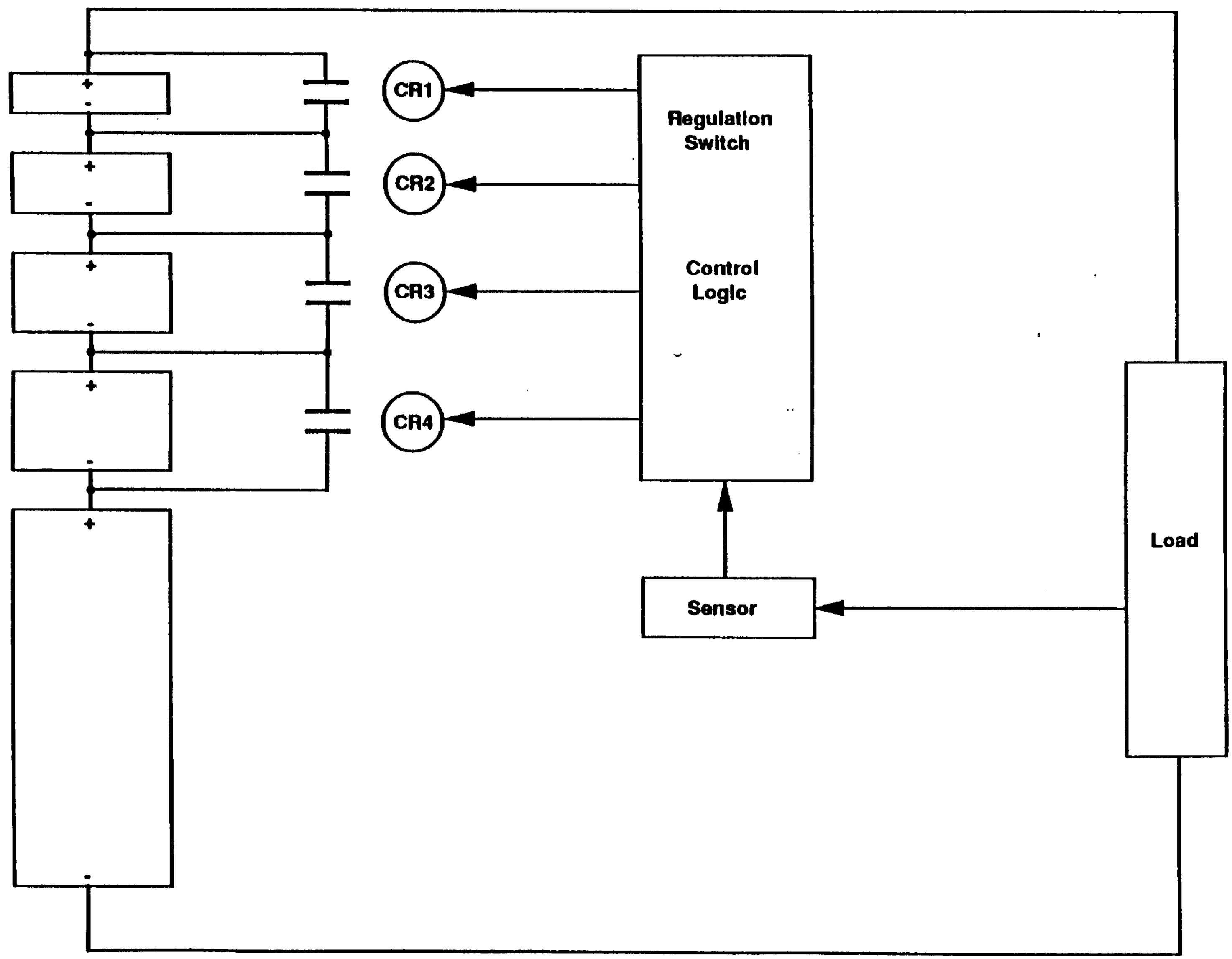
Sized for:

EOL
Max. Temp.
Max. Load

Unregulated Array

Sized for:

BOL
Min. Temp.
Min. Load



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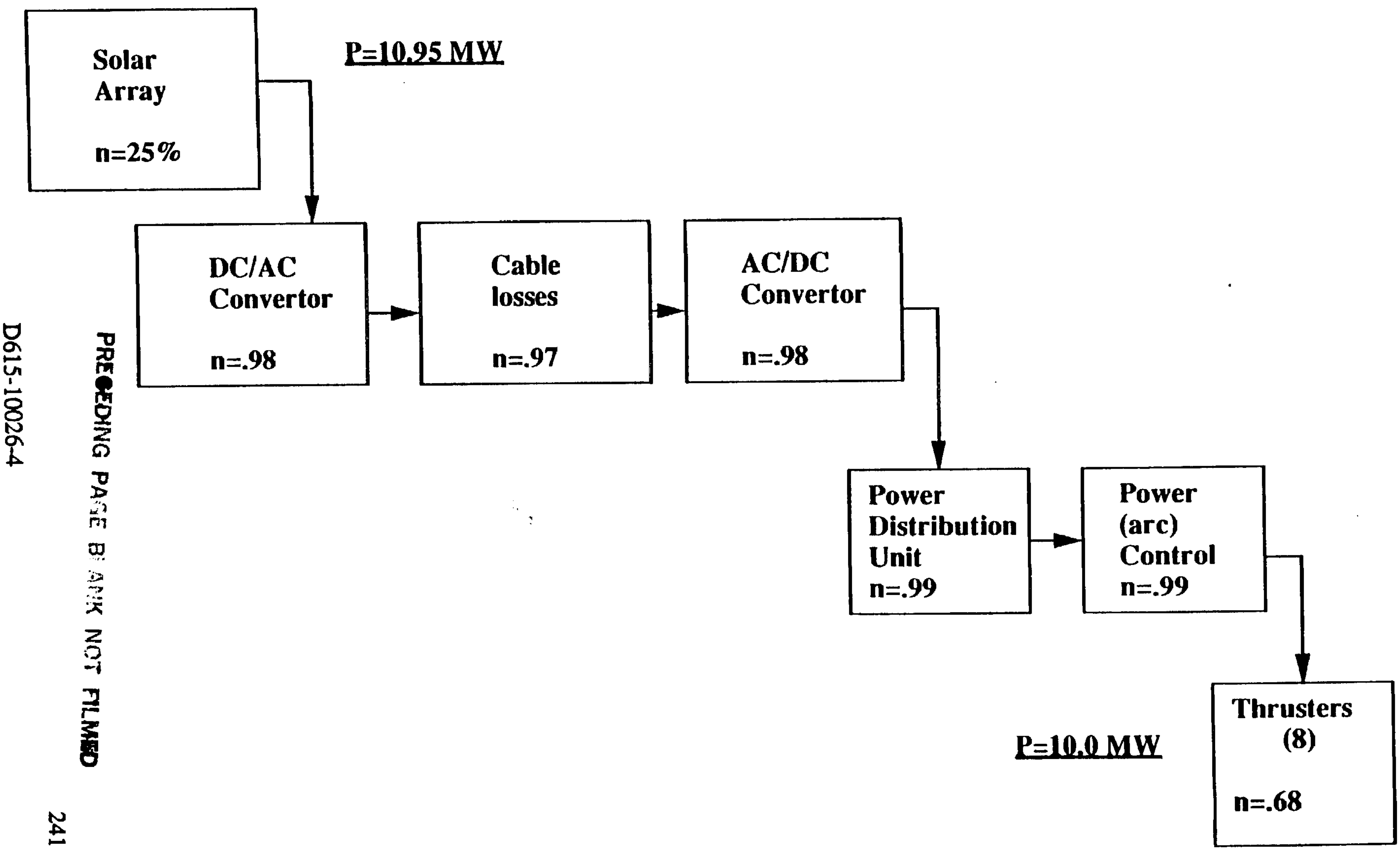
Power Path Efficiency



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Parts Description

SEP Parts List

Component	Subsystem	Description	Size (m)	Mass (t)	Qty.
Array Blanket	Power	Multi-junction CLEFT GaAs/ CIS, 26 % eff., 460 W/kg.	20 X 2 cyl	26.2	14
Array Structure	Structure	Erectable graphite composite tetrahedral truss	7 X 5 X 3	3.6	2
Communications	Avionics	Two comm dishes for long distance applications	2 X 2 X 1	.6	2
Experiment Platforms	Structure	Provide accommodations for long-term deep space exper	5 X 2 X 1	1.7	2
Attitude Control	Propulsion	Resistojets	2 X 2 X 2	5.7	2
Avionics	Avionics	SSF derived command, control & data, GN&C platforms	2 X 2 X 2	2.5	1
Power Processing Unit	Power	Power Processing for array power to usable thruster power	2 X 2 X 1	4.7	2
Power distribution & control	Power	Power network from array to thrusters, includes power control for thrusters	5 X 2 X 1	8.2	2

STCAEM/brc/02Aug90

SEP Parts List

Component	Subsystem	Description	Size (m)	Mass (t)	Qty.
Radiators	Thermal	Heat rejection for power conditioning system, heat pipe, ~400 K	8 X 2 X 1	6.9	2
Thruster Pods	Propulsion	Ion thrusters, composed of 12, 1 X 5 meter thrusters	13 X 7 X 2	13.7	2
Propellant & tanks	Propulsion	Argon propellant, 2 % tankage fraction	4.2 sphere	190	4
Transfer array *Optional	Power	Option used for spiral through Van Allen belts, saves chem boost stage	20 X 2 cyl	26	14
Spiral Propellant *Optional	Propulsion	Optional propellant for Van Allen belt spiral	4.2 sphere	40	1

SEP Parts List

Component	Subsystem	Description	Size (m)	Mass (t)	Qty.
MEV Descent Stage	Structure	Thrust struct, etc. frame 10% inert mass landing legs - 3% landed mass	* See current design		
	Thermal	Passive			
	Aerobrake	Dimensions as chem ref, mass 13% of PL			
	Avionics	Main prop C&I, aerobrake att control			
	Power	Parasitic from host, fuel cell backup			
	Propulsion	4-34k lb w/EO, ext/rect, Isp=460 sec			
MEV Ascent Stage	Structure	Thrust structure, tanks w/ vac shell	* See current design		
	Thermal	Passive			
	Avionics	Main prop C&I, Cryo prop monitor sys			
	Power	Parasitic from host, fuel cell backup			
	Propulsion	2-34k lb w/EO, ext/rect, Isp=460 sec			
MEV Crew Module	ECLSS	Open loop Apollo type	* See current design		
	Crew Accom.	54 cubic meters, spartan accom, 3 day			
	Structures	4.4m D 0.5 ellipsoidal Al shell			
	Thermal	Water/Glycol w/ext. panel radiator			

STCAEM/brc/02Aug90

SEP Parts List

Component	Subsystem	Description	Size (m)	Mass (t)	Qty.
MEV Crew Module	Avionics	Apollo/LEM type complete flight ctrl sys Onboard health monitoring equip	* See current design		
	Power	2.3 kW fuel cell for descent/ascent solar arrays for surface			
ECCV	ECLSS	Open loop Apollo type	* See current design		
	Crew Accom	8 cubic meters, Apollo type crew accom 3 day nominal occupancy			
	Structure	3.9m X 2.7m Apollo type			
	Thermal	Water/Glycol			
	Avionics	Apollo Command Module type			
	Power	Battery Storage			

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Weights Statement

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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
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Solar Array Blanket

Photovoltaic Cell	840mg/cell	* 23,464,336 cells
Reinforced SiO2	211mg/cell	=27.7 t
adhesive (fiber/cell)	60mg/cell	
Kevlar Support Structure	68mg/cell	

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
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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

/STCAEM/crf/17DEC90

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;1

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

/STCAEM/bbd/31May90

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*

Group	Ref	Item	Mass	Rationale
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

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	CO_2 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 O_2 & N_2 and tanks
	Atmos. Composition & Monitor Assem.	55	O_2 & N_2 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

/STCAEM/bbd/31May90

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Artificial Gravity Option

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Solar Electric Propulsion Vehicle Artificial Gravity Configuration

The solar electric vehicle (SEP) artificial gravity (g_a) concept presents complications not present in the lower-performance propulsion concepts. For full-fledged g_a conditions, EP vehicles pose the problem of spinning while thrusting. [An alternative, operational solution may be to fly μg for most of the trajectory, spinning only during the midflight coast intervals (25 to 60 days) and upon arrival at Mars. For STCAEM purposes, however, it is essential to pursue the outcome of a vehicle required to provide artificial gravity for the entire flight.] Because the thrust vector must average tangential to the flight path, the fundamental configuration trade-off is between rotating, high-power transfer assemblies (for the spin vector normal to the ecliptic) and spin-vector precession (for any other orientation).

Of the many possible configuration options identified by STCAEM, the one was chosen that is similar both to the μg SEP and to the NEP g_a concept. This configuration concept, called an *eccentric rotator*, avoids tethers, complex extendible booms or deployable trusses. All components are rigid and the design is simple.

The fundamental concept is that the large solar array is split in two, leaving a gap or slot within which spins a rigid boom supporting the habitable systems. The optimal shape of the two solar array halves has not yet been determined. A single, double-ended slipring assembly (which transmits only habitation-system power levels) is used to despin the vehicle bus. No deployment mechanism is required to change the habitat system separation when the MEV mass is lost. Instead, the rotation rate is adjusted to provide 1g in the center of the long-duration habitat, *according to the habitat's actual separation from the current vehicle mass center*, which shifts after MEV operations. Thus the mass center is not necessarily axially aligned with the engine outrigger or geometric center of the solar array, although it always remains at the zenith relative to the habitat floors. When the mass center is not along the outrigger axis, the outrigger and solar array also orbits the mass center. The engine assemblies therefore trace out circles as they thrust, although the thrust vector orientation remains fixed. For low-thrust systems in particular, this is expected to cause no problems. The solar array, main structure and engine assemblies are used as the countermass to the crew systems.

④

Artificial Gravity (g_a) Assessment Assumptions

A 1g gravity level was assumed for this study over partial g because the minimum gravity level required to offset physiological deterioration is not known. The rotation rate was set to be no more than 4 rpm, which is based on experimental data in the Pensacola Slow Rotation Room (1960's) on human adaptation. The crew compartments are contiguously pressurized during all mission phases, and the crew modules are to be oriented with the long axis parallel to the spin vector to offset the Coriolis effect along major circulation paths. Connections between habitation and the countermass are either tethers or a truss rather than a pressurized tunnel because, since all crew compartments are contiguous, the is no need for an IVA transfer.

Artificial Gravity (g_a) Assessment

Assumptions	Rationale
1g gravity level	• Earth-normal conditioning for exploration in surface EMU
Rotation rate ≤ 4 rpm (56 m)	• Generally accepted range for vestibular disturbance tolerance
Contiguous crew compartments	• Maximize available volume • In-flight simulation and training • Contingency operations
Truss and tether connections • Tethers are "ribbon" shaped	• Avoids mass penalty • Not needed for contiguous volumes • Facilitates conductors
Module orientation parallel to spin vector	• g level consistency; minimizing vestibular disturbance • Mass properties quasi-isotropic to first order

g_a Solar Electric Propulsion Vehicle

The next 7 charts show the options being considered for the SEP artificial gravity configuration. The SEP is in the preliminary stage of trading configurations, and one concept has not yet been chosen.

Artificial Gravity for Continuous Thrust Systems

- SEP vehicles lack obvious countermass:
  - inert mass trades with long tether
  - using array as countermass requires heavier array structure

- Spinning while thrusting poses configuration complications:
  - high-power sliprings (single-point failure route)
  - cross-product engines (10 - 15% mass penalty)

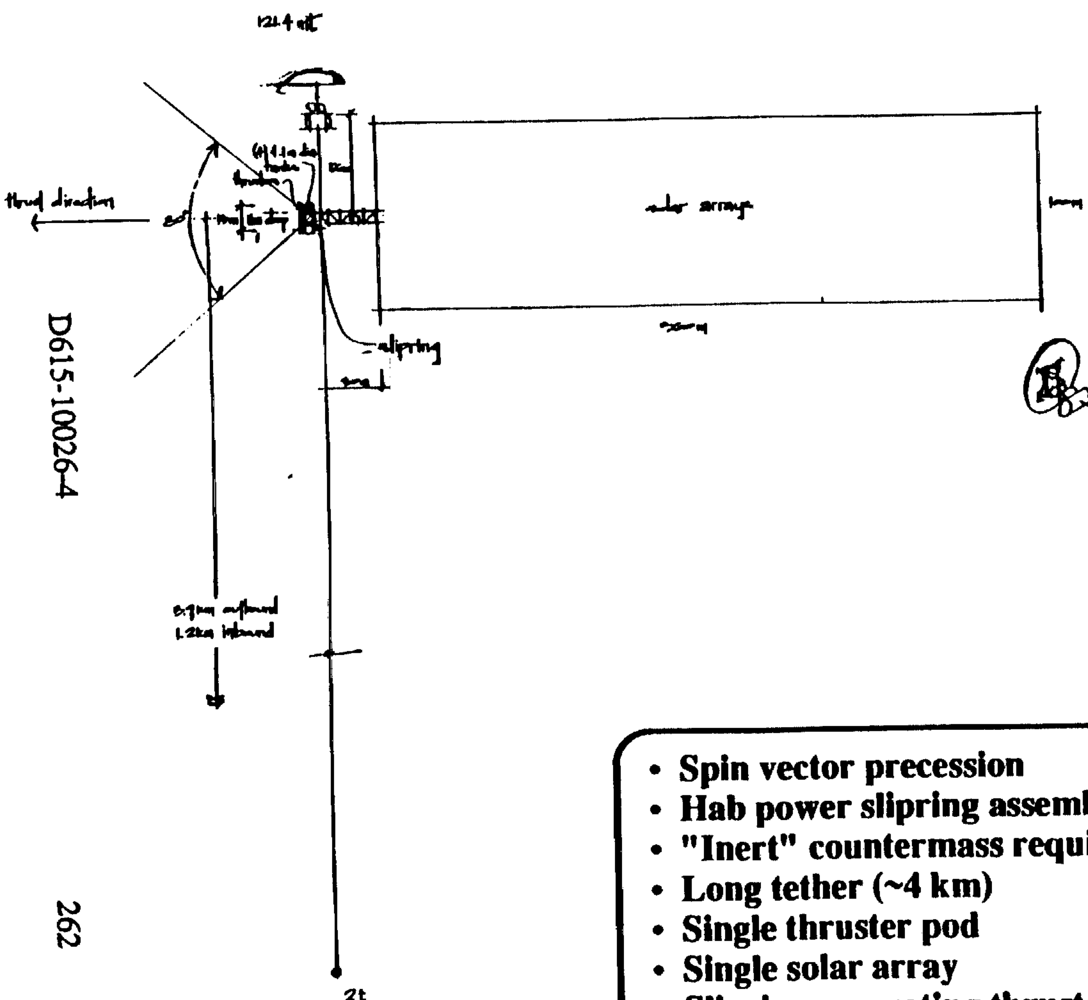
- Least-mass, least mechanically complicating solution
  - 0 g for most of trajectory
  - g_a (no precession) for last 15d before Mars arrival (re-conditioning period)
  - g_a possible for mid-course no-thrust interval (25-60d)
  - full conditioning not perfomance - driven for Earth return

g_a Solar Electric Propulsion Vehicle
Option 1

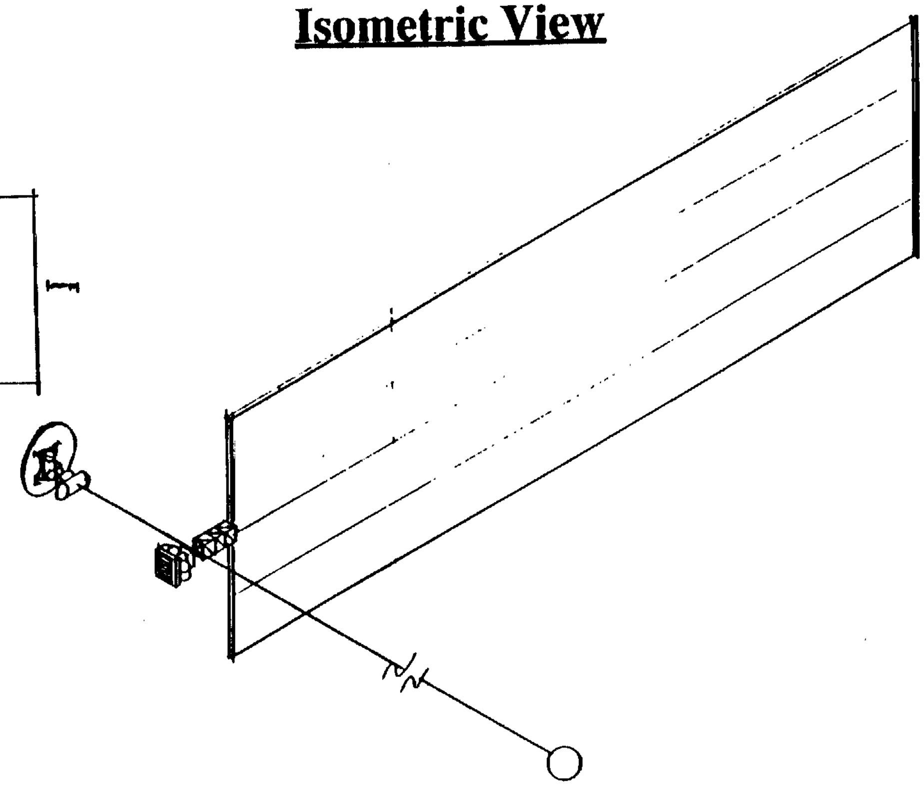
ADVANCED CIVIL SPACE SYSTEMS

BOEING
STCAEM/sdc/22May90

Orthogonal View



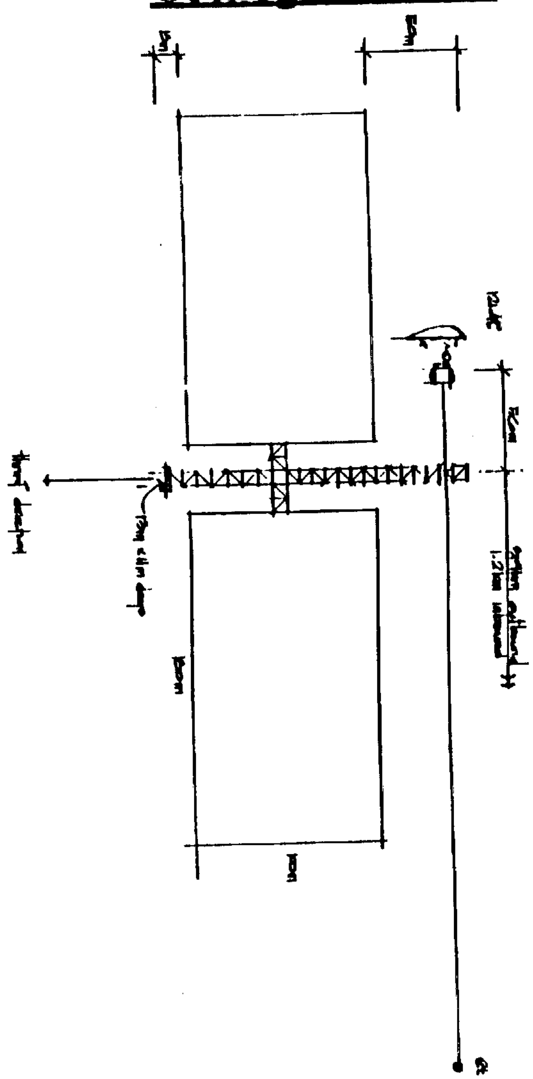
Isometric View



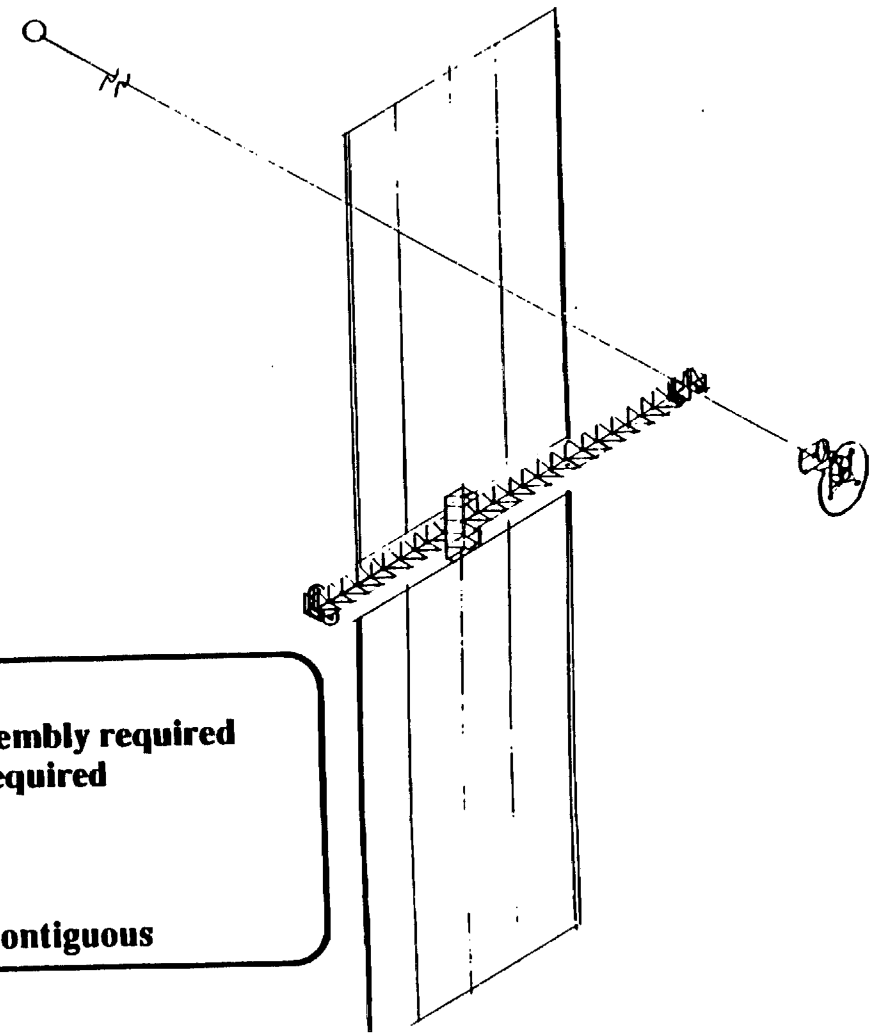
- Spin vector precession
- Hab power slipring assembly required
- "Inert" countermass required
- Long tether (~4 km)
- Single thruster pod
- Single solar array
- Slipring separating thrusters and arrays

g_a Solar Electric Propulsion Vehicle Option 2A

Orthogonal View



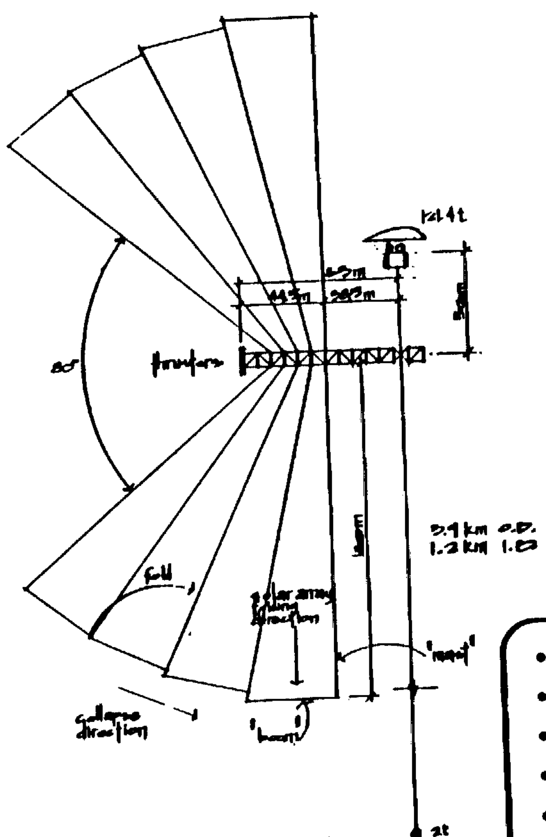
Isometric View



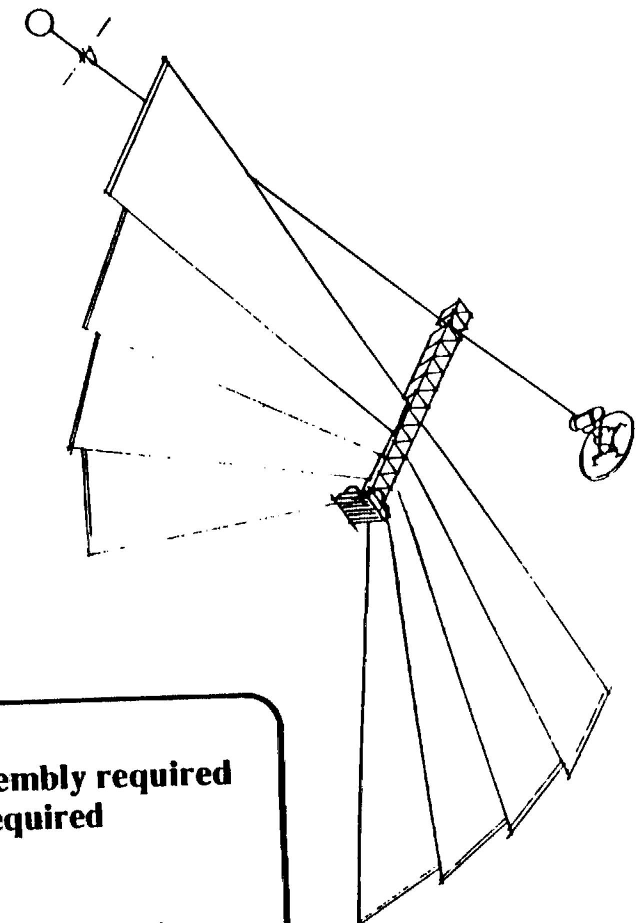
- Spin vector precession
- Hab power slipring assembly required
- "Inert" countermass required
- Long tether (~4 km)
- Single thruster pod
- 2 solar arrays
- Thrusters and arrays contiguous

g_a Solar Electric Propulsion Vehicle Option 2B

Orthogonal View



Isometric View



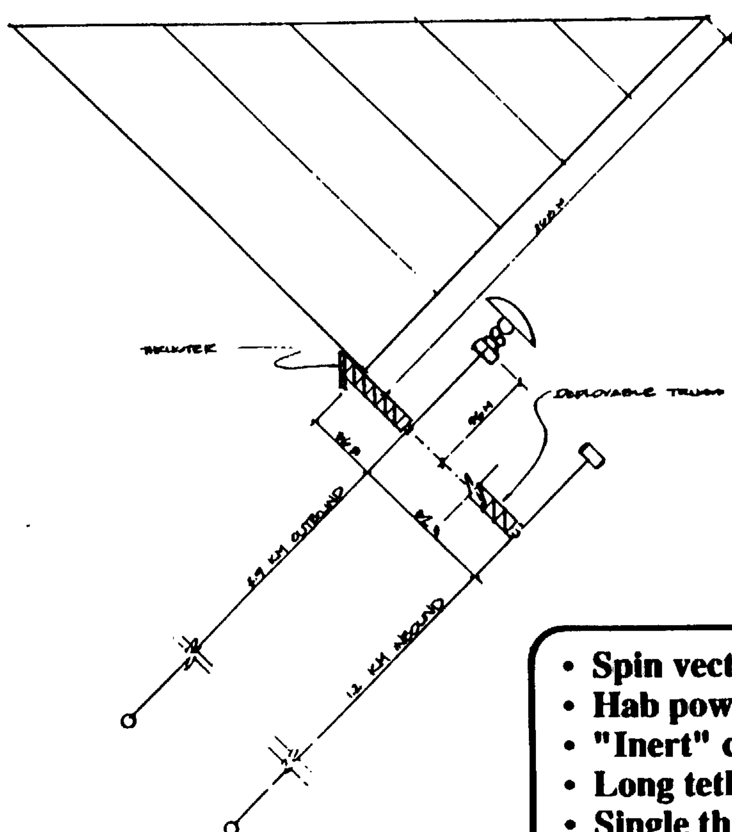
- Spin vector precession
- Hab power slipring assembly required
- "Inert" countermass required
- Long tether (~4 km)
- Single thruster pod
- 8 solar arrays - "fan" deployment
- Thrusters and arrays contiguous
- 50% truss savings over option 2A

g_a Solar Electric Propulsion Vehicle
Option 2C

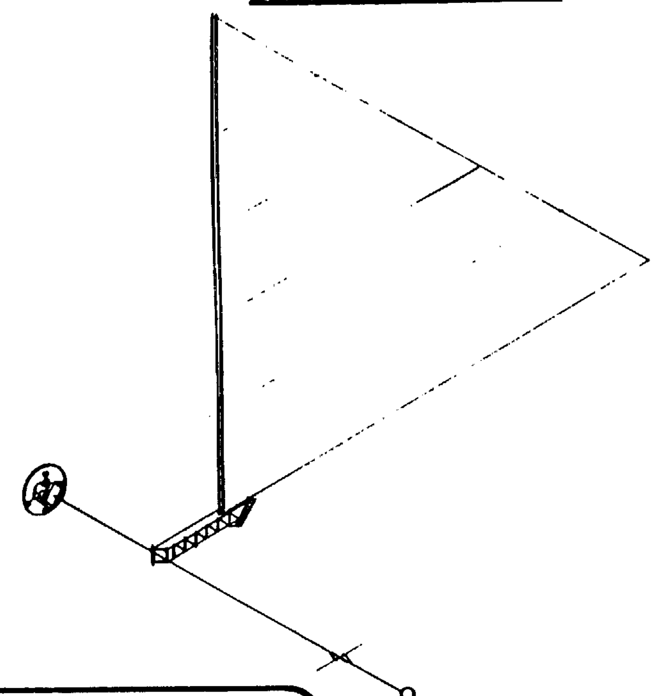
ADVANCED CIVIL SPACE SYSTEMS

BOEING
STCAEM/sdc/24May90

Orthogonal View



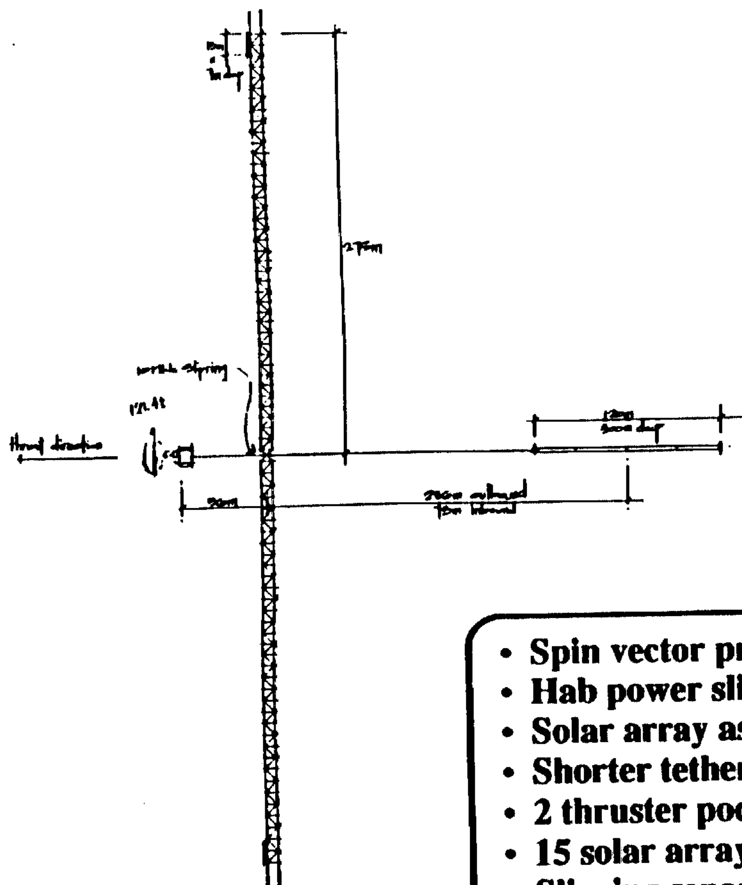
Isometric View



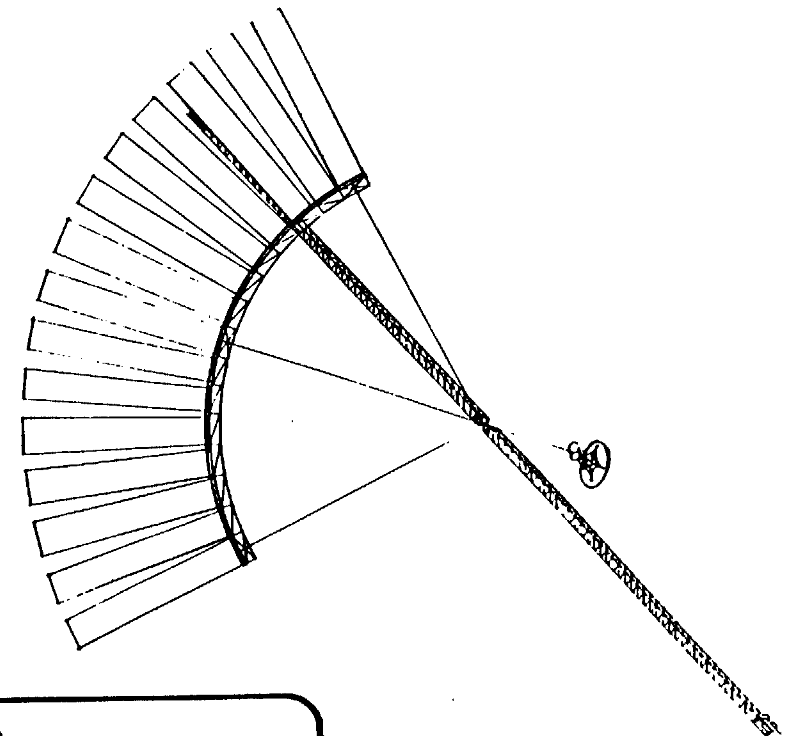
- Spin vector precession
- Hab power slipring assembly required
- "Inert" countermass required
- Long tether (~4 km)
- Single thruster pod
- Single solar array
- Thrusters and arrays contiquous
- Deployable truss - 35% less than option 2A

g_a Solar Electric Propulsion Vehicle Option 3

Orthogonal View



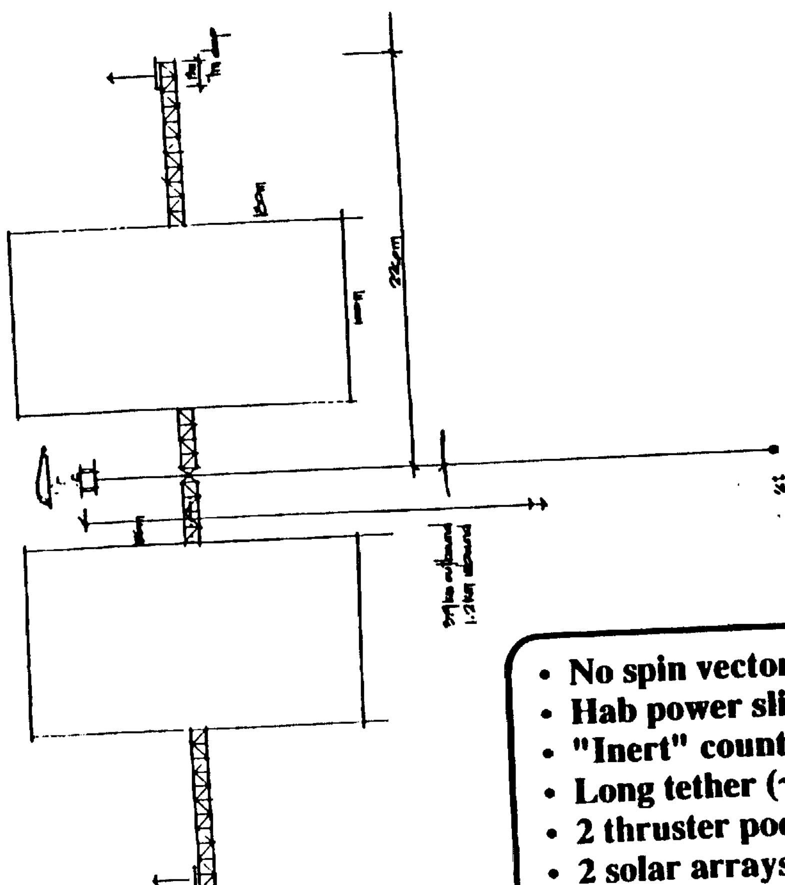
Isometric View



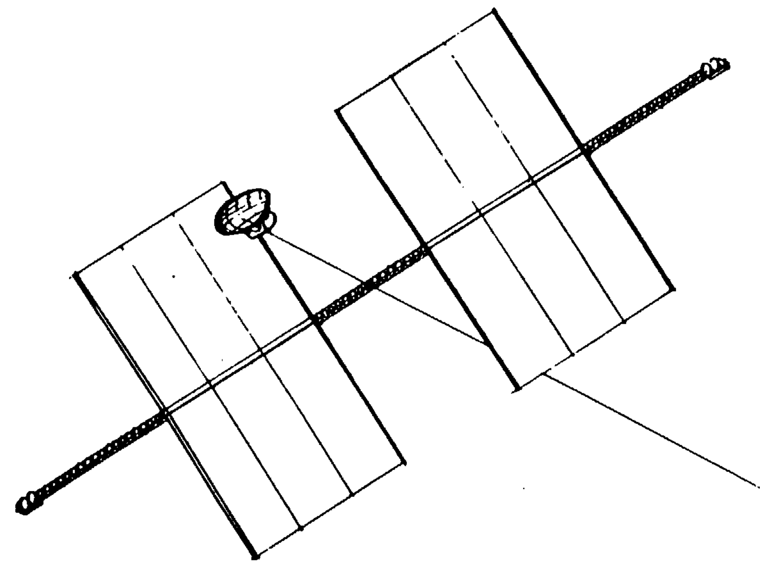
- Spin vector precession
- Hab power slipring assembly required
- Solar array as countermass
- Shorter tether (~250 m)
- 2 thruster pods
- 15 solar arrays - "radial" deployment
- Slipring separating thrusters and arrays
- 550 m of truss

g_a Solar Electric Propulsion Vehicle Option 4

Orthogonal View



Isometric View



- No spin vector precession
- Hab power slipring assembly required
- "Inert" countermass required
- Long tether (~4 km)
- 2 thruster pods
- 2 solar arrays
- Slipring separating 1 thruster pod and 1 array
- 450 m truss

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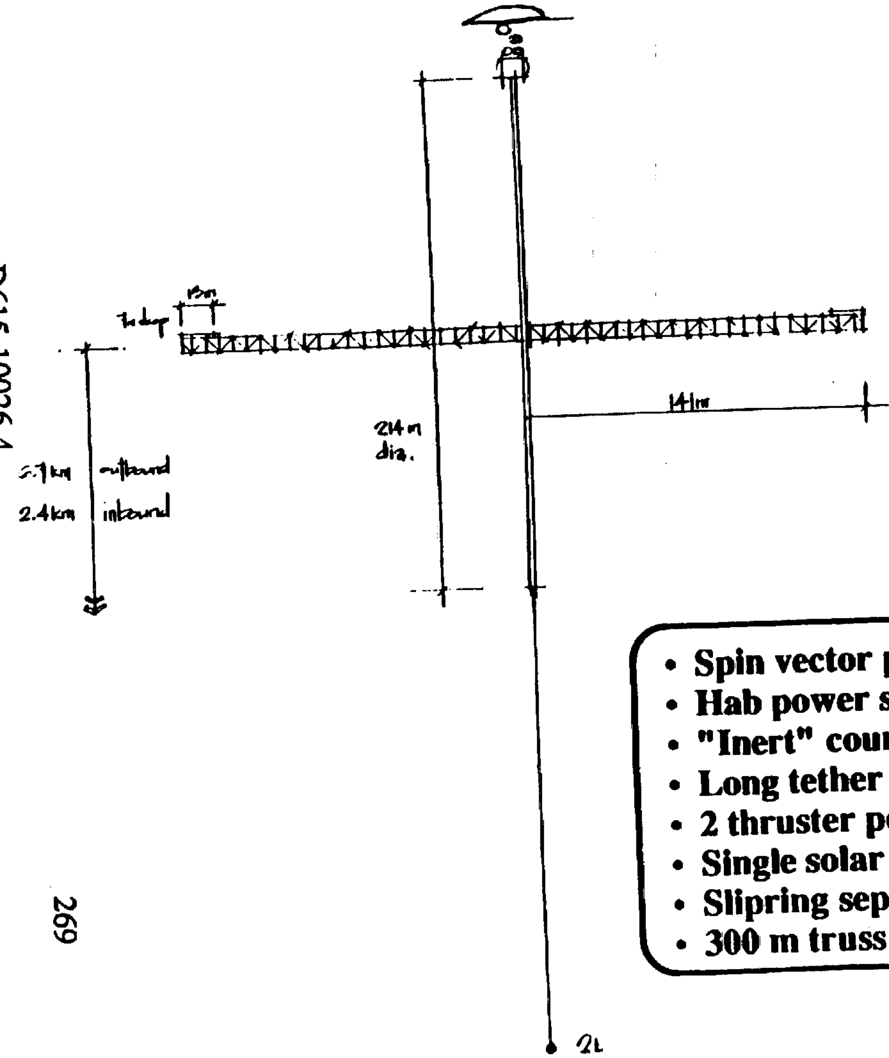
ADVANCED CIVIL SPACE SYSTEMS

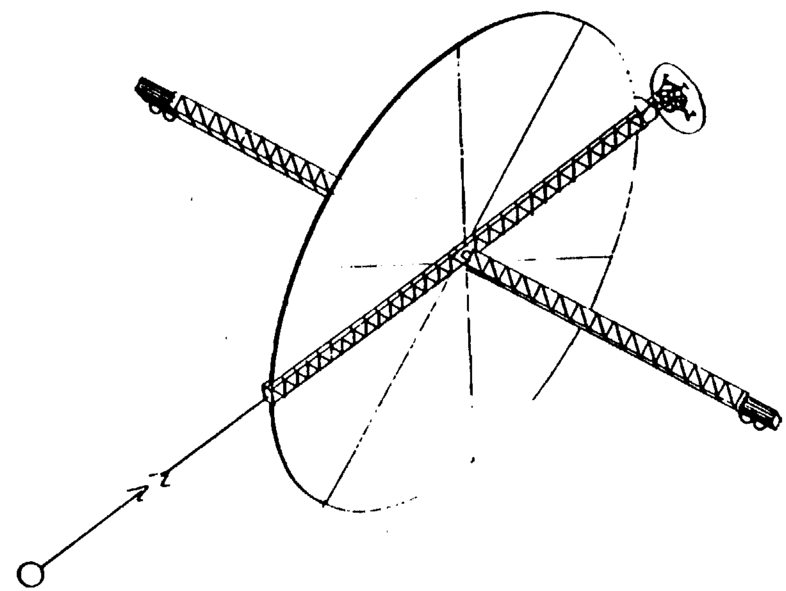
g_a Solar Electric Propulsion Vehicle
Option 5

BOEING
STCAEM/sdc/22May90

Orthogonal View



Isometric View



- Spin vector precession
- Hab power slipring assembly required
- "Inert" countermass required
- Long tether (~6.6 km)
- 2 thruster pods
- Single solar array - circular configuration
- Slipring separating thrusters and array
- 300 m truss

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Eccentric Thrusting for g_a Electric Vehicles

Through studying the g_a SEP problem (in which there is no obvious countermass for the swinging habitation system), a new concept for g_a low-thrust vehicles was developed, called the *eccentric rotator*. It works equally well for the NEP system, and has been chosen as the baseline approach. The vehicle is comprised of rigid structure, with no deployable components. It rotates about its center of mass, which shifts as payload is dropped during the mission. Dual engine assemblies, located on outriggers to prevent ion plume impingement, are despun. Although their thrust vector remains pointed in the same direction, the *source of the thrust (the engine assemblies themselves), orbits the vehicle mass center also*. For low-thrust systems, this is not a problem, because the vehicle's rotational inertia is much greater than can be practically affected by disturbances with a period equal to the gravity-inducing rotation. The moment caused by misalignment of the thrust vector and the CM is cancelled with each 2π rotation. The proper gravity level (1g is baselined) is maintained in the habitation system by adjusting the spin rate.

In a properly balanced vehicle design, the CM remains in the vicinity of the outrigger axis even though it shifts, so rotation-induced moments on the outrigger are always kept small compared to the impulse loading expected from normal assembly, maintenance, attitude-control and berthing maneuvers. As for all rotating vehicles, the structure's stiffness properties must be designed to suppress vibration modes close to the approximately 4 rpm forcing function frequency.

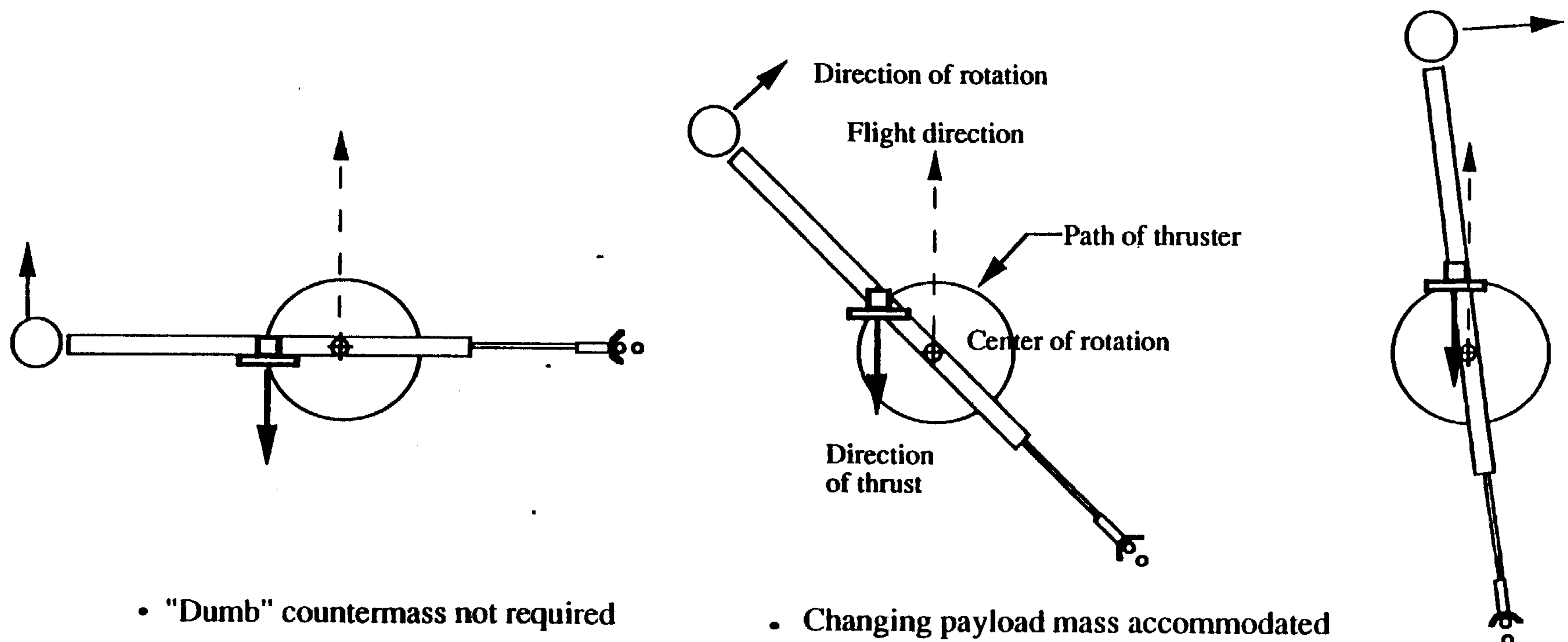
ADVANCED CIVIL SPACE SYSTEMS

Eccentric Thrusting for Artificial Gravity Electric Vehicles

BOEING

Rationale: Thruster pods and either solar arrays or radiators can be used as countermass

Conclusion: Continuous low thrust (10^{-4} g) produces an average thrust vector through both the center of mass and rotation



- "Dumb" countermass not required
- Direction of thrust remains constant
- Habitation radius can be fixed

- Changing payload mass accommodated through changing rotation rate
- Minimizes payload structure

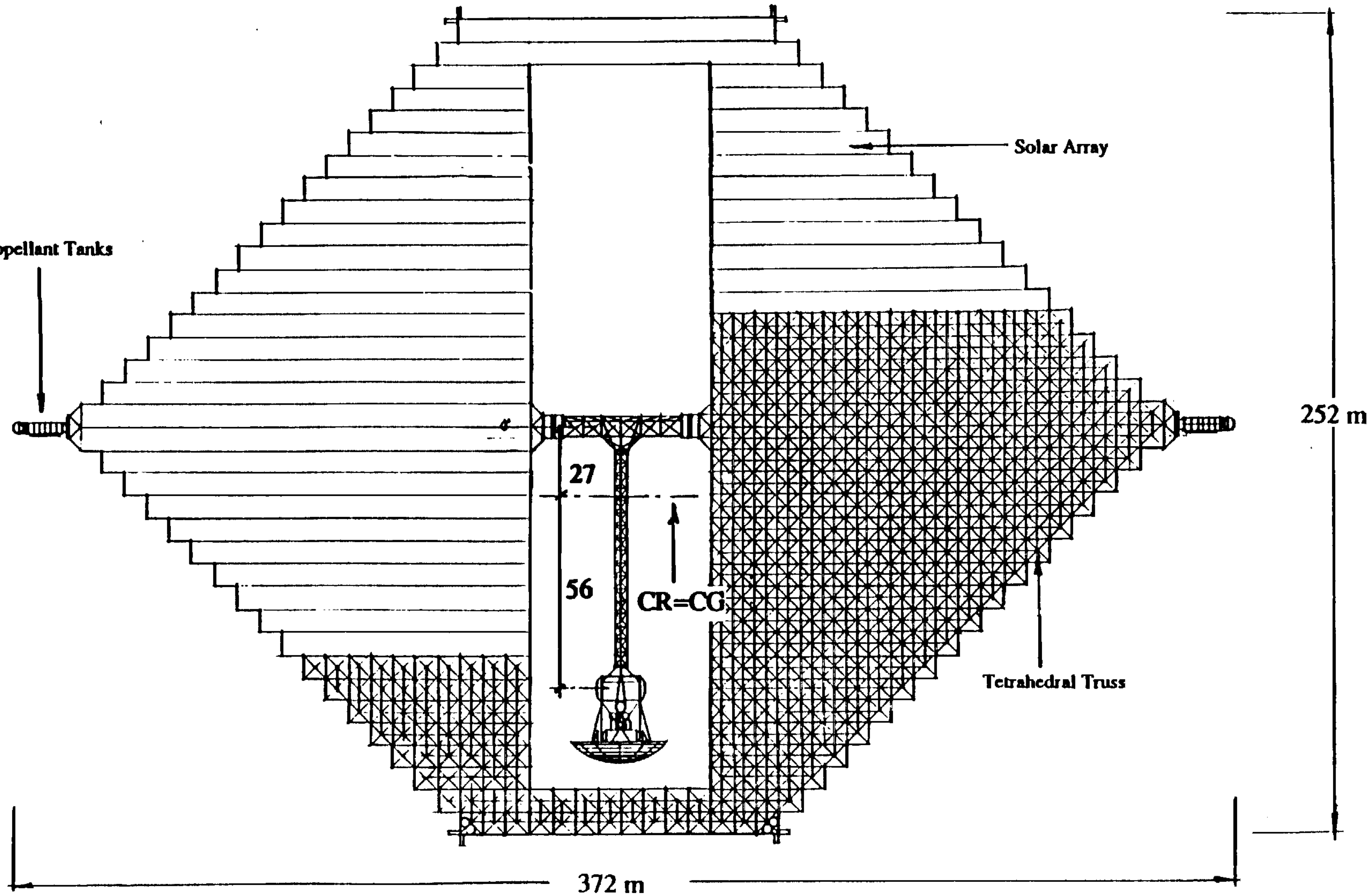
'STCAEM/crf/28Sep90

g_a SEP Eccentric Rotator Requirement

Shown is the artificial gravity version for the solar electric vehicle. This vehicle uses the eccentric rotating thruster concept, which allows the solar array and truss structure to act as a countermass for the habitat / payload, which revolves around the center of gravity of the entire vehicle at a distance of 56 meters and 4 rpm.

Cyclic loading of the tetrahedral truss structure will have an impact on the total mass of the vehicle, however, this impact has not as yet been determined.

ga SEP Eccentric Rotator Requirement



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V. Support Systems

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Support Systems for the Mars Solar Electric Propulsion Vehicle.

The support systems necessary for the Mars Solar Electric Propulsion Vehicle are very similar in nature to those of the Mars Cryo/Aerobrake Transfer Vehicle. The discussion provided for the latter vehicle also applies generally for the SEP; however, detailed analysis for the specific systems needed to support the SEP have not been completed. It is currently assumed that this study will mainly consist of only deltas from the Cryo/Aerobrake Vehicle. Some manifesting work has been done for the major components of the SEP (as given on the following pages) using two different HLLV scenarios (each assumes the integrated aerobrake "Ninja Turtle" launch concept):

1) 10 meter x 30 meter shroud, 140 metric ton payload capacity

2) Mixed fleet consisting of:
   a) 7.6 meter x 30 meter shroud, 120 metric ton payload capacity; and,
   b) 10 meter x 30 meter shroud, 84 metric ton payload capacity

The total number of assembly missions for Scenario One is 5, while Scenario Two requires 6 flights. For the mixed fleet option, only the first assembly mission utilizes the 120 mt payload carrier. This is due to SEP launch packages being much more limited by volume rather than by mass. Scenario One and Two also differ in that the first assumed that the MTV Hab should come up early (to assist in man-tended assembly operations) and the second brought up the MTV Hab late (for use in ground test and verification). The constraints of volume and MTV Hab sequencing were the major factors in the additional launch for Scenario Two.

The manifests given within have not yet been based on detailed ground processing and on-orbit assembly analyses. The philosophies and facilities chosen for ground operations (test and verification plans, payload processing, integrated assembly & checkout facilities, etc.) and assembly operations (Assembly Node location and capabilities, robotic and man-tended provisions, etc.) will obviously mature this manifesting.

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Space

SEP Earth Orbital Operations Trade

The objective of this trade was to determine optimum assembly/departure node locations as well as operation modes. The node locations under consideration were LEO, MEO, GEO, L_1, and L_2. The MEO node location was dismissed early due to unacceptable debris and radiation environments. Due to the slow spiral times associated with electric propulsion, environmental impacts become a primary driver in node selection. A top level trade is presented that points out the major criteria for a scenario selection. Several options are presented and evaluated based on the judgement criteria.

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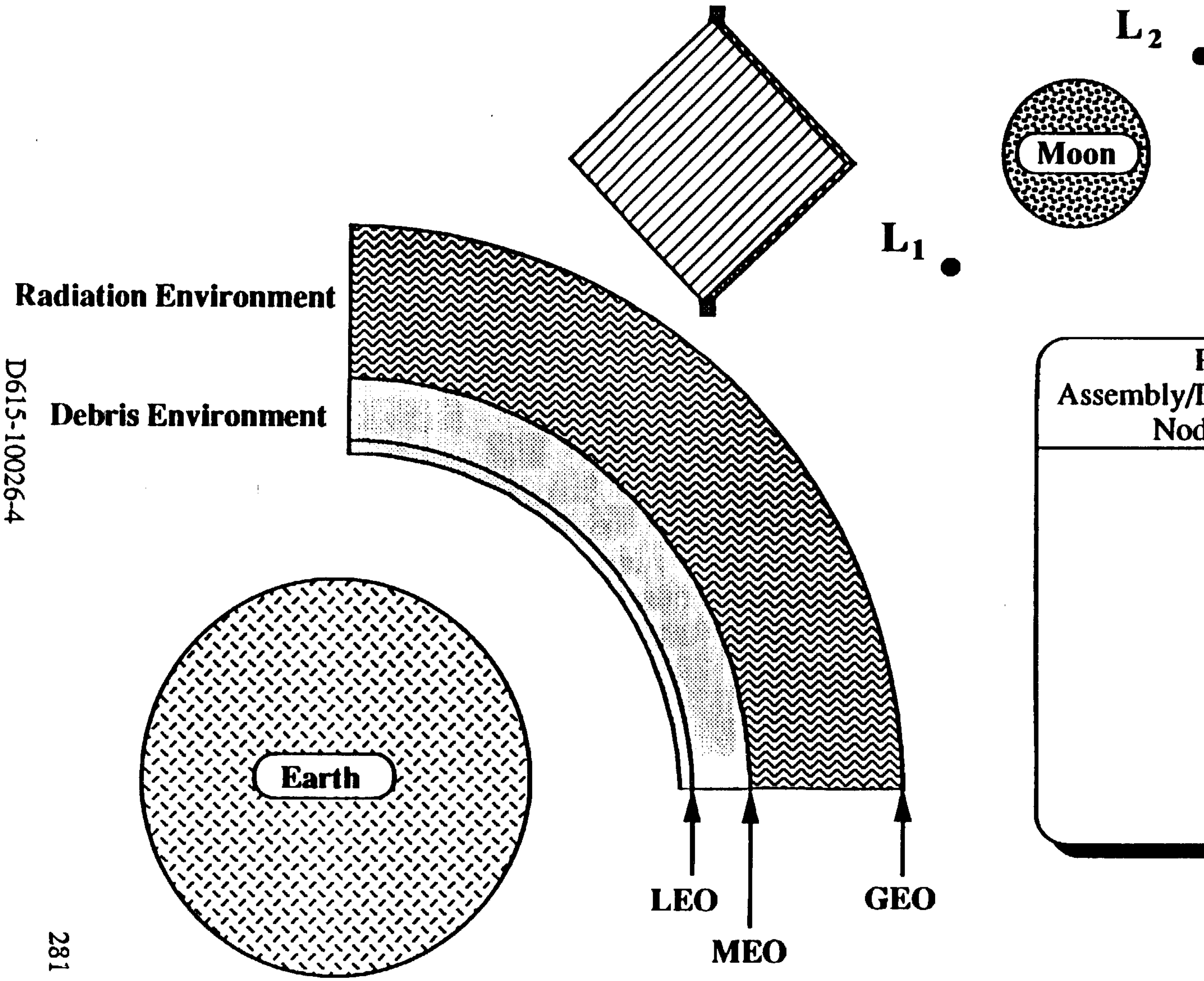
SEP Earth Orbital Operations Trade



Possible
Assembly/Departure/Return
Node Locations

LEO

MEO

GEO

L_1

L_2

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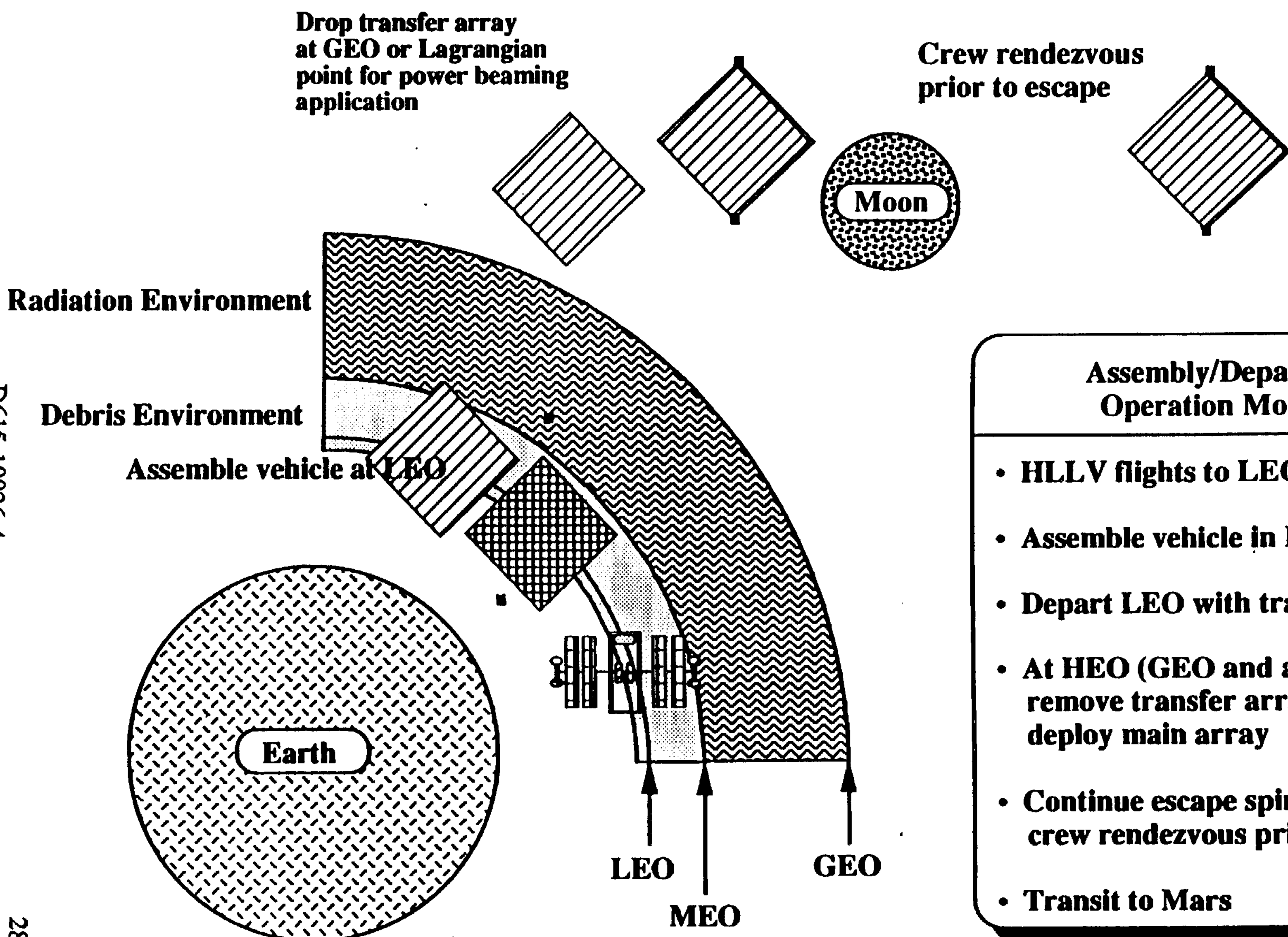
Option #1

The following two charts describe the assembly/departure operation modes and the advantages and disadvantages associated with option #1. In option #1, the vehicle is assembled in in LEO with the transfer array deployed and the main array in a stowed configuration. The vehicle spirals out of the van Allen belts with its own propulsion system where the transfer array is left for another possible application. At this point, the vehicle deploys the main array and initiates the earth escape sequence. Prior to Earth escape, the crew will rendezvous with the SEP vehicle via LTV a few days prior to Earth escape.

Option #1



Assembly/Departure Operation Modes

- HLLV flights to LEO
- Assemble vehicle in LEO
- Depart LEO with transfer array
- At HEO (GEO and above), remove transfer array and deploy main array
- Continue escape spiral with crew rendezvous prior to escape
- Transit to Mars

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Option #1 Advantages and Disadvantages

Advantages	Disadvantages
• Use of transfer array eliminates the issue of radiation degradation to the main array. • Transfer array eliminates the large mass penalty associated with a chemical boost stage. • This option does not require any other support vehicles for the van Allen belt transfer. • Option requires no extra structural support requirements since the transfer employs its own propulsion system. • Transfer array can be used at a high altitude node for another application, as power beaming.	• Solar arrays are a major cost in the program. Current costs are on the order of $1250 per watt. If costs cannot be reduced due to the large procurement, this will be a costly penalty. • Slow spiral through the debris and radiation environment will increase probability of damage to vehicle.

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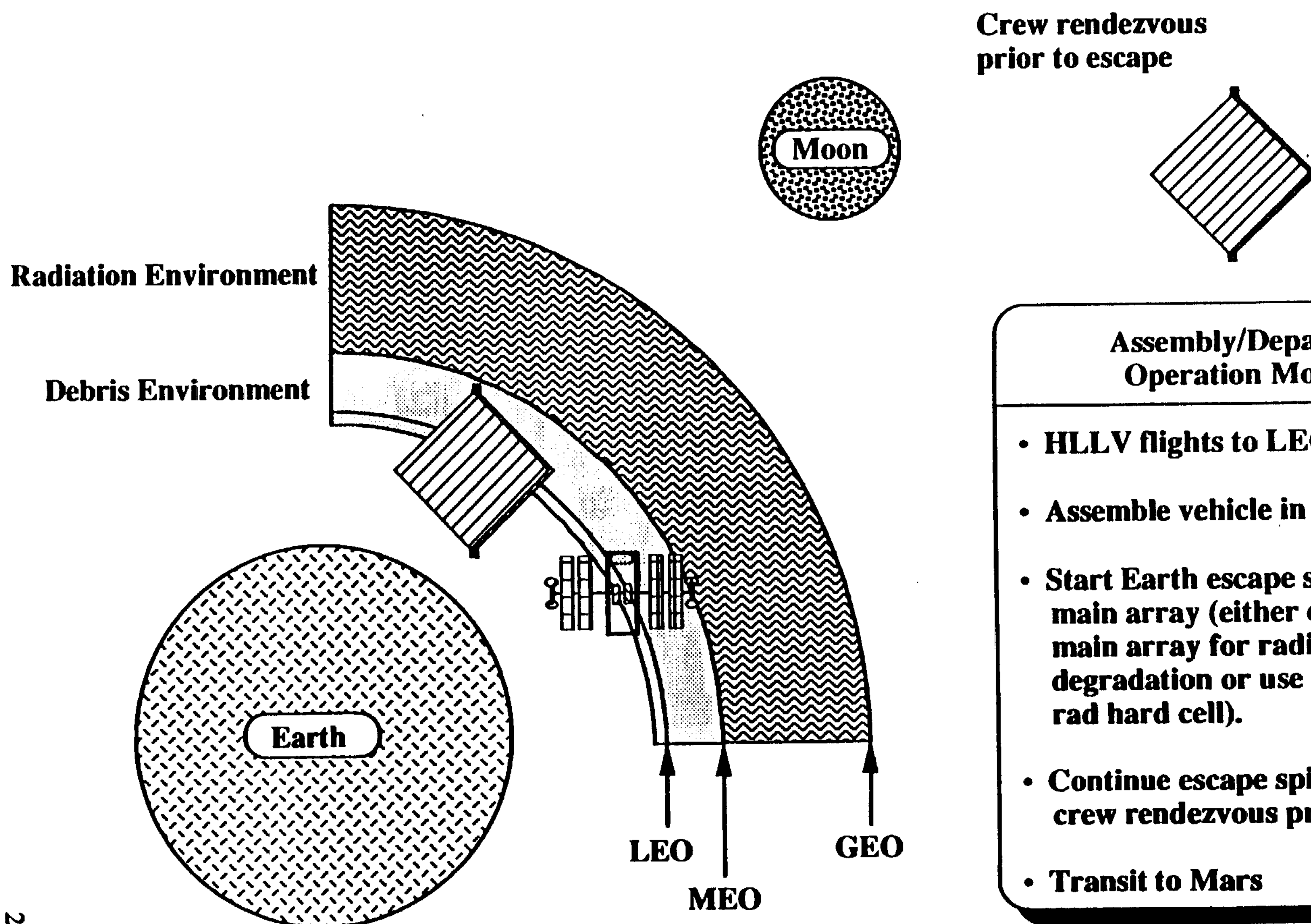
Option #2

The following two charts describe the assembly/departure operation modes and the advantages and disadvantages associated with option #2. In option #2, the vehicle is fully assembled in LEO with the main array. The main array can be oversized to account for the radiation degradation or the array can be composed of a radiation resistant solar cell. The radiation resistant solar cell (I_nP) will tolerate more passes through the van Allen belts than would an oversized array, therefore the radiation resistant solar cell should be favored. After assembly, the vehicle initiates the Earth escape sequence with crew rendezvous via LTV A few days prior to Earth escape.

Option #2



Assembly/Departure
Operation Modes

- HLLV flights to LEO
- Assemble vehicle in LEO
- Start Earth escape spiral with main array (either oversize main array for radiation degradation or use less efficient rad hard cell).
- Continue escape spiral with crew rendezvous prior to escape
- Transit to Mars

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Option #2 Advantages and Disadvantages

Advantages	Disadvantages
• Transfer from LEO to HEO with electric propulsion eliminates the large mass penalty associated with a chemical boost stage. • This option does not require any other support vehicles for the van Allen belt transfer. • Option requires no extra structural support requirements since the transfer employs its own propulsion system. • Use of the vehicles dedicated propulsion system eliminates cost of transfer array or another form of boost stage. • Operations are more simple than other options.	• If the main array is used for the spiral through the van Allen belts, the array will experience considerable radiation degradation which will result in the vehicle having to transfer useless mass to Mars and back. The array will also be oversized by ~35% to make up for the damage. or If a rad hard cell is used that will not experience radiation degradation, the cells (indium phosphide) will have a lower efficiency, resulting in a larger array area and a higher vehicle mass. • Slow spiral through the debris and radiation environment will increase probability of damage to vehicle.

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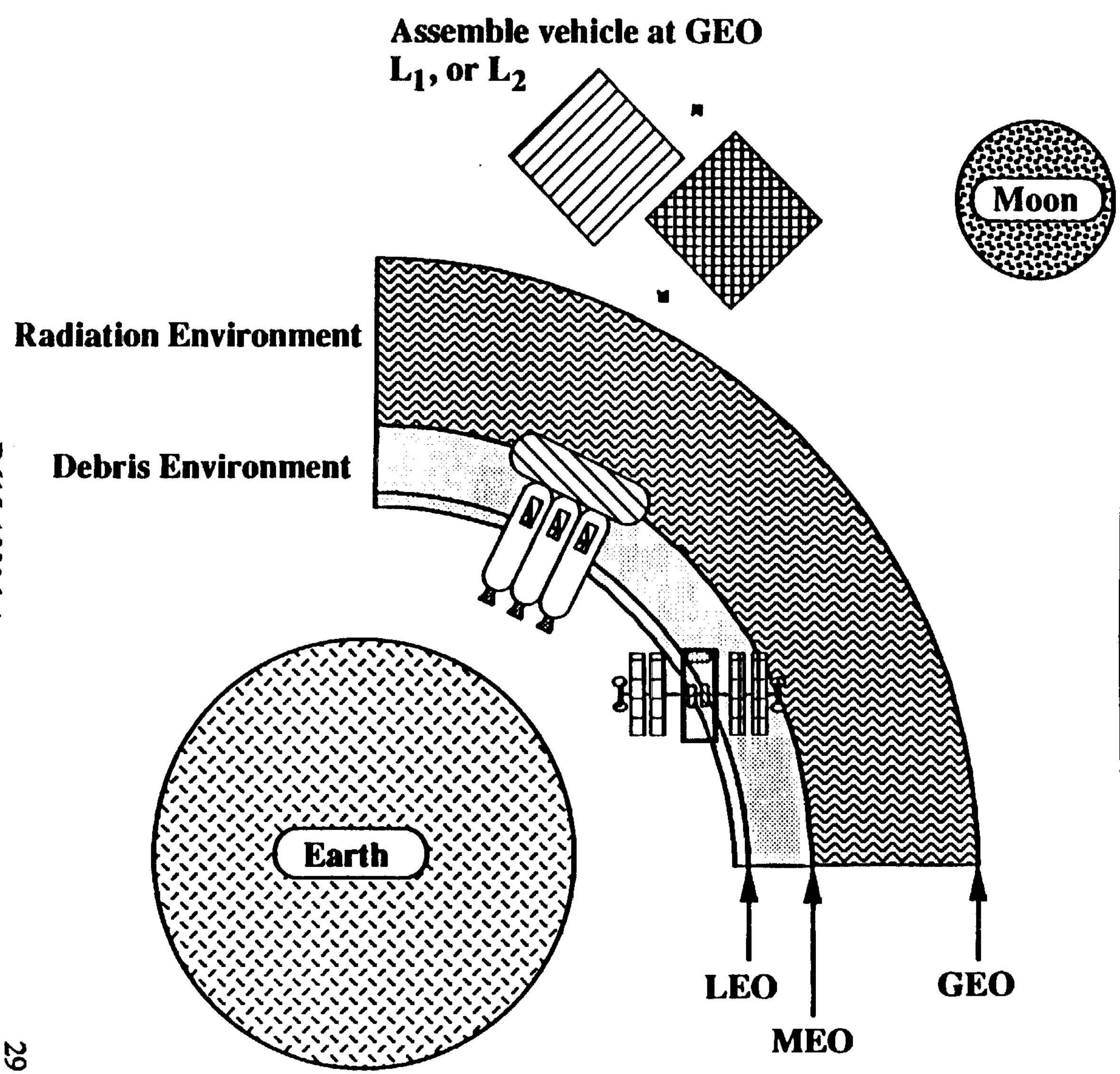
Option #3

The following two charts describe the assembly/departure operation modes and the advantages and disadvantages associated with option #3. In option #3, the vehicle components are boosted with a chemical stage to a high Earth orbit (out past the van Allen belts) for assembly. Once the vehicle is assembled at this node, the vehicle initiates the Earth escape sequence. Prior to Earth escape, the crew will rendezvous with the SEP vehicle via LTV a few days prior to Earth escape.

Option #3



Assembly/Departure Operation Modes

- HLLV flights to LEO
- Boost components to HEO (GEO, L_1, L_2) with chemical stage
- Assemble vehicle at this node
- Start escape spiral
- Continue escape spiral with crew rendezvous prior to escape
- Transit to Mars

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Option #3 Advantages and Disadvantages

Advantages	Disadvantages
• There is no radiation degradation to the main array. • An extra array is not required. • Option requires no extra structural support requirements since the vehicle is not assembled during the boost phase. • The SEP vehicle does not spent a lengthly amount of time in the radiation or debris environment, reducing the risk of damage.	• A chemical boost stage is heavy. The effective IMLEO of this option will be twice that of the SEP vehicle IMLEO. • More support hardware and transfer vehicles are associated with this scenario.

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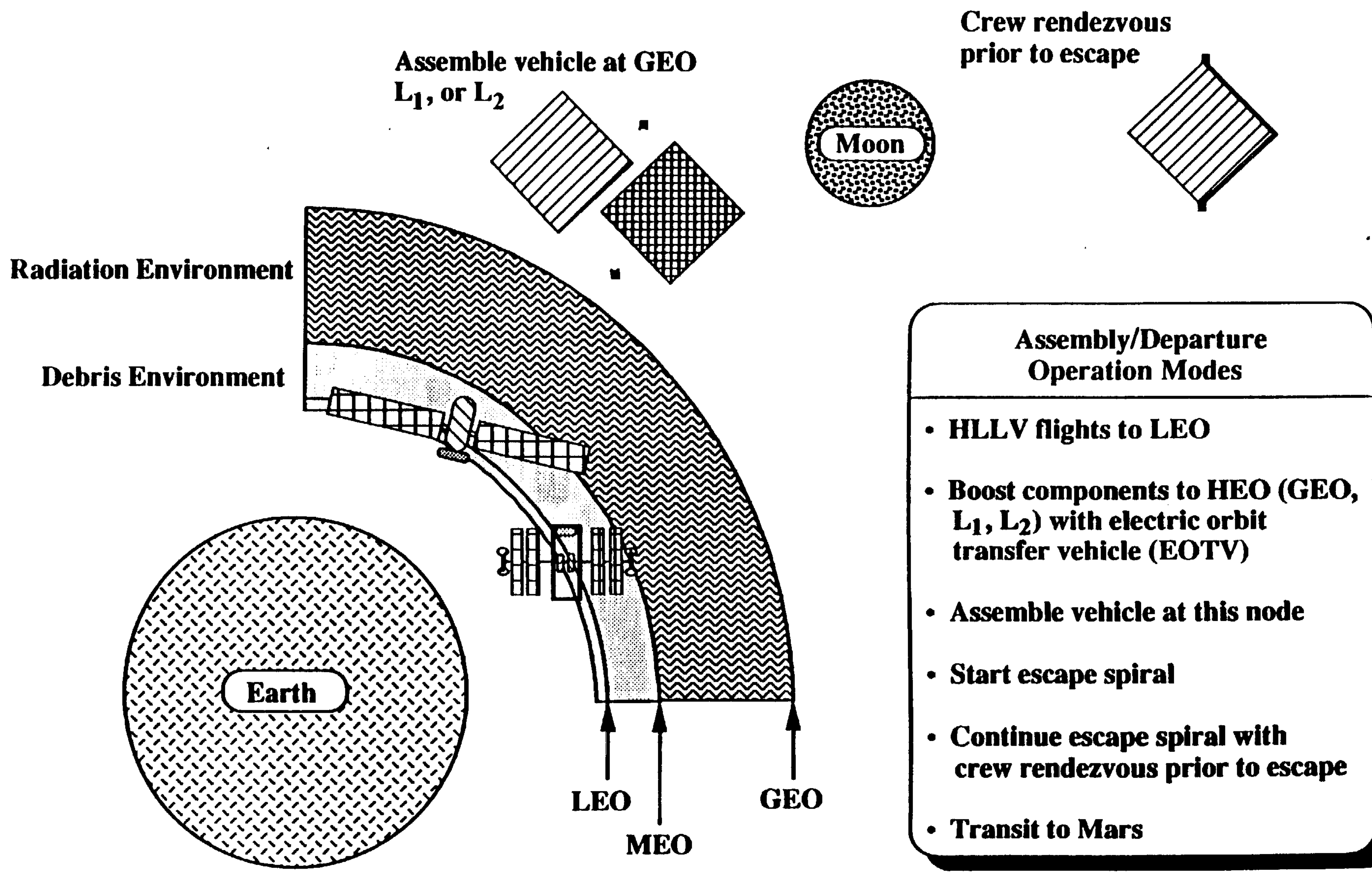
Option #4

The following two charts describe the assembly/departure operation modes and the advantages and advantages associated with option #4. In option #4 , the vehicle components are boosted with an electrical orbital transfer vehicle (EOTV) to a high Earth orbit (out past the van Allen belts) for assembly. The EOTV should be a beamed power EOTV to avoid occultation and drag associated with a solar powered ETOV and radiation impacts associated with a nuclear powered vehicle. Once the vehicle is assembled at this node, the vehicle initiates the Earth escape sequence. Prior to Earth escape, the crew will rendezvous with the SEP vehicle Via LTV a few days prior to Earth escape.

Option #4



Assembly/Departure Operation Modes

- HLLV flights to LEO
- Boost components to HEO (GEO, L_1, L_2) with electric orbit transfer vehicle (EOTV)
- Assemble vehicle at this node
- Start escape spiral
- Continue escape spiral with crew rendezvous prior to escape
- Transit to Mars

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Option #4 Advantages and Disadvantages

Advantages	Disadvantages
• Use of an EOTV reduces the effective IMLEO when compared to a chemical boost stage. • An infrastructure that includes an EOTV will provide an excellent testbed and data base for Mars missions that employ electric propulsion. • The EOTV can be used for other orbital transfer applications, reducing the cost burden on the Mars program. • The EOTV can also provide the most economical means of furnishing resupply requirements for future missions. • Option requires no extra structural support requirements since the vehicle is not assembled during the boost phase.	• More support hardware and transfer vehicles are associated with this scenario. • An electric propulsion orbit transfer is a relatively slow transfer, which will require modest power levels to keep up with the HLLV manifest.

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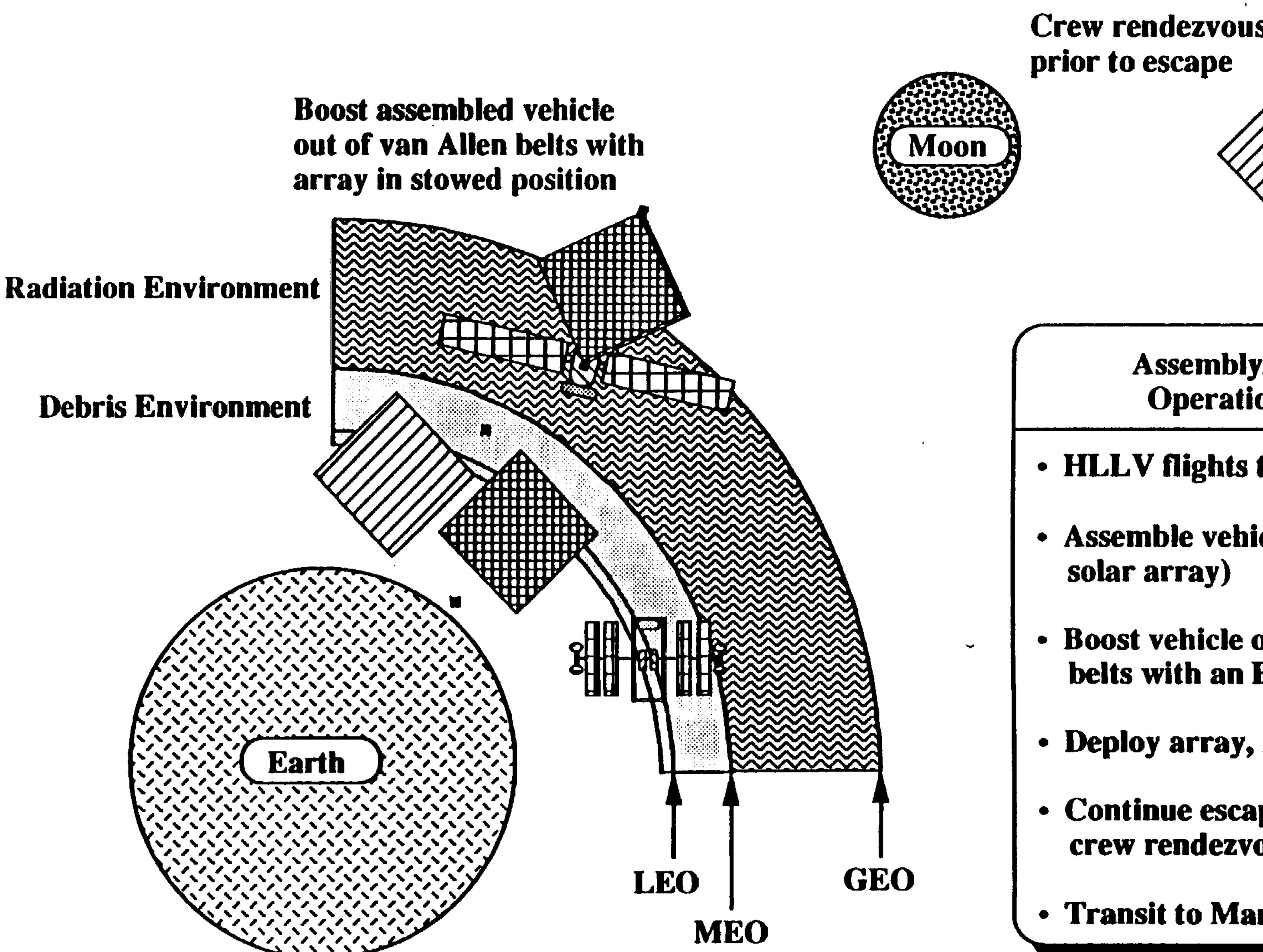
Option #5

The following two charts describe the assembly/departure operation modes and the advantages and disadvantages associated with option #5. In option #5, the vehicle is assembled in LEO with the main array in stowed configuration. The vehicle is then "tugged" to a higher Earth orbit by an EOTV (powered as an option #4). The EOTV will be much larger than in option #4 due to the increased payload. Once the vehicle is clear of the radiation belts, the main array will be deployed and the vehicle will initiate the Earth escape sequence. Prior to Earth escape, the crew will rendezvous with the SEP vehicle via LTV a few days prior to Earth escape.

Option #5



Assembly/Departure Operation Modes

- HLLV flights to LEO
- Assemble vehicle in LEO (except solar array)
- Boost vehicle out past van Allen belts with an EOTV.
- Deploy array, initiate spiral
- Continue escape spiral with crew rendezvous prior to escape
- Transit to Mars

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Option #5 Advantages and Disadvantages

Advantages	Disadvantages
• Use of an EOTV reduces the effective IMLEO when compared to a chemical boost stage. • An infrastructure that includes an EOTV will provide an excellent testbed and data base for Mars missions that employ electric propulsion. • The EOTV can be used for other orbital transfer applications, reducing the cost burden on the Mars program. • The EOTV can also provide the most economical means of furnishing resupply requirements for future missions.	• More support hardware and transfer vehicles are associated with this scenario. • An electric propulsion orbit transfer is a relatively slow transfer, which will require modest power levels to keep up with the HLLV manifest. • Option requires extra structural support requirements since the vehicle is assembled during the boost phase. • There is two assembly points, ie. the vehicle is assembled at LEO and the solar array is deployed at GEO.

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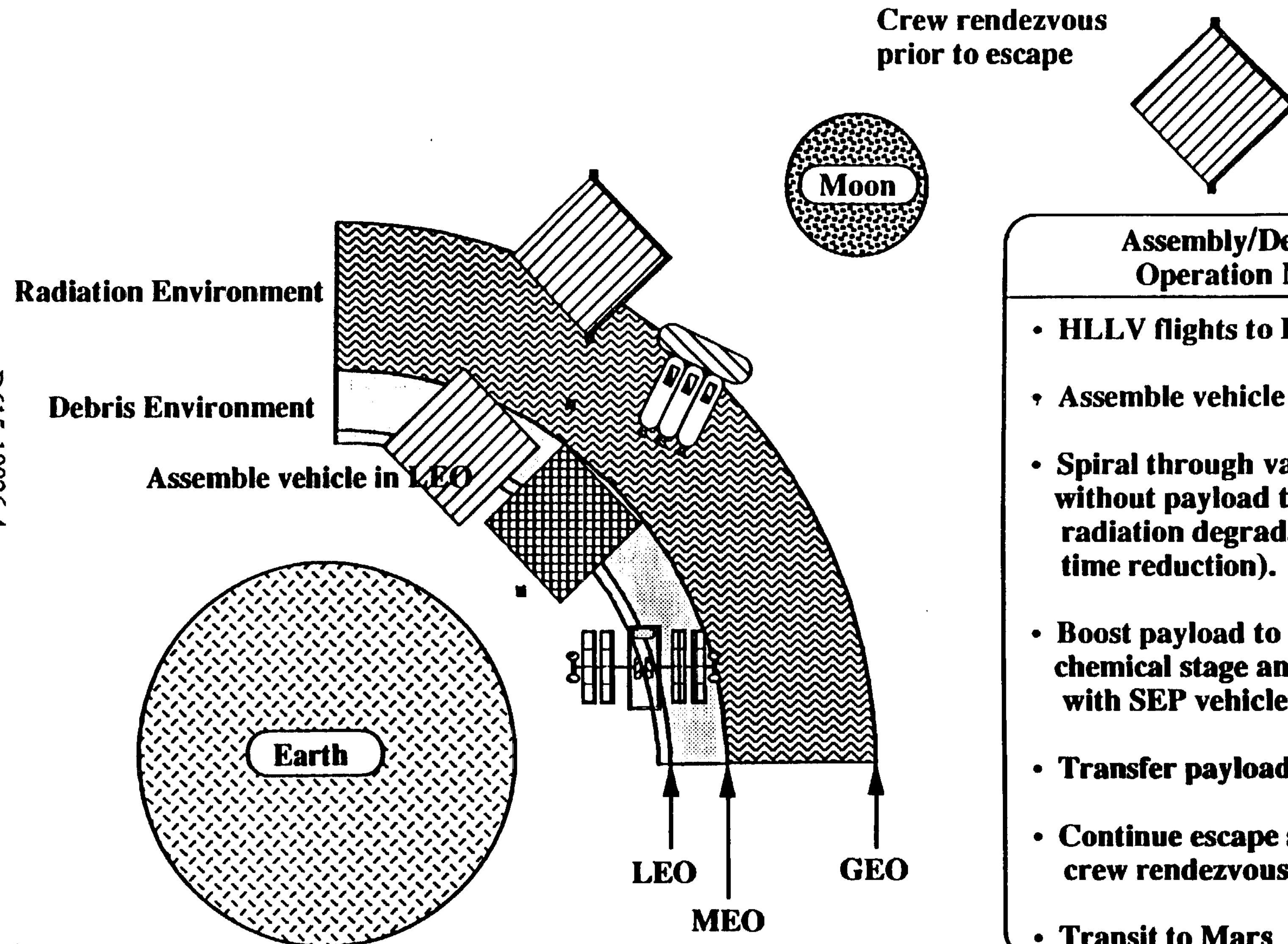
Option #6

The following two charts describe the assembly/departure operation modes and the advantages and disadvantages associated with option #6. In option #6, the vehicle is fully assembled in LEO without the payload. The vehicle then spirals out past the radiation belts to wait for payload rendezvous. This transfer without the payload allows for a fast spiral time which reduces the amount of debris and radiation degradation to the SEP vehicle. The payload is then sent to the SEP vehicle by means of a chemical boost stage. Once the payload has been attached, the SEP vehicle begins the Earth escape sequence. Prior to Earth escape, the crew will rendezvous with the SEP vehicle via LTV a few days prior to Earth escape.

Option #6



Assembly/Departure Operation Modes

- HLLV flights to LEO
- Assemble vehicle in LEO
- Spiral through van Allen belts without payload to reduce radiation degradation (spiral time reduction).
- Boost payload to GEO with chemical stage and rendezvous with SEP vehicle.
- Transfer payload, start spiral
- Continue escape spiral with crew rendezvous prior to escape
- Transit to Mars

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Option #6 Advantages and Disadvantages

Advantages	Disadvantages
• Use of a chemical boost stage for the payload will result in less damage to the SEP vehicle due to the decreased time in the debris and radiation environments. • Option requires no extra structural support requirements since the transfer employs its own propulsion system.	• Delta V requirements for the LEO to GEO transfer will result in a heavy mass penalty for the use of a chemical boost stage for the payload. • More support hardware and transfer vehicles are associated with this scenario. • Although the SEP vehicle will spiral through the debris and radiation environments much faster without the payload, the vehicle will still experience some damage.

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SEP Mission Selection Criteria

The following chart is a list of selection criteria on which the different options are judged. Along with the criteria is a rational behind the criteria selection. The criteria are used in the trade matrix for option evaluation.

SEP Missions Selection Criteria

Criteria	Rationale
1. Total IMLEO	• Missions with less IMLEO are favored.
2. Orbital Debris Shield Mass	• Missions operating in LEO will require greater debris shielding
3. Mass @ Spiral Initiation	• Missions where payload and main vehicle are integrated separately may impact this parameter.
4. Mass @ Departure	• Total mass at departure.
5. Isp (transfer stage)	• Isp directly correlates with propellant mass.
6. Spiral Time	• The less the spiral time, the less the array degradation, the lower the probability of debris hits, the greater the mission efficiency.
7. DeltaV	• Mission with less total deltaV is favored.
8. Days Exposure to Radiation	•The greater the time the SEP spends in the Van Allen belts, the greater the degree of solar array degradation.
9. Days Exposure to Orbital Debris	•The greater the time the SEP spends in LEO, the damage to the solar array due to orbital debris.
10. Total Mission Time	• Driven by assembly time and spiral time.
11. Resiliency	• Time required to recover from failure mode. Higher nodes require greater time to recover from a failure.
12. Infrastructure Cost	• Assembly/departure node locations may drive mission costs
13. Infrastructure Complexity	• Missions requiring extensive space based support hardware, are more complex and are subject to increased chances of failure.
14. % Degradation of Solar Array	• Each mission will result in varying degrees of array degradation.
15. Total HLLV Flights	• Certain mission options may result in a less number of total HLLV assembly missions.
16. Reusability of Used Hardware	• Transfer array on certain mission options may be reused for power beaming applications.
17. GCR Exposure to crew	• Operating in higher nodes increases amount of GCR received by crew during assembly and checkout.
18. Flight Proven Technology	• An EOTV provides a proving ground for SEP technology prior to mission.

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SEP Mission Options Trade

The following chart is a trade matrix that shows how the options were judged against a set of weighted criteria. The criteria were assigned a normalized weighting factor based on priority determination. The options were then compared to each other with actual numbers used wherever possible. The options were then assigned a number of 1-5, with 5 being the most favorable, based on the comparison. The 1-5 numbers were multiplied by the weights, with this result being summed for the total score. Changing the criteria priority can result in a different outcome. The result of this trade indicates that the two favored options are the transfer array scenario and a vehicle with a radiation resistant solar array. Option 5 would also look favorable if an EOTV infrastructure was already in place. The results from a refurbishment node location/operations mode trade should be integrated into this analysis to establish a more solid near Earth orbit operations baseline.

SEP Mission Options Trade

Criteria	SEP Mission Options												
	Option 1		Option 2		Option 3		Option 4		Option 5		Option 6		Weights
1. Total IMLEO	5	4.15	5	4.15	1	.83	4	3.32	3	2.49	3	2.49	.83
2. Orbital Debris Shield Mass	1	.56	1	.56	5	2.80	5	2.80	1	.56	2	1.12	.56
3. Mass @ Spiral Initiation	2	.66	2	.66	3	.99	3	.99	3	.99	2	.66	.33
4. Mass @ Departure	3	.84	1	.28	3	.84	3	.84	3	.84	3	.84	.28
5. Isp (transfer stage)	5	3.60	5	3.60	1	.72	5	3.60	5	3.60	3	2.16	.72
6. Spiral Time	1	.50	2	1.00	5	2.50	5	2.50	5	2.50	4	2.00	.50
7. DeltaV	5	3.90	5	3.90	2	1.56	1	.78	5	3.90	4	3.12	.78
8. Days Exposure to Radiation	1	.39	2	.78	5	1.95	5	1.95	5	1.95	4	1.56	.39
9. Days Exposure to Orbital Debris	1	.44	1	.44	5	2.20	5	2.20	1	.44	2	.88	.44
10. Total Mission Time	3	1.83	1	.61	5	3.05	2	1.22	1	.61	4	2.44	.61
11. Resiliency Time	5	4.10	5	4.10	1	.82	1	.82	5	4.10	4	3.28	.82
12. Infrastructure Cost	4	3.80	5	4.75	3	2.85	2	1.90	3	2.85	4	3.80	.95
13. Infrastructure Complexity	4	.68	5	.85	2	.34	1	.17	2	.34	3	.51	.17
14. % Degradation of Solar Array	5	3.35	4	2.68	5	3.35	5	3.35	5	3.35	2	1.34	.67
15. Total HLLV Flights	5	4.45	5	4.45	1	.89	4	3.56	3	2.67	3	2.67	.89
16. Reusability of Used Hardware	3	.03	1	.01	1	.01	5	.05	5	.05	1	.01	.01
17. GCR Exposure to Crew	3	.33	5	.55	1	.11	1	.11	3	.33	4	.44	.11
18. Flight Proven Technology	1	.06	1	.06	1	.06	5	.30	5	.30	1	.06	.06
Total Scores	33.67		33.43		25.87		30.46		31.87		29.38		

1 unfavorable 2 3 average 4 5 favorable

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Key Findings - Operations Trade

- A change in criteria priority can change the trade outcome.
- Assumptions for the trade assumed that no infrastructure was in existence that included an EOTV.
- If an EOTV is already in place, the favored option could change.
- The trade revealed that using a transfer array or the use of a less efficient, radiation resistant cell are the most favored options.
- If there is a use for the expended transfer array, such as a power beaming application, the transfer array option would be favored.
- If there is not a use for the expended array, the radiation resistant cell option may be favored.
- Options with more support hardware seem to be less favored.

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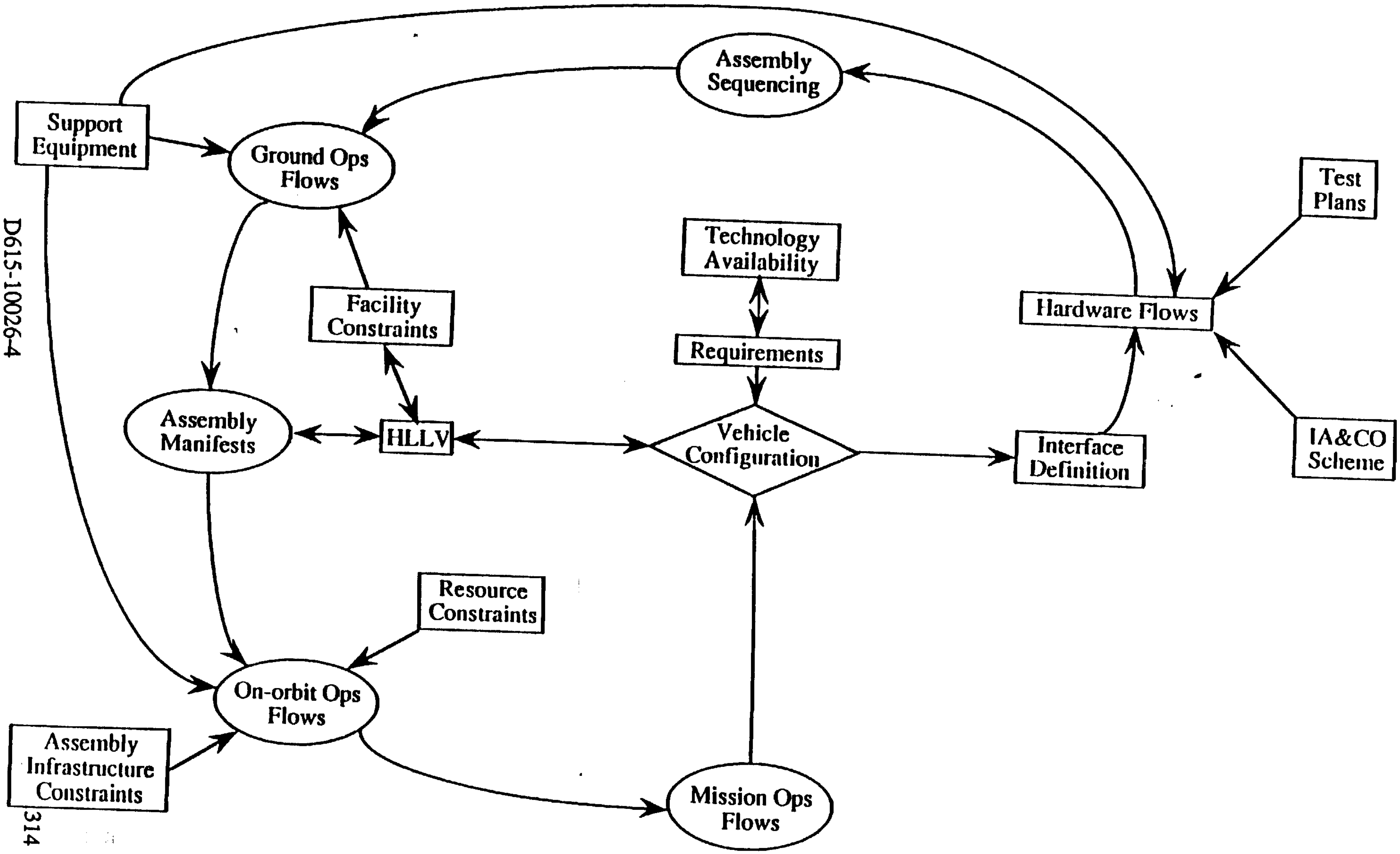
Operations Analyses and On-Orbit Assembly Concepts for NTR, NEP, and SEP

- Groundrules and Assumptions
- Assembly Node Concepts
- Manifesting/Packaging
- Assembly Flows
- Ground Processing
- Summary

Manifesting and Assembly Operations

This is an iterative, interdependent process

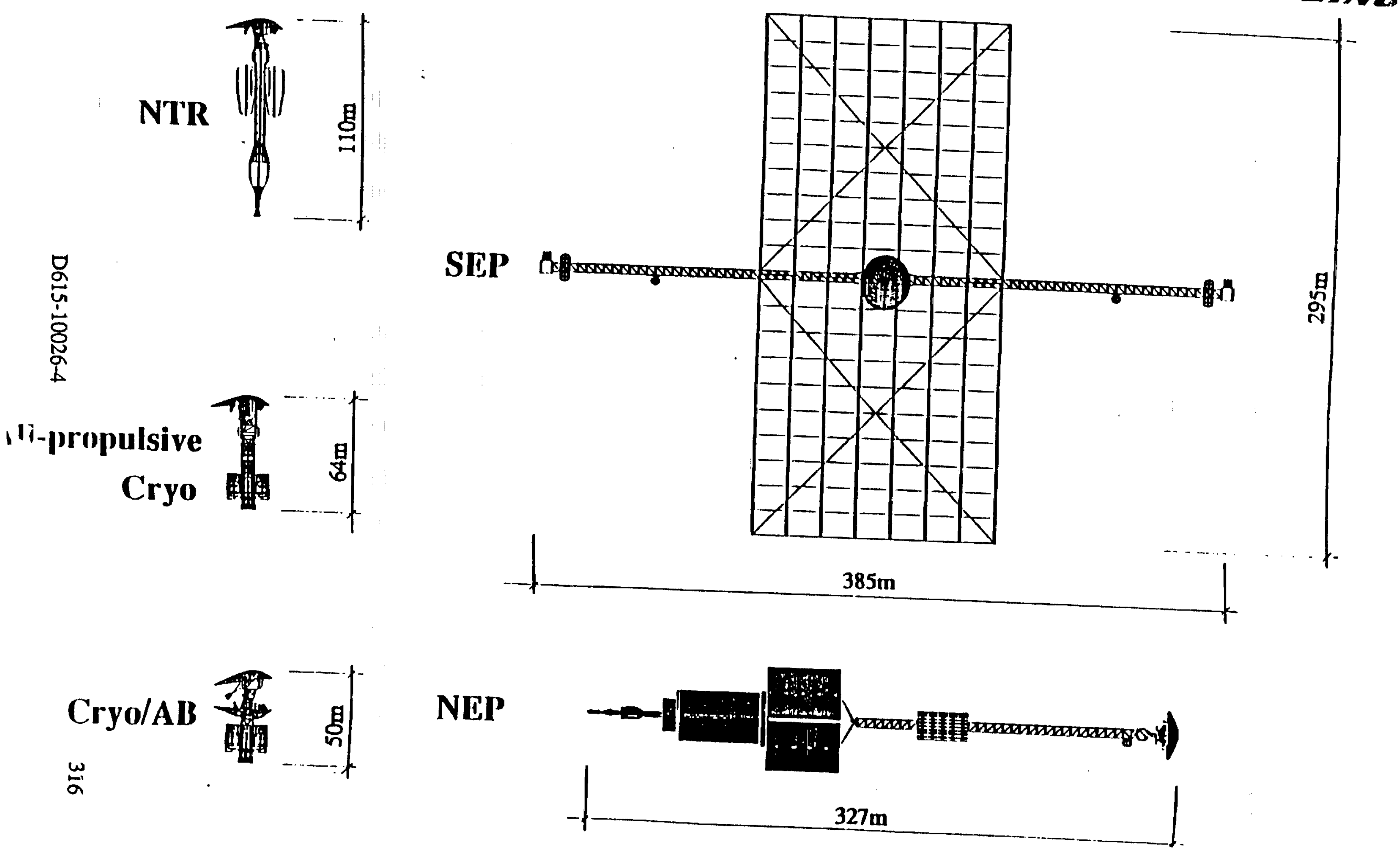


Manifesting and Assembly Operations- continued

Generic Assumptions and Ground Rules

- Based on Mars Vehicle (NEP, SEP, and NTR) configurations as of 3rd Quarterly Briefing with updates through 8/15/90
- Baseline Earth-to-Orbit (ETO) Vehicle (HLLV) has 10m x 30m shroud with 140 mt payload capability
- HLLV nosecone has some additional TBD volume for launch element packaging
- Nominal 85% payload packaging and mass factors used for HLLV manifesting (propellant tanks may be excepted)
- HLLV has a nominal 3 to 7 day station-keeping ability
- HLLV unloaded piece by piece by Cargo Transfer Vehicle (CTV)
- Crew transported to Assembly Location from SSF via ACRV
- CTV will be designed to support all identified manned/unmanned operations (on-orbit refueling may be available via on-orbit depot, HLLV provisioning, the Mars Vehicle itself, or SSF)
- HLLV launched on 90 day centers = time constraint for on-orbit assembly operations
- All Mars Vehicles are assumed to be launched February 2016
- Any localized debris shielding is removed from Mars Vehicle prior to departure from Earth (micrometeoroid shielding is assumed to be needed for the mission duration

Reference Vehicles
Size Comparison



On-Orbit Assembly Considerations

Required System/Service for Assembly	Available from Vehicle Itself ?
• Power	Yes
• Thermal Control	Yes
• Communications	Yes
• Micrometeoroid/Debris Protection	Possible with Localized/Temporary Shielding
• Reboost	Limited by Propellant
• Attitude Control	Limited by Propellant (gravity gradient should improve)
• GN&C	Yes
• Crew Volumes	
- Pressurized	Yes
- Unpressurized	Yes
• Robotics	Yes
• Test and Special Assembly Equipment	Yes
• Storage	Limited (but will be required for spares, etc.)
• Viewing/Proximity Operations	Undefined (but will be required)
• Consumable Resupply	Undefined (but will be required)
• SSF-compatible Interfaces	Limited (but will be required for crew transfer, etc.)
• Redundancy	Assembly-related Redundancy Undefined
• Disassembly/Refurbishment Accommodations	Possible Design Goal

Systems/Services Indicated as Available from Vehicle Exist *Only* After They Have Been Assembled

On-Orbit Assembly Considerations - continued

Vehicle's First Element Launch is Major Configuration Driver for Assembly Concept:

- No vehicle systems yet in place
- Even deployable vehicle systems require power and data
- Vehicle-independent HLLV "unloader" needed (this continues to be a need if HLLV is not brought to Assembly Site
- Vehicle-independent "assembler" may be needed
- Autonomous or external control of both HLLV and vehicle needed during assembly
- Constraints exist for assembly operation durations as well as for HLLV on-orbit lifetime

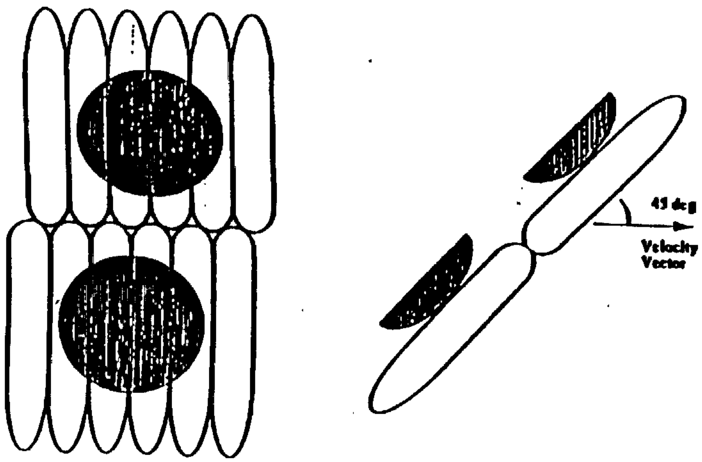
Assembly Mode Options:

- Robotic
- Ground-based Telerobotic + Automation
- Ground and Space-based Telerobotic + Automation
- **Ground and Space-based Telerobotic + Automation + EVA**
- Ground or Space-based Telerobotic + EVA
- Automation + EVA
- EVA

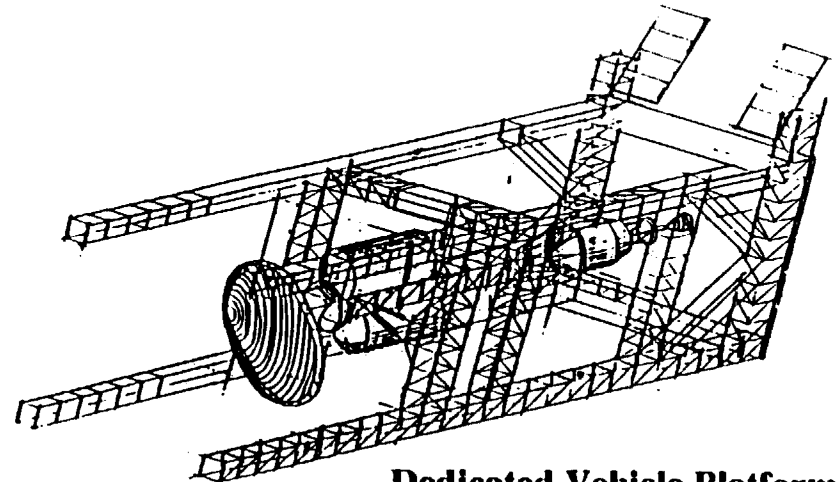
On-Orbit Assembly Concepts Summary

On-orbit Assembly Concept	Vehicle Applicability				
	CAB	CAP	NTR	NEP	SEP
• Vehicle as Its Own Platform	----	----	X	X	X
• ET-based Platform	X	X	MEV only	MEV only	MEV only
• Dedicated Vehicle Platform	X	X	X	?	----
• "I-Beam"	----	----	X	X	?
• "Smart" HLLV	FEL	FEL	FEL	FEL	FEL
• Flexible (Hinged) Truss	----	----	?	X	?
• Assembly Flyer	?	?	X	X	X
• SSF-based FEL Assembly	FEL, MEV (MTV)	FEL, MEV	FEL, MEV	FEL, MEV	FEL, MEV
• Tethered-Off-SSF Assembly	X	X	X	----	----

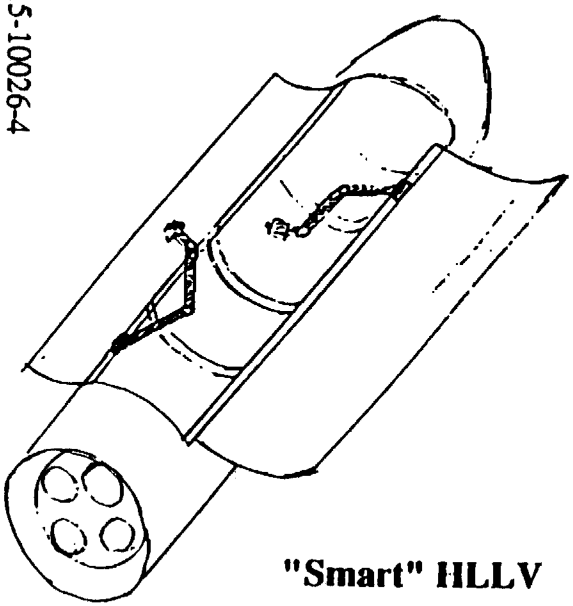
On-orbit Assembly Concepts



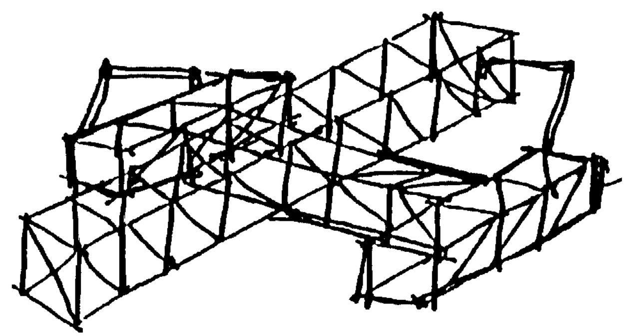
ET-based Platform



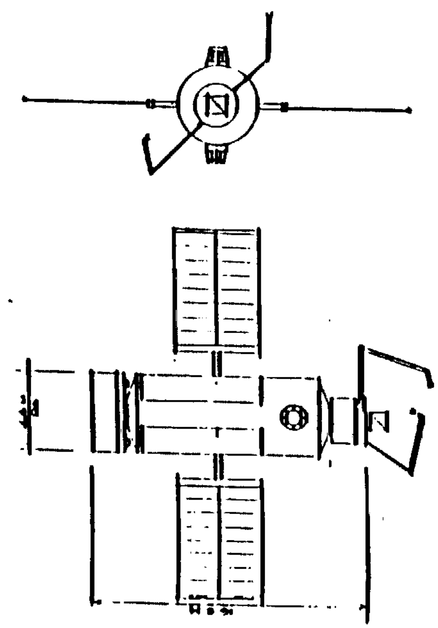
Dedicated Vehicle Platform



"Smart" HLLV

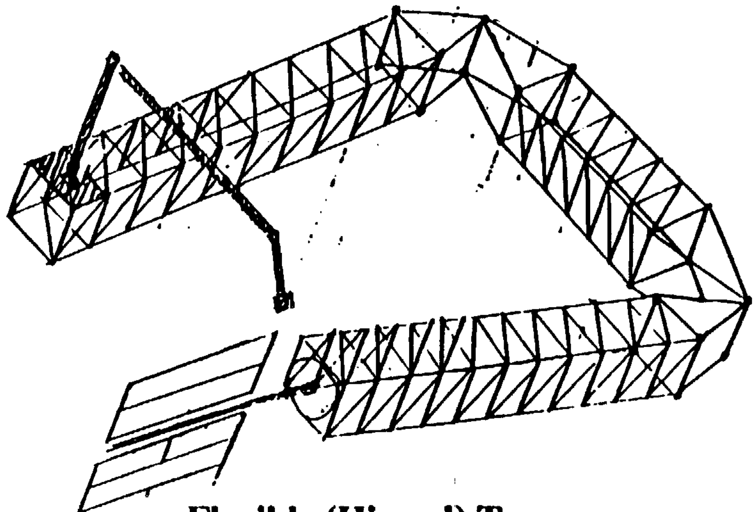


"I-Beam"

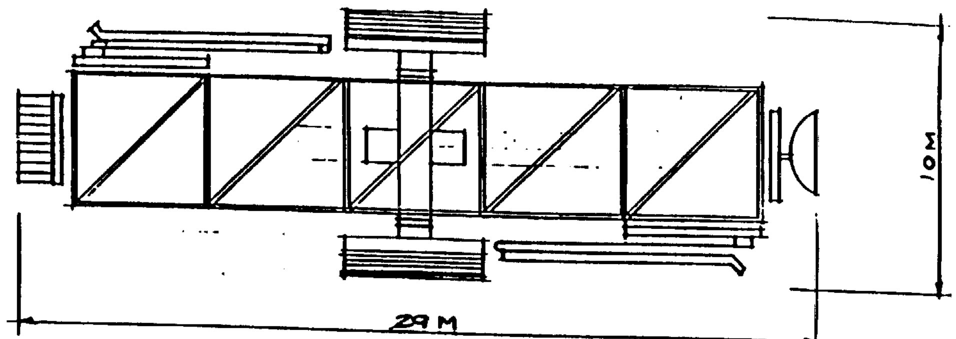


Assembly Flyer

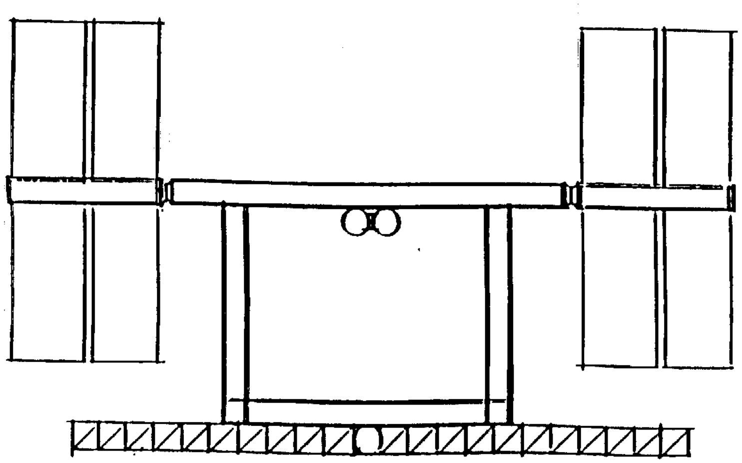
On-orbit Assembly Concepts - continued



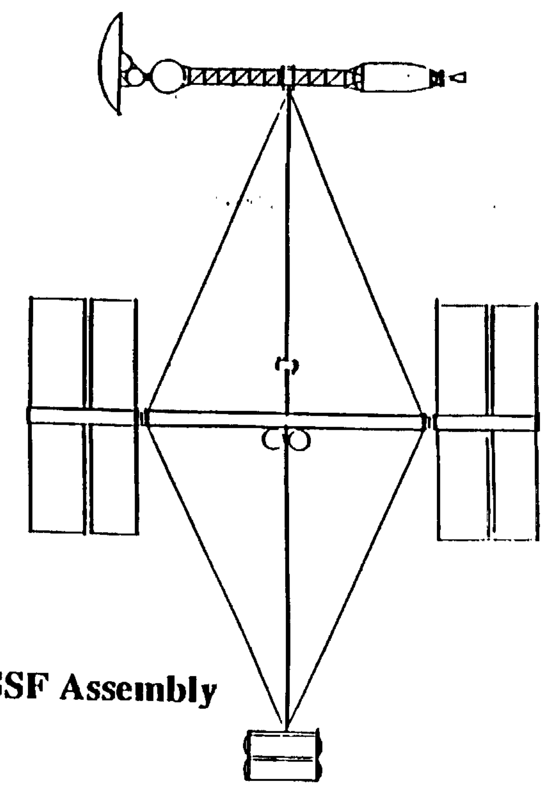
Flexible (Hinged) Truss



Vehicle as Its Own Platform



SSF-based FEL Assembly

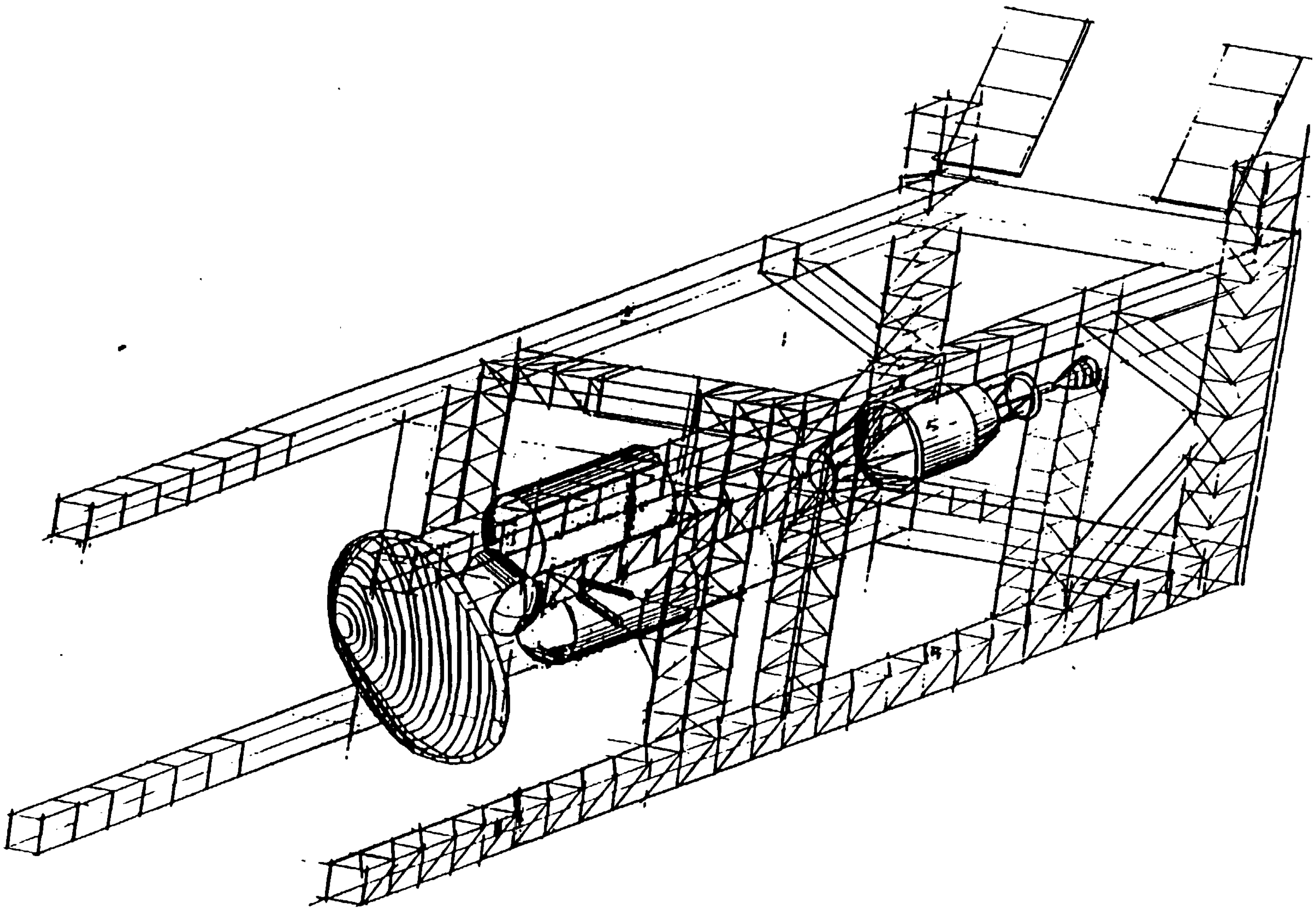


Tethered-Off-SSF Assembly

Dedicated Vehicle Assembly Platform Features

- Uses SSF type truss structure
- Dimensions 130m x 50m x 50m
- Movable and adjustable sections; can accommodate dual MEV configurations
- Aerobrake held from inside structure; TPS end is clear of obstructions. Allows unimpeded assembly and repair of TPS
- To release MEV from assembly platform, Aerobrake Assembly Section slides out longitudinally to the end of the platform, holding structure releases aerobrake, MEV moves out . MMV drops out from below the platform
- Pressurized Control Station with a logistics module and airlock
- Reboost system; occasional refueling needed and can be supported by CTV
- Gravity gradient stable
- Local debris shielding required
- Robot manipulator arms move longitudinally along tracks on platform truss
- Photo-voltaic arrays to provide power for platform and/or vehicle systems
- Storage fixtures are located along side the platform trusswork to store sections of the vehicle
- Platform can be controlled from SSF, from a ground station, and from the platform itself

Dedicated Vehicle Assembly Platform



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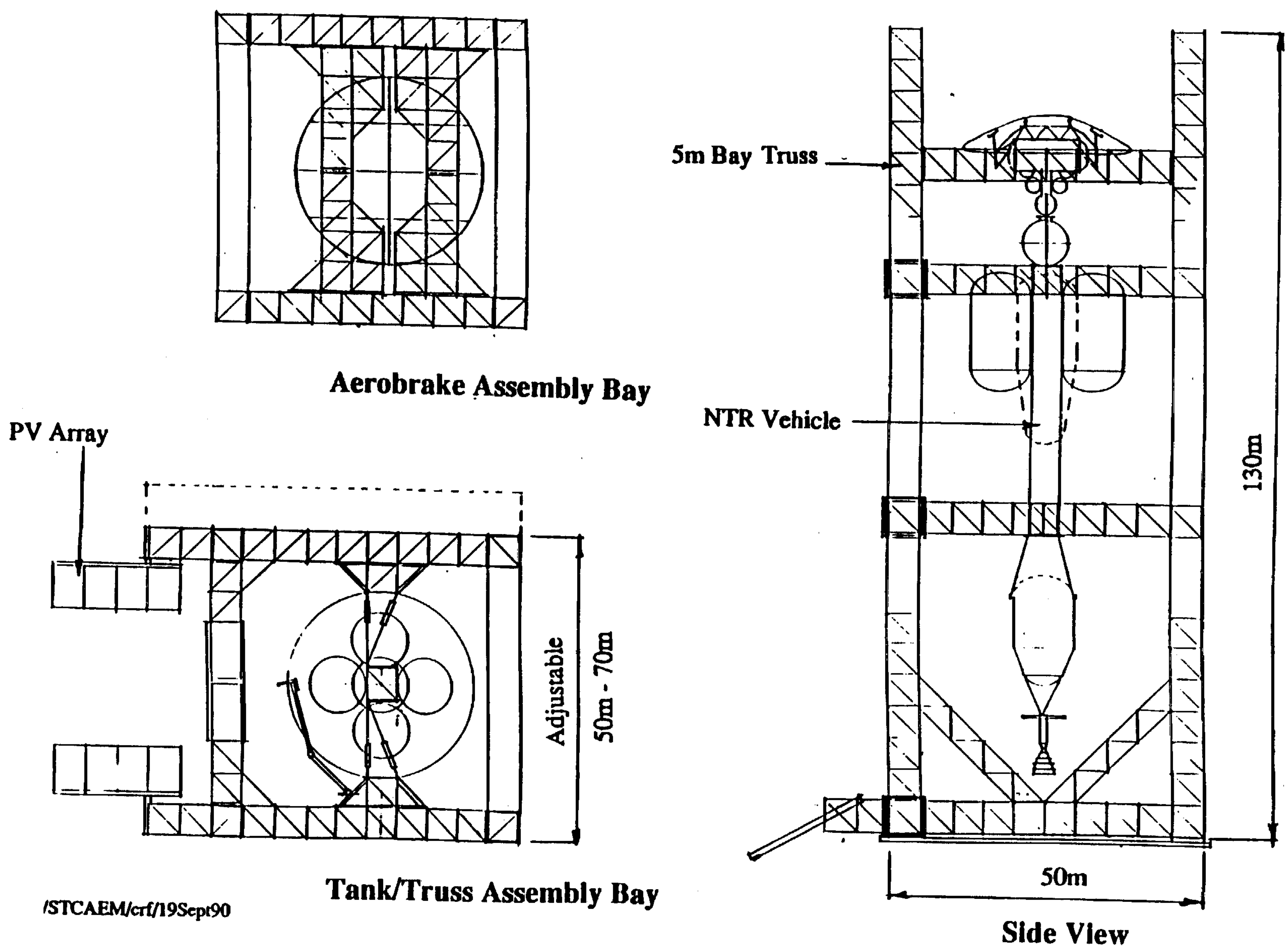
Dedicated Vehicle Assembly Platform



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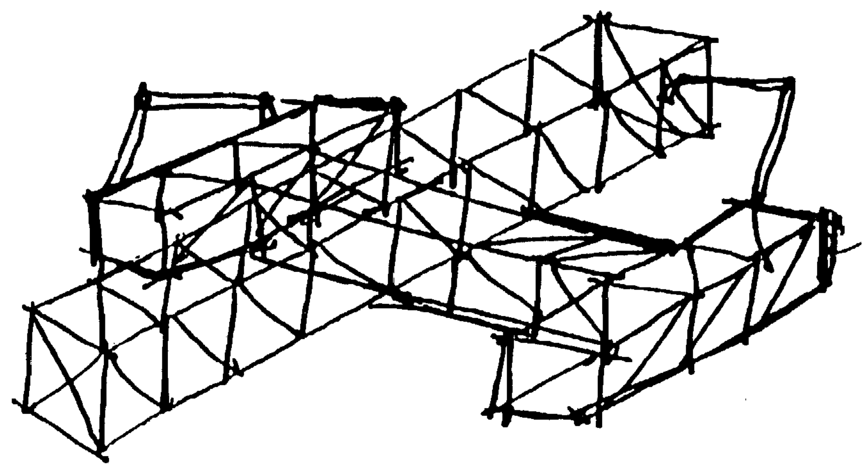
I-Beam Assembly Platform

- I-beam platform is carried up in first HLLV flight along with vehicle truss, both of which are self deploying
- I-beam platform attaches to one plane of vehicle truss
- Two robot arms that can move linearly on a base on side beams of i-beam platform
- Reboost, communication, avionics capabilities will be provided by vehicle being assembled
- Flies gravity gradient stable
- Debris shielding will have to be locally supplied to needed areas; minimum vehicle cross section facing debris
- "Pre"-assembly mission will be needed to set up vehicle and I-beam trusses (interfaces, cables, wires, conduits, communication, data, reboost, etc.) prior to main vehicle assembly

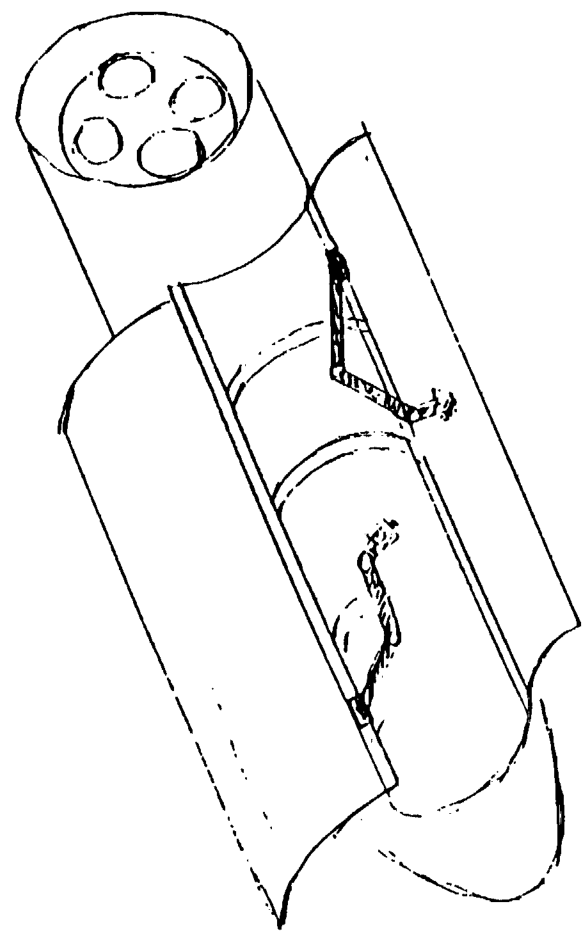
I-Beam Assembly Platform



"Smart" HLLV Assembly Platform

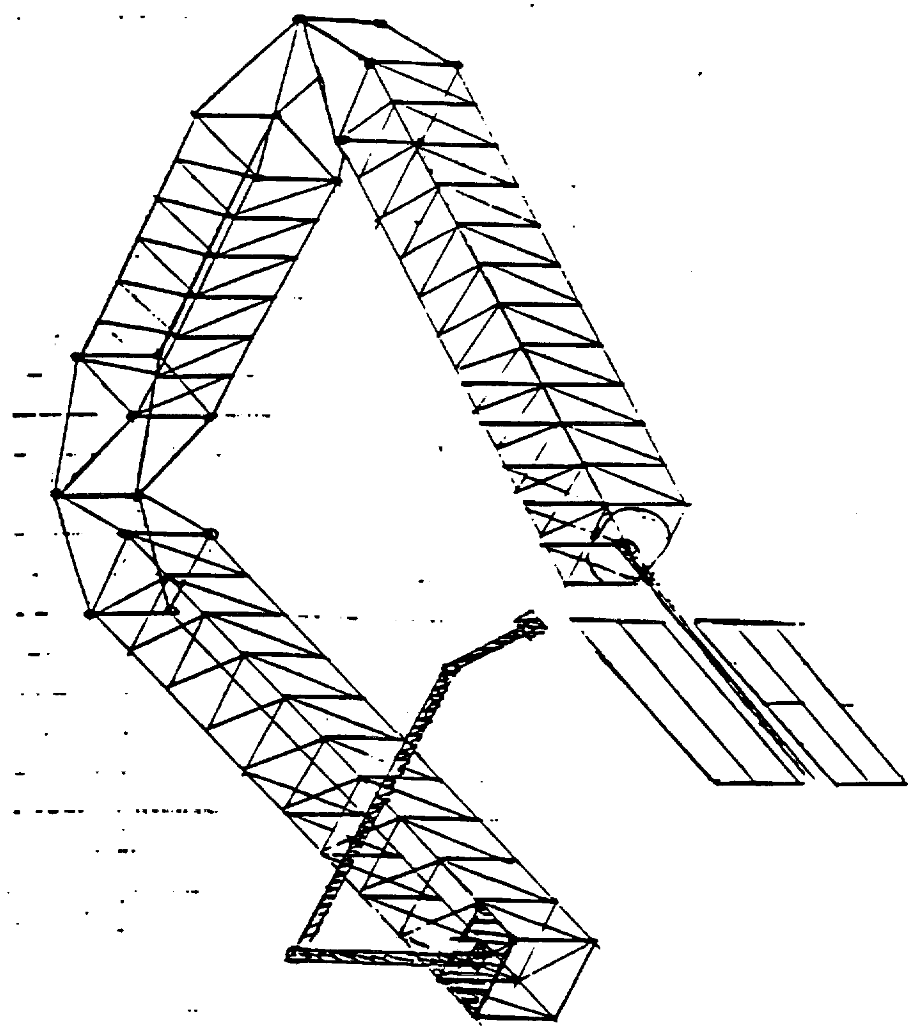
- Eliminates need for any additional platform
- Two robot arms similar to the current shuttle RMS, that can move lineraly on HLLV payload bay tracks
- HLLV provides partial debris shielding; supplemental local shielding will be required
- Telescopic mooring struts to attach vehicle to HLLV
- Reboost is provided by HLLV; refueling can be supported by CTV
- Vehicle's transit hab is used by crew during assembly operations
- All power for communication, avionics, robotics, RCS, etc., will be provided by vehicle systems
- "Pre"-assembly mission will be need to set truss interfaces, power, cables, wires, conduits, etc. Vehicle assembly proceeds after truss is readied for assembly operations
- Robot arms are transferred to vehicle from HLLV after a particular phase of assembly
- HLLV flies gravity gradient stable
- Only first assembly mission involves a "smart" HLLV; all others are cargo structures only

"Smart" HLLV Assembly Platform



Flexible (Hinged) Vehicle Truss as Assembly Platform

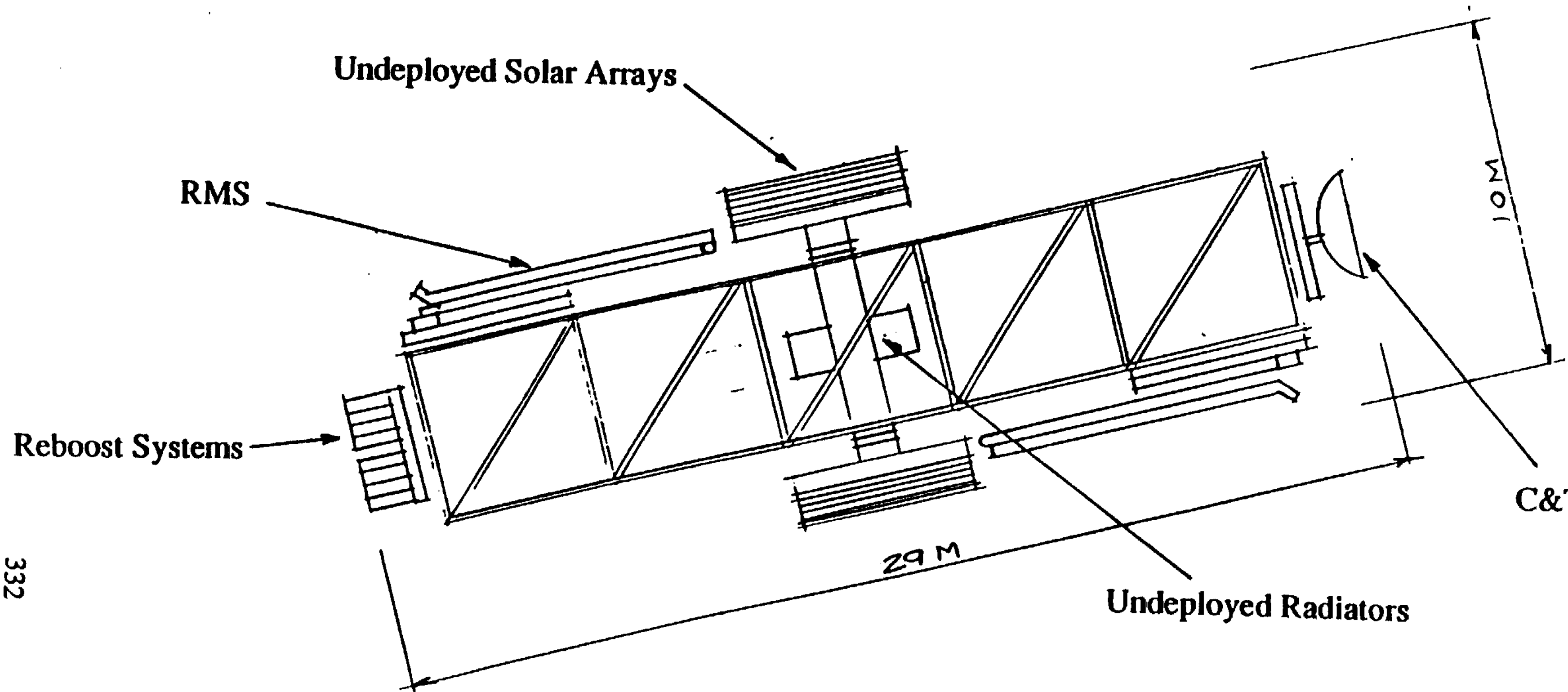
- NTR or NEP truss truss itself seves as assembly platform; truss can however flex at hinge points to provide reach behind the vehicle
- Minimum of two hinges to allow angular motion in one plane
- Eliminates need for any additional platform
- Two robot arms can be affixed to longest sections of hinged truss; robot arm can move along truss
- Hinges are modular and locking. Upon assembly completion, hinges lock and provide structural rigidity
- Local debris shielding required; vehicle is oriented such that minimum cross section faces debris
- Reboost is provided by vehicle's own reboost system with refuel support provided by CTV
- Vehicle's transit hab is used during assembly operations by crew
- All power for communication, avionics, robotics, RCS, etc., will be provided by vehicle's own systems
- "Pre"-assembly mission will be need to set up flex-truss, interfaces, power, cables, wires, conduits, hinge operation, communications data, reboost, etc.



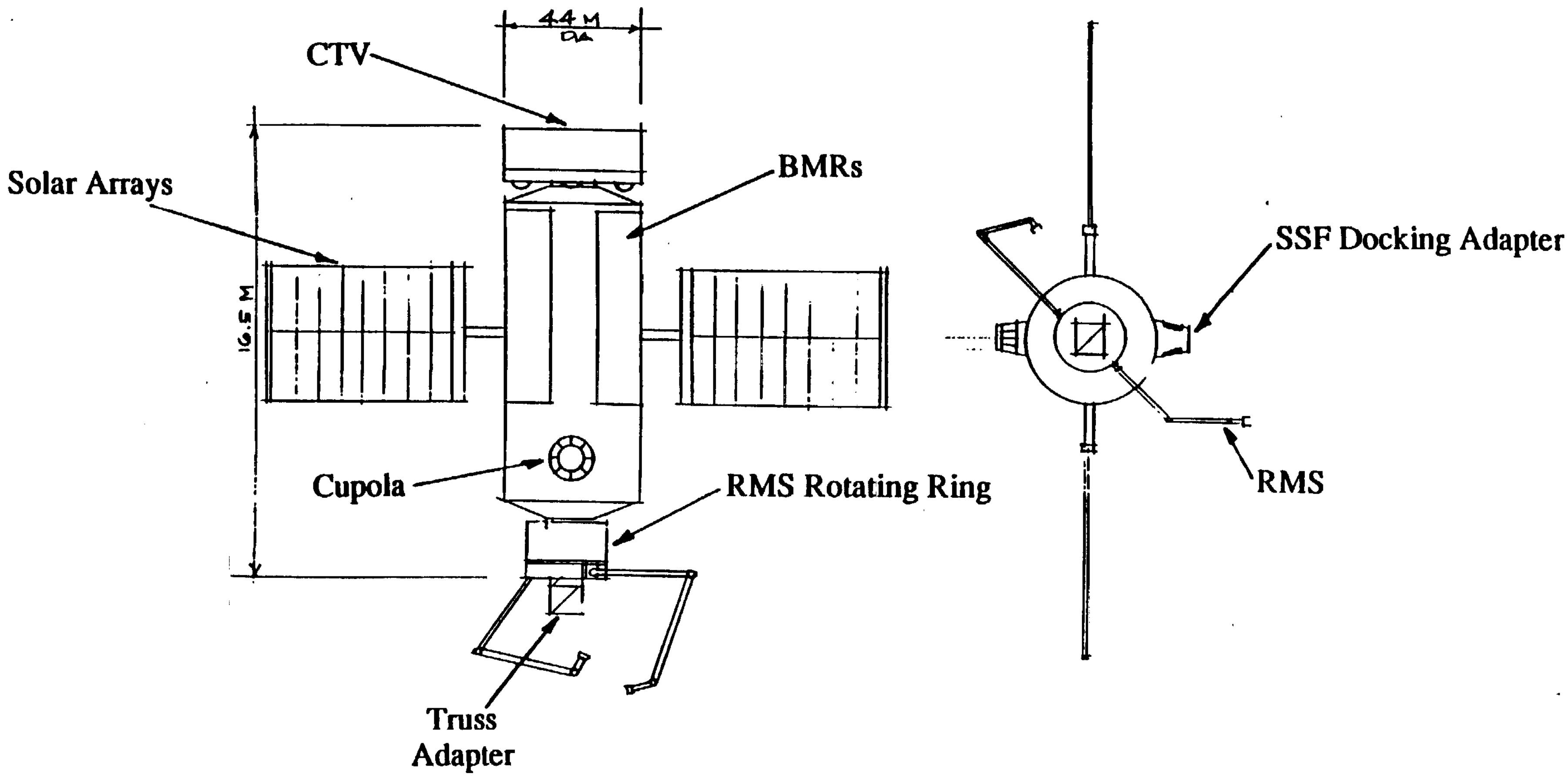
Using Vehicle as its Own Assembly Platform

- First Element Launch (FEL) delivers compact, fully integrated spacecraft:
  - Sized to be launched within 10m x 30m shroud
  - Contains self-sustaining and assembly support equipment necessary until the next element launch
  - FEL is integral part of the vehicle itself
  - HLLV releases automatically at proper attitude and orbit
  - On-board batteries deploy necessary power, radiator, and communication systems
  - Includes appropriate control and reboost systems

  } These functions may also be accomplished by a CTV
  - Succeeding assembly missions are initially based from this element and expand with the vehicle
  - MMD shielding would be localized, integrated at launch, and removed prior to Earth departure



Assembly Flyer Concept



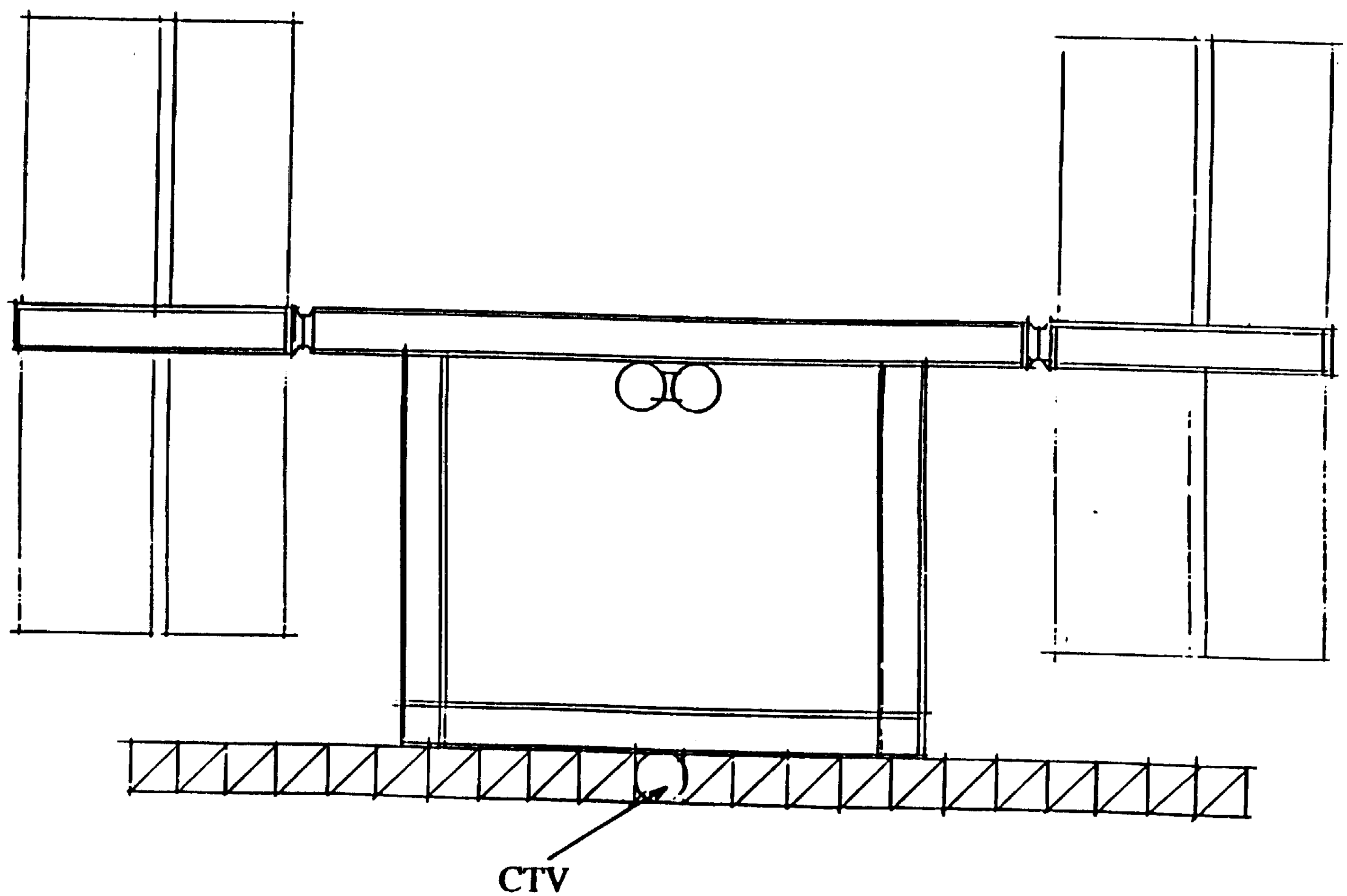
- Self-contained Assembly Flyer:
  - Performs assembly operations in any of three modes:
    - Free Flying: use for unloading HLLV, transfer of equipment/crew, etc.
    - Tandem Flying: use for handing off to vehicle, inspection, general assembly
    - Attached Operations: use for detailed and/or long duration assembly tasks (attachment may be directly to vehicle structure or to some temporary scaffolding)
  - Capable of manned and/or autonomous operations
  - Derivative from Industrial Space Facility (ISF)

SSF-Based Assembly of First Element Concept

- First Element of Mars Vehicle is assembled at SSF
  - Primary Truss
  - Power Systems
  - Thermal Control System
  - Communications
  - Avionics
  - Reboost and Attitude Control Systems
  - Remote Manipulator System
  - Utilities

- Once First Element is complete, the vehicle itself or a CTV docked to the vehicle transports it to an off-SSF location where remainder of vehicle is assembled:
  - Vehicle is enabled to assemble remainder itself
  - If needed, CTV aids with reboost and control until supplemental systems arrive
  - Debris shielding may be localized
  - MEV is assembled prior to Aerobrake/Aeroshell assembly
  - Temporary scaffolding may be used as needed

- First Element may be assembled with its orientation parallel (as shown) or perpendicular to SSF, depending on:
  - Drag effects
  - Controllability
  - Microgravity effects on SSF
  - RMS reach



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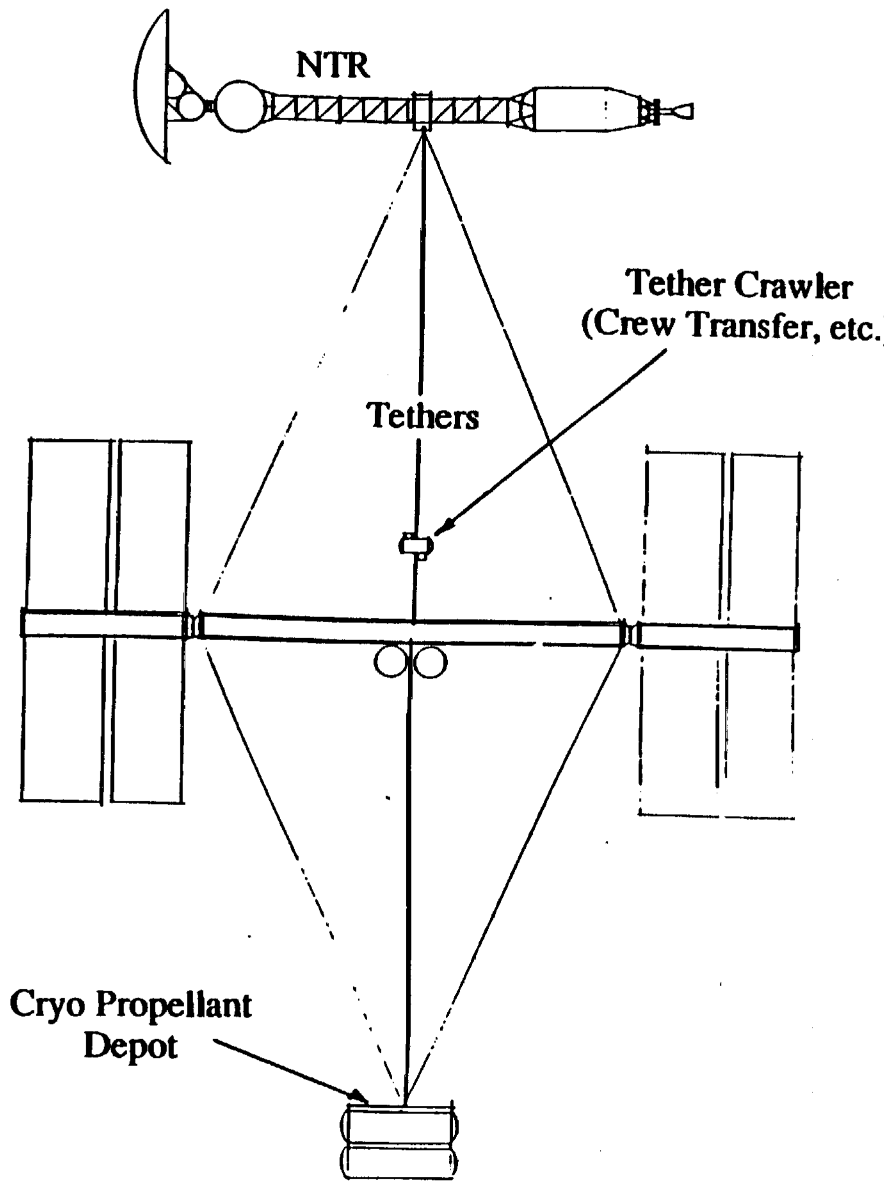
Tethered-Off-SSF Assembly Concept

- Allows easy crew and logistics access (pressurized or unpressurized) between SSF and assembly area
- Removes hazardous operations and materials from SSF
- SSF facilities (with upgrades) may be available to both vehicle and on-orbit depot:
  - Power
  - Data
  - Communications

  } via tether
  - Attitude and Reboost Systems
- Center of mass may be maintained in the SSF Labs by moving the vehicle and depot along the tether as the vehicle is built up, propellant is transferred, etc.
- Tether also serves to mitigate dynamic disturbances to SSF caused by assembly or propellant operations



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Assembly Node Concepts Pros and Cons

Node Concepts	Key Features/Advantages	Key Disadvantages
Dedicated Assembly Node	• Abundant storage • Totally self-contained • Vehicle systems unused • Multiple robot arms • Sections of vehicle may be assembled simultaneously	• Larger than SSF • Will take long time to construct • Excessive reboost requirements • Mechanically complex • Local debris shielding required • Must be in place prior to vehicle assembly
I-Beam Platform	• Can be carried up in first HLLVflight • Can easily reach most parts of vehicle with two robot arms • Uses vehicle for comm., data, RCS, power after initial deployment • Can serve as base for experiments	• Fuel cells, batteries required for initial deployment • Limited storage area • Precursor mission required for deployment
"Smart" HLLV Platform	• No additional platform required • HLLV shroud provides limited debris shielding • HLLV provides for communication, data, RCS, GNC, etc. • Robot arms transferable to NTR	• Increased HLLV complexity • Reboost fuel has to be replenished • Limited storage • Vehicle must be detached from HLLV prior to assembly complete • Local debris shielding required
Hinged Truss Platform	• Uses vehicle truss as assembly platform; no other platform needed • Reach to remote engine section of vehicle provided by flexing truss at hinges • Vehicle subsystems used; no additional systems necessary	• Requires a precursor mission to deploy truss • Batteries, fuel cells necessary for initial deployment • Reboost, comm., data, power, must be in place prior to assembly start • Limited storage • Local debris shielding required
Vehicle as its own Platform	• Reduces needed on-orbit infrastructure • Deletes additional facilities and resources needed for designing, builiding, launching, and maintaining separate assembly platform	• Requires dedicated HLLV flight for non-optimized packaged first element • Requires vehicle to have additional control, reboost • No additional storage • Requires batteries or fuel cells for initial deployment • Requires localized debris shielding

Assembly Node Concepts Pros and Cons
(continued)

Node Concepts	Key Features/Advantages	Key Disadvantages
Assembly Flyer Platform	• Performs HLLV unloading, payload/crew transport, and assembly with one vehicle • Compatible with SSF • Capable of manned/robotic operations • Uses CTV for main P/A • Can serve as free flying platform between assemblies	• No additional storage • Requires vehicle to have additional control and reboost systems • Requires development and production of sophisticated man-rated space vehicle • Requires localized debris shielding
SSF Based Assembly of First Element	• Uses planned SSF growth concept • Provides quick and easy crew logistics access to initial assembly operations • Allows verification and checkout of critical systems prior to independent vehicle operations • Does not disrupt SSF operations beyond first assembly mission (remainder of assembly based from vehicle itself)	• Impact to SSF (resources, microgravity, drag, etc.) • Eventually requires vehicle to have additional control and reboost systems • Requires localized debris shielding • No additional storage beyond first element
Tethered off-SSF Assembly Platform	• Compatible with current SSF design • Provides quick and easy crew and logistics access to entire assembly and propellant transfer operations • Microgravity and dynamic loads impacts to SSF minimized by tether • Removes hazardous operations and materials to SSF standoff distance	• Impact to SSF resources • Requires localized debris shielding • No additional storage • Requires additional reboost and control systems on SSF

SEP Component Manifest Data

NEP Component	Quantity	Dimensions (meters)	Total Mass (metric tons)
• MEV			
Aeroshell	1	28 x 30 x 7 box	9.51
Descent System (incl 2 rovers)	1	9.5 x 20 x 4 box *	32.83
Ascent System	1	9.5 x 9.5 x 5.5 box	24.83
Surface Payload Module	1	13 x 4.4 (dia) cylinder	25.00
Surface Payload Module Airlock	1	2.9 x 3 (dia) cylinder	4.50
		Subtotal =	**96.67**
• MTV			
MTV Hab Module	1	10 x 8 (dia) cylinder	40.30
MCRV	1	3 x 4 (dia) cylinder	7.00
MTV-to-MEV Tunnel and Airlock	1	6 x 3 (dia) cylinder	7.00
Main Truss	3	7 x 7 x 7 box (deployable) *	9.20
Power Processing Unit	2	2 x 2 x 1 box	4.70
Communications	2	2 x 1 x 2 box	0.60
Attitude Control	2	2 x 2 x 2 box	5.70
Avionics	1	2 x 2 x 2 box	1.70
Power Distribution and Control	2	5 x 2 x 1 box	8.20
Thruster Pods	2	13 x 7 x 2 box	13.70
Propellant and Tanks	4	4.1 (dia) sphere	154.00
		Subtotal =	**252.10**

* These represent launch package dimensions, not mission configuration

SEP Component Manifest Data- continued

NEP Component	Quantity	Dimensions (meters)	Total Mass (metric tons)
• Solar Power Generation			
Mission Array Blanket	18	15 x 2.2 (dia) cylinder *	26.2
Spiral Transfer Array	18	15 x 2.2 (dia) cylinder *	26.0
Array Structure	2	7 x 3 x 5 box *	3.6
Array Deployment Mechanism	1	2 x 2 x 1 box	2.6
Radiators	2	8 x 2 x 1 box	6.9
		Subtotal =	65.3
• Other			
Experimental Platforms	2	5 x 2 x 1 box	1.7
		NEP Total =	415.77

* These represent launch package dimensions, not mission configuration
** Total Power Generation and Heat Transport launch package dimensions = 15 x 9 (dia) cylinder

SEP - Manifesting and Packaging

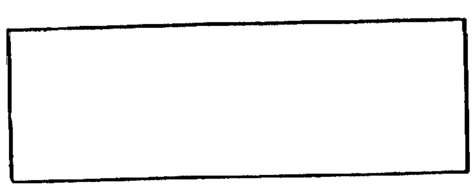
(10m x 30m Shroud, 140 mt HLLV using ET-Based Platform)

Ground Rules and Assumptions

- Heavy Lift Launch Vehicle (HLLV) with 140 metric ton capability and 10m x 30m shroud
- Sequencing based on External Tank-derived Assembly Platform concept
- Some TBD volume is available in nosecone of HLLV
- No specific FSE/OSE or CG constraints identified (heavier payload located at bottom of stack)
- SEP configuration (volume and mass) current as of 3rd Quarterly Briefing
- MEV Aeroshell is assembled on orbit (in ten pieces) and requires two HLLV flights

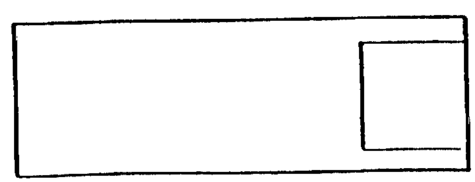
Assembly Mission One

- MEV Aeroshell (5 out of 10 pieces)



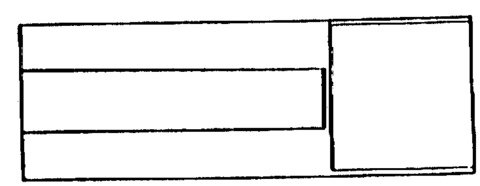
Assembly Mission Two

- MEV Aeroshell (5 out of 10 pieces)
- Main Truss (1 of 3)



Assembly Mission Three

- MEV Descent System (incl 2 rovers)
- MEV Ascent System



ADVANCED CIVIL SPACE SYSTEMS

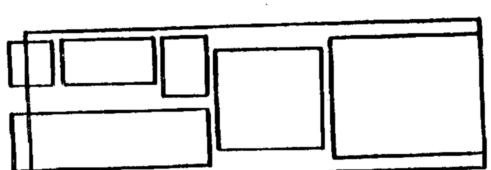
SEP - Manifesting and Packaging (continued)

(10m x 30m Shroud, 140 mt HLLV using ET-Based Platform)

BOEING

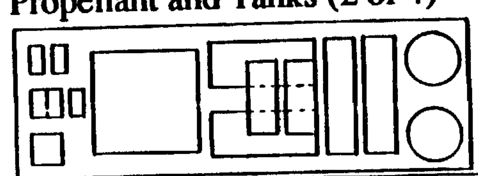
Assembly Mission Four

- Mars Surface Payload Module
- Mars Surface Module Airlock
- MTV-to-MEV Airlock and Tunnel
- MCRV
- Main Truss (2nd of 3)
- MTV Hab Module



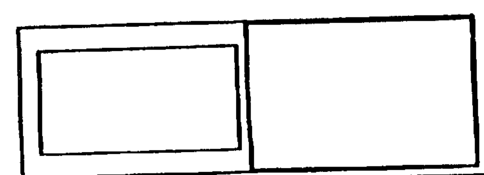
Assembly Mission Five

- Array Deployment Mechanism (1)
- Attitude Control (2)
- Communications (2)
- Avionics (1)
- Power Processing Units (2)
- Power Distribution and Control (2)
- Main Truss (3rd of 3)
- Array Structure (2)
- Experimental Platforms (2)
- Radiators (2)
- Propellant and Tanks (2 of 4)



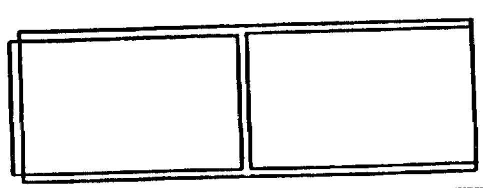
Assembly Mission Six

- Thruster Pods (2)
- Spiral Transfer Arrays (10 of 18)



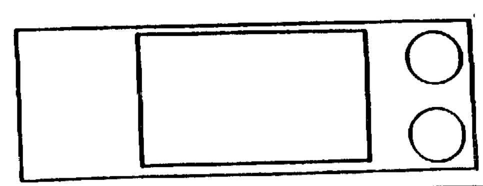
Assembly Mission Seven

- Spiral Transfer Arrays (8 of 18)
- Mission Array Blanket (12 of 18)



Assembly Mission Eight

- Mission Array Blanket (6 of 18)
- Propellant and Tanks (3rd, 4th of 4)



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SEP - Manifesting and Packaging

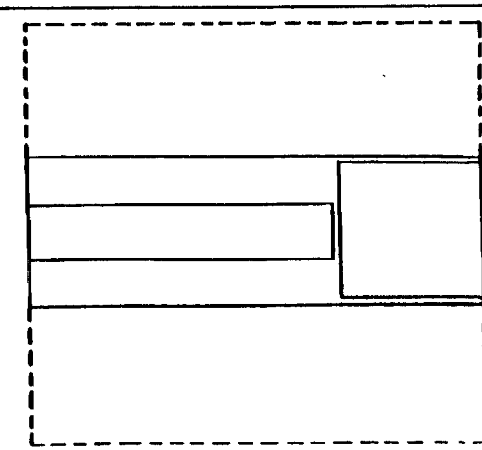
(Mixed HLLV Fleet, using ET-Based Platform)

Ground Rules and Assumptions

- Heavy Lift Launch Vehicle (HLLV) mixed fleet consists of:
  - HLLV #1: 84 metric ton payload capability with 10m x 30m shroud
  - HLLV #2: 120 metric ton payload capability with 7.6m x 30m shroud
- Sequencing based on External Tank-derived Assembly Platform concept
- Some TBD volume is available in nosecone of HLLV
- No specific FSE/OSE or CG constraints identified (heavier payload located at bottom of stack)
- SEP configuration (volume and mass) current as of 3rd Quarterly Briefing
- MEV Aeroshell is assumed to be integrated at launch ("Ninja Turtle" concept) with other payload packaged in shroud

Assembly Mission One (HLLV #1)

- MEV Aeroshell (externally mounted)
- MEV Descent System (incl 2 rovers)
- MEV Ascent System



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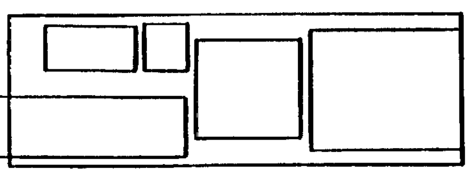
SEP - Manifesting and Packaging (continued)

(Mixed HLLV Fleet, using ET-Based Platform)

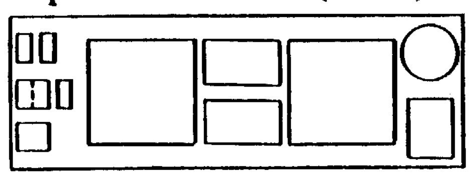
Assembly Mission Two (HLLV #1)

- Mars Surface Payload Module
- Mars Surface Module Airlock
- MTV-to-MEV Airlock and Tunnel
- Main Truss (1 of 3)
- MTV Hab Module



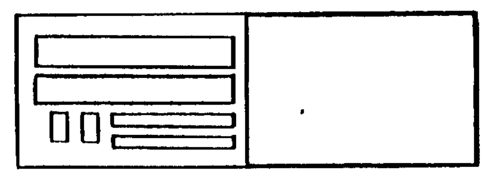
Assembly Mission Three (HLLV #1)

- Array Deployment Mechanism (1)
- Attitude Control (2)
- Communications (2)
- Avionics (1)
- Power Processing Units (2)
- Power Distribution and Control (2)
- Main Truss (2nd of 3)
- Array Structure (2)
- MCRV
- Propellant and Tanks (1 of 4)



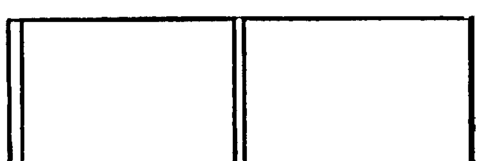
Assembly Mission Four (HLLV #1)

- Experimental Platforms (2)
- Radiators (2)
- Thruster Pods (2)
- Spiral Transfer Arrays (12 of 18)



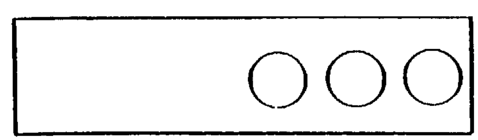
Assembly Mission Five (HLLV #1)

- Spiral Transfer Arrays (6 of 18)
- Mission Array Blanket (18 of 18)



Assembly Mission Six (HLLV #2)

- Propellant and Tanks (3 of 4)



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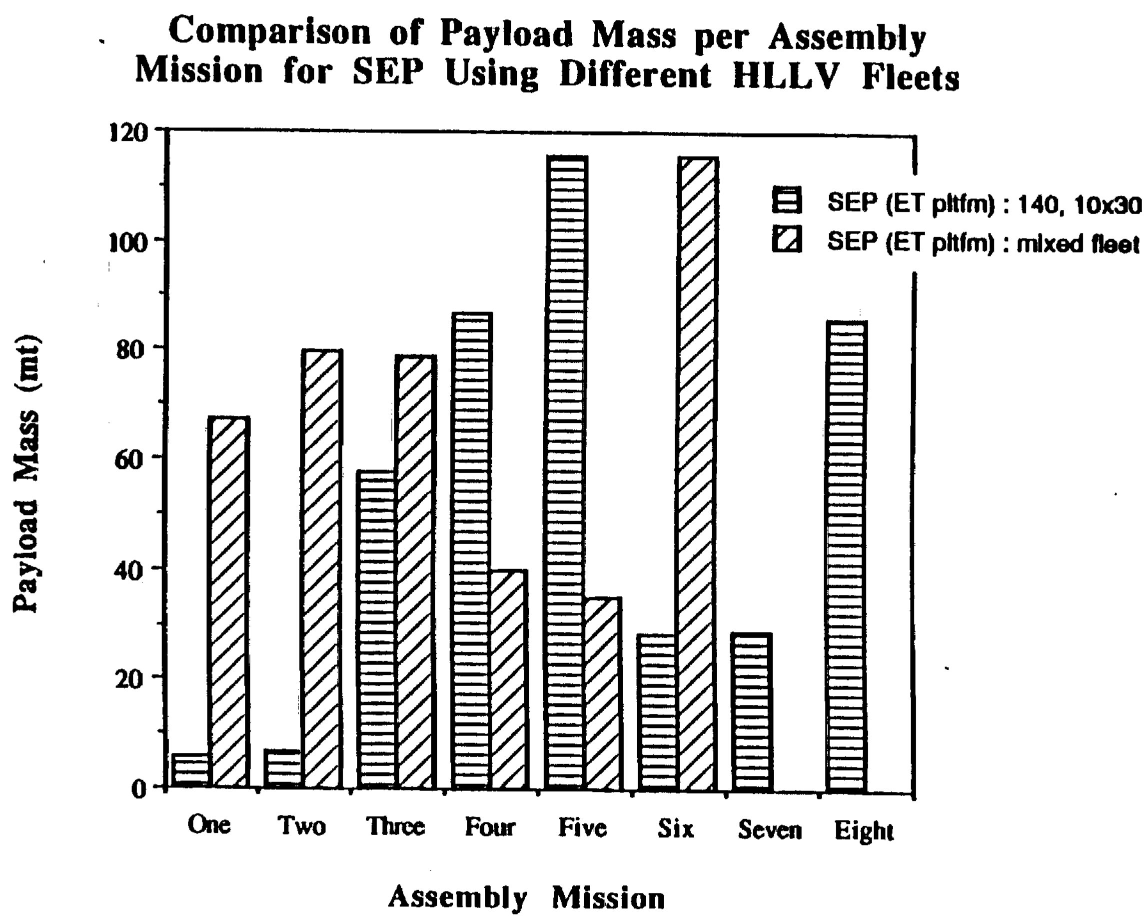
SEP Manifesting and Packaging



SEP Assembly Ground Rules and Assumptions

- SEP configuration and component list current as of 8/30/90
- Heavy Lift Launch Vehicle (HLLV) assumed with 10 meter x 30 meter shroud and 140 metric ton capacity
- Cargo Transfer Vehicle (CTV) capable of maneuvering maximum possible payloads for unloading HLLV, transporting vehicle elements to assembly area, and propulsive assists
- Assembly accomplished mainly through use of ground-based and autonomous robotics; crew presence for monitoring, contingency, and crew systems checkout only
- Crew assumed to be based at Space Station Freedom and are transported to assembly area by ACRV (crew presence is represented in flows from start of assembly mission until end; however, crew support may not be necessary for the duration of the mission). ACRV serves as both pressurized and unpressurized operations base until crew modules available
- Assumes Space Shuttle Program External Tanks (ET) based assembly platform concept available and functional to support initial assembly missions (later assembly utilizes this platform mainly for storage while vehicle systems are used to complete construction)
- Robotic systems as defined for 2nd and 3rd Quarter Cryo/Aerobrake Vehicle assembly (PRMS, RAMS, ASF) with addition of the SEP-based Remote Truss Manipulator System (RTMS) which is used for both assembly and mission ops
- Mars transfer launch on February 2016 (final assembly mission ends two months prior to this to allow spiral out of Earth orbit)
- MEV Aeroshell divided into 10 pieces and assembled in orbit (two dedicated flights assumed necessary to completely deliver Aeroshell)
- MTV Hab System launched after MEV complete (remaining on-ground non-mechanical interface verification utilize simulators)

SEP Assembly Ground Rules and Assumptions - cont

- SEP Main Truss is fully deployable (Utility Distribution Systems contained within launch package and are also deployed or attached robotically)

- Array structure for transfer and mission arrays is compacted into 2 separate packages which self deploy once attached to the Main Truss (all necessary systems deploy with the structure)

- Deployment of all array structures facilitated by deployment mechanism

- Arrays are attached to structureand then self-deploy and self-latch:
    - Mission arrays may not be deployed until spiral out of Earth's Van Allen Belts is complete

- Arrays themselves compact into cylindrical bundles which contain panels and systems

- Truss-mounted systems may be unloaded and transported from the HLLV in groups

- SEP Thruster Pods carried in two 13x7x2 meter sections which attach mainly to Main Truss, not to each other

- Mission spares storage is provided on SEP (TBD)

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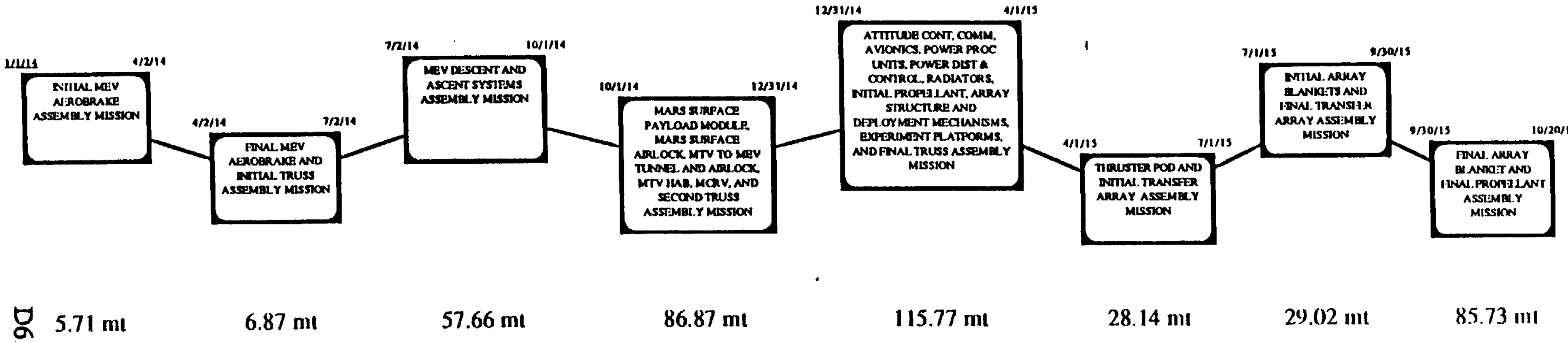
SEP ASSEMBLY SCHEDULE



- Smallest unit of time is 1 hour
- 16 hours = 1 day of Assembly Duration

BASELINE DURATIONS:
- HLLV Launch = .5 day
- HLLV achieves stable orbit = .25 day
- MV deploys from/to Freedom = .5 day
- MV berths to components = .25 day
- Unstow and power up Robotics = .06 day
- Robotic verification = .12 day
- HLLV deploys components = .06 day
- MV transfers components = .25 day
- Robotic tasks = .06 day
- EVA/Robotic Contingency = .5 day
- Component Inspection = .12 day
- Component Test = .25 day
- Subassemblies to stand-by mode = .5 day
- Mechanical Fastening of components = .18 day

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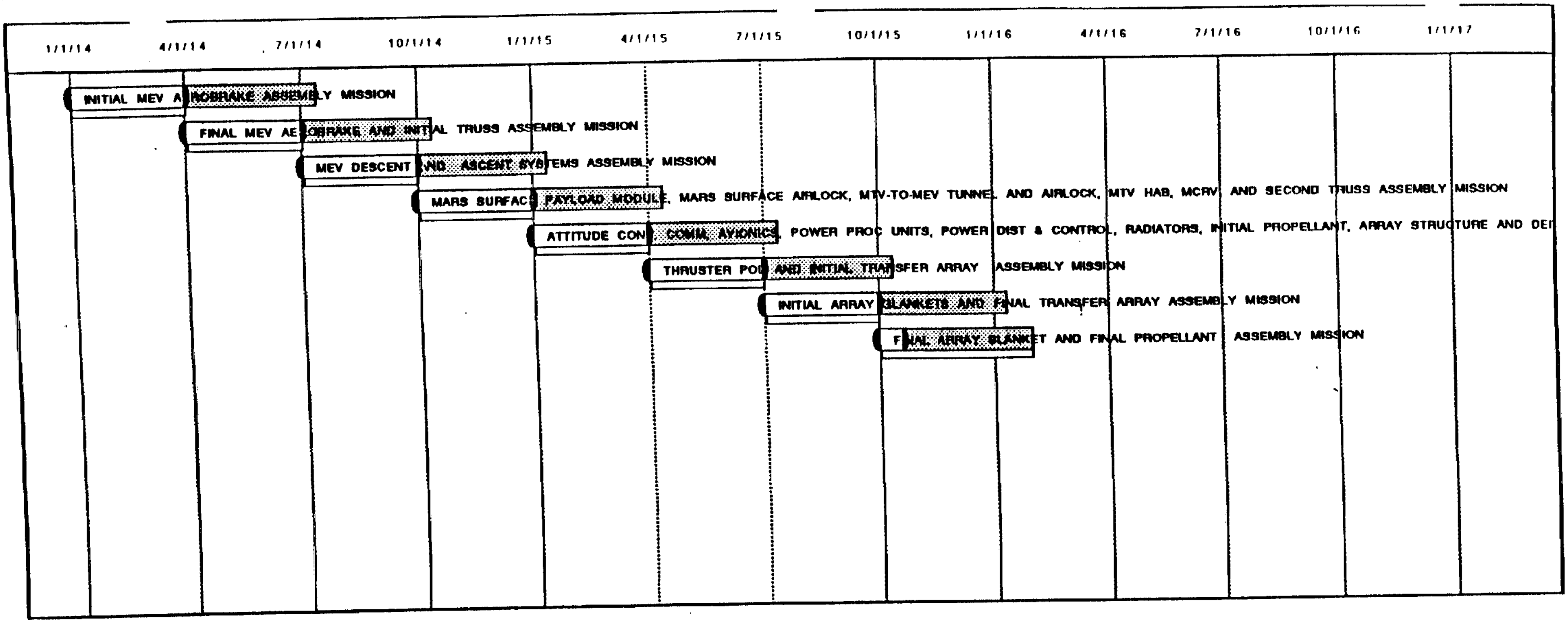
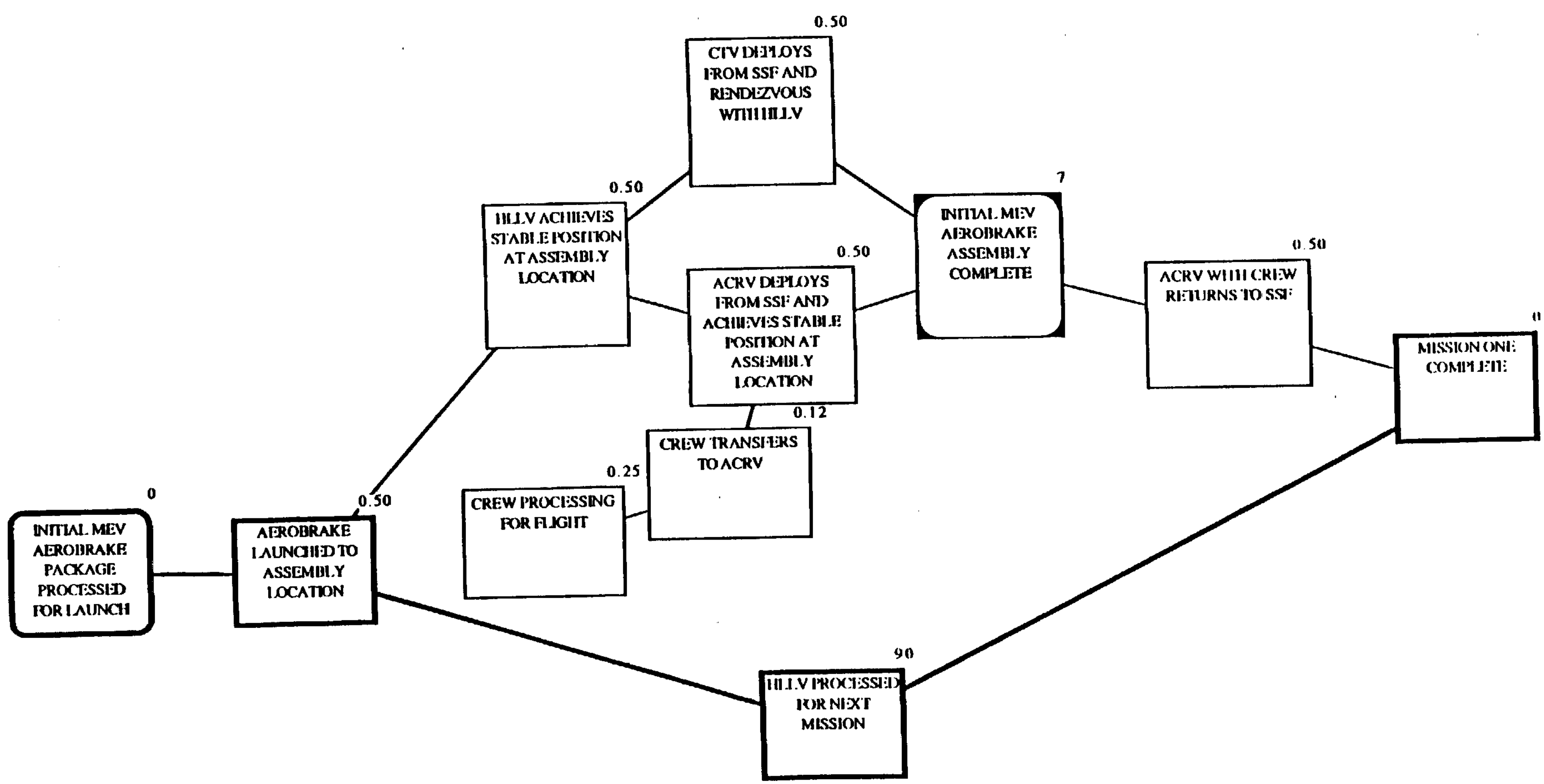
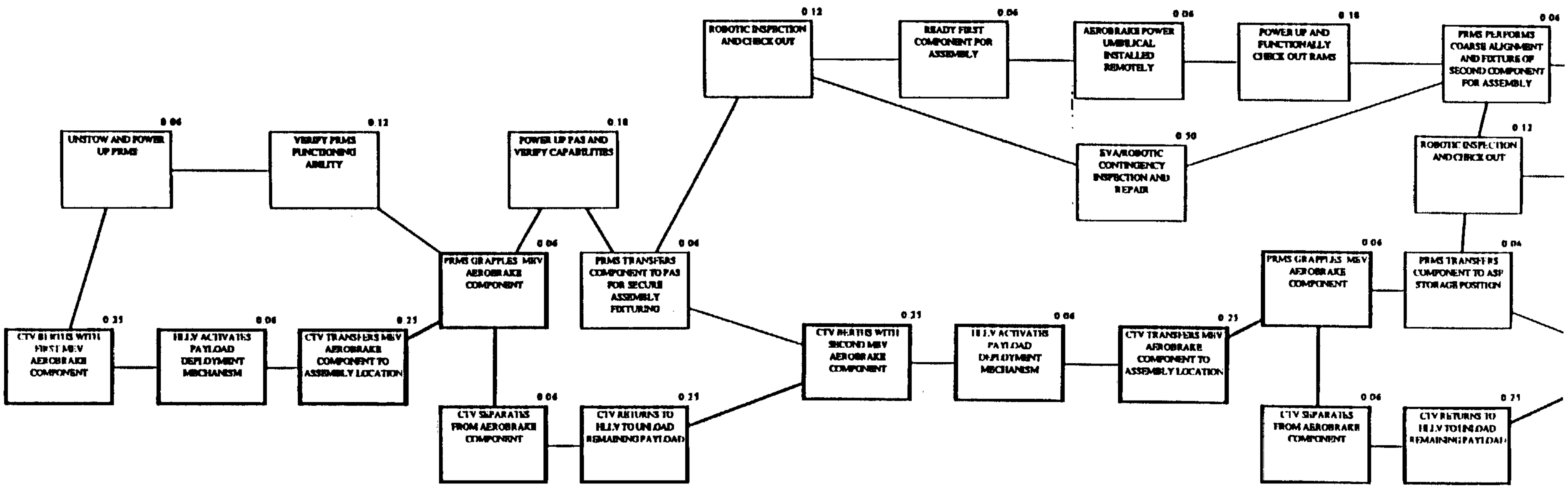


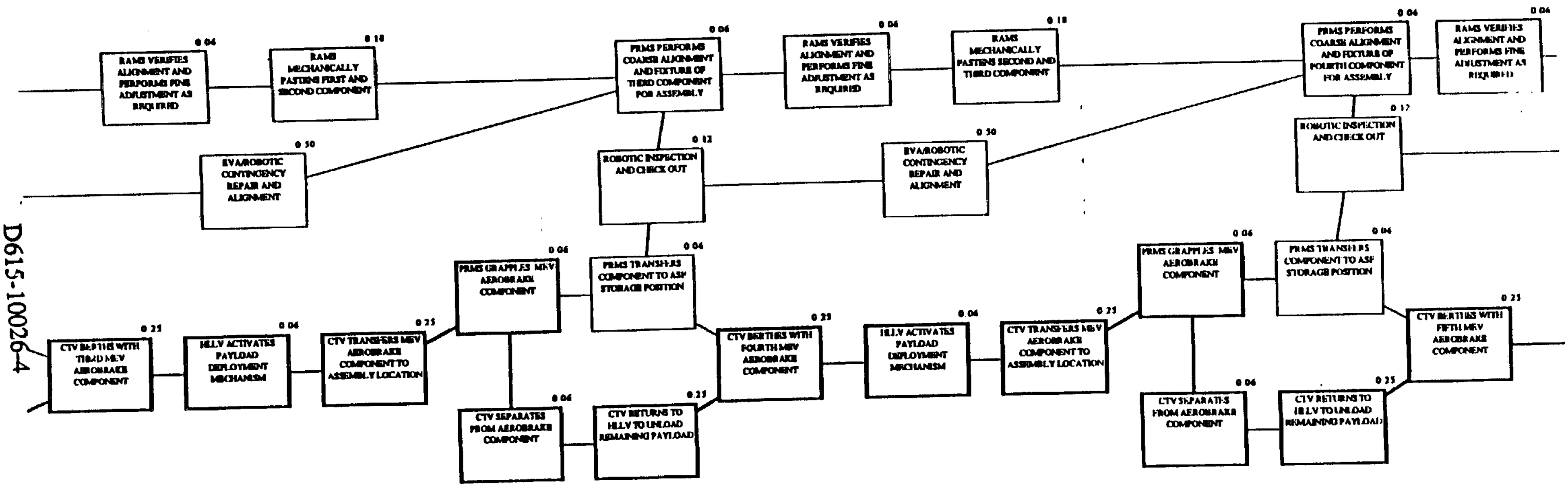
1/1/14
4/1/14
7/1/14
10/1/14
1/1/15
4/1/15
7/1/15
10/1/15
1/1/16
4/1/16
7/1/16
10/1/16
1/1/17
INITIAL MEV AEROBRAKE ASSEMBLY MISSION
FINAL MEV AEROBRAKE AND INITIAL TRUSS ASSEMBLY MISSION
MEV DESCENT AND ASCENT SYSTEMS ASSEMBLY MISSION
MARS SURFACE PAYLOAD MODULE, MARS SURFACE AIRLOCK, MTV-TO-MEV TUNNEL AND AIRLOCK, MTV HAB, MCRV AND SECOND TRUSS ASSEMBLY MISSION
ATTITUDE CONT, COMM, AVIONICS, POWER PROC UNITS, POWER DIST & CONTROL, RADIATORS, INITIAL PROPELLANT, ARRAY STRUCTURE AND DEI
THRUSTER POD AND INITIAL TRANSFER ARRAY ASSEMBLY MISSION
INITIAL ARRAY BLANKETS AND FINAL TRANSFER ARRAY ASSEMBLY MISSION
FINAL ARRAY BLANKET AND FINAL PROPELLANT ASSEMBLY MISSION

SEP ASSEMBLY MISSION ONE



SEP MEV AEROSHELL INITIAL ASSEMBLY




0.04
RAMS VERIFIES ALIGNMENT AND PERFORMS FINE ADJUSTMENT AS REQUIRED
0.18
RAMS MECHANICALLY FASTENS FIRST AND SECOND COMPONENT
0.50
EVA/ROBOTIC CONTINGENCY REPAIR AND ALIGNMENT
0.06
PRMS PERFORMS COARSE ALIGNMENT AND FIXTURE OF THIRD COMPONENT FOR ASSEMBLY
0.04
RAMS VERIFIES ALIGNMENT AND PERFORMS FINE ADJUSTMENT AS REQUIRED
0.18
RAMS MECHANICALLY FASTENS SECOND AND THIRD COMPONENT
0.12
ROBOTIC INSPECTION AND CHECK OUT
0.50
EVA/ROBOTIC CONTINGENCY REPAIR AND ALIGNMENT
0.06
PRMS PERFORMS COARSE ALIGNMENT AND FIXTURE OF FOURTH COMPONENT FOR ASSEMBLY
0.04
RAMS VERIFIES ALIGNMENT AND PERFORMS FINE ADJUSTMENT AS REQUIRED
0.17
ROBOTIC INSPECTION AND CHECK OUT
0.25
CTV BERTHS WITH THIRD MEV AEROBRAKE COMPONENT
0.04
HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM
0.25
CTV TRANSFERS MEV AEROBRAKE COMPONENT TO ASSEMBLY LOCATION
0.06
PRMS GRAPPLES MEV AEROBRAKE COMPONENT
0.04
PRMS TRANSFERS COMPONENT TO ASF STORAGE POSITION
0.06
CTV SEPARATES FROM AEROBRAKE COMPONENT
0.25
CTV RETURNS TO HLLV TO UNLOAD REMAINING PAYLOAD
0.25
CTV BERTHS WITH FOURTH MEV AEROBRAKE COMPONENT
0.04
HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM
0.25
CTV TRANSFERS MEV AEROBRAKE COMPONENT TO ASSEMBLY LOCATION
0.06
PRMS GRAPPLES MEV AEROBRAKE COMPONENT
0.04
PRMS TRANSFERS COMPONENT TO ASF STORAGE POSITION
0.06
CTV SEPARATES FROM AEROBRAKE COMPONENT
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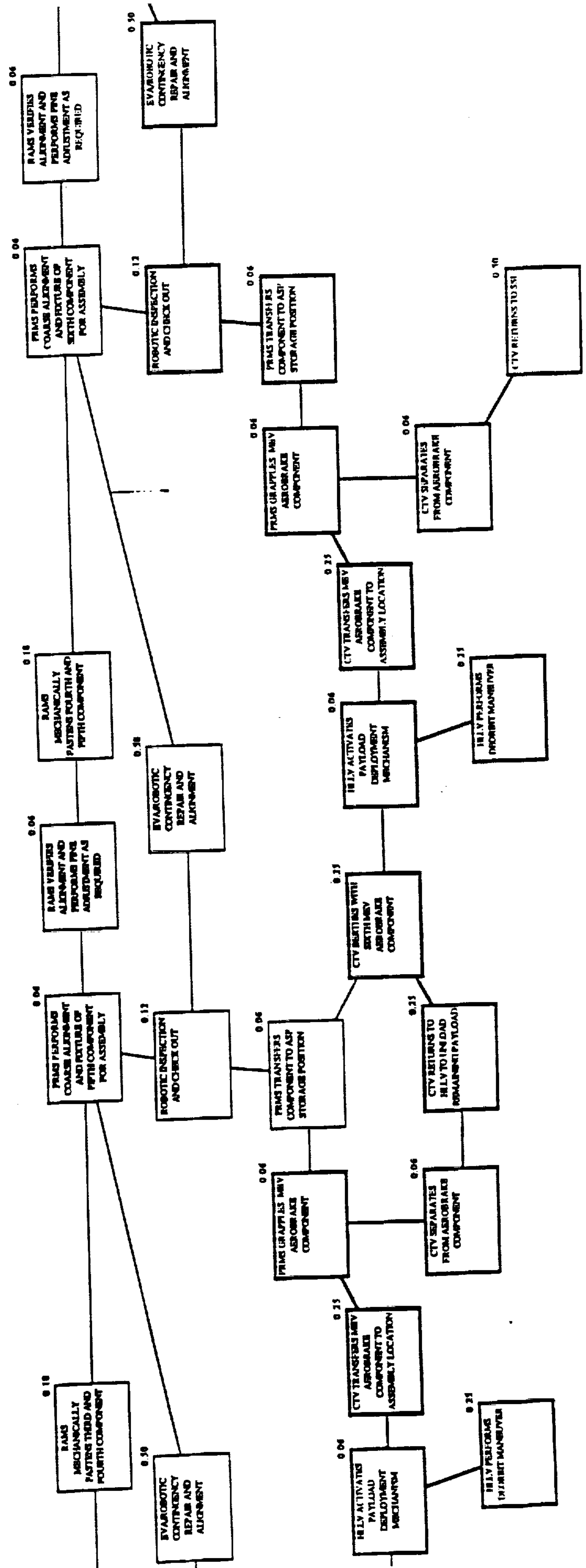
CTV RETURNS TO HLLV TO UNLOAD REMAINING PAYLOAD
0.25
CTV BERTHS WITH FIFTH MEV AEROBRAKE COMPONENT


0.06
RAMS VERIFIES ALIGNMENT AND PERFORMS FINE ADJUSTMENT AS REQUIRED
0.50
EVA/ROBOTIC CONTINGENCY REPAIR AND ALIGNMENT
0.04
PRMS PERFORMS COARSE ALIGNMENT AND FIXTURE OF SIXTH COMPONENT FOR ASSEMBLY
0.12
ROBOTIC INSPECTION AND CHECK OUT
0.06
PRMS TRANSFERS COMPONENT TO ASP STORAGE POSITION
0.10
CTV RETURNS TO SSF
0.04
PRMS GRAPPLES MEV AEROBRAKE COMPONENT
0.06
CTV SEPARATES FROM AEROBRAKE COMPONENT
0.25
CTV TRANSFERS MEV AEROBRAKE COMPONENT TO ASSEMBLY LOCATION
0.25
HLLV PERFORMS DEORBIT MANEUVER
0.04
HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM
0.10
RAMS MECHANICALLY FASTENS FOURTH AND FIFTH COMPONENT
0.50
EVA/ROBOTIC CONTINGENCY REPAIR AND ALIGNMENT
0.06
RAMS VERIFIES ALIGNMENT AND PERFORMS FINE ADJUSTMENT AS REQUIRED
0.25
CTV BERTHS WITH SIXTH MEV AEROBRAKE COMPONENT
0.04
PRMS PERFORMS COARSE ALIGNMENT AND FIXTURE OF FIFTH COMPONENT FOR ASSEMBLY
0.12
ROBOTIC INSPECTION AND CHECK OUT
0.06
PRMS TRANSFERS COMPONENT TO ASP STORAGE POSITION
0.25
CTV RETURNS TO HLLV TO UNLOAD REMAINING PAYLOAD
0.04
PRMS GRAPPLES MEV AEROBRAKE COMPONENT
0.06
CTV SEPARATES FROM AEROBRAKE COMPONENT
0.25
CTV TRANSFERS MEV AEROBRAKE COMPONENT TO ASSEMBLY LOCATION
0.10
RAMS MECHANICALLY FASTENS THIRD AND FOURTH COMPONENT
0.50
EVA/ROBOTIC CONTINGENCY REPAIR AND ALIGNMENT
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HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM
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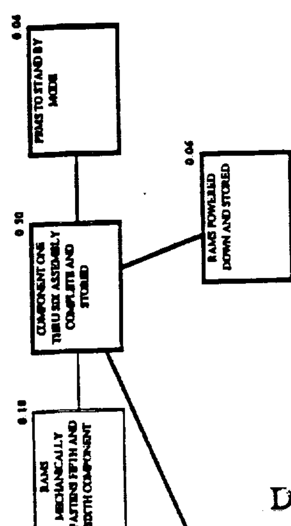
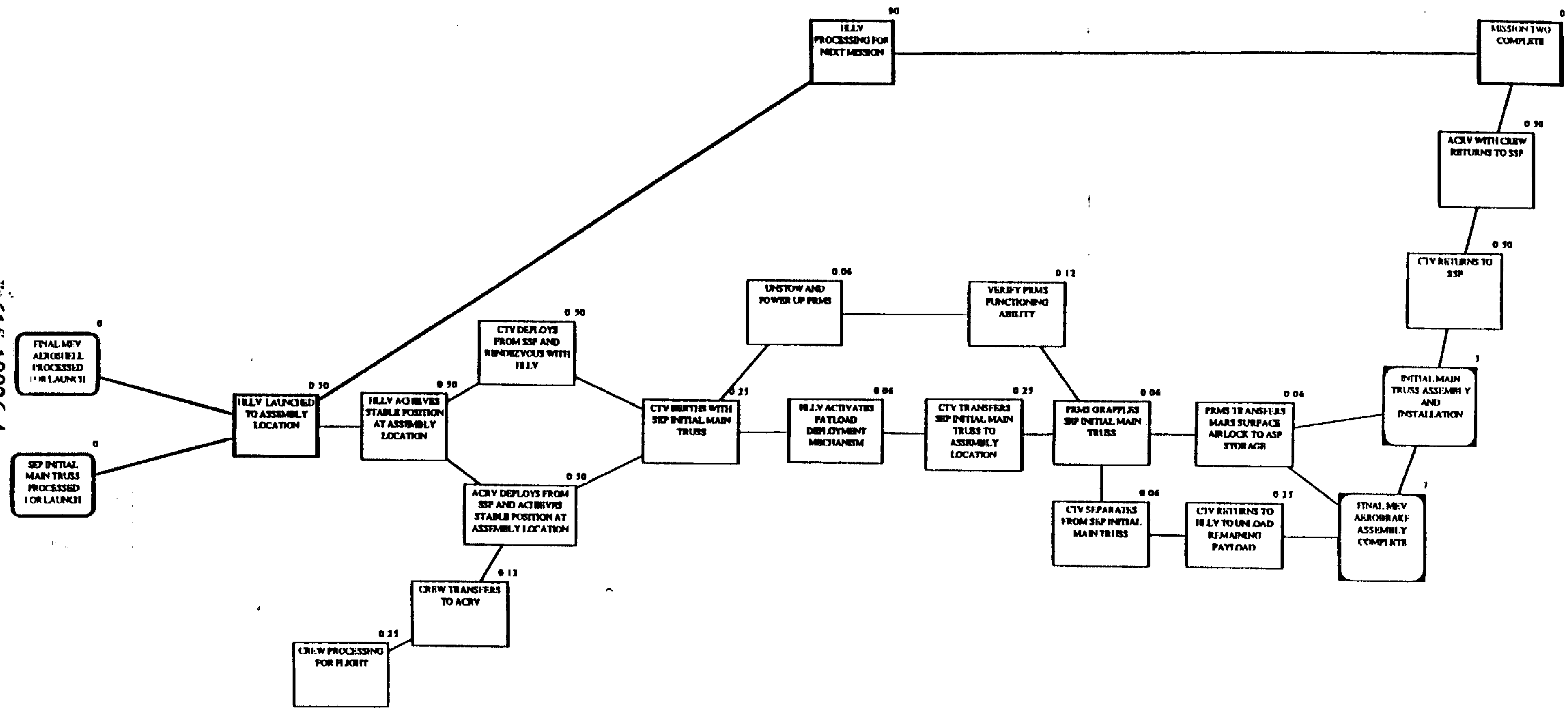
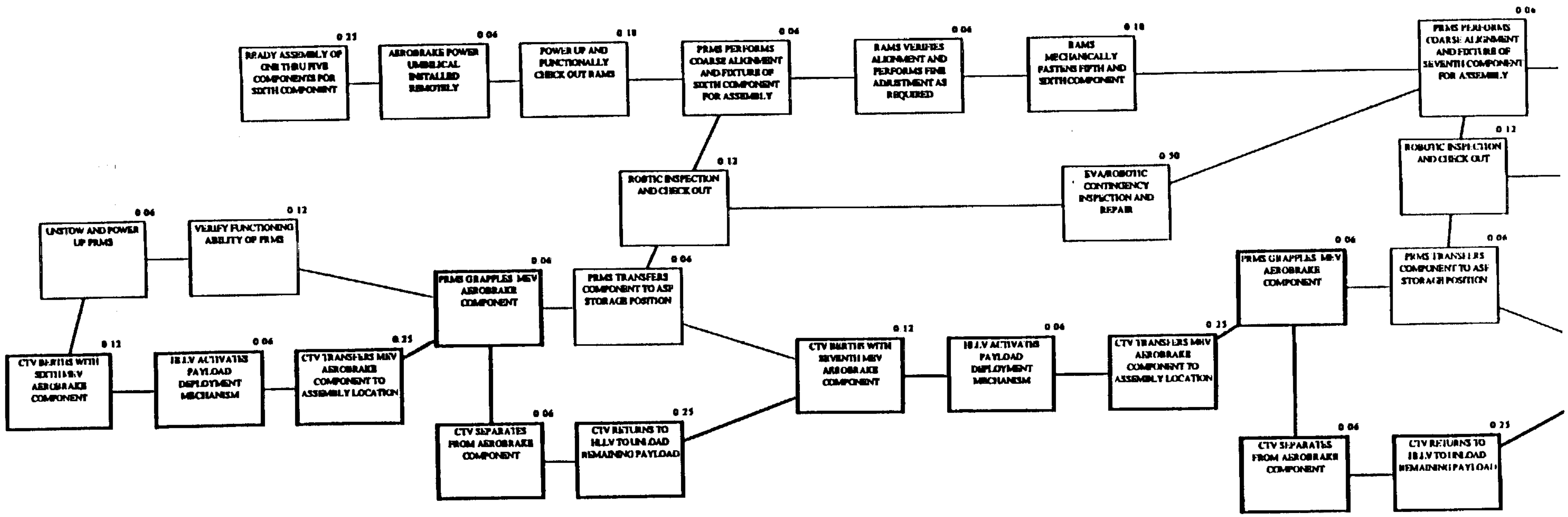
HLLV PERFORMS DEORBIT MANEUVER

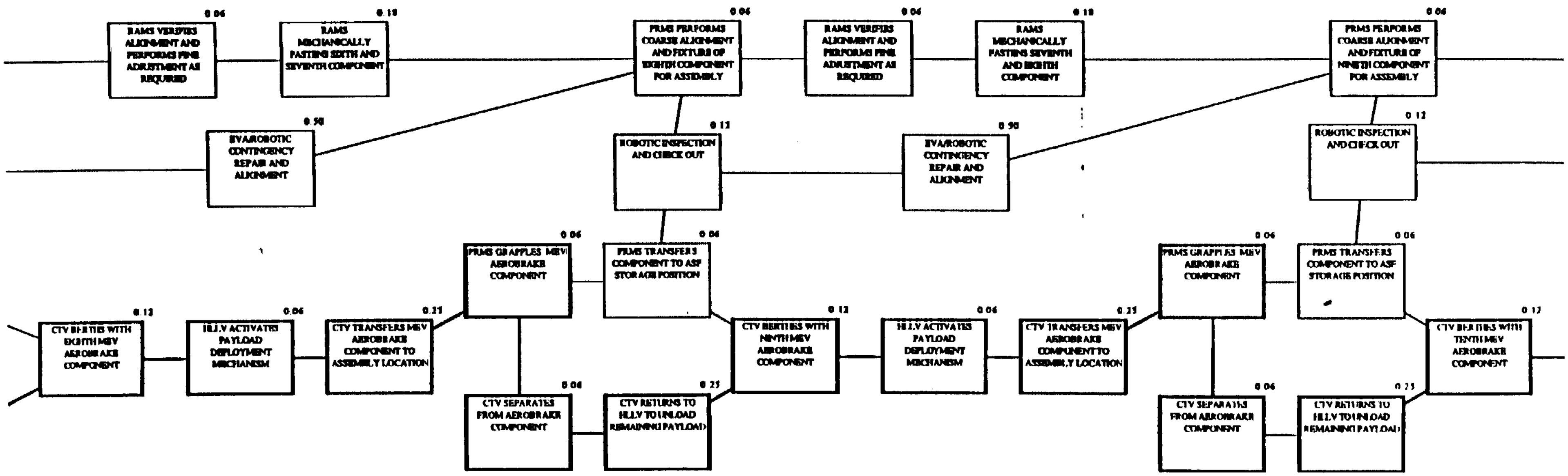

0.04
PRAMS TO STAND BY MODE
0.50
COMPONENT ONE THRU SIX ASSEMBLY COMPLETE AND STORED
0.04
RAMS POWERED DOWN AND STORED
0.10
RAMS MECHANICALLY FASTENS FIFTH AND SIXTH COMPONENT

SEP ASSEMBLY MISSION TWO



SEP MEV AEROBRAKE FINAL ASSEMBLY

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RAMS VERIFIES ALIGNMENT AND PERFORMS FINE ADJUSTMENT AS REQUIRED
0.18
RAMS MECHANICALLY FASTENS SIXTH AND SEVENTH COMPONENT
0.50
EVA/ROBOTIC CONTINGENCY REPAIR AND ALIGNMENT
0.06
PRMS PERFORMS COARSE ALIGNMENT AND FIXTURE OF EIGHTH COMPONENT FOR ASSEMBLY
0.12
ROBOTIC INSPECTION AND CHECK OUT
0.06
RAMS VERIFIES ALIGNMENT AND PERFORMS FINE ADJUSTMENT AS REQUIRED
0.18
RAMS MECHANICALLY FASTENS SEVENTH AND EIGHTH COMPONENT
0.50
EVA/ROBOTIC CONTINGENCY REPAIR AND ALIGNMENT
0.06
PRMS PERFORMS COARSE ALIGNMENT AND FIXTURE OF NINETH COMPONENT FOR ASSEMBLY
0.12
ROBOTIC INSPECTION AND CHECK OUT
0.12
CTV BERTHS WITH EIGHTH MEV AEROBRAKE COMPONENT
0.06
HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM
0.25
CTV TRANSFERS MEV AEROBRAKE COMPONENT TO ASSEMBLY LOCATION
0.06
PRMS GRAPPLES MEV AEROBRAKE COMPONENT
0.06
PRMS TRANSFERS COMPONENT TO ASF STORAGE POSITION
0.06
CTV SEPARATES FROM AEROBRAKE COMPONENT
0.25
CTV RETURNS TO HLLV TO UNLOAD REMAINING PAYLOAD
0.12
CTV BERTHS WITH NINTH MEV AEROBRAKE COMPONENT
0.06
HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM
0.25
CTV TRANSFERS MEV AEROBRAKE COMPONENT TO ASSEMBLY LOCATION
0.06
PRMS GRAPPLES MEV AEROBRAKE COMPONENT
0.06
PRMS TRANSFERS COMPONENT TO ASF STORAGE POSITION
0.06
CTV SEPARATES FROM AEROBRAKE COMPONENT
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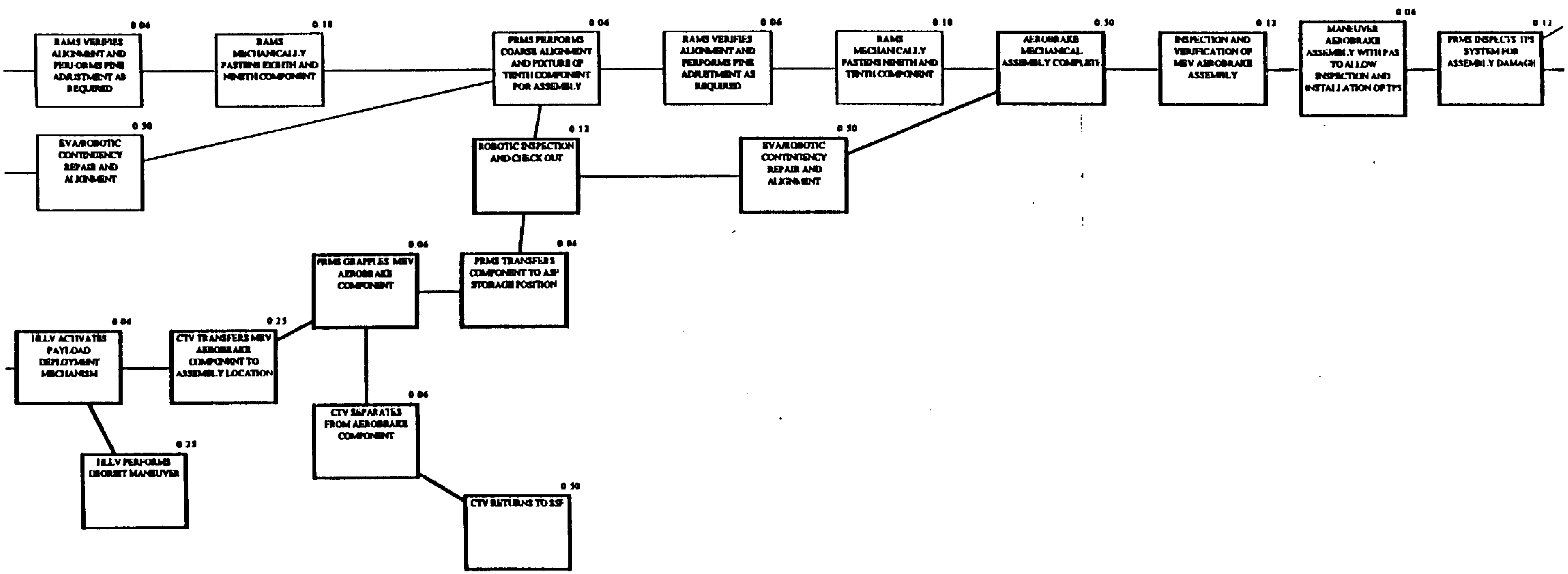
CTV RETURNS TO HLLV TO UNLOAD REMAINING PAYLOAD
0.12
CTV BERTHS WITH TENTH MEV AEROBRAKE COMPONENT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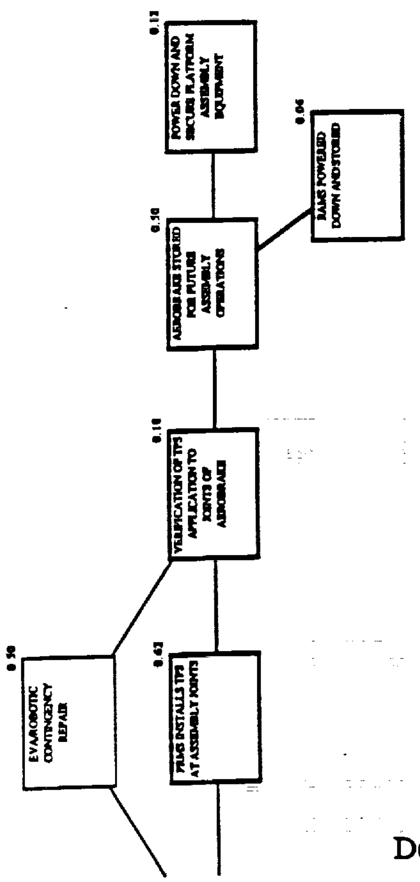
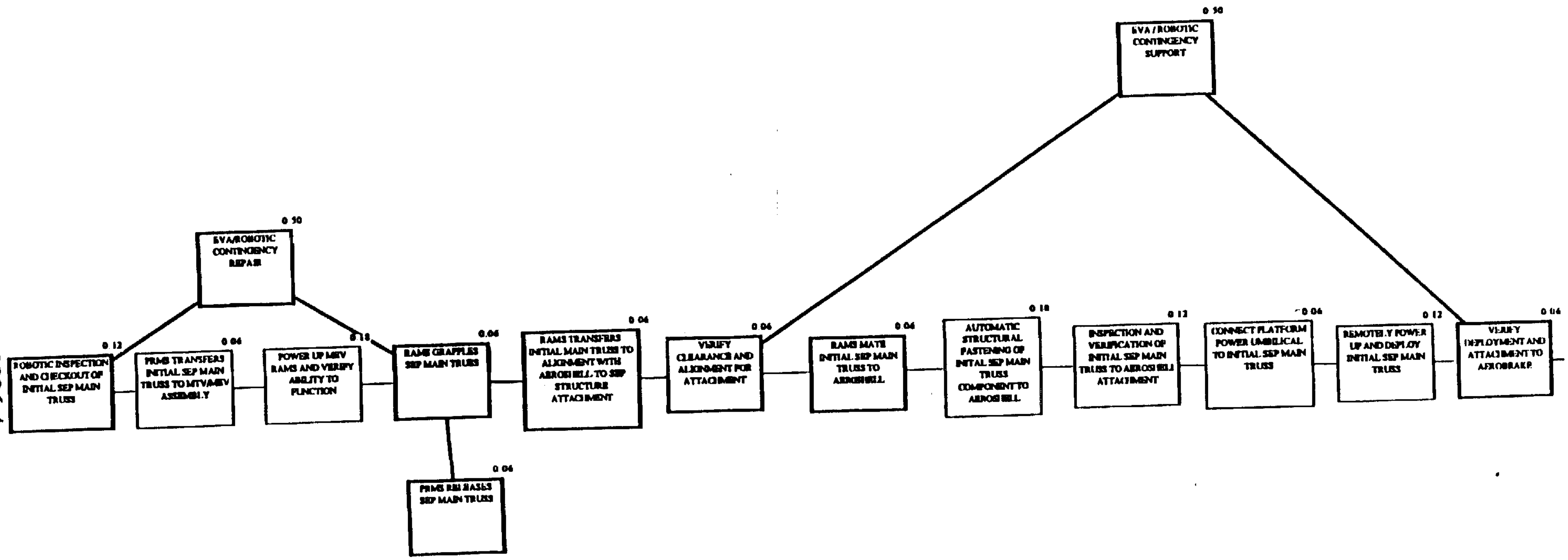
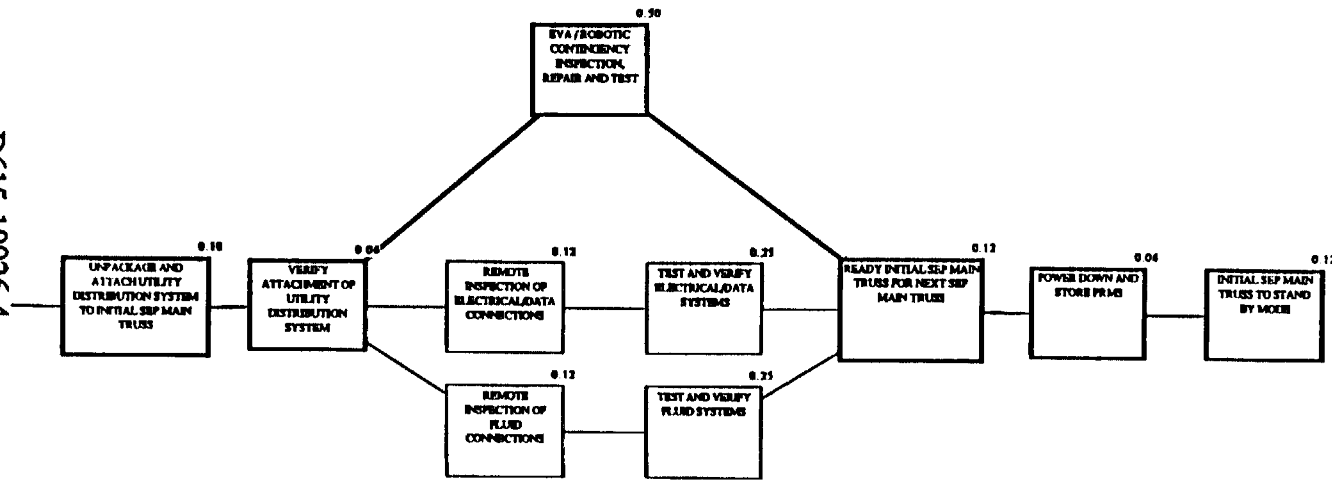
RAMS VERIFIES ALIGNMENT AND PERFORMS FINE ADJUSTMENT AS REQUIRED
0.18
RAMS MECHANICALLY FASTENS EIGHTH AND NINETH COMPONENT
0.04
PRMS PERFORMS COARSE ALIGNMENT AND FIXTURE OF TENTH COMPONENT FOR ASSEMBLY
0.06
RAMS VERIFIES ALIGNMENT AND PERFORMS FINE ADJUSTMENT AS REQUIRED
0.18
RAMS MECHANICALLY FASTENS NINETH AND TENTH COMPONENT
0.50
AEROBRAKE MECHANICAL ASSEMBLY COMPLETE
0.12
INSPECTION AND VERIFICATION OF MEV AEROBRAKE ASSEMBLY
0.04
MANEUVER AEROBRAKE ASSEMBLY WITH PAS TO ALLOW INSPECTION AND INSTALLATION OF TPS
0.12
PRMS INSPECTS TPS SYSTEM FOR ASSEMBLY DAMAGE
0.50
EVA/ROBOTIC CONTINGENCY REPAIR AND ALIGNMENT
0.12
ROBOTIC INSPECTION AND CHECK OUT
0.50
EVA/ROBOTIC CONTINGENCY REPAIR AND ALIGNMENT
0.06
PRMS GRAPPLES MEV AEROBRAKE COMPONENT
0.04
PRMS TRANSFERS COMPONENT TO ASP STORAGE POSITION
0.04
HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM
0.25
CTV TRANSFERS MEV AEROBRAKE COMPONENT TO ASSEMBLY LOCATION
0.04
CTV SEPARATES FROM AEROBRAKE COMPONENT
0.25
HLLV PERFORMS DEORBIT MANEUVER
0.50
CTV RETURNS TO SSF

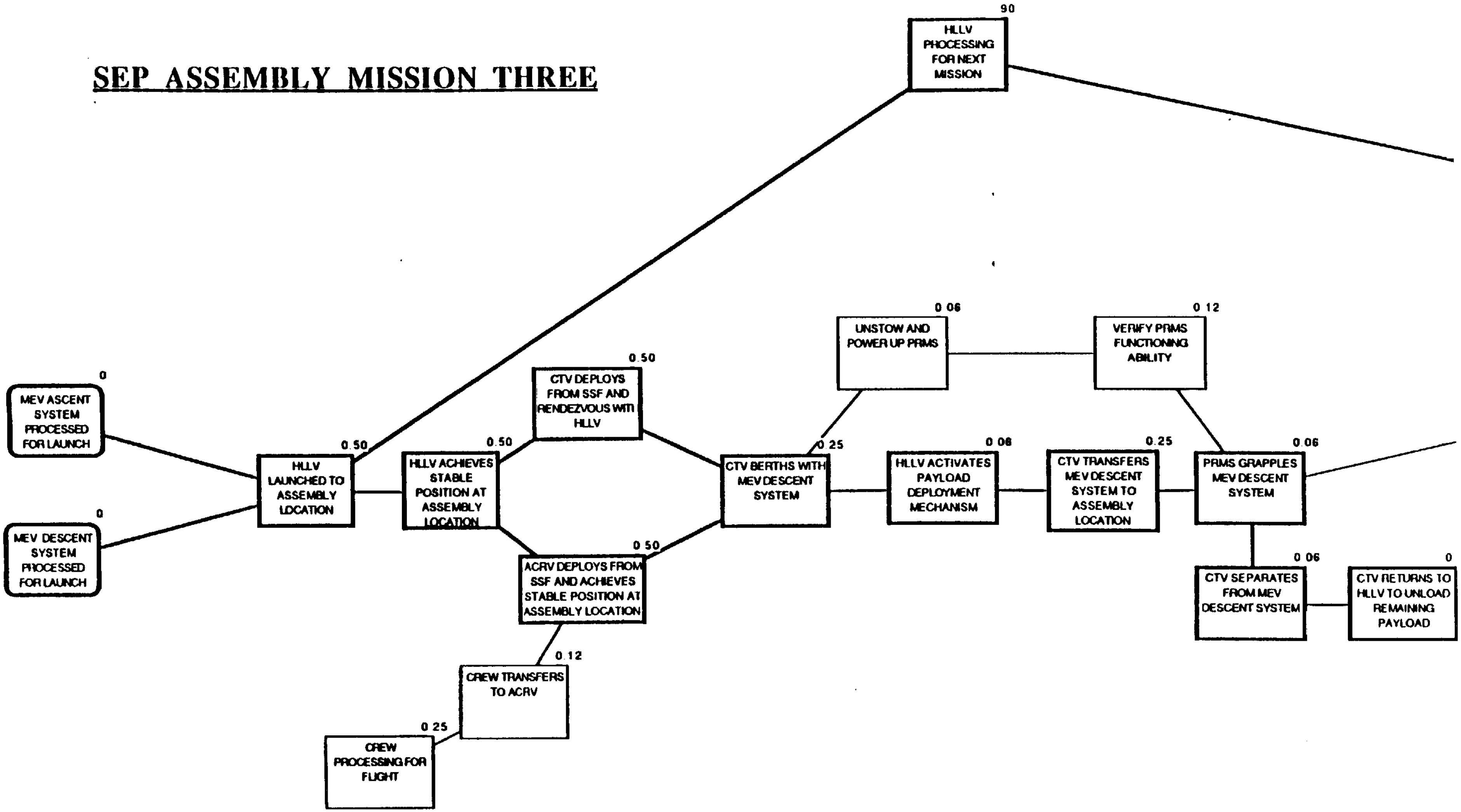

POWER DOWN AND SECURE PLATFORM ASSEMBLY EQUIPMENT
AEROBRAKE STORED FOR FUTURE ASSEMBLY OPERATIONS
RAMS POWERED DOWN AND STORED
VERIFICATION OF TPS APPLICATION TO JOINTS OF AEROBRAKE
EVA/ROBOTIC CONTINGENCY REPAIR
PRMS INSTALLS TPS AT ASSEMBLY JOINTS

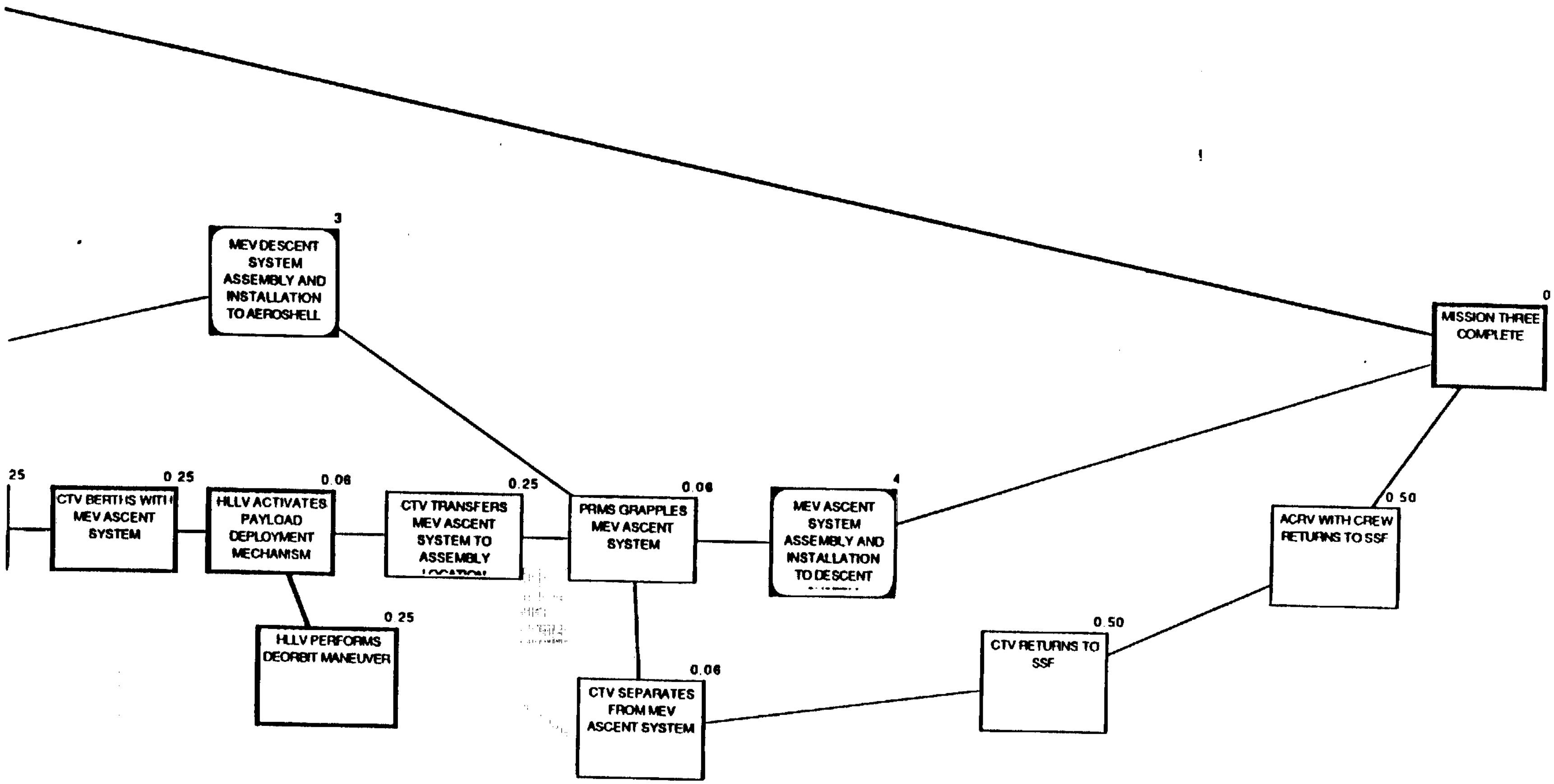
SEP INITIAL MAIN TRUSS ASSEMBLY




0.10
UNPACKAGE AND ATTACH UTILITY DISTRIBUTION SYSTEM TO INITIAL SEP MAIN TRUSS
0.04
VERIFY ATTACHMENT OF UTILITY DISTRIBUTION SYSTEM
0.50
EVA / ROBOTIC CONTINGENCY INSPECTION, REPAIR AND TEST
0.12
REMOTE INSPECTION OF ELECTRICAL/DATA CONNECTIONS
0.25
TEST AND VERIFY ELECTRICAL/DATA SYSTEMS
0.12
REMOTE INSPECTION OF FLUID CONNECTIONS
0.25
TEST AND VERIFY FLUID SYSTEMS
0.12
READY INITIAL SEP MAIN TRUSS FOR NEXT SEP MAIN TRUSS
0.04
POWER DOWN AND STORE PRMS
0.12
INITIAL SEP MAIN TRUSS TO STAND BY MODE

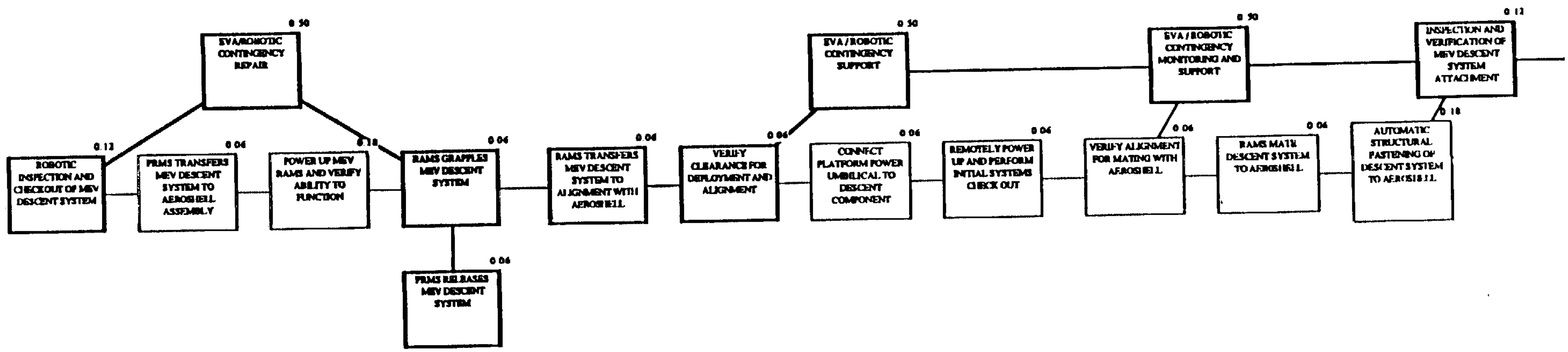
SEP ASSEMBLY MISSION THREE

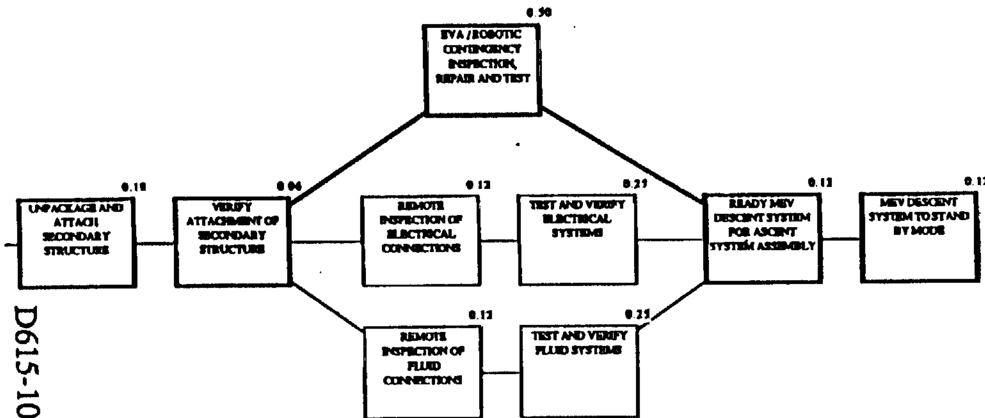




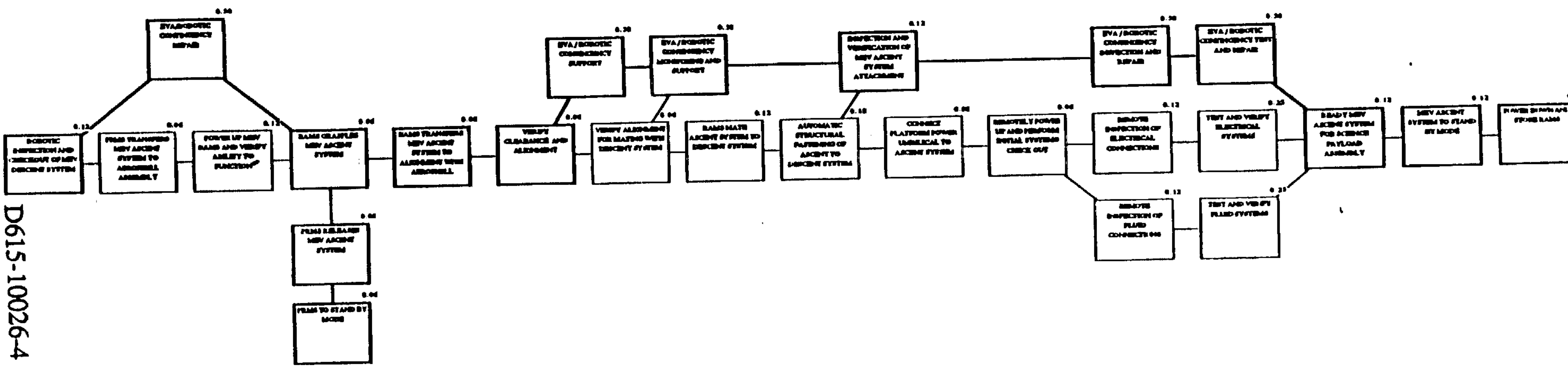
25
CTV BERTHS WITH MEV ASCENT SYSTEM
0.25
HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM
0.06
HLLV PERFORMS DEORBIT MANEUVER
0.25
CTV TRANSFERS MEV ASCENT SYSTEM TO ASSEMBLY
0.25
MEV DESCENT SYSTEM ASSEMBLY AND INSTALLATION TO AEROSHELL
3
PRMS GRAPPLES MEV ASCENT SYSTEM
0.06
CTV SEPARATES FROM MEV ASCENT SYSTEM
0.06
MEV ASCENT SYSTEM ASSEMBLY AND INSTALLATION TO DESCENT
4
CTV RETURNS TO SSF
0.50
ACRV WITH CREW RETURNS TO SSF
0.50
MISSION THREE COMPLETE
0

SEP MEV DESCENT SYSTEM ASSEMBLY

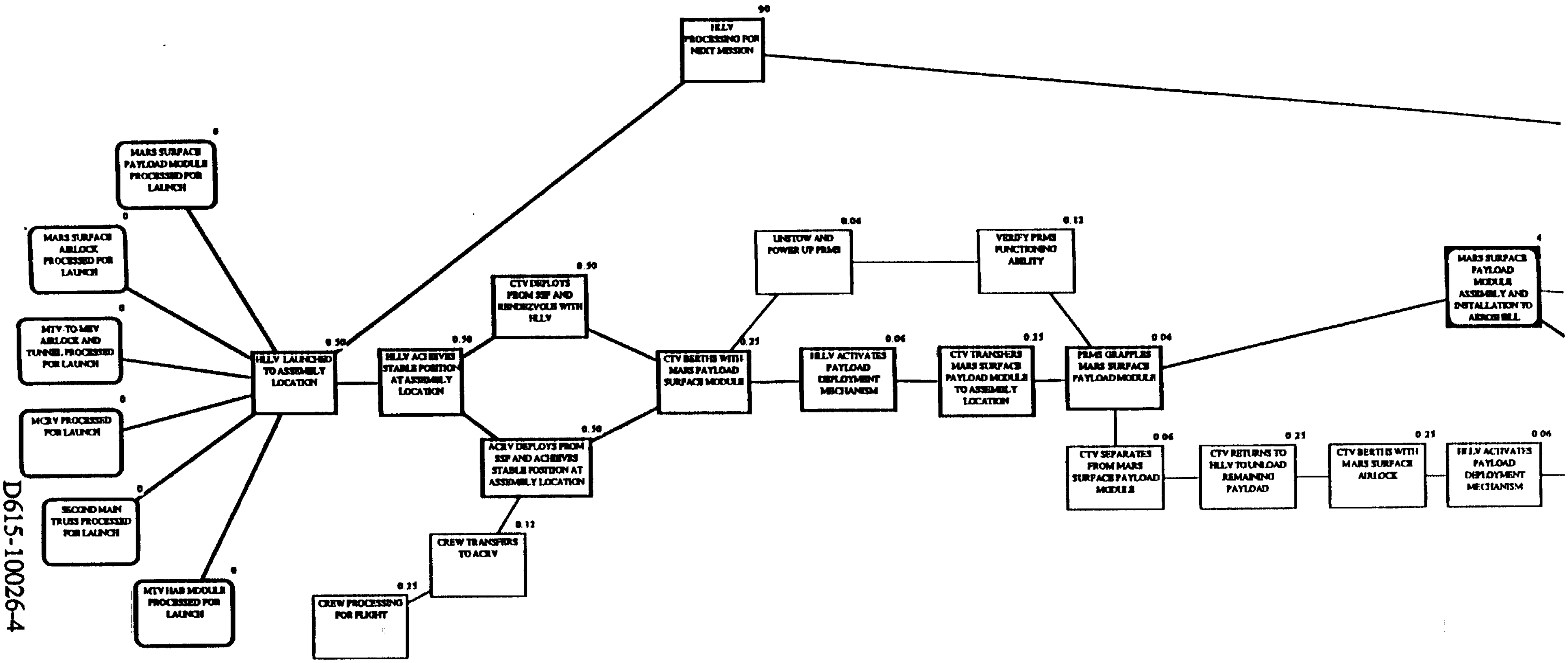


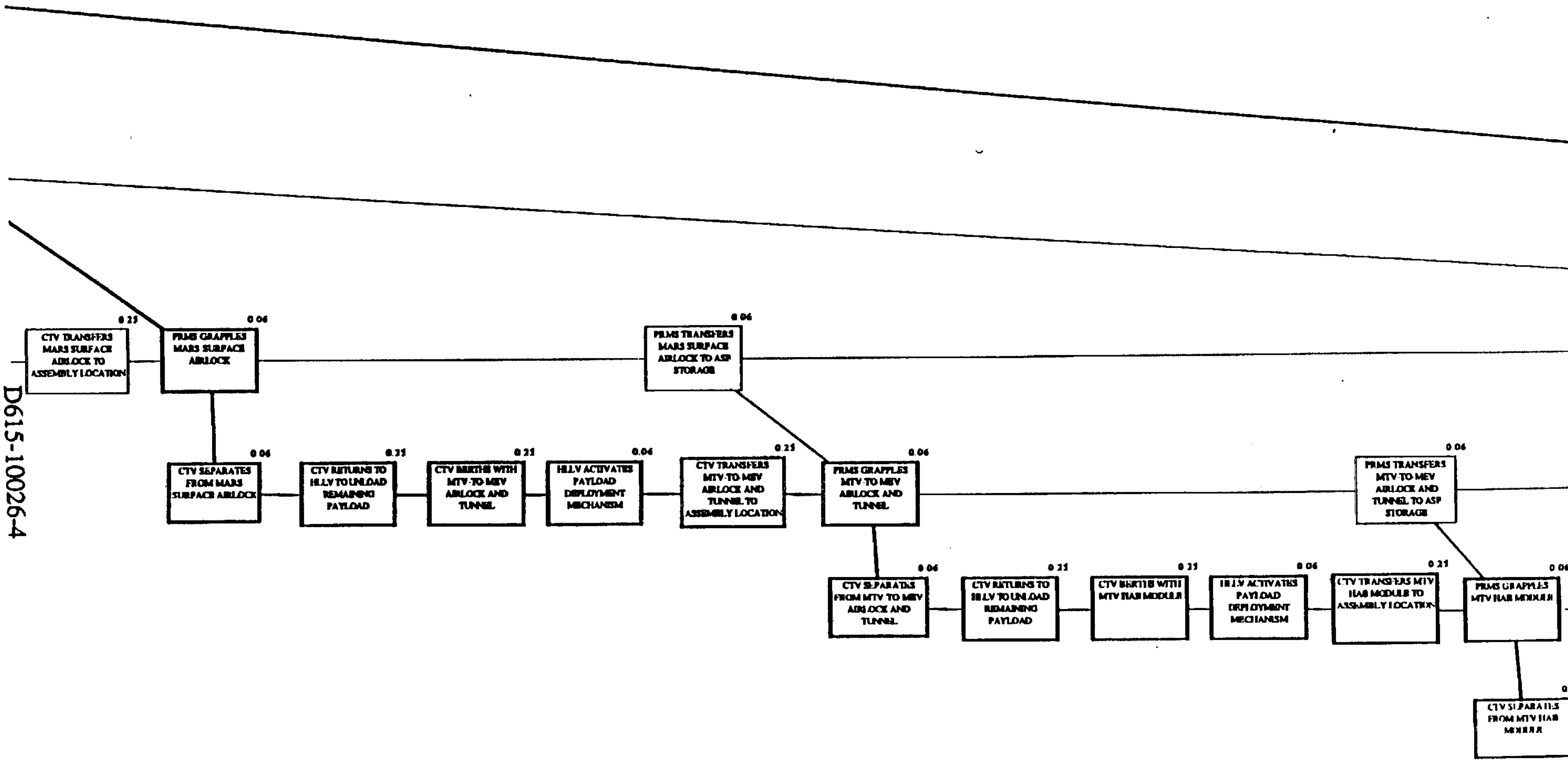

0.18
UNPACKAGE AND ATTACH SECONDARY STRUCTURE
0.06
VERIFY ATTACHMENT OF SECONDARY STRUCTURE
0.50
EVA / ROBOTIC CONTINGENCY INSPECTION, REPAIR AND TEST
0.12
REMOTE INSPECTION OF ELECTRICAL CONNECTIONS
0.25
TEST AND VERIFY ELECTRICAL SYSTEMS
0.12
REMOTE INSPECTION OF FLUID CONNECTIONS
0.25
TEST AND VERIFY FLUID SYSTEMS
0.12
READY MEV DESCENT SYSTEM FOR ASCENT SYSTEM ASSEMBLY
0.12
MEV DESCENT SYSTEM TO STAND BY MODE

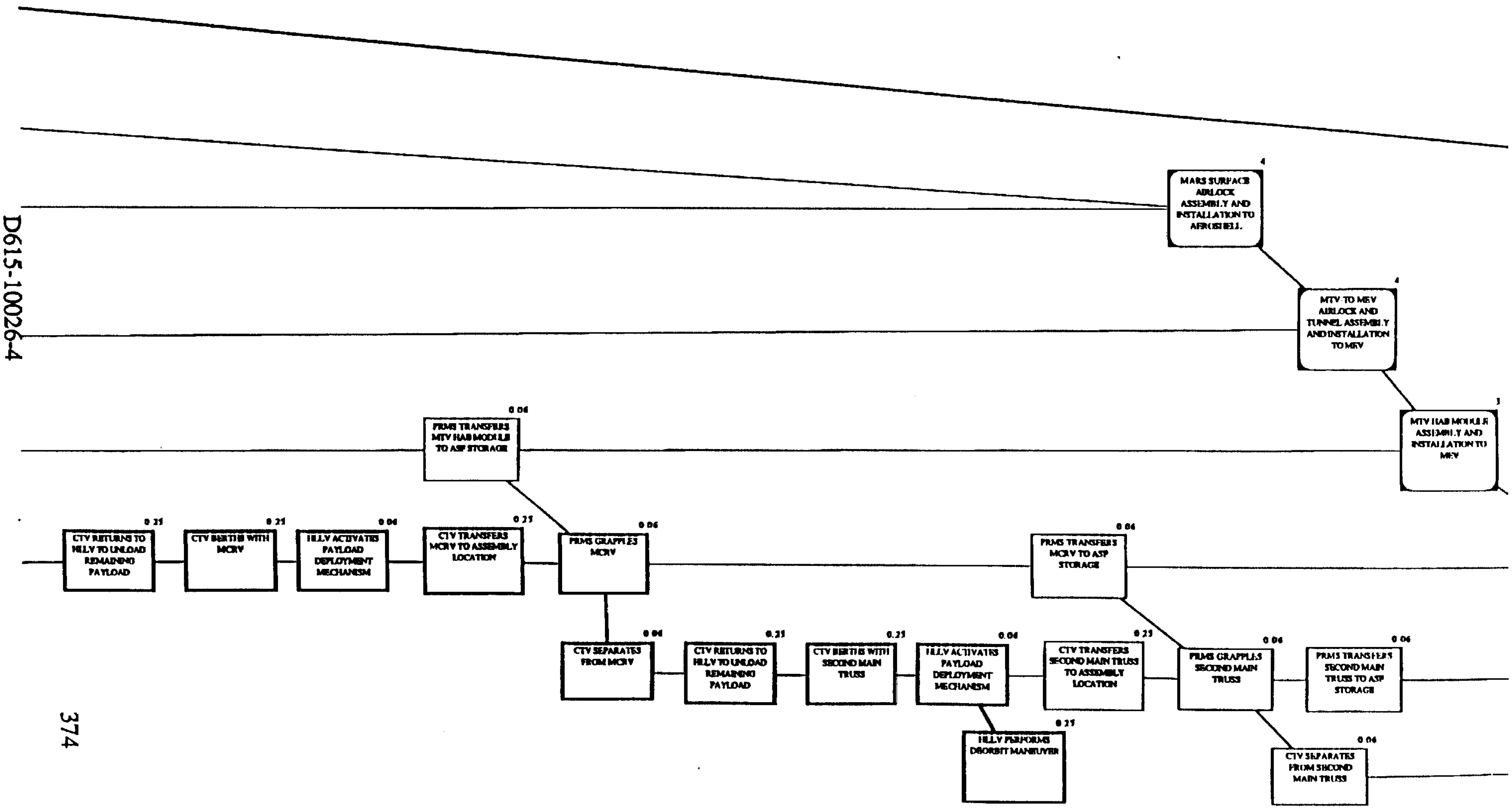
SEP MEV ASCENT SYSTEM ASSEMBLY



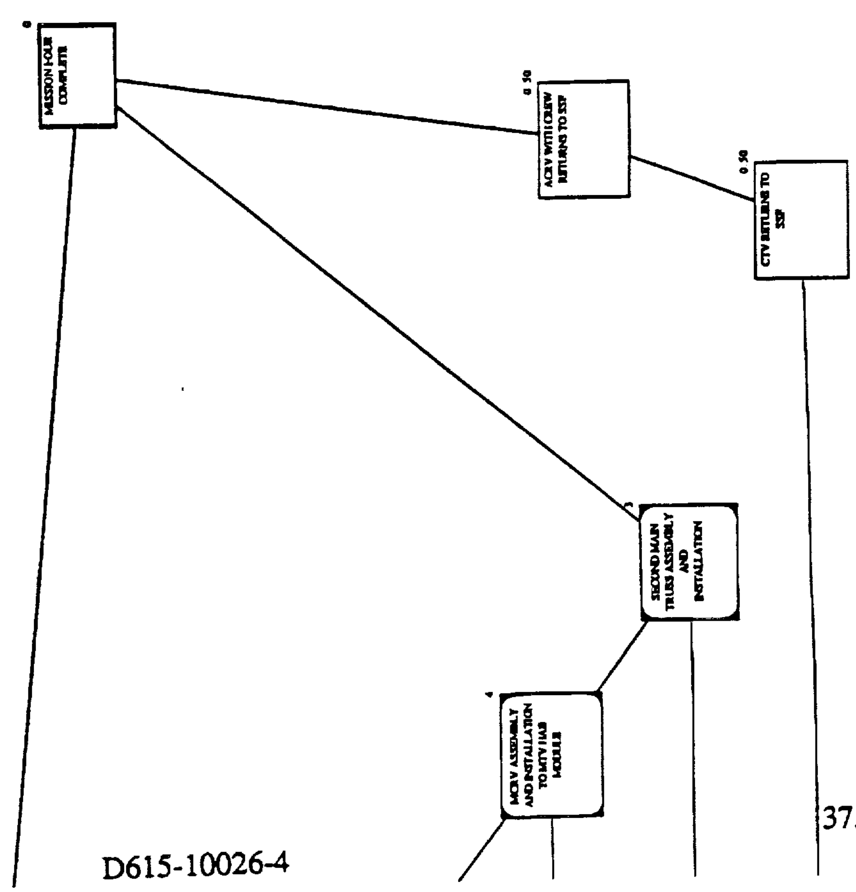
SEP ASSEMBLY MISSION FOUR



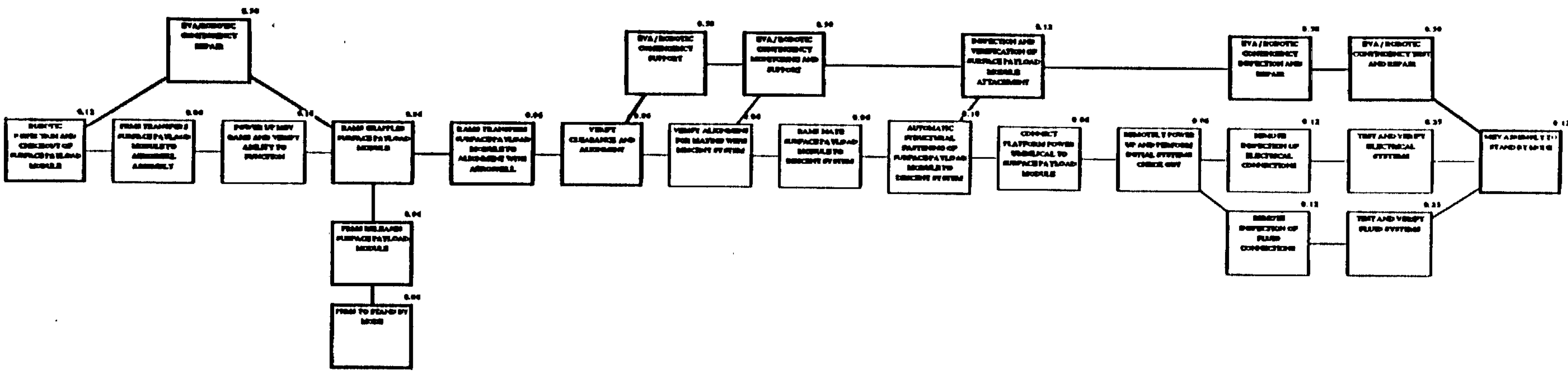

0.25
CTV TRANSFERS MARS SURFACE AIRLOCK TO ASSEMBLY LOCATION
0.06
PRMS GRAPPLES MARS SURFACE AIRLOCK
0.06
CTV SEPARATES FROM MARS SURFACE AIRLOCK
0.25
CTV RETURNS TO HLLV TO UNLOAD REMAINING PAYLOAD
0.25
CTV BERTHS WITH MTV-TO-MEV AIRLOCK AND TUNNEL
0.06
HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM
0.25
CTV TRANSFERS MTV-TO-MEV AIRLOCK AND TUNNEL TO ASSEMBLY LOCATION
0.06
PRMS TRANSFERS MARS SURFACE AIRLOCK TO ASP STORAGE
0.06
PRMS GRAPPLES MTV-TO-MEV AIRLOCK AND TUNNEL
0.06
CTV SEPARATES FROM MTV TO MEV AIRLOCK AND TUNNEL
0.25
CTV RETURNS TO HLLV TO UNLOAD REMAINING PAYLOAD
0.25
CTV BERTHS WITH MTV HAB MODULE
0.06
HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM
0.25
CTV TRANSFERS MTV HAB MODULE TO ASSEMBLY LOCATION
0.06
PRMS TRANSFERS MTV-TO-MEV AIRLOCK AND TUNNEL TO ASP STORAGE
0.06
PRMS GRAPPLES MTV HAB MODULE
0.06
CTV SEPARATES FROM MTV HAB MODULE



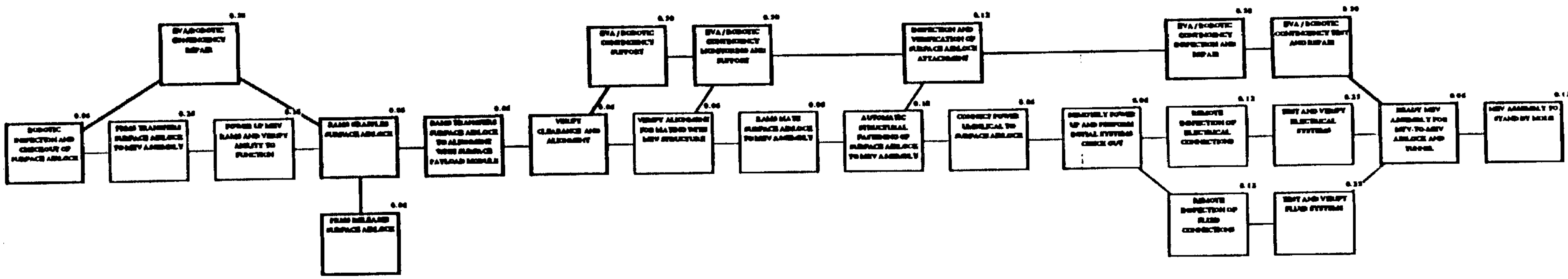
4
MARS SURFACE AIRLOCK ASSEMBLY AND INSTALLATION TO AEROSHELL
4
MTV TO MEV AIRLOCK AND TUNNEL ASSEMBLY AND INSTALLATION TO MEV
3
MTV HAB MODULE ASSEMBLY AND INSTALLATION TO MEV
0.06
PRMS TRANSFERS MTV HAB MODULE TO ASP STORAGE
0.25
CTV RETURNS TO HLLV TO UNLOAD REMAINING PAYLOAD
0.25
CTV BERTHS WITH MCRV
0.06
HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM
0.25
CTV TRANSFERS MCRV TO ASSEMBLY LOCATION
0.06
PRMS GRAPPLES MCRV
0.06
PRMS TRANSFERS MCRV TO ASP STORAGE
0.06
CTV SEPARATES FROM MCRV
0.25
CTV RETURNS TO HLLV TO UNLOAD REMAINING PAYLOAD
0.25
CTV BERTHS WITH SECOND MAIN TRUSS
0.06
HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM
0.25
HLLV PERFORMS DEORBIT MANEUVER
0.25
CTV TRANSFERS SECOND MAIN TRUSS TO ASSEMBLY LOCATION
0.06
PRMS GRAPPLES SECOND MAIN TRUSS
0.06
PRMS TRANSFERS SECOND MAIN TRUSS TO ASP STORAGE
0.06
CTV SEPARATES FROM SECOND MAIN TRUSS


MISSION FOUR COMPLETE
0 50
ACRV WITH CREW RETURNS TO SSF
0 50
CTV RETURNS TO SSF
SECOND MAIN TRUSS ASSEMBLY AND INSTALLATION
MCRV ASSEMBLY AND INSTALLATION TO MTV HAB MODULE

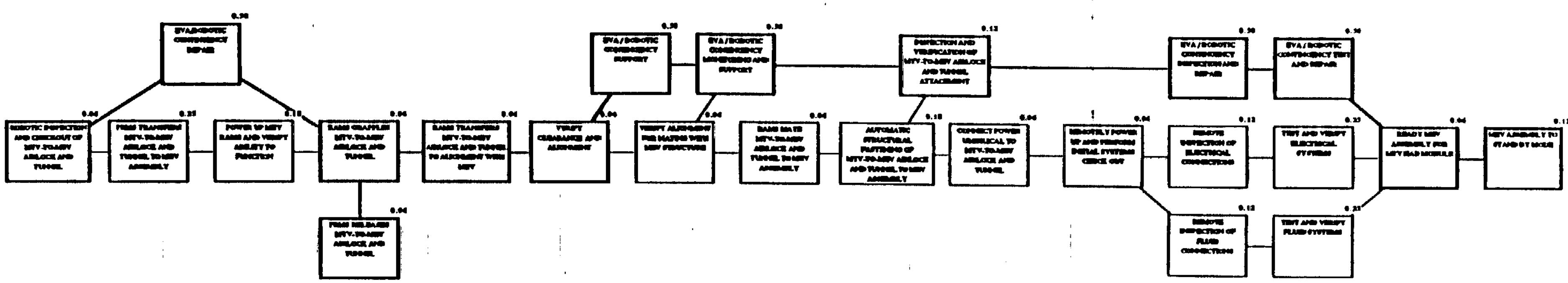
SEP MARS SURFACE PAYLOAD MODULE ASSEMBLY



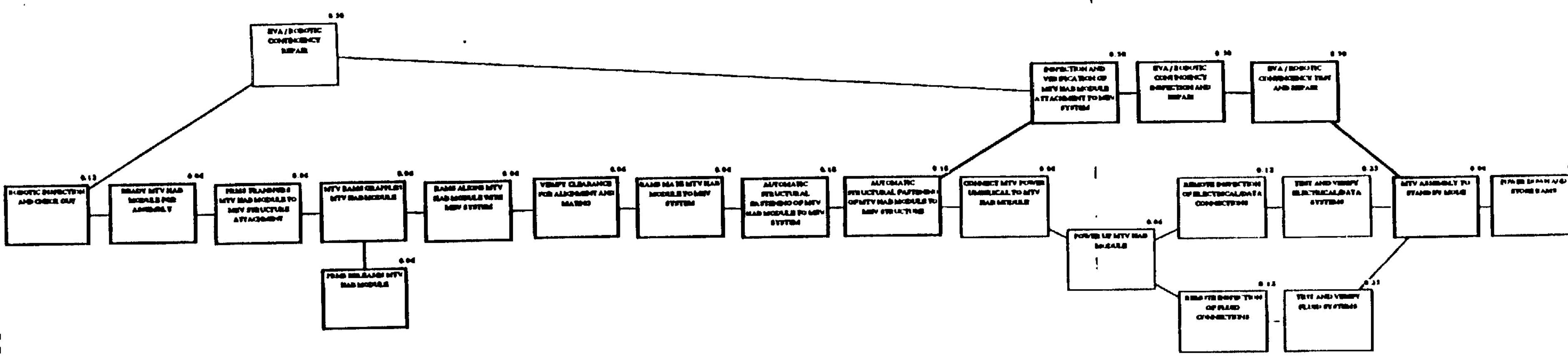
SEP MARS SURFACE AIRLOCK ASSEMBLY

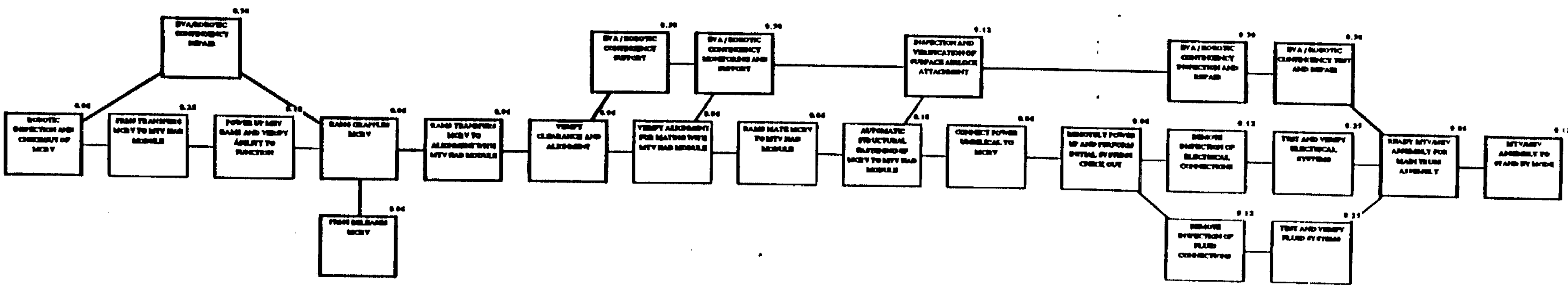


SEP MTV-TO-MEV AIRLOCK AND TUNNEL ASSEMBLY

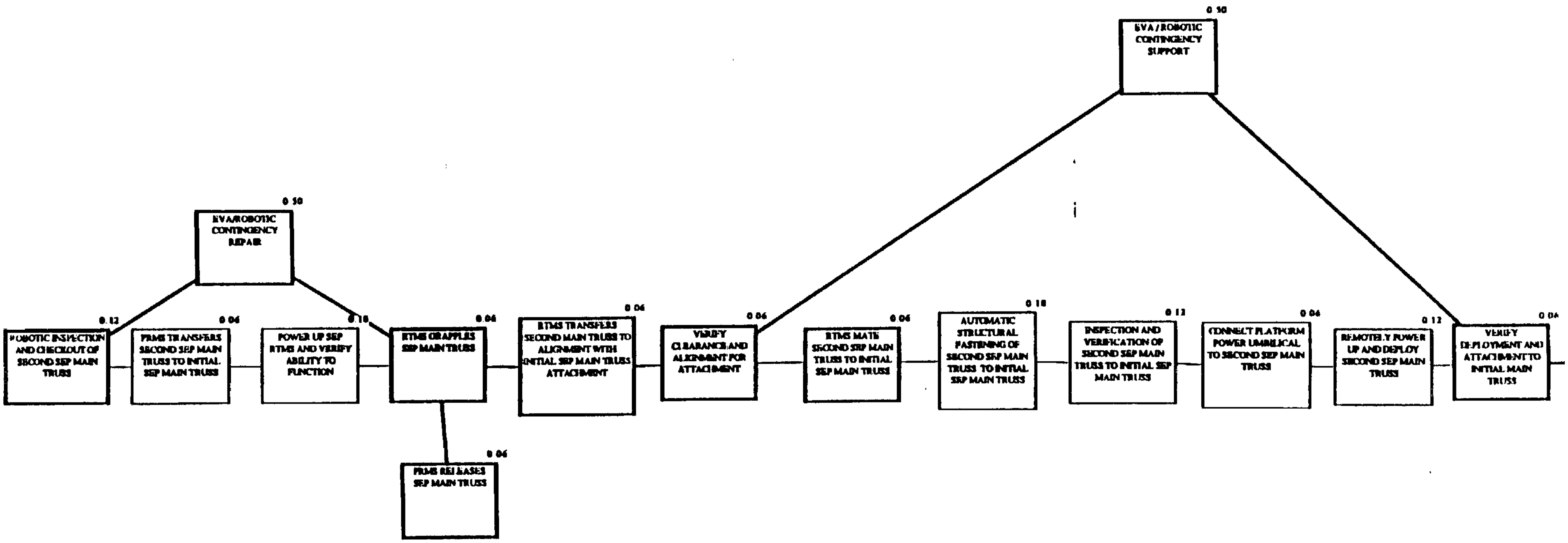


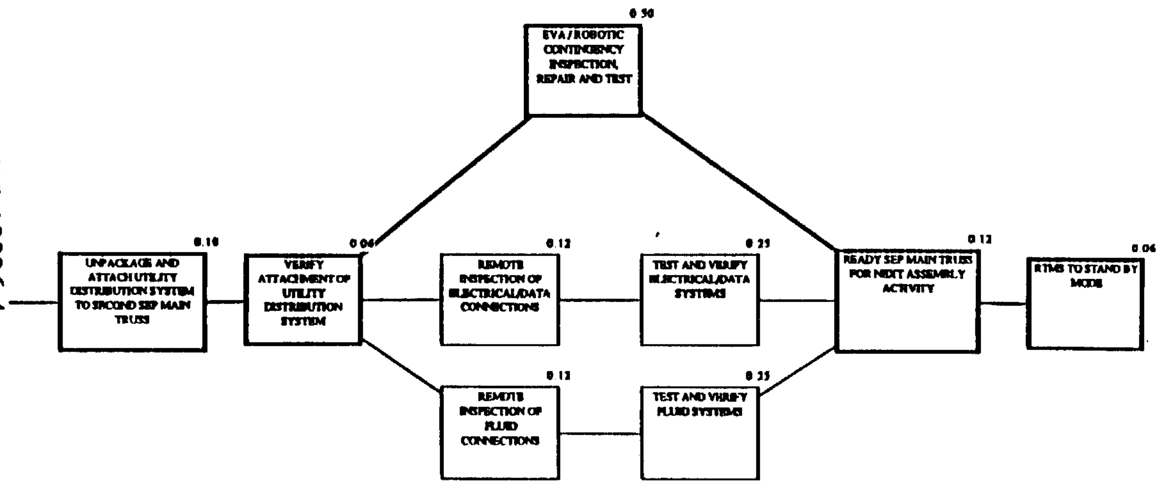
SEP MTV HAB MODULE ASSEMBLY





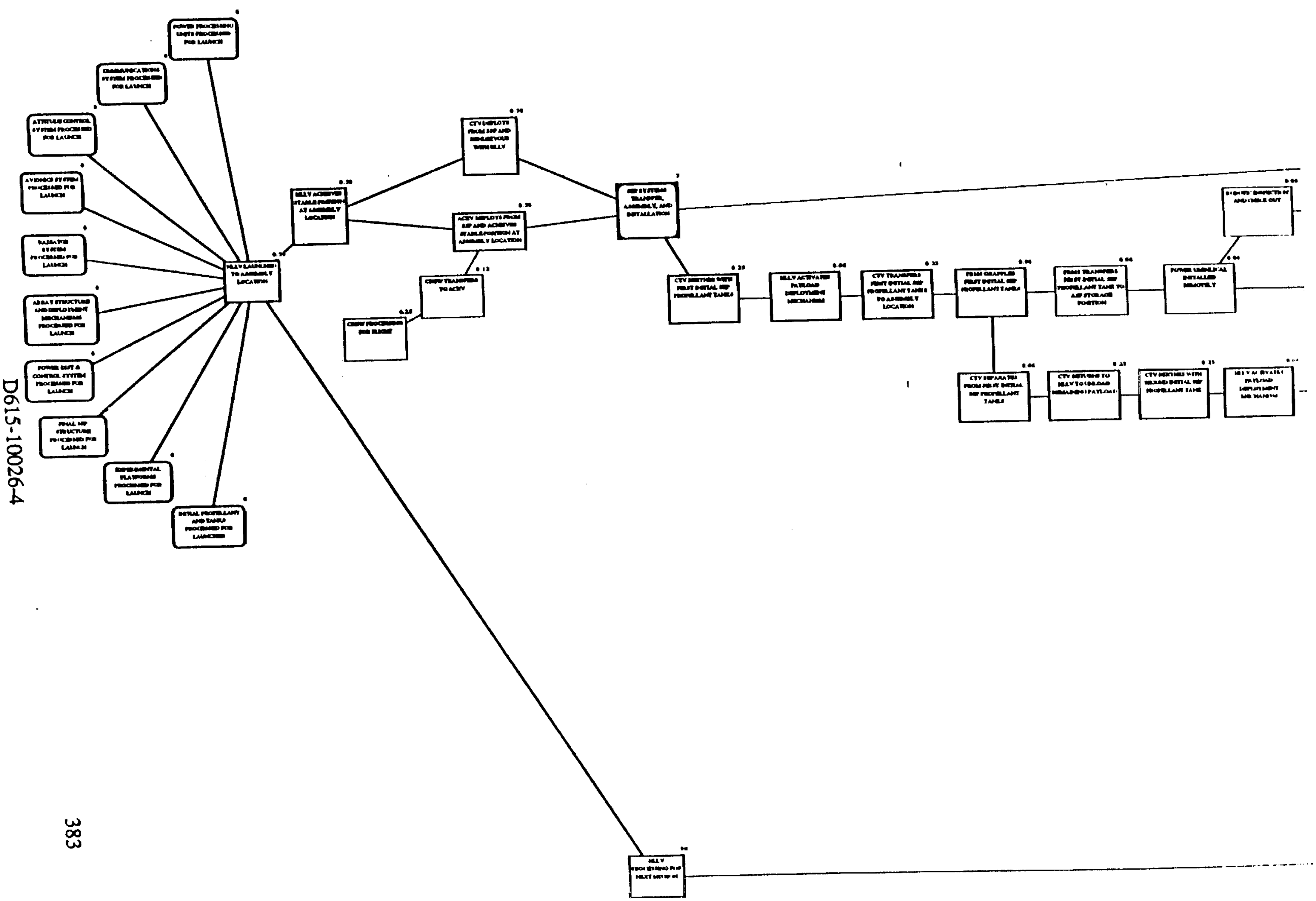
SEP SECOND MAIN TRUSS ASSEMBLY

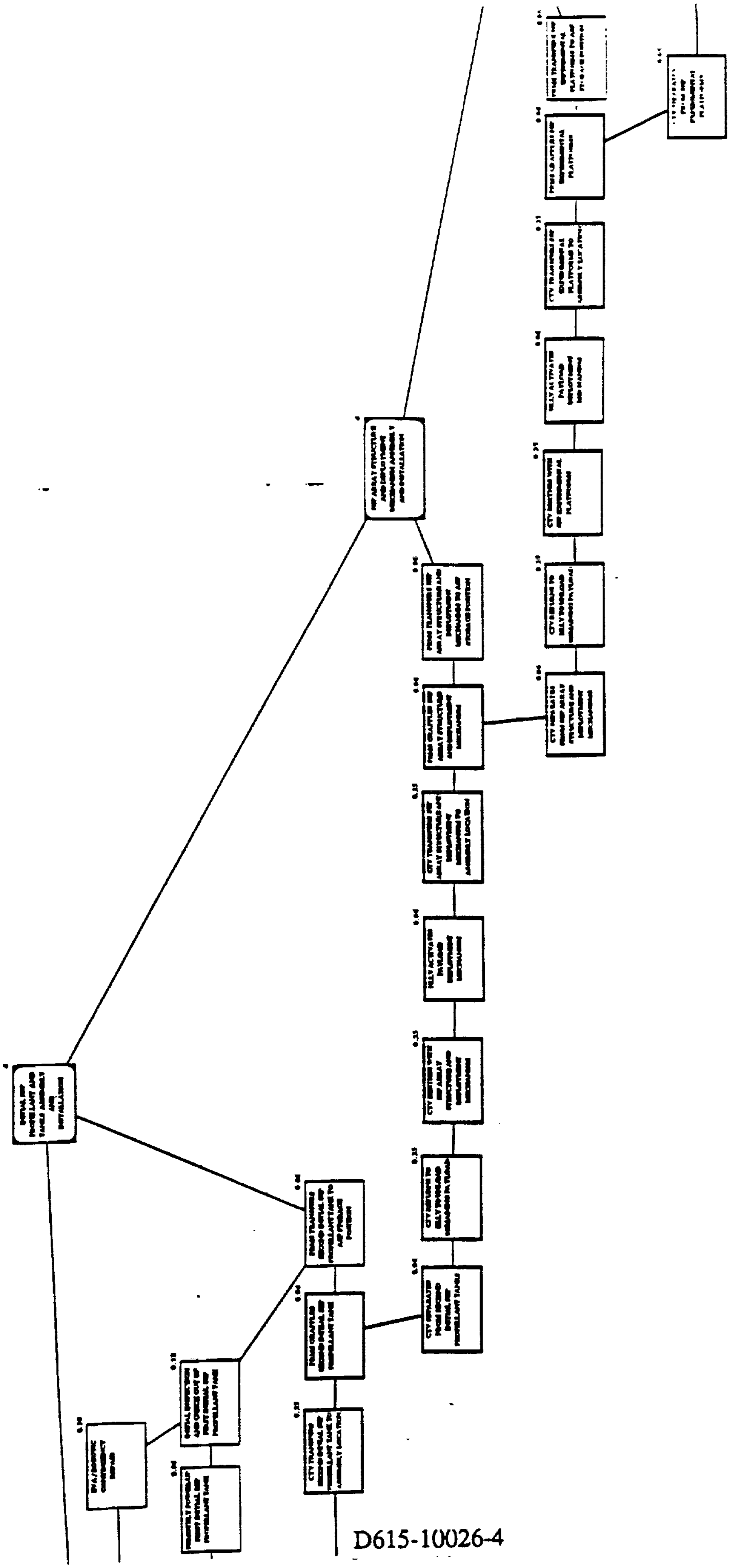


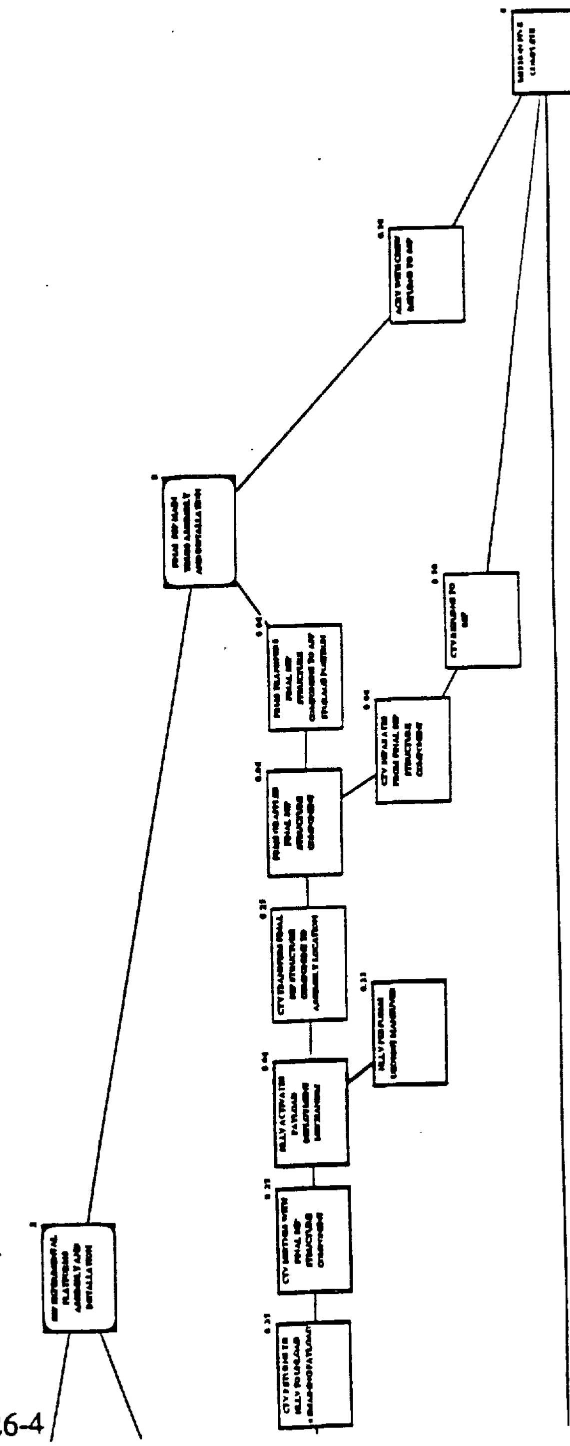


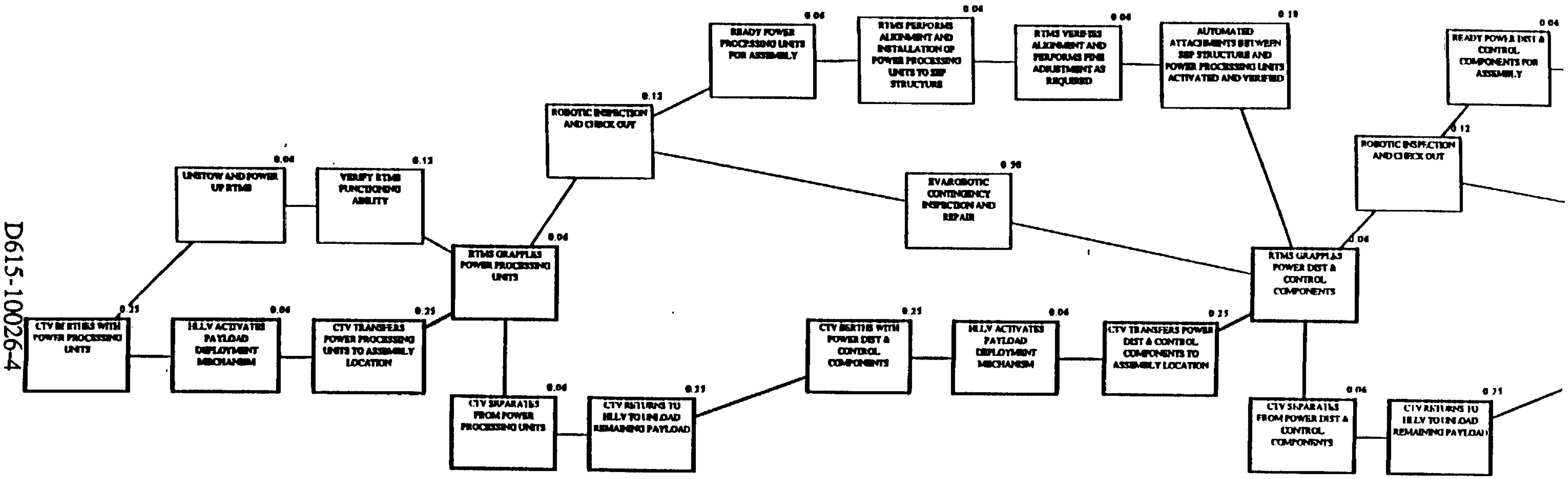
0.10
UNPACKAGE AND ATTACH UTILITY DISTRIBUTION SYSTEM TO SECOND SEP MAIN TRUSS
0.04
VERIFY ATTACHMENT OF UTILITY DISTRIBUTION SYSTEM
0.50
EVA / ROBOTIC CONTINGENCY INSPECTION, REPAIR AND TEST
0.12
REMOTE INSPECTION OF ELECTRICAL/DATA CONNECTIONS
0.25
TEST AND VERIFY ELECTRICAL/DATA SYSTEMS
0.12
REMOTE INSPECTION OF FLUID CONNECTIONS
0.25
TEST AND VERIFY FLUID SYSTEMS
0.12
READY SEP MAIN TRUSS FOR NEXT ASSEMBLY ACTIVITY
0.06
RTMS TO STAND BY MODE

SEP ASSEMBLY MISSION FIVE

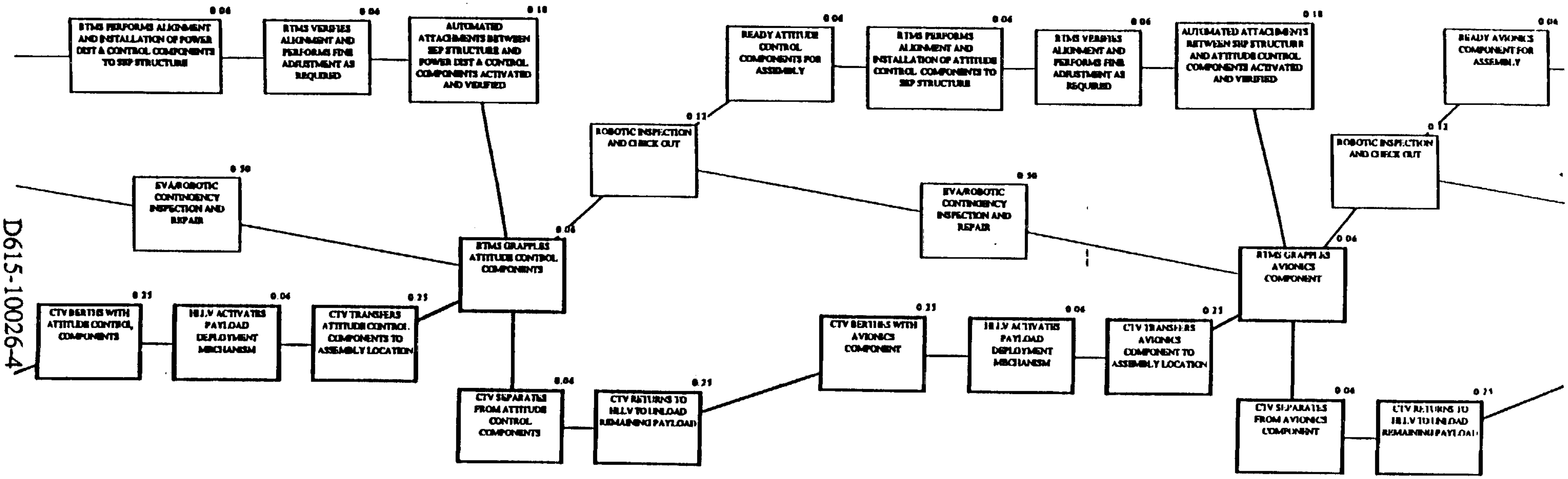


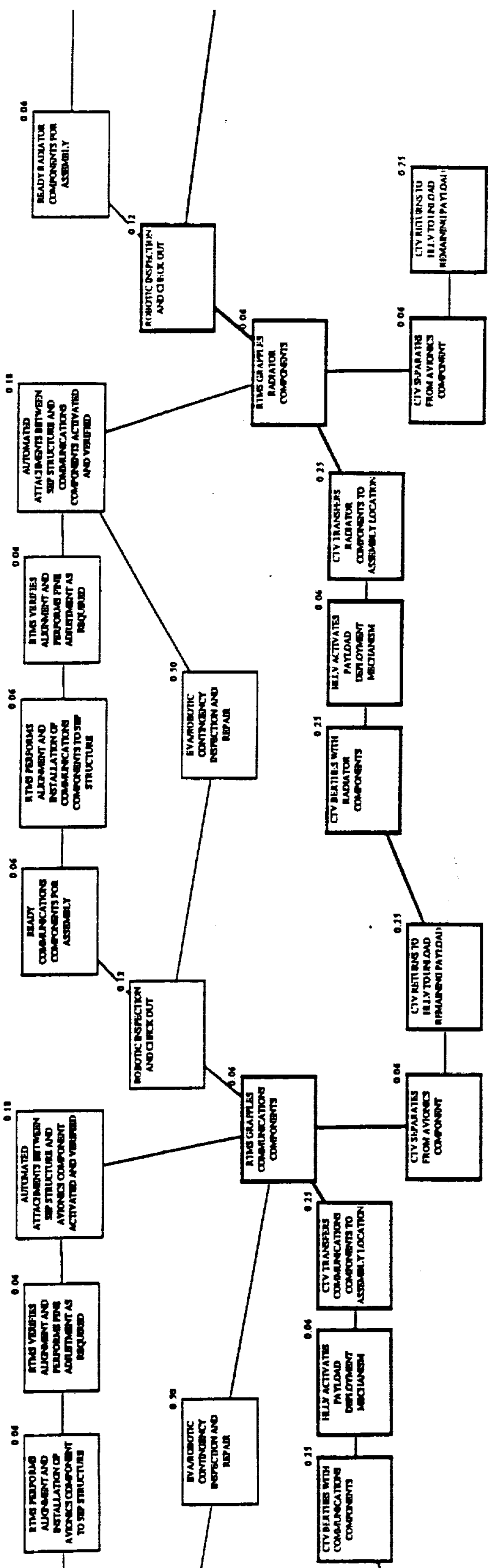


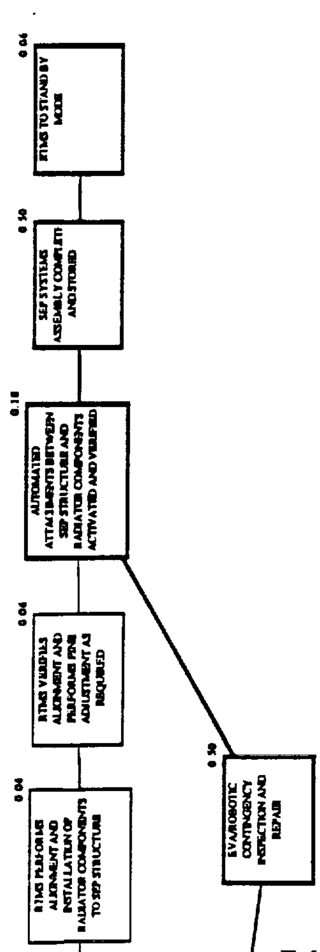




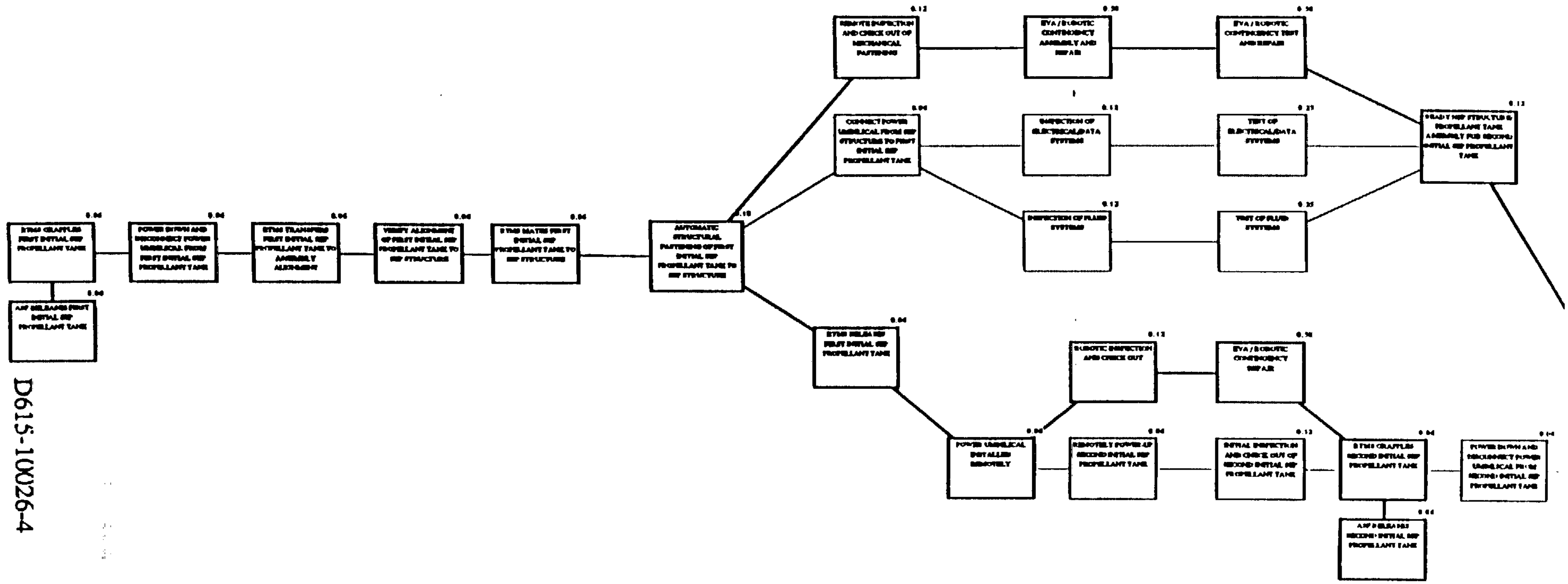
CTV BERTHS WITH POWER PROCESSING UNITS
0.25
UNSTOW AND POWER UP RTMS
0.04
VERIFY RTMS FUNCTIONING ABILITY
0.12
HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM
0.04
CTV TRANSFERS POWER PROCESSING UNITS TO ASSEMBLY LOCATION
0.25
RTMS GRAPPLES POWER PROCESSING UNITS
0.04
CTV SEPARATES FROM POWER PROCESSING UNITS
0.04
CTV RETURNS TO HLLV TO UNLOAD REMAINING PAYLOAD
0.25
ROBOTIC INSPECTION AND CHECK OUT
0.12
READY POWER PROCESSING UNITS FOR ASSEMBLY
0.04
RTMS PERFORMS ALIGNMENT AND INSTALLATION OF POWER PROCESSING UNITS TO SEP STRUCTURE
0.04
RTMS VERIFIES ALIGNMENT AND PERFORMS FINE ADJUSTMENT AS REQUIRED
0.04
AUTOMATED ATTACHMENTS BETWEEN SEP STRUCTURE AND POWER PROCESSING UNITS ACTIVATED AND VERIFIED
0.10
EVA/ROBOTIC CONTINGENCY INSPECTION AND REPAIR
0.50
CTV BERTHS WITH POWER DIST & CONTROL COMPONENTS
0.25
HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM
0.04
CTV TRANSFERS POWER DIST & CONTROL COMPONENTS TO ASSEMBLY LOCATION
0.25
RTMS GRAPPLES POWER DIST & CONTROL COMPONENTS
0.04
CTV SEPARATES FROM POWER DIST & CONTROL COMPONENTS
0.04
CTV RETURNS TO HLLV TO UNLOAD REMAINING PAYLOAD
0.25
ROBOTIC INSPECTION AND CHECK OUT
0.12
READY POWER DIST & CONTROL COMPONENTS FOR ASSEMBLY
0.04

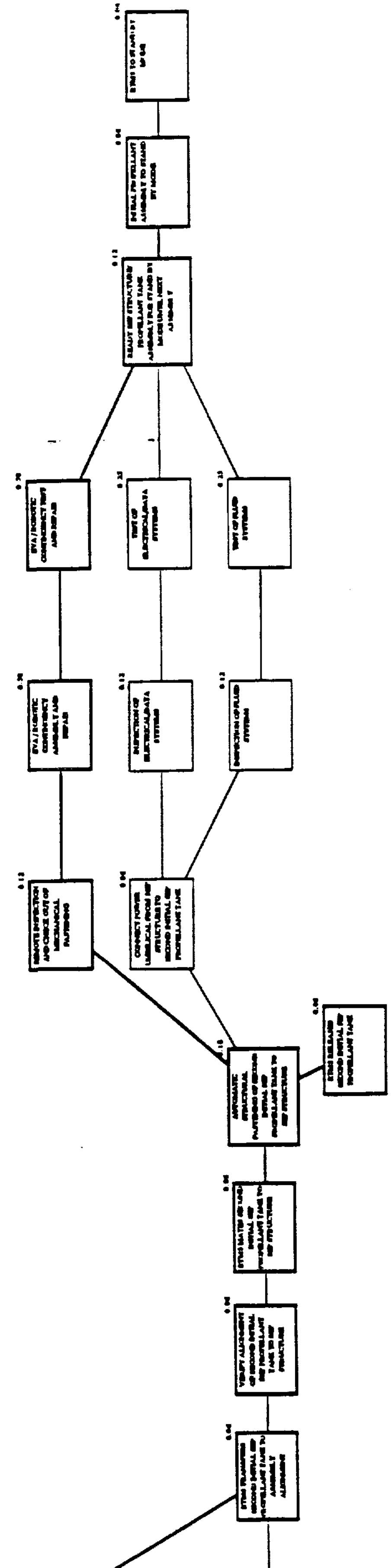

0.06
RTMS PERFORMS ALIGNMENT AND INSTALLATION OF POWER DIST & CONTROL COMPONENTS TO SEP STRUCTURE
0.06
RTMS VERIFIES ALIGNMENT AND PERFORMS FINE ADJUSTMENT AS REQUIRED
0.18
AUTOMATED ATTACHMENTS BETWEEN SEP STRUCTURE AND POWER DIST & CONTROL COMPONENTS ACTIVATED AND VERIFIED
0.50
EVA/ROBOTIC CONTINGENCY INSPECTION AND REPAIR
0.06
RTMS GRAPPLES ATTITUDE CONTROL COMPONENTS
0.25
CTV BERTHS WITH ATTITUDE CONTROL COMPONENTS
0.04
HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM
0.25
CTV TRANSFERS ATTITUDE CONTROL COMPONENTS TO ASSEMBLY LOCATION
0.04
CTV SEPARATES FROM ATTITUDE CONTROL COMPONENTS
0.25
CTV RETURNS TO HLLV TO UNLOAD REMAINING PAYLOAD
0.12
ROBOTIC INSPECTION AND CHECK OUT
0.04
READY ATTITUDE CONTROL COMPONENTS FOR ASSEMBLY
0.04
RTMS PERFORMS ALIGNMENT AND INSTALLATION OF ATTITUDE CONTROL COMPONENTS TO SEP STRUCTURE
0.06
RTMS VERIFIES ALIGNMENT AND PERFORMS FINE ADJUSTMENT AS REQUIRED
0.18
AUTOMATED ATTACHMENTS BETWEEN SEP STRUCTURE AND ATTITUDE CONTROL COMPONENTS ACTIVATED AND VERIFIED
0.50
EVA/ROBOTIC CONTINGENCY INSPECTION AND REPAIR
0.25
CTV BERTHS WITH AVIONICS COMPONENT
0.04
HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM
0.25
CTV TRANSFERS AVIONICS COMPONENT TO ASSEMBLY LOCATION
0.06
RTMS GRAPPLES AVIONICS COMPONENT
0.12
ROBOTIC INSPECTION AND CHECK OUT
0.04
READY AVIONICS COMPONENT FOR ASSEMBLY
0.04
CTV SEPARATES FROM AVIONICS COMPONENT
0.25
CTV RETURNS TO HLLV TO UNLOAD REMAINING PAYLOAD


READY RADIATOR COMPONENTS FOR ASSEMBLY
0.06
ROBOTIC INSPECTION AND CHECK OUT
0.12
RTMS GRAPPLES RADIATOR COMPONENTS
0.06
CTV SEPARATES FROM AVIONICS COMPONENT
0.06
CTV RETURNS TO HLLV TO UNLOAD REMAINING PAYLOAD
0.25
AUTOMATED ATTACHMENTS BETWEEN SEP STRUCTURE AND COMMUNICATIONS COMPONENTS ACTIVATED AND VERIFIED
0.18
CTV TRANSFERS RADIATOR COMPONENTS TO ASSEMBLY LOCATION
0.25
RTMS VERIFIES ALIGNMENT AND PERFORMS FINE ADJUSTMENT AS REQUIRED
0.06
HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM
0.06
EVA/ROBOTIC CONTINGENCY INSPECTION AND REPAIR
0.50
RTMS PERFORMS ALIGNMENT AND INSTALLATION OF COMMUNICATIONS COMPONENTS TO SEP STRUCTURE
0.06
CTV BERTHES WITH RADIATOR COMPONENTS
0.25
READY COMMUNICATIONS COMPONENTS FOR ASSEMBLY
0.06
CTV RETURNS TO HLLV TO UNLOAD REMAINING PAYLOAD
0.25
ROBOTIC INSPECTION AND CHECK OUT
0.12
RTMS GRAPPLES COMMUNICATIONS COMPONENTS
0.06
CTV SEPARATES FROM AVIONICS COMPONENT
0.06
AUTOMATED ATTACHMENTS BETWEEN SEP STRUCTURE AND AVIONICS COMPONENT ACTIVATED AND VERIFIED
0.18
CTV TRANSFERS COMMUNICATIONS COMPONENTS TO ASSEMBLY LOCATION
0.25
RTMS VERIFIES ALIGNMENT AND PERFORMS FINE ADJUSTMENT AS REQUIRED
0.06
HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM
0.06
EVA/ROBOTIC CONTINGENCY INSPECTION AND REPAIR
0.50
RTMS PERFORMS ALIGNMENT AND INSTALLATION OF AVIONICS COMPONENT TO SEP STRUCTURE
0.06
CTV BERTHES WITH COMMUNICATIONS COMPONENTS
0.25

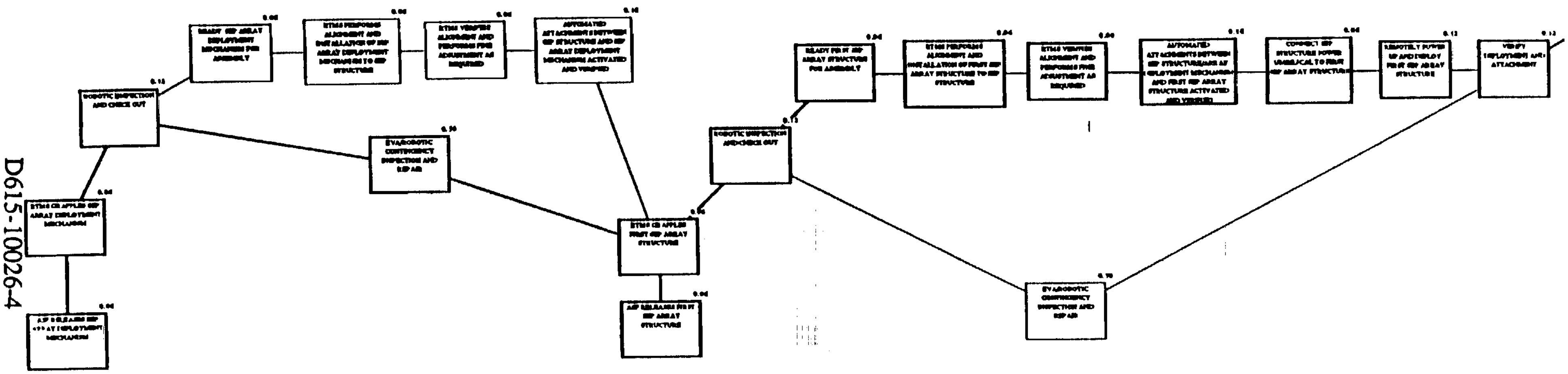

0.04
RTMS PERFORMS ALIGNMENT AND INSTALLATION OF RADIATOR COMPONENTS TO SEP STRUCTURE
0.04
RTMS VERIFIES ALIGNMENT AND PERFORMS FINE ADJUSTMENT AS REQUIRED
0.18
AUTOMATED ATTACHMENTS BETWEEN SEP STRUCTURE AND RADIATOR COMPONENTS ACTIVATED AND VERIFIED
0.50
SEP SYSTEMS ASSEMBLY COMPLETE AND STORED
0.04
RTMS TO STAND BY MODE
0.50
EVA/ROBOTIC CONTINGENCY INSPECTION AND REPAIR

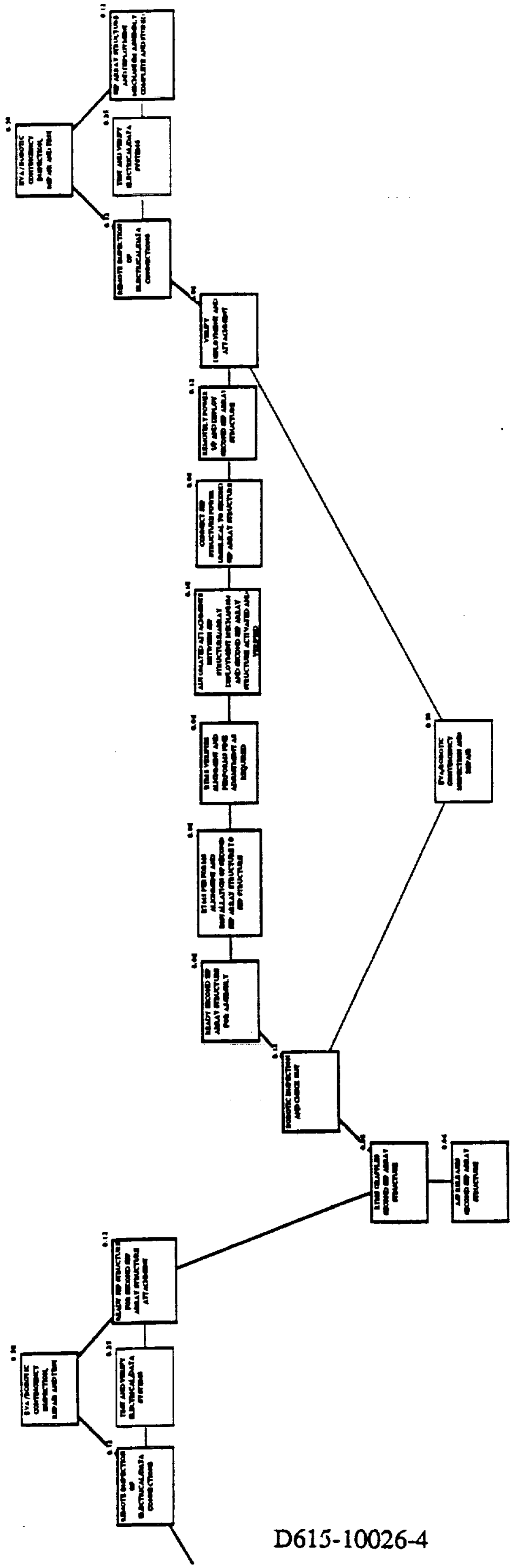
INITIAL SEP PROPELLANT AND TANKS ASSEMBLY



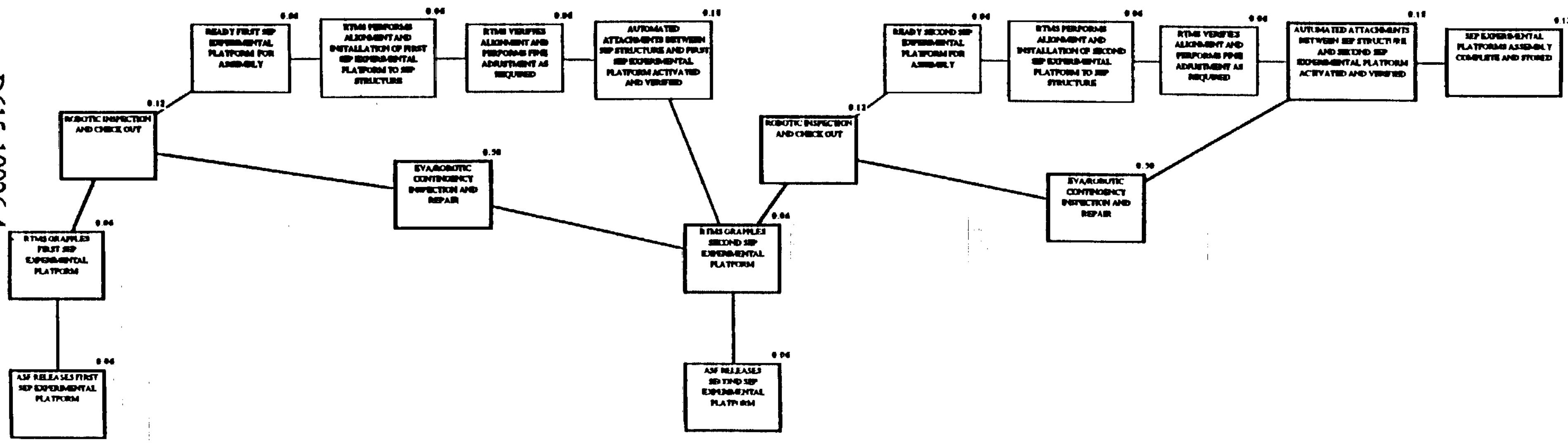


SEP ARRAY STRUCTURE AND DEPLOYMENT MECHANISM ASSEMBLY

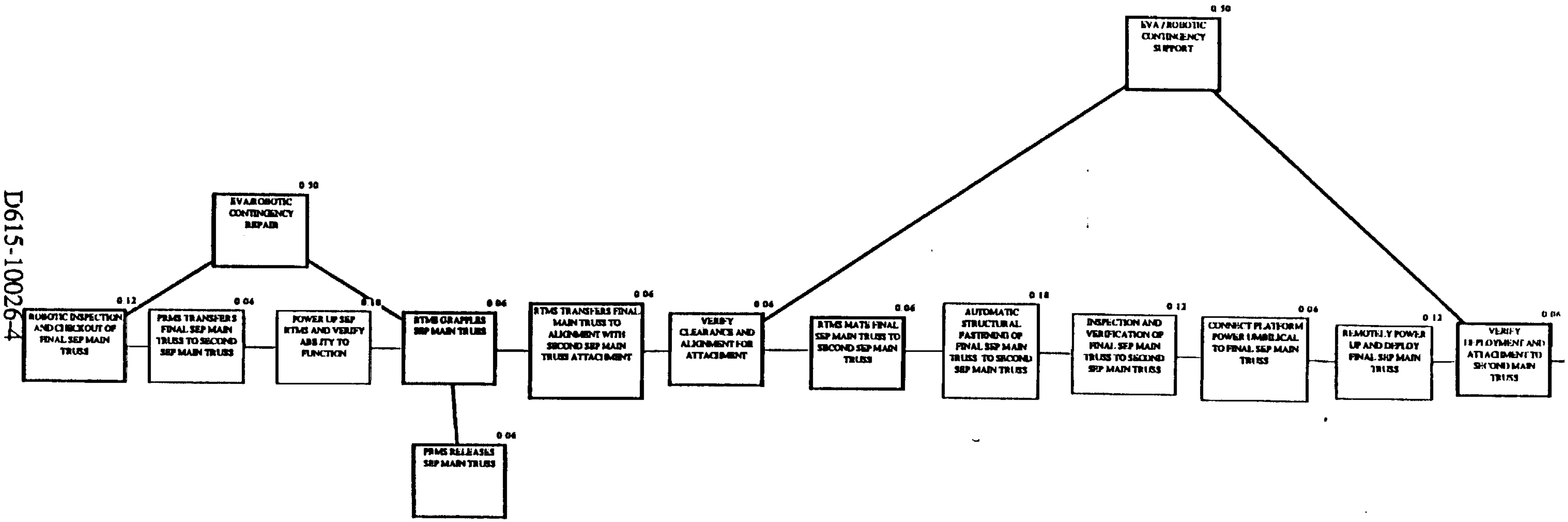


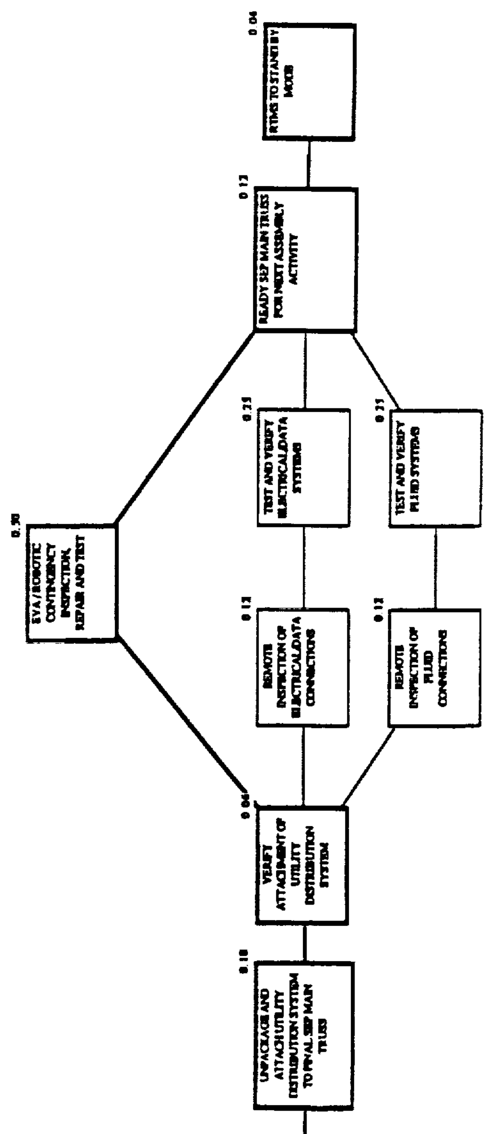


SEP EXPERIMENTAL PLATFORMS ASSEMBLY



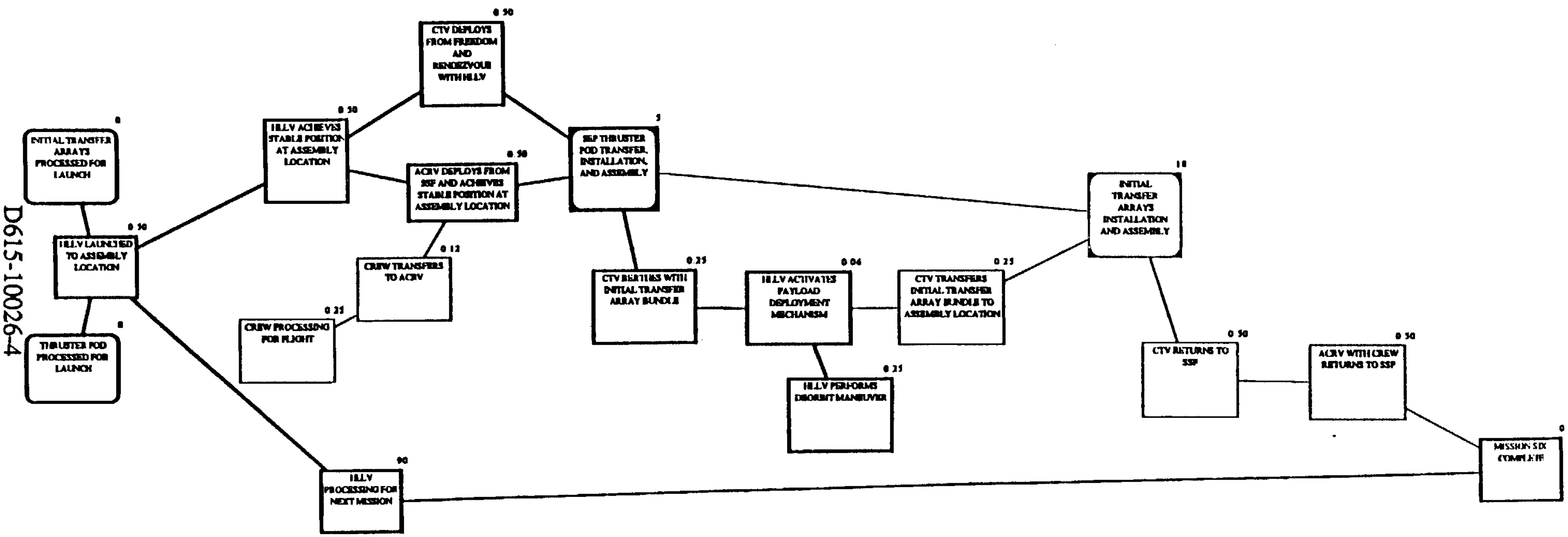
SEP FINAL MAIN TRUSS ASSEMBLY



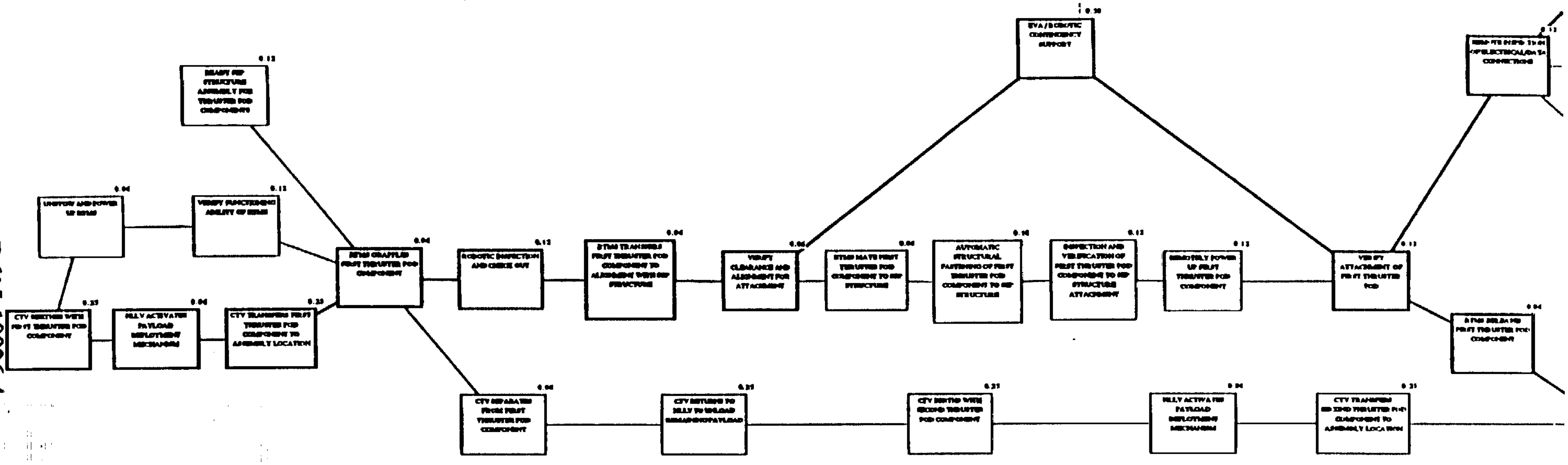


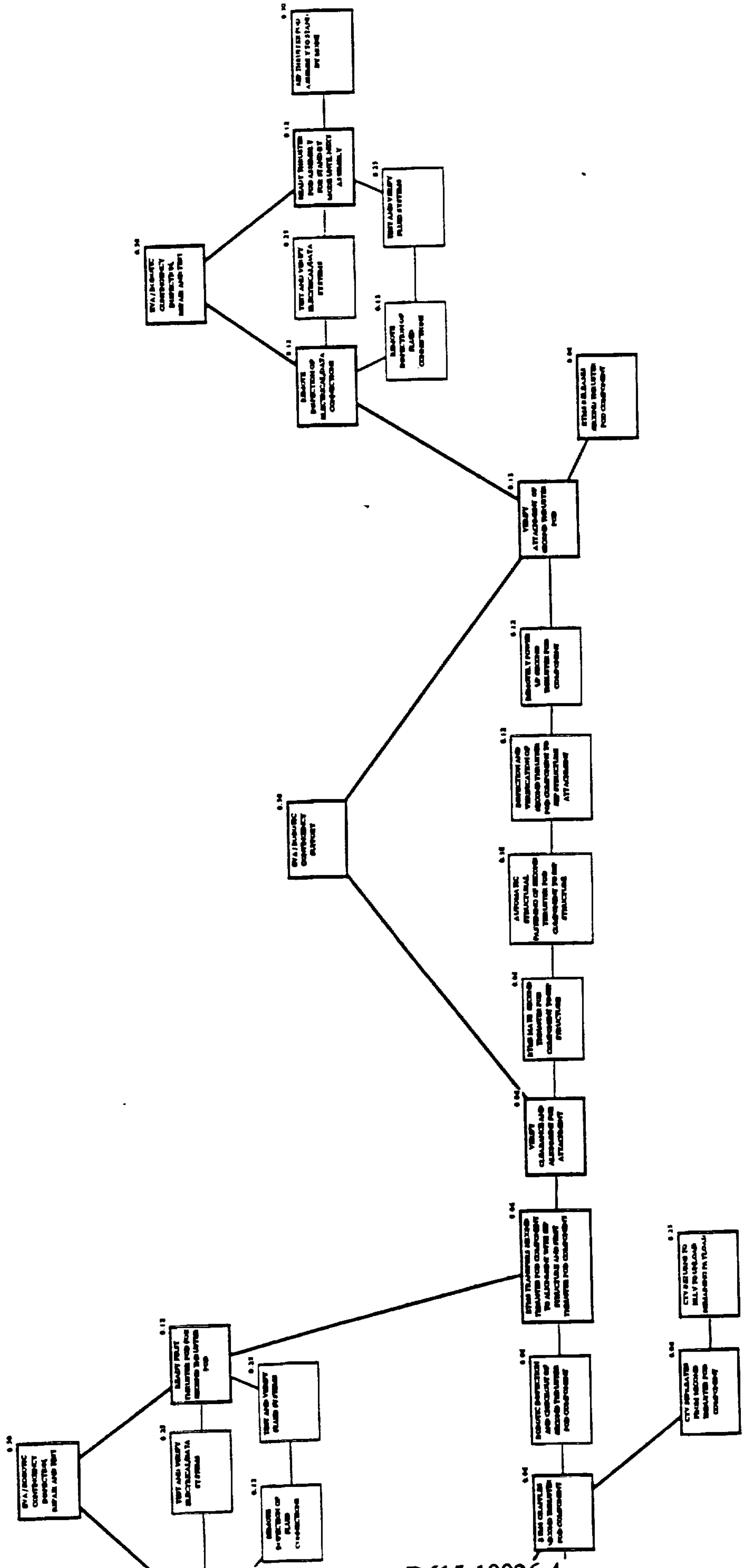
0.18
UNPACKAGE AND ATTACH UTILITY DISTRIBUTION SYSTEM TO FINAL SEP MAIN TRUSS
0.04
VERIFY ATTACHMENT OF UTILITY DISTRIBUTION SYSTEM
0.50
EVA / ROBOTIC CONTINGENCY INSPECTION, REPAIR AND TEST
0.12
REMOTE INSPECTION OF ELECTRICAL/DATA CONNECTIONS
0.25
TEST AND VERIFY ELECTRICAL/DATA SYSTEMS
0.12
REMOTE INSPECTION OF FLUID CONNECTIONS
0.25
TEST AND VERIFY FLUID SYSTEMS
0.12
READY SEP MAIN TRUSS FOR NEXT ASSEMBLY ACTIVITY
0.04
RTMS TO STANDBY MODE

SEP ASSEMBLY MISSION SIX

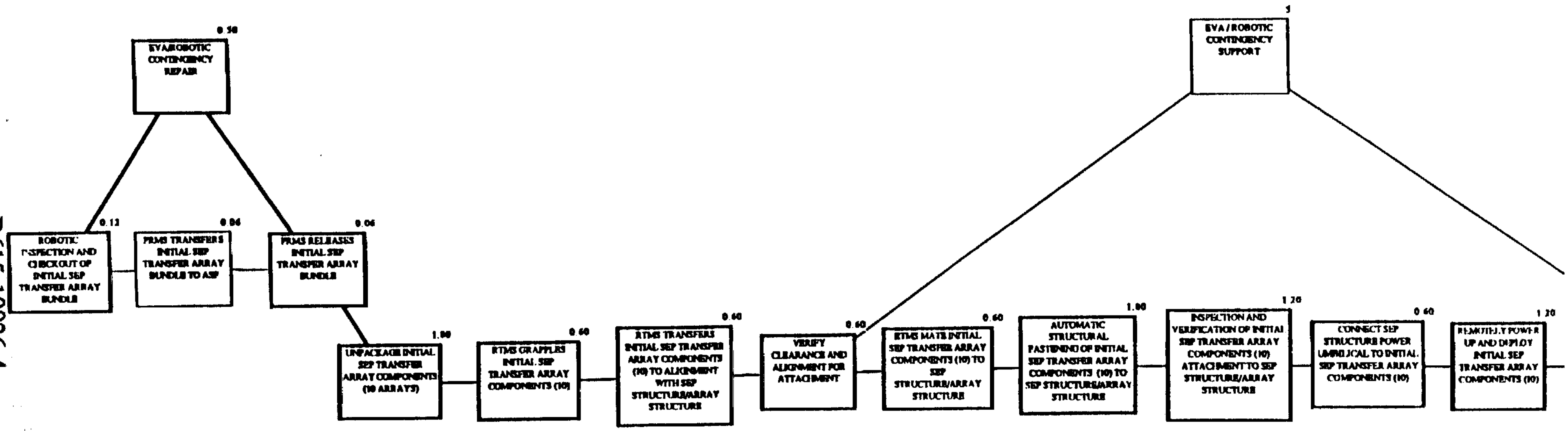


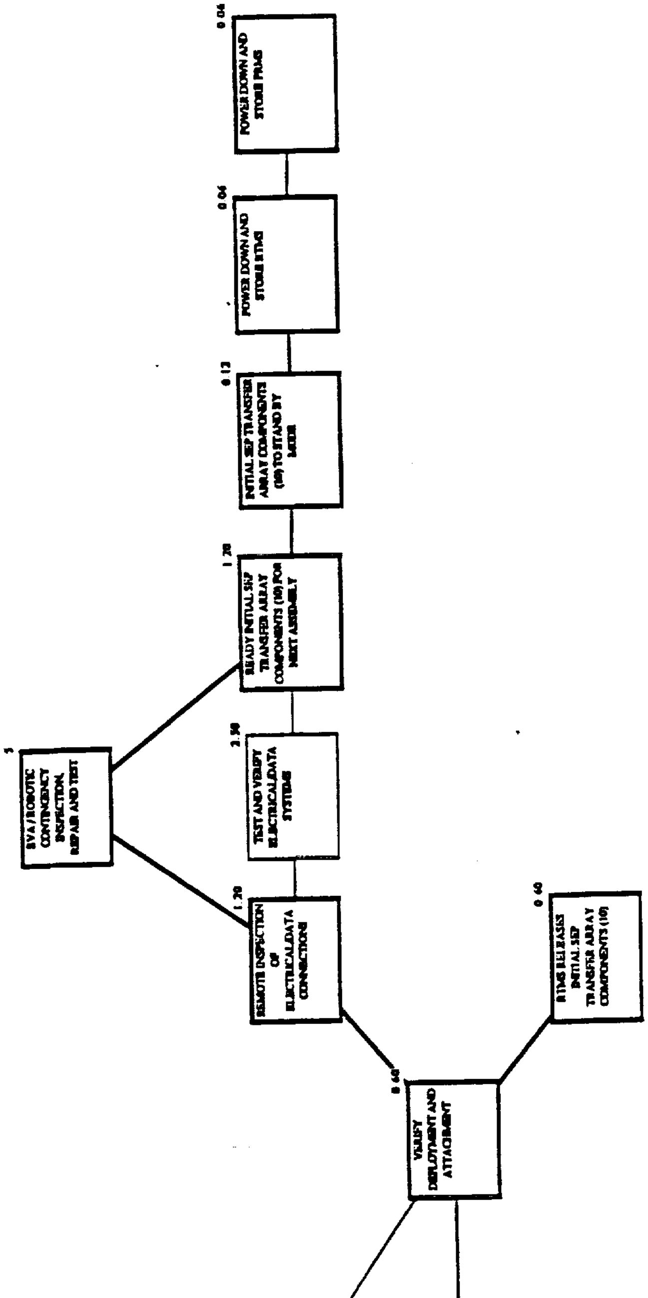
SEP THRUSTER POD ASSEMBLY





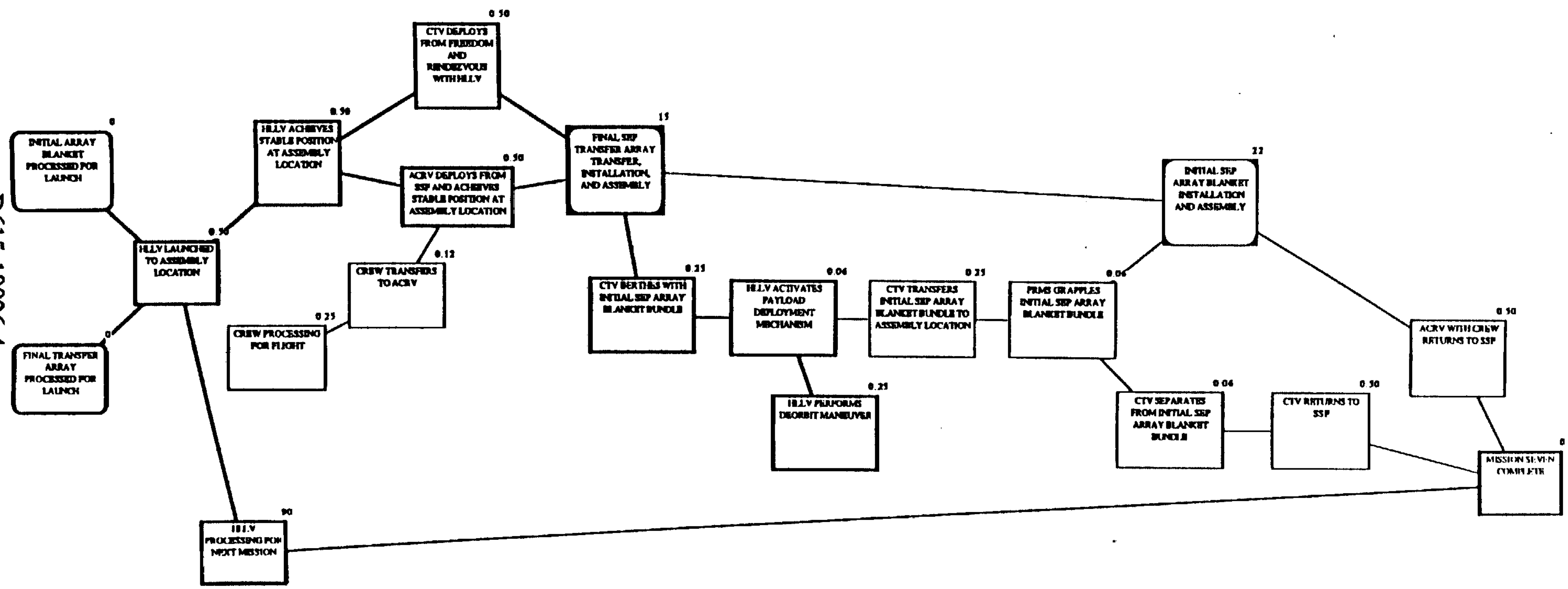
INITIAL SEP TRANSFER ARRAY ASSEMBLY



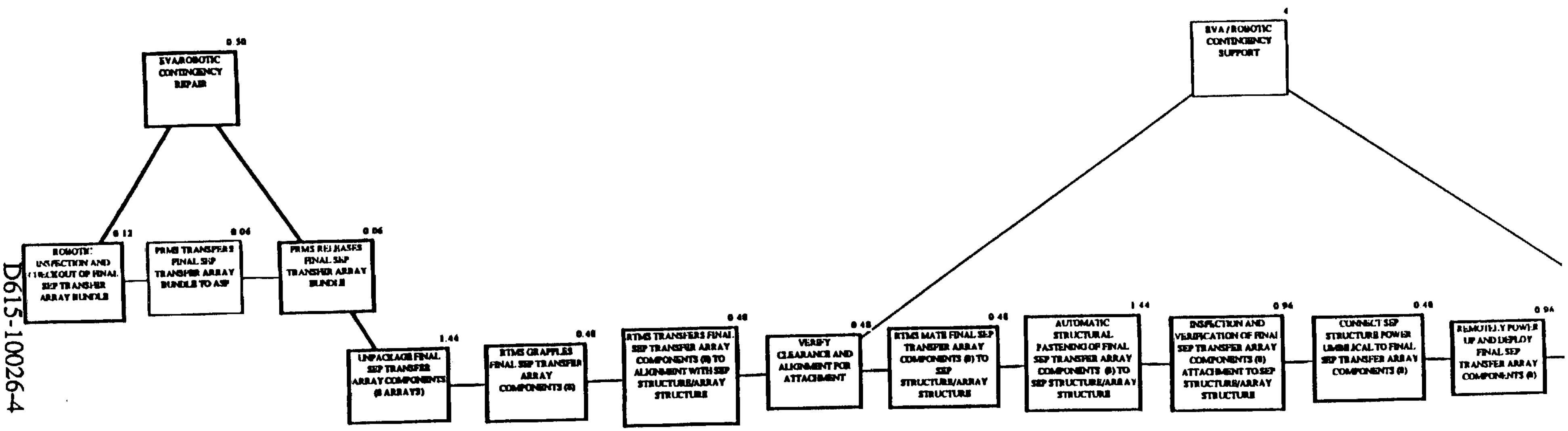


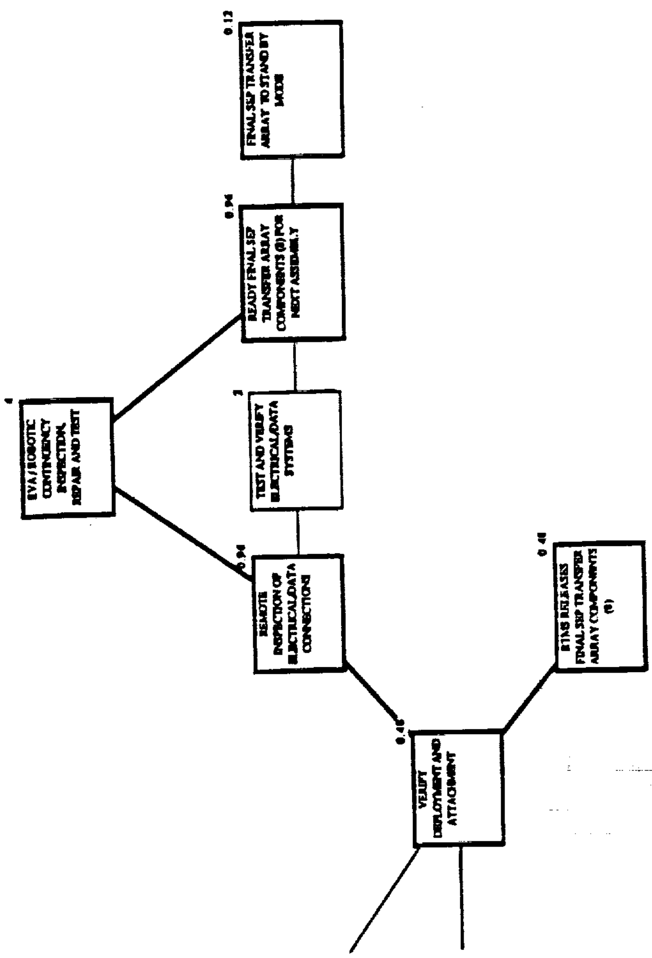
VERIFY DEPLOYMENT AND ATTACHMENT
RTMS RELEASES INITIAL SEP TRANSFER ARRAY COMPONENTS (10)
1.20
REMOTE INSPECTION OF ELECTRICAL/DATA CONNECTIONS
5
EVA / ROBOTIC CONTINGENCY INSPECTION, REPAIR AND TEST
2.50
TEST AND VERIFY ELECTRICAL/DATA SYSTEMS
1.20
READY INITIAL SEP TRANSFER ARRAY COMPONENTS (10) FOR NEXT ASSEMBLY
0.12
INITIAL SEP TRANSFER ARRAY COMPONENTS (10) TO STAND BY MODE
0.06
POWER DOWN AND STORE RTMS
0.04
POWER DOWN AND STORE PRMS

SEP ASSEMBLY MISSION SEVEN

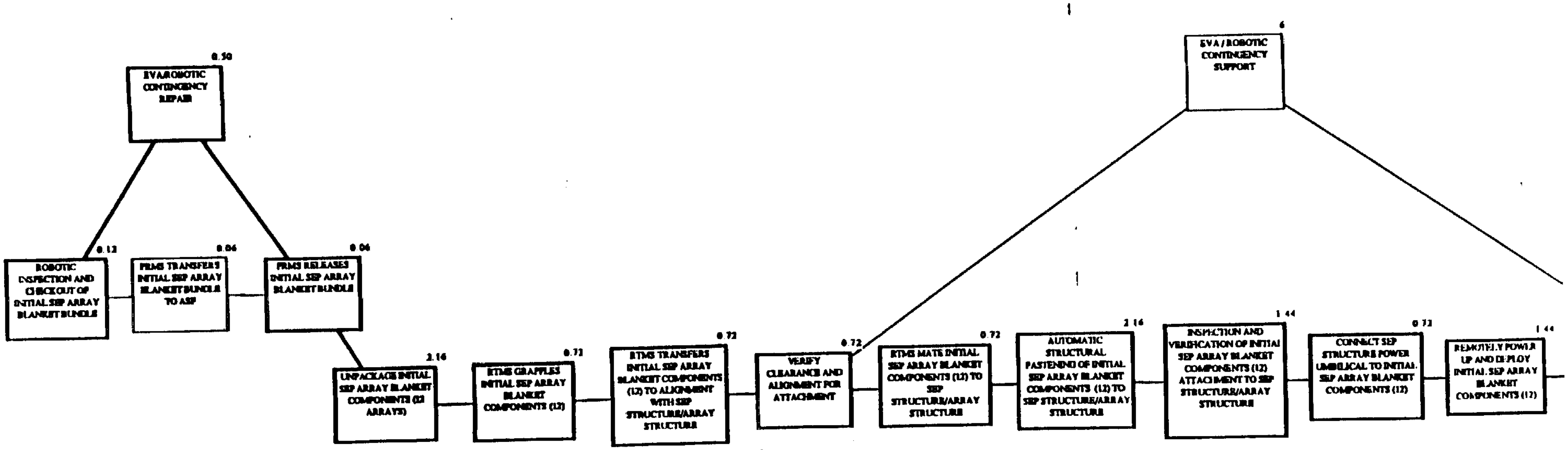


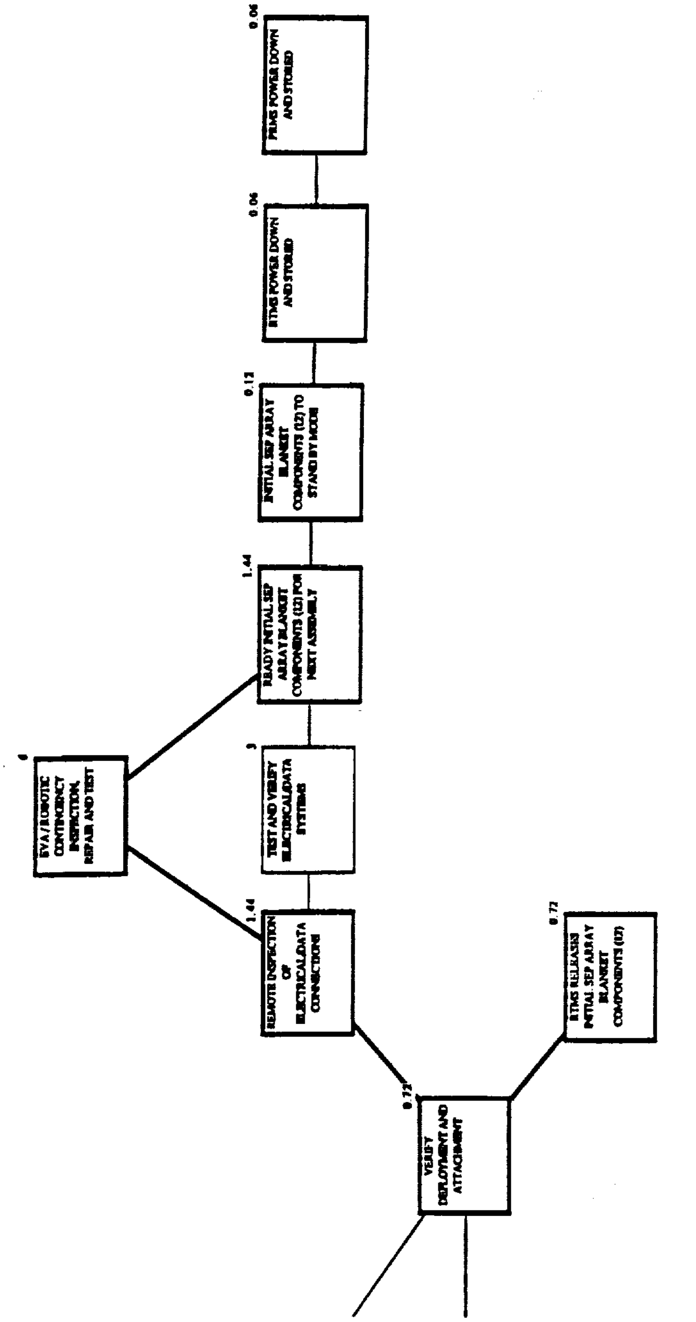
FINAL SEP TRANSFER ARRAY ASSEMBLY



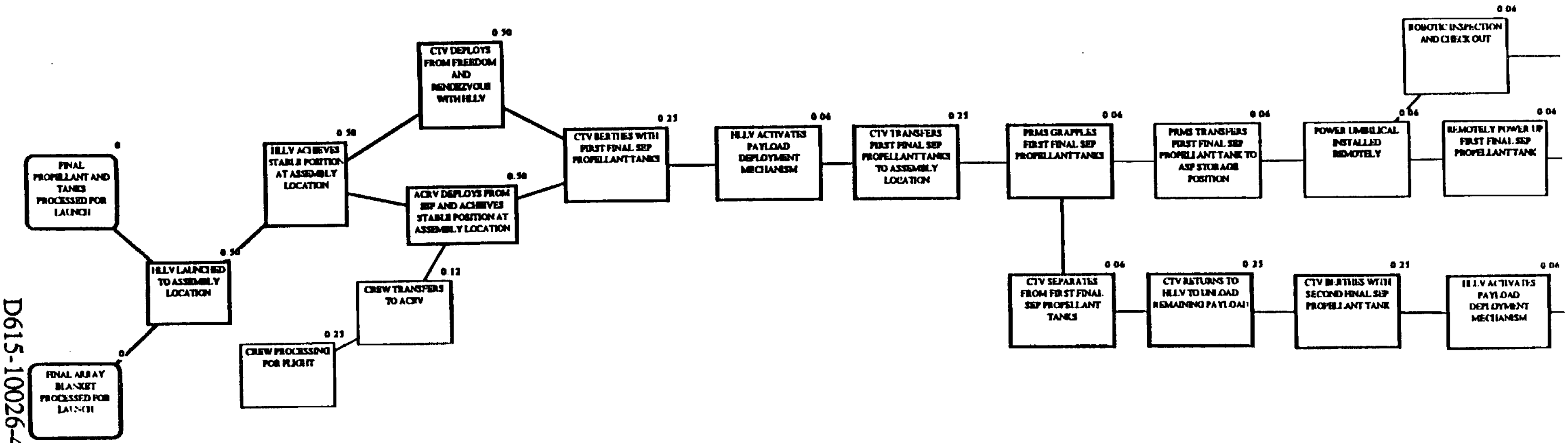

EVA / ROBOTIC CONTINGENCY INSPECTION, REPAIR AND TEST
4
REMOTE INSPECTION OF ELECTRICAL/DATA CONNECTIONS
0.94
TEST AND VERIFY ELECTRICAL/DATA SYSTEMS
2
READY FINAL SEP TRANSFER ARRAY COMPONENTS (8) FOR NEXT ASSEMBLY
0.94
FINAL SEP TRANSFER ARRAY TO STAND BY MODE
0.12
VERIFY DEPLOYMENT AND ATTACHMENT
0.48
RTMS RELEASES FINAL SEP TRANSFER ARRAY COMPONENTS (8)
0.48

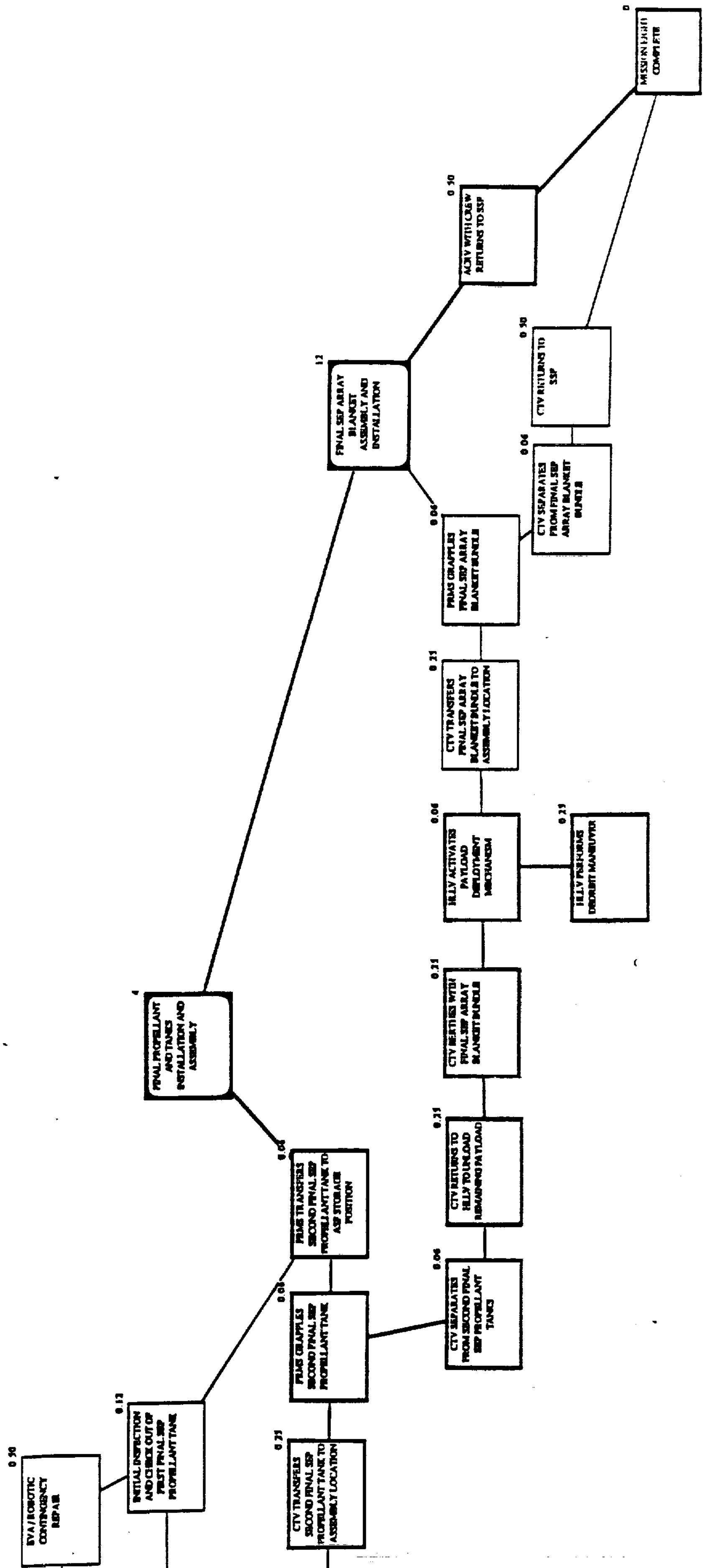
INITIAL SEP ARRAY BLANKET ASSEMBLY




VERIFY DEPLOYMENT AND ATTACHMENT
0.72
RTMS RELEASES INITIAL SEP ARRAY BLANKET COMPONENTS (12)
0.72
REMOTE INSPECTION OF ELECTRICAL/DATA CONNECTIONS
1.44
EVA / ROBOTIC CONTINGENCY INSPECTION, REPAIR AND TEST
TEST AND VERIFY ELECTRICAL/DATA SYSTEMS
3
READY INITIAL SEP ARRAY BLANKET COMPONENTS (12) FOR NEXT ASSEMBLY
1.44
INITIAL SEP ARRAY BLANKET COMPONENTS (12) TO STAND BY MODE
0.12
RTMS POWER DOWN AND STORED
0.06
PRMS POWER DOWN AND STORED
0.06

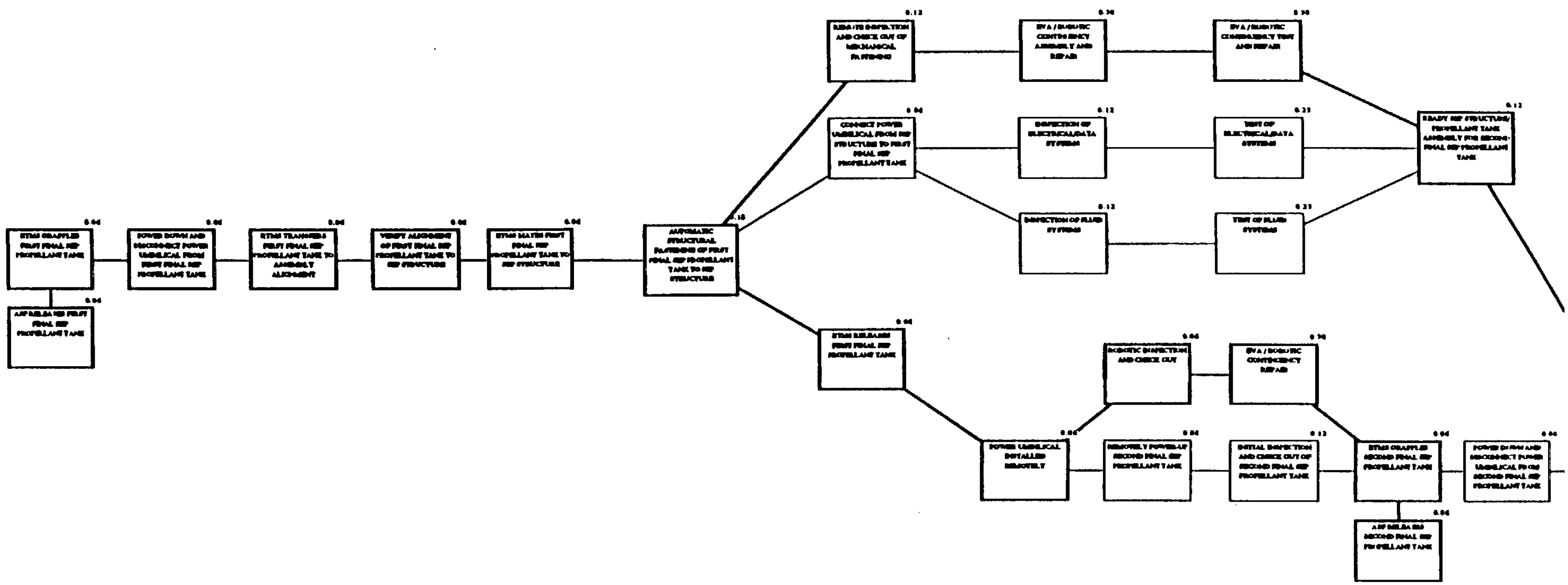
SEP ASSEMBLY MISSION EIGHT

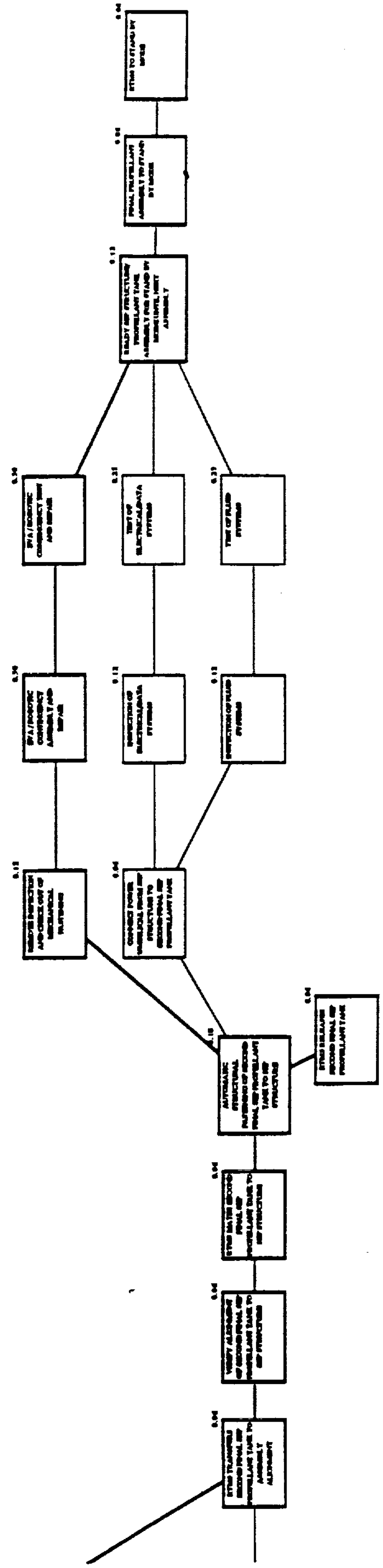




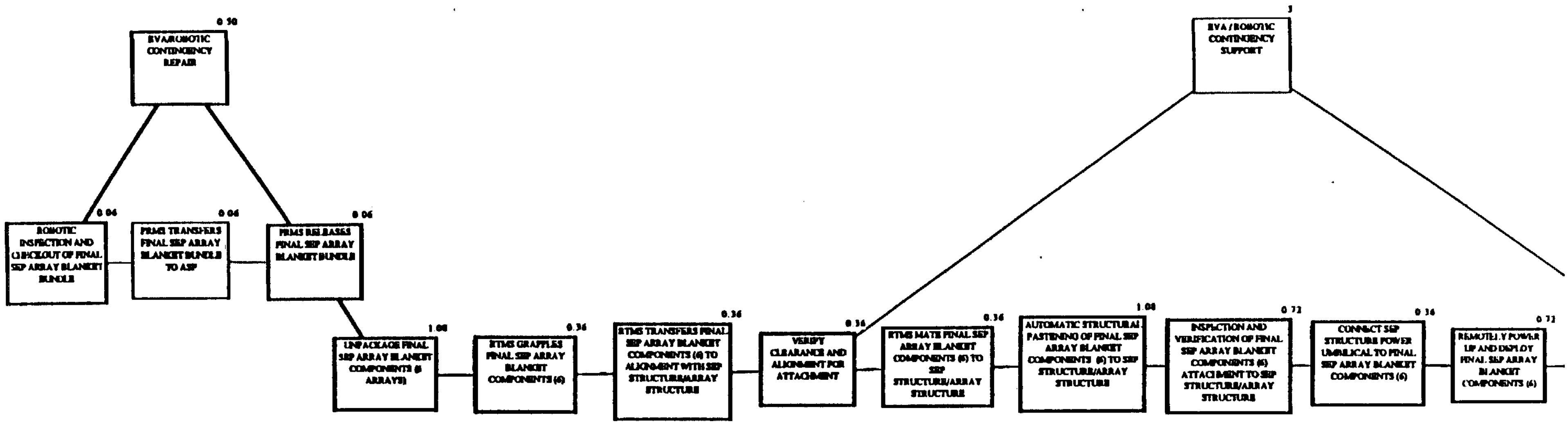
0.50
EVA / ROBOTIC CONTINGENCY REPAIR
0.12
INITIAL INSPECTION AND CHECK OUT OF FIRST FINAL SEP PROPELLANT TANK
0.25
CTV TRANSFERS SECOND FINAL SEP PROPELLANT TANK TO ASSEMBLY LOCATION
0.08
PRMS GRAPPLES SECOND FINAL SEP PROPELLANT TANK
0.04
PRMS TRANSFERS SECOND FINAL SEP PROPELLANT TANK TO ASP STORAGE POSITION
4
FINAL PROPELLANT AND TANKS INSTALLATION AND ASSEMBLY
0.06
CTV SEPARATES FROM SECOND FINAL SEP PROPELLANT TANKS
0.25
CTV RETURNS TO HLLV TO UNLOAD REMAINING PAYLOAD
0.25
CTV BERTHES WITH FINAL SEP ARRAY BLANKET BUNDLE
0.06
HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM
0.25
HLLV PERFORMS DEORBIT MANEUVER
0.25
CTV TRANSFERS FINAL SEP ARRAY BLANKET BUNDLE TO ASSEMBLY LOCATION
0.04
PRMS GRAPPLES FINAL SEP ARRAY BLANKET BUNDLE
0.06
CTV SEPARATES FROM FINAL SEP ARRAY BLANKET BUNDLE
0.50
CTV RETURNS TO SSP
12
FINAL SEP ARRAY BLANKET ASSEMBLY AND INSTALLATION
0.50
ACRV WITH CREW RETURNS TO SSP
0
MISSION EIGHT COMPLETE

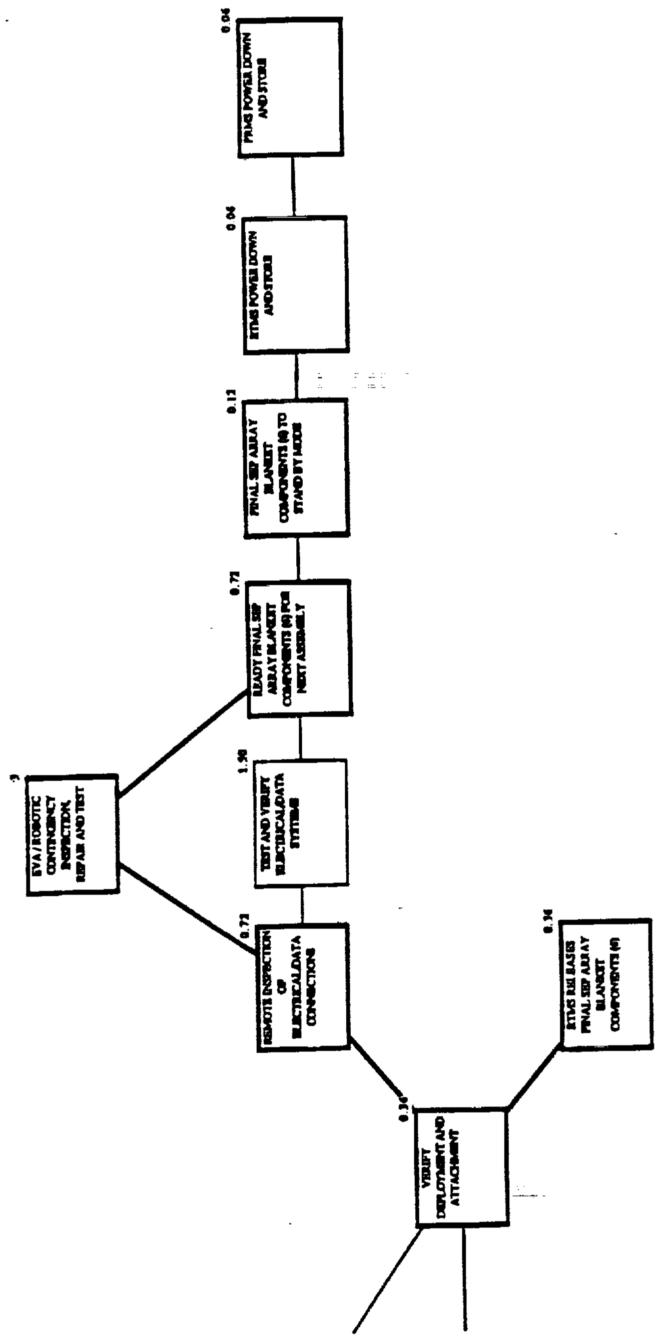
FINAL SEP PROPELLANT TANKS ASSEMBLY





FINAL SEP ARRAY BLANKET ASSEMBLY




VERIFY DEPLOYMENT AND ATTACHMENT
0.36
REMOTE INSPECTION OF ELECTRICAL/DATA CONNECTIONS
0.72
EVA / ROBOTIC CONTINGENCY INSPECTION, REPAIR AND TEST
3
TEST AND VERIFY ELECTRICAL/DATA SYSTEMS
1.50
READY FINAL SEP ARRAY BLANKET COMPONENTS (6) FOR NEXT ASSEMBLY
0.72
FINAL SEP ARRAY BLANKET COMPONENTS (6) TO STAND BY MODE
0.12
RTMS POWER DOWN AND STORE
0.04
PRMS POWER DOWN AND STORE
0.04
FINAL SEP ARRAY BLANKET COMPONENTS (6)
0.36

Ground

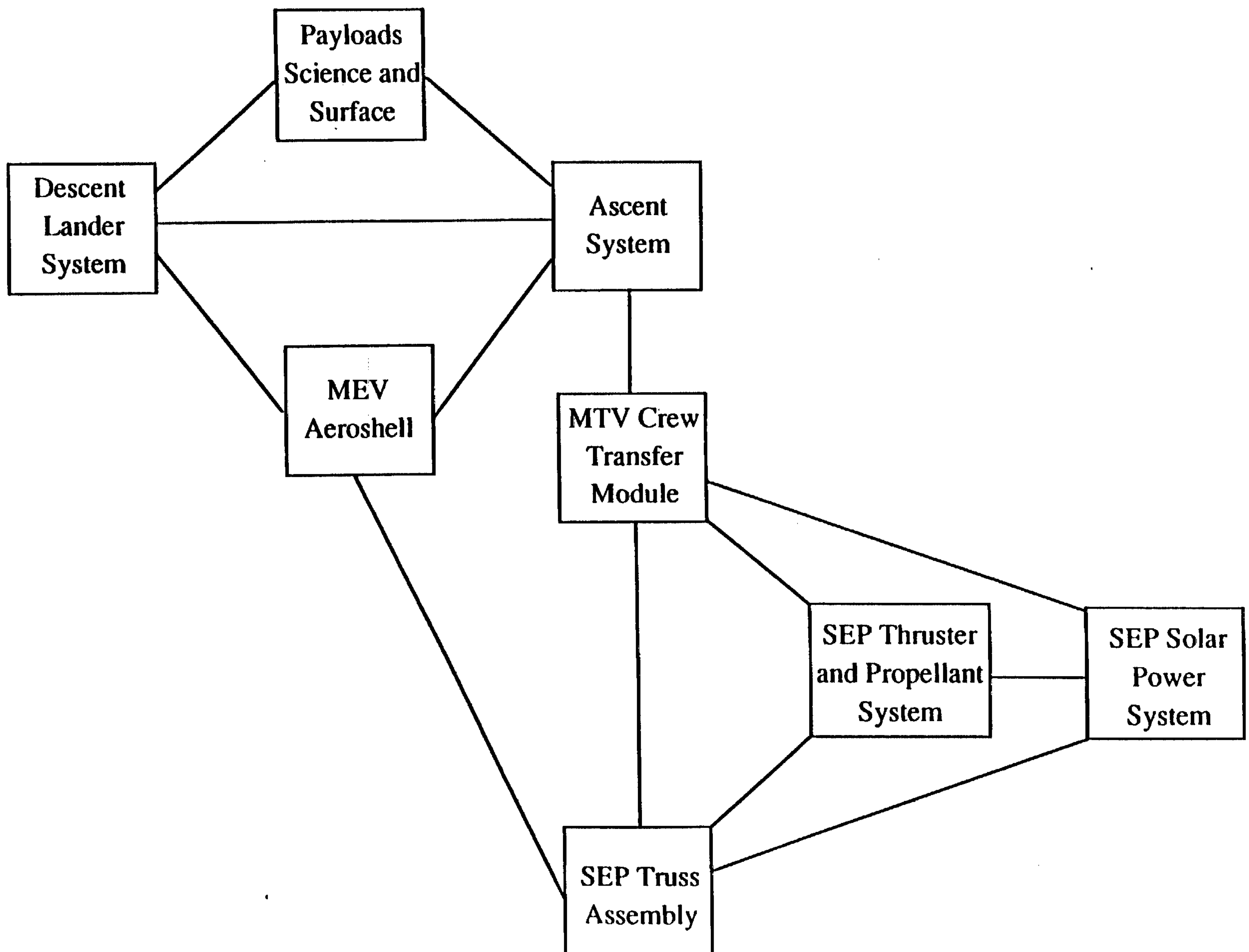
NTR, NEP, and SEP Assembly Flow Summary

- First mission of NTR assembly will require truss to be deployed and secured to dedicated assembly platform to lend additional stability to the platform during vehicle assembly
- The two TMI and two MOC tanks of NTR vehicle are brought up in staggered configuration starting with TMI tank #1 in the fifth HLLV flight. The reason for this is that the off-loaded propellant tankers that come up with the MOC tanks, (flights 6 and 8) will not have to be stored for a prolonged period of time
- The NTR in-line tank (or EOC tank) is integrated with the shield and engine along with associated structures; further the engine nozzle is mounted in reverse to improve packaging efficiency. Portion of reverse-mounted nozzle protrudes into HLLV nosecone space. Engine assembly to the NTR vehicle will first require properly assembling the nozzle to the engine.
- Assembly of NEP Heat Transport and Rejection Systems (Missions 5, 7, and 8) requires nearly the full 90 days between Assembly Flights:
    - Due mainly to number of pieces and connections
    - Increasing number of assembly robots and multi-tasking may reduce this some; however, since this is a serial task, it must be done in steps
    - It is expected that welding pipes, instead of fastening with clamps, may reduce required time (including necessary verification procedures)
- NEP configuration should include robotic access to aft end of vehicle (later configurations include truss for the length of the NEP)
- If ACRV can not accommodate crew assembly operations, some type of control station must exist at assembly site until MTV Hab arrives:
    - ET-based platform devised for Cryo/Aerobrake Vehicle included SSF Node and airlock
    - MTV Hab could come up first (using ground simulators for remainder of interface verification)
- Integrated Aeroshell launch would reduce flights and on-orbit assembly time
- HLLV payload may need to be unloaded in groups rather than individually to prevent violation of HLLV on-orbit stay time

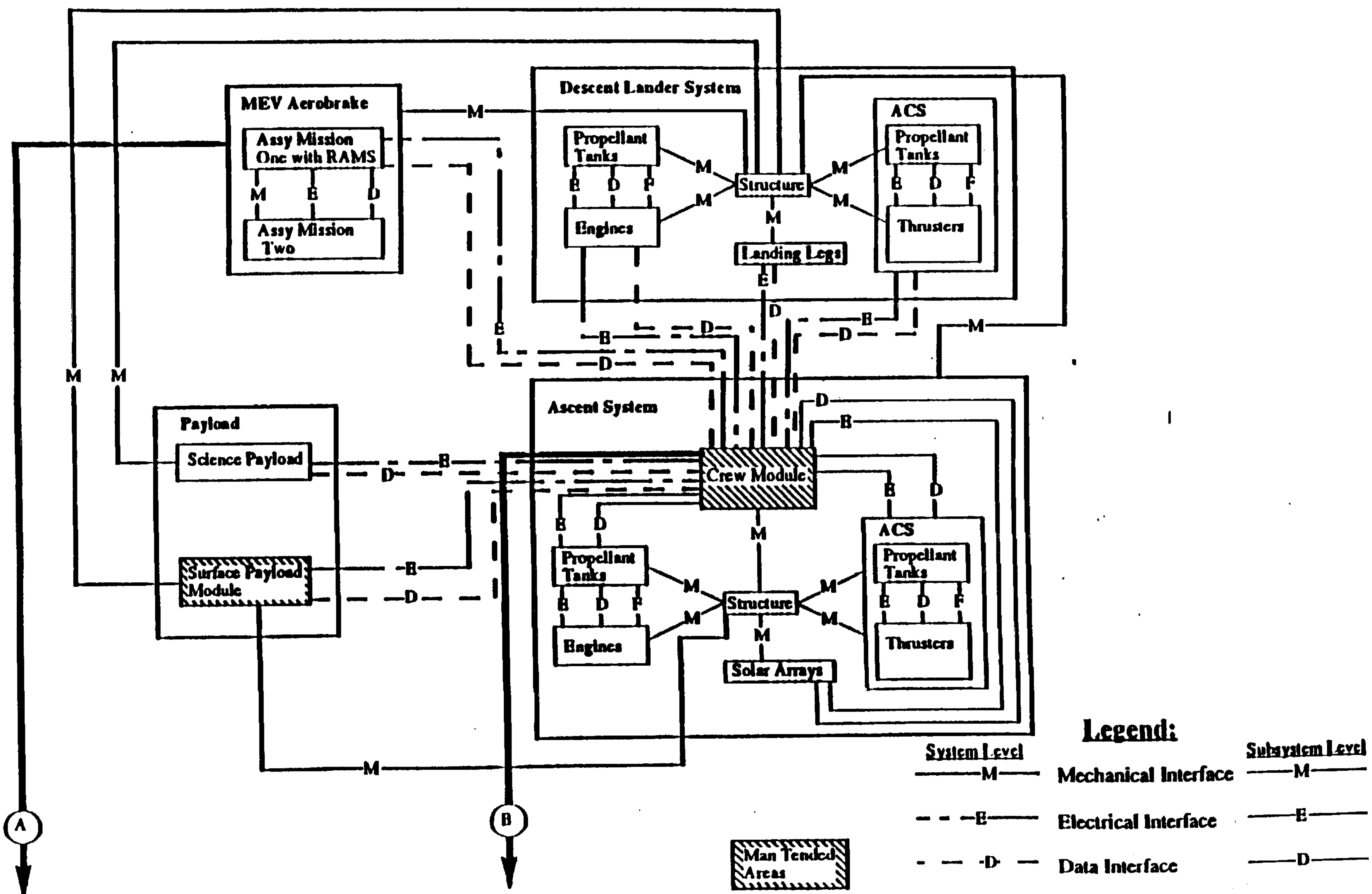
Ground Rules and Assumptions for Ground Processing

- A system is a group of components and supporting structure that is integrated by a contractor and delivered as a unit to the processing facility (e.g. MEV Aerobrake, MEV descent lander, ascent system, etc.).
- System interfaces are those which transmit data, power, or fluids across the system's boundaries and mechanically secure one system to another.
- Subsystems interfaces are those which are internal to a system.
- Subsystem interfaces are verified by the manufacturer prior to system integration.
- Component interfaces are those which are internal to a subsystem.
- Component interfaces are verified by the manufacturer during subsystem assembly.
- Interfaces verified prior to a system level integration will be accepted with no repetition of tests.
- Flight hardware will be used to verify system interfaces.
- Ground facilities will simulate assembly node operations and limitations.
- Certain non-mechanical interfaces to NTR, NEP, and SEP are simulated to allow desired launch sequencing.

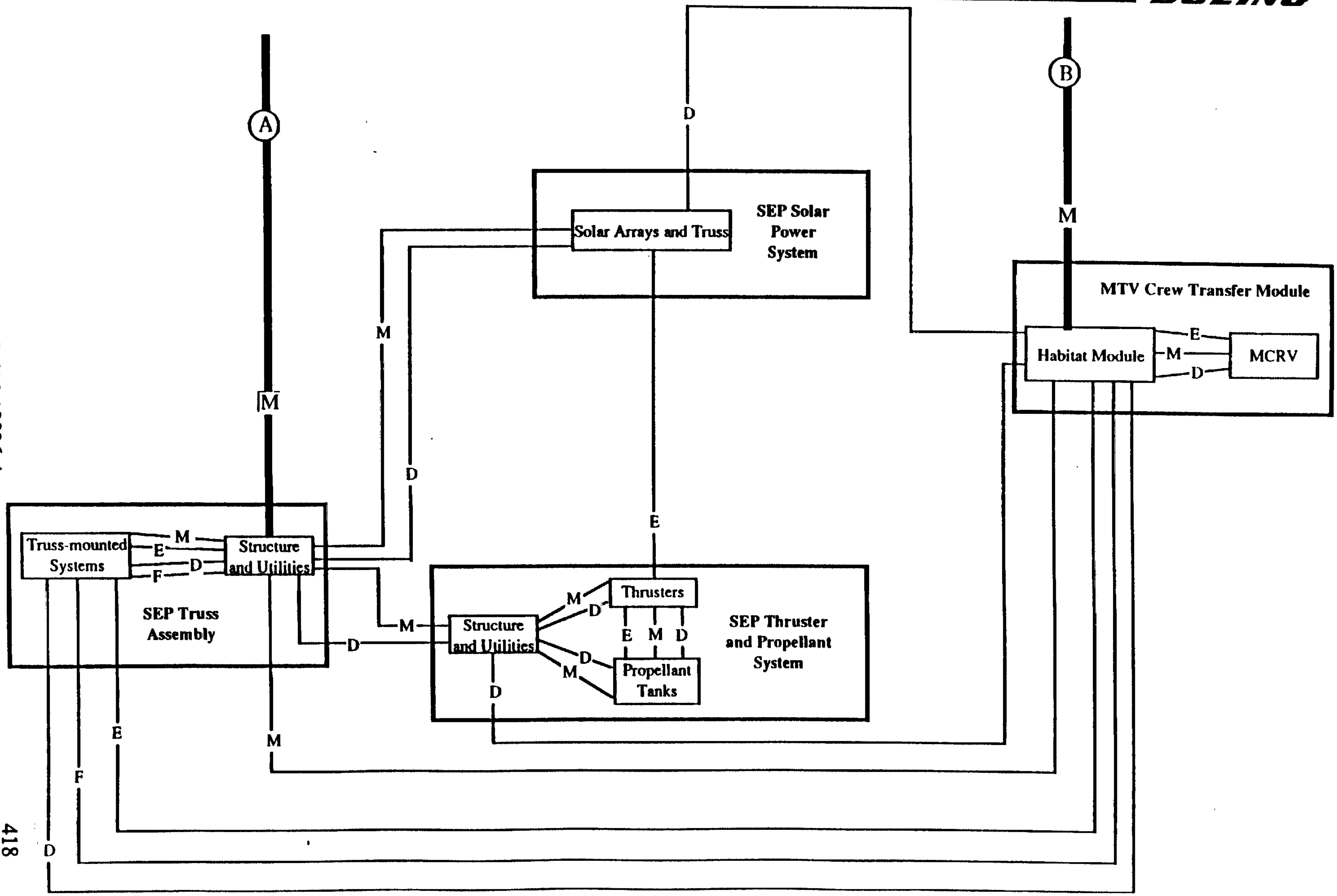
SEP Interfaces



SEP Interfaces (MEV)



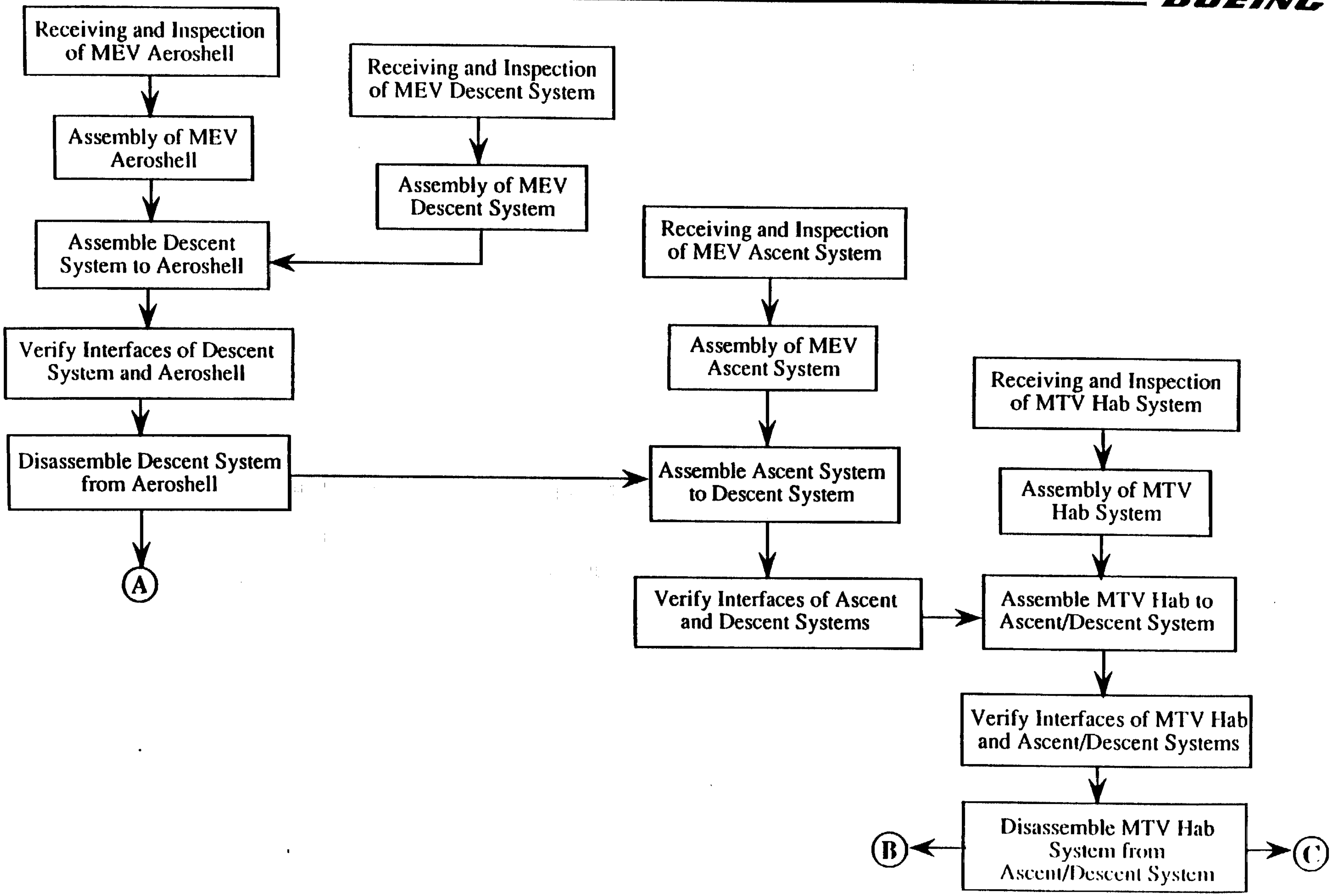
SEP Interfaces - continued



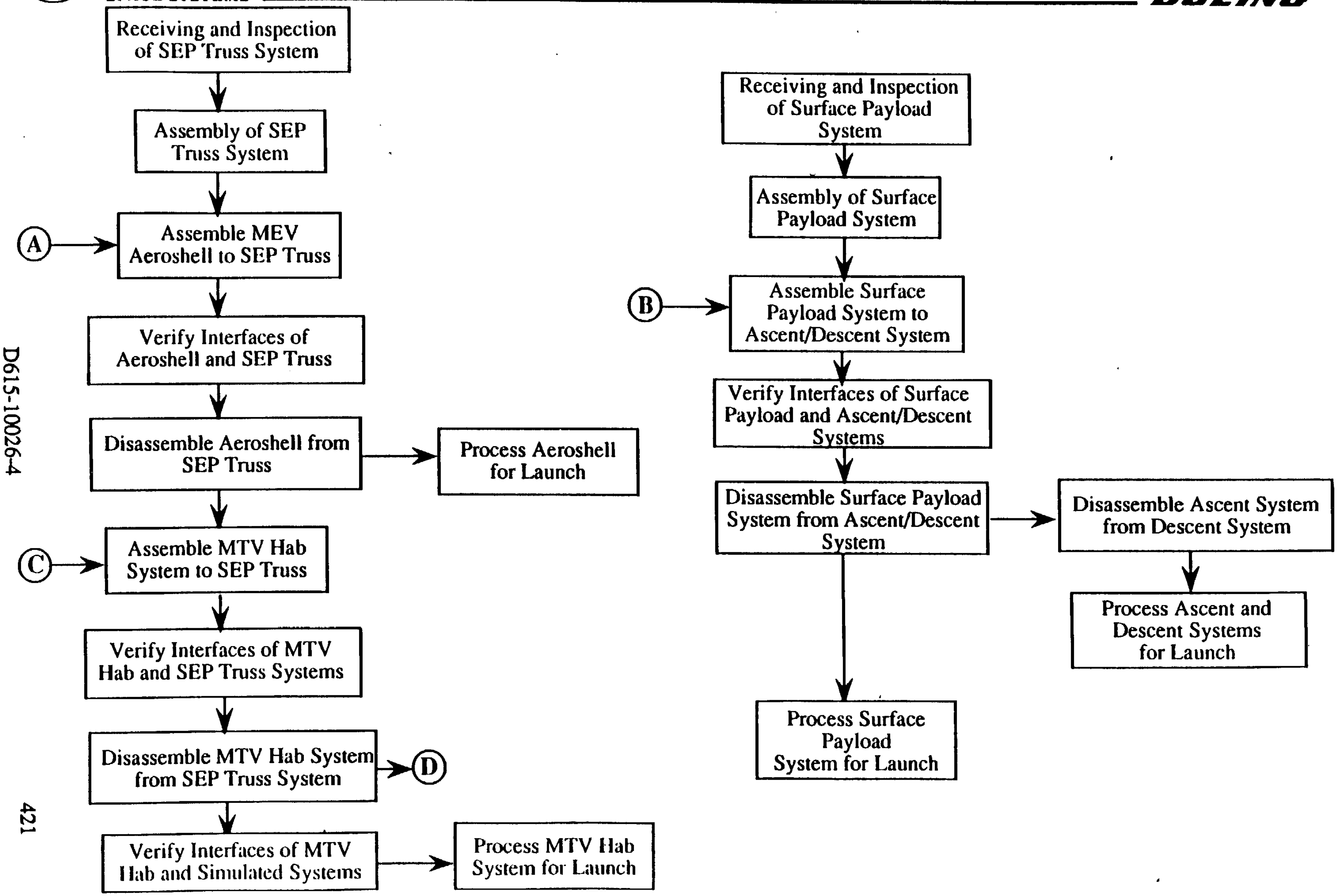
Sequential Interface Verification

- Process of verifying the interfaces of the Mars Mission Vehicles elements without complete assembly.
- Elements are received and inspected at the assembly area.
- Internal test performed and certified by the contractor will not be repeated.
- Elements will be assembled to the level required to verify the interfaces from one element to another.
- Interfaces will be verified by flight hardware when feasible or by match mate devices/prototypes when necessary.
- Elements will be disassembled to payload configurations and processed for launch.

SEP Ground Processing Functional Flow



SEP Ground Processing Functional Flow - continued



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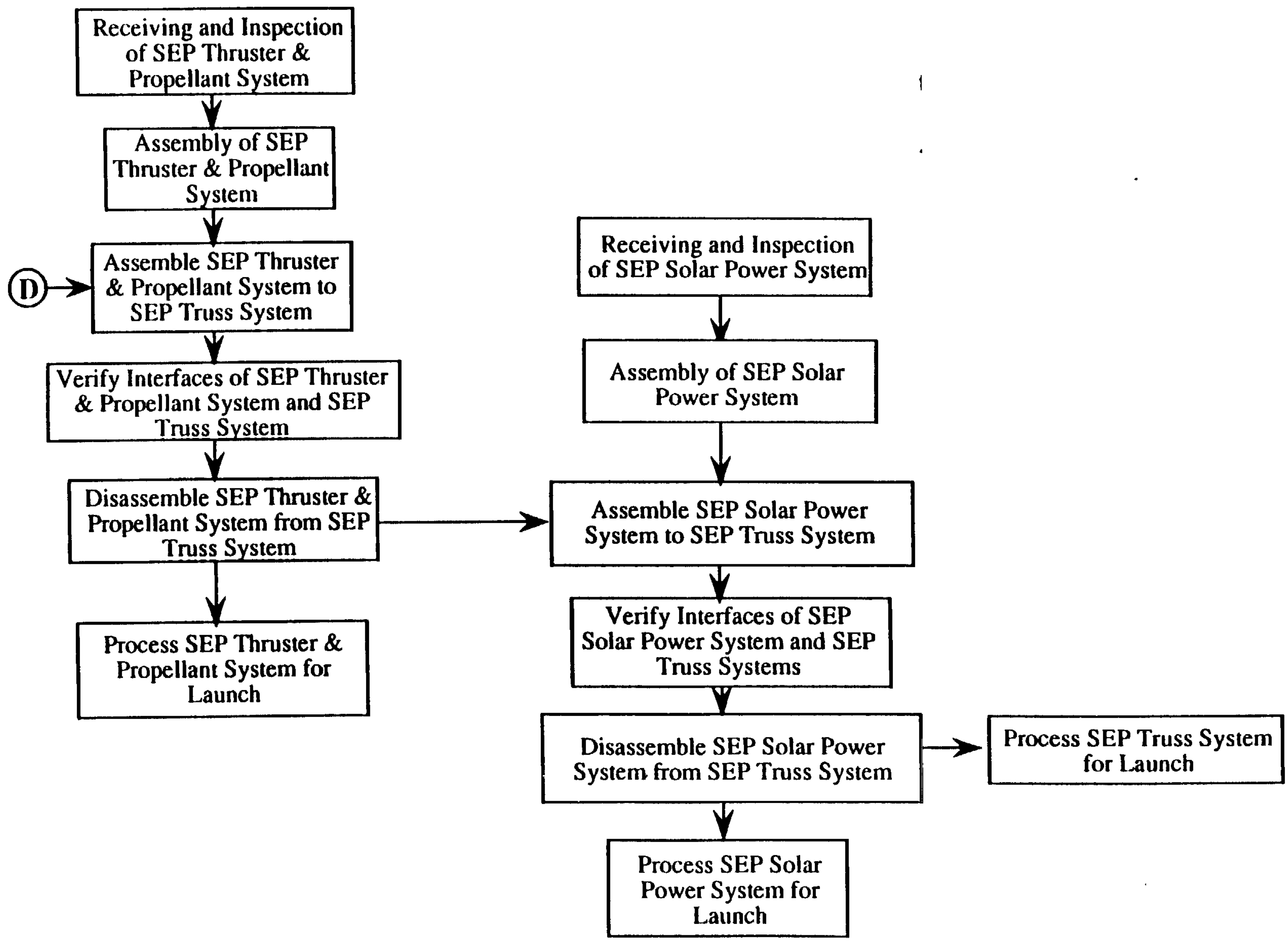
SEP Ground Processing Functional Flow - continued



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Special Ground and On-Orbit Processing Facility and Equipment Requirements

Facilities/Equipment	NTR	NEP	SEP
Ground			
• Reactor/engine mating and processing facility	X	X	
• Nuclear fuel loading facility	X	X	
• Contaminated materials storage and disposal facility	X	X	
• Solar array/radiator packing and storage facility	X	X	X
• Alkali metals materials and transferring facility		X	
• Radiation/hazardous materials contamination treatment facility	X	X	
• Robotics to handle radioactive fuels and hazardous chemicals/materials and components	X	X	
• Vehicle truss processing and packaging facility	X	X	X
On-Orbit			
• On-orbit robotic welding and certification equipment	X	X	
• On-orbit alkali metal heating capability		X	
• On-orbit robotic repair/maintenance equipment	X	X	X

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Summary (Ground Processing, Manifesting, On-Orbit Assembly)

- Ground processing flows are very interdependent upon the launch vehicle and assembly concept assumed
- Non-hardware interface verification may require simulators to better schedule hardware deliveries
- Assembly flights are mainly volume, not mass, dependent
  - Most flights underutilize relative mass capability
  - A mixed fleet may improve launch packaging efficiency for NEP and SEP
  - Integrated aerobrake launch provides advantage in terms of number of flight and orbital assembly
- Capabilities, requirements of first element launch (FEL) of Mars vehicles drives on-orbit assembly infrastructure
- Two of the NEP assembly stages require nearly the full 90 days allotted between flights
  - Radiators and heat transport system require a large number of operations
  - Changes in assumptions used for number of pieces and method of attachment could easily violate 90 day limit
- Assumed deployable truss for NEP, SEP, and NTR reduces on-orbit times
- Assumed extensive assembly robotics tends to decrease crew time and needed infrastructure

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VI. Implementation Plan

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Technology Needs and Advanced Plans

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Technology Issues - SEP

I. Introduction

Technology issues relating to the SEP vehicle are presented in this section. Some of the charts are also included in the Cryo, NTR, and NEP IP&ED documents. The focus of this section will be to bring out those issues important to the SEP from these charts, and to present a series of technology level requirements necessary for the reference SEP vehicle.The most important technology development needs for SEP are in the areas of power production and handling, and electric propulsion.

II. Technology commonality Issues

The following nine charts lay out the important technology commonality issues between the major propulsion options as well as across the seven major mission architectures identified in this study. The SEP vehicle exhibits commonality to the other vehicles in several important areas. The transfer crew module is substantially the same as for all the other options, especially those flying conjunction missions. The MEV is identical across all vehicle options, except for the cryogenic propellant management and storage system, which must provide storage for the outbound trip, instead of transferring it from larger tanks prior to landing. The argon propellant storage system will be similar to the oxygen storage system employed on the cryogenic vehicles (Lunar & Mars). The ion propulsion system will employ the same thrusters as the NEP vehicle, which increases the amount of parallel development which can take place before a full scale development decision must be made.

The seven identified Lunar/Mars mission architectures verses the required component technologies, enabling and enhancing, are shown on the next set of charts and facing page text. Many of these component technology issues are common across the listed architectures. These issues are for the entire integrated architectures, and do not necessarily refer specifically to the SEP vehicle in all cases. The areas of high power solar arrays and power distribution (at the multi-MW level), are the primary areas of technology development concern for the SEP option. Many of the other areas, however, are common to the initial cryogenic vehicles, indicating that the SEP could become an attractive Mars growth option.

III. Technology Development Concerns

As noted before, many of the identified critical and high leverage technology development issues are common across all four major vehicle options. Common critical technology issues include low-g human factors, autonomous system health monitoring, long term cryogenic storage and management (Argon and lander H_2 & O_2), long duration ECLSS, radiation shelter material and configuration, and in-space assembly. Unique SEP technology issues include low cost solar cells, and low mass, efficient, power conversion equipment. Enhancing technologies include cryogenic refrigeration (lander tanks), O_2-H_2 RCS, advanced in-space assembly techniques, and advanced materials development.

IV. SEP Vehicle Technology Requirements

Technology performance levels required for the SEP reference vehicle are outlined in the next six charts. These are not intended to be the levels needed for any SEP vehicle, but serve mainly to document the levels required to accomplish the identified reference mission profile with the vehicle model as configured. Changes to these specifications would not necessarily affect the feasibility of a SEP mission, but would change the reference vehicle configuration. The list also includes operational requirements which could drive technology development or advanced development. An example of this would be requirements for in-space assembly and testing which could drive in-space assembly facility design and capability.

V. Ion engine & Solar Array Technology Development

The technology issues relating to an integrated ion propulsion subsystem are presented, along with performance projections for both ion engines and solar arrays. The performance parameters include present levels, near term and far term levels, and in the case of the ion engine, a conservative and optimistic projection of future specific masses (kg/kW).

VI. SEP Technology Development Schedule

The final chart in this section is a proposed technology development schedule for the solar electric propulsion option. The schedule shows that, given a FY '91 start, the SEP vehicle could be ready for a Mars mission in the 2010 timeframe. A full scale decision point is also highlighted at the beginning of year 7. This is the point where a commitment should be made for full scale funding and development of the program.

Technology Commonality and Differences

System/Subsystem	Reference (Cryo-A/B)	NTR Vehicle	NEP Vehicle	SEP Vehicle
Crew Systems/Habitats Life Support, rad. prot., hab. struct., & airlock/EVA	Long duration life support system derived from SSF proven system. LTV crew module evolves to MTV; common LEV/MEV habitat system. Mars surface habitat derived from proven Lunar design. Mars surface TCS requires additional technology advances to deal with unique heat rejection problems. All extended missions (>2-3 d) require solar flare radiation protection. Hab systems common across mission architecture. Shorter mission LSS sized for free return abort contingency. Minimum mass airlock could be shuttle-evolved.			
Power System & Thermal Control	Deployed solar array system; low power (~50-75 kW). Low temp heat rejection (~400°K)	Common to reference vehicle system	Nuc. /Rankine or Brayton cycle energy conv. sys. Very high power level (up to 200 MW). High temp heat rejection (~1000°K-main cycle).	Solar-electric energy conversion. High power (~10 MW or greater) level. Moderate temperature radiators (400 - 650 K).
Propellant Management & Storage	Long term storage of H_2 & O_2 for Earth & Mars orbit, and deep space environ. necessary with minimal boiloff. Low-g fluid gaging, acquisition, and transfer highly enhancing or enabling for all missions. NTR requires common techniques for LH_2 fuel.		Argon propellant management system can be similar to LOX storage system, but without the safety constraints associated with an oxidizer.	
Propulsion System	Advanced cryogenic space engines with >475 sec Isp, and ~30 klb to ~200 klb thrust.	NERVA derived /advanced NTR system with higher Isp (up to 1050 sec vs. 850 sec.)	Rankine or Brayton cycle conversion system driving cluster of Ion thrusters for NEP. Same thrusters for SEP. Number of thrusters depends on available thruster size and required redundancy.	
Aerobraking Lunar	Low L/D - AFE derived for Earth capture.	Not needed for Lunar NTR (propulsive capture@ Earth)	Not needed for NEP or SEP.	
Aerobraking Mars	Higher L/D necessary - structure and TPS technology base.	Only low energy lander aerobrake needed, since entire vehicle, including MEV is propulsively captured at Mars. Can be common with earlier cryo A/B vehicle, unless crossrange constraints require higher L/D design.		
Avionics	Avionics system hardware may be common for Lunar or Mars (or L/M growth)		Avionics system required for low & continuous thrust vehicles are lower than for Cryo A/B or NTR vehicle.	
Assembly & Checkout	Common assembly facility & equip. for most mission vehicles. Assembly time in LEO, and thus M/D protection level is varied. Mars vehicle requires launch & assembly of large (~ 30 m vs. 20 m for Lunar) aeroshell. Nuclear vehicles (NTR & NEP) may face political constraints on launch & assembly of vehicle. Assembly & operation may be necessary from nuclear safe orbit.			Severe LEO debris environ. damaging to solar arrays. Spare set of arrays may be necessary. MEV A/B launch & assembly needed.

Required Technologies vs. Alternative Mission Architecture

A set of required technologies for the seven identified alternative mission architectures outlined in the evolotionary concepts section is presented. The purpose of this matrix is to provide a preliminary comparison of technology development needs for the alternative architectures. The matrix also serves to better define the architectures. From this top level matrix, a more detailed set of technology requirements can be derived. A set of accommodating technologies can be compiled for needs areas where options exist. Finally, the technology areas can be prioritized as enabling and enhancing, and a return on investment performed for identified high leverage technologies. This portion of the matrix includes most of the cryogenic management issues. Enabling technologies are represented by the filled circle, and enhancing technologies by the open circle. Extensive low - g cryogenic propellant launch, acquisition and transfer refers to the Mars conjunction case, and the mass driver option, where propellant will be used for the transfer vehicles, which will be parked in a low - g environment (Lunar or Mars orbit, or libration staging point). The Mars cycler orbit case includes a question mark for the long term cryogenic storage system, because the necessary thrust levels and type of propulsion system are undetermined at this time.

Required Technologies vs. Alternative Mission Architecture

	Low boiloff cryogenic propellant storage system (1-3 yr)	Low boiloff cryogenic propellant storage system (15 - 60 d)	Low - g fluid acquisition and transfer	Extensive low - g cryogenic propellant launch, acquisition, and transfer	Cryogenic tank integrity monitor	Cryo fluid rematible umbilical	Lunar LOX production, liquification, and transfer technology	Mars O2 production, liquification, and transfer technology
Mars NEP Alternative Architecture	○	●	●		●	●	●	●
Lunar/Mars NTR Alternative Architecture	●	●	●		●	●	●	
Mars SEP Alternative Architecture	○	●	●		●	●	●	●
L2 Node / Mass Driver Alternative Architecture	●	●	●	●	●	●	●	
Mars Cycler Orbits Alternative Architecture	?	●	●		●	●	●	
Mars Conjunction/Direct Alternative Architecture	●	●	●	●	●	●	●	● + H2
Lunar / Mars NEP Alternative Architecture	○	●	●		●	●	●	●

● - Enabling

○ - Enhancing

Required Technologies vs. Alternative Mission Architecture (Cont.)

This matrix section represents the major aerobraking concerns. The aerobraking energy columns for Mars and Earth capture digresses from the format in order to illustrate the energy levels, and therefore, the level of technology development needed for the various architectures. Aeroheating predictions, reusable aerobrake TPS, advanced GN&C, and TT&C follow along with the high and medium energy missions. Again, a question mark is shown for the Mars cycler orbit case. Reusable TPS for Earth return cannot be determined as a technology development concern until the aeroheating load at Mars can be determined for the cycler orbits. Further mission design efforts must be carried out before an estimate on this can be made.

Required Technologies vs. Alternative Mission Architecture (Cont.)

	Earth return aerobrake energy	Mars capture aerobrake energy	Mars lander aerobrake	High performance aerobrake structure	Aerobrake assembly and test	Aeroheating prediction (Earth and/or Mars)	Reusable aerobrake TPS for Earth return	GN & C to protect TPS	Advanced high accuracy and rate TT & C	In space AR&D / assembly
Mars NEP Alternative Architecture	Low	Low	●	●	●					●
Lunar/Mars NTR Alternative Architecture			●	●	●					●
Mars SEP Alternative Architecture	Low	Low	●	●	●					●
L2 Node / Mass Driver Alternative Architecture	High	High	●	●	●	●	●	●	●	●
Mars Cycler Orbits Alternative Architecture	High	High	●	●	●	●	?	●	●	●
Mars Conjunction/Direct Alternative Architecture	Medium	Medium	●	●	●	●	●	●	●	●
Lunar / Mars NEP Alternative Architecture	Low	Low	●	●	●					●

● - Enabling

○ - Enhancing

Required Technologies vs. Alternative Mission Architecture (Cont.)

This matrix area represents the major propulsion issues, with the exception of the radiation protection system, for the baseline and alternative mission architectures. The system to inert and can waste for radiation shielding can be enhancing, while a GCR and ALSPE shelter is enabling for all mission architectures. Again, due to the undefined Mars cycler orbit trajectories, it is questionable as to the need for a large cryogenic space engine. A H2 - O2 ACS/RCS system is noted as enabling for each option, as it will be for any option over a baseline storable system. A Lunar orbital momentum storage and transfer device such as a bolo can be enhancing for all missions, after an initial launch and assembly penalty for the massive (~ 1000 Mt) device.

Required Technologies vs. Alternative Mission Architecture (Cont.)

	Large (150 - 200 klb) cryogenic advanced space engine	Small (15 - 30 klb) cryogenic advanced space engine	H2 - O2 ACS/RCS	Multi - MW space based nuclear electric power	Multi - MW space based nuclear thermal power	Surface nuclear electric power	Multi MW solar power system (arrays and handling equip.)	Radiation protection (system to inert & can waste)	Mass driver / rail gun technology	Lunar orbital momentum transfer device (Bolo)
Mars NEP Alternative Architecture		●	○	●		●		●		○
Lunar/Mars NTR Alternative Architecture		●	○		●	●		●		○
Mars SEP Alternative Architecture		●	○				●	●		○
L2 Node / Mass Driver Alternative Architecture		●	○			●		●	●	○
Mars Cycler Orbits Alternative Architecture	?	●	○			●	●	●		○
Mars Conjunction/Direct Alternative Architecture	●	●	○			●		●		○
Lunar / Mars NEP Alternative Architecture		●	○	●		●		●		○

● - Enabling

○ - Enhancing

Required Technologies vs. Alternative Mission Architecture (Cont.)

The final section of the matrix is not as illustrative as the others, in that all of the listed technologies are enabling, with the exception of a closed ecological life support system, which is significantly enhancing for all identified mission architectures.

Required Technologies vs. Alternative Mission Architecture (Cont.)

	Autonomous health monitoring and check - out	High data rate comm. or high performance compression	DMS/system diagnostics. Art. intell/neural nets/high processing rate GN&C	Long duration refurbishable crew habitat	Long duration ECLSS	CELSS
Mars NEP Alternative Architecture	●	●	●	●	●	●
Lunar/Mars NTR Alternative Architecture	●	●	●	●	●	●
Mars SEP Alternative Architecture	●	●	●	●	●	●
L2 Node / Mass Driver Alternative Architecture	●	●	●	●	●	●
Mars Cycler Orbits Alternative Architecture	●	●	●	●	●	●
Mars Conjunction/Direct Alternative Architecture	●	●	●	●	●	●
Lunar / Mars NEP Alternative Architecture	●	●	●	●	●	●

Mars SEP Vehicle Technology Requirements

I. TMIS / MTV

A. Propulsion

1. Isp = 5000 s.
2. Power level: 10 MW.
3. Thruster type: ion; Engine efficiency = 68%.
4. Thrust = 122 N.
5. Burn lifetime: 10000 hr projected (replaced after 1 mission).
6. No throttling requirements.
7. Gimbal angle (nominal) = 20°
8. Space exposure life = 3 yr.
9. Propellant: Argon.
10. In-space changeout capability.
11. Off vehicle preflight checks.
12. No retraction / extension required.

B. Power system

1. Type: Solar Cell.
2. Type: CLEFT GaAs/CIS .
3. Efficiency = 26%.
4. Power level: 10 MW.
5. Array blanket specific power: 460 W/kg.

C. Cryogenic storage system

1. Thermal protection system: MLI over foam (for launch) 2" MLI over 1/2" - 1" foam.
2. Tanks launched wet.
3. Thermodynamic vent coupled to a single vapor cooled shield.
4. Topoff before Earth departure.
5. ~ 6 - 12 months in LEO before use.
6. Negligible boiloff loss after topoff.

Mars SEP Vehicle Technology Requirements (cont.)

D. Structure
1. Material - metal matrix composites, advanced alloys, and organic matrix composites.
2. Meteor/debris protection provided for tanks and plumbing.

E. Avionics
Piggybacked on MTV.

F. Power
1. Level : < 10 kW (crew systems only); Propulsion system : 10 MW.
2. System: Solar panels and conditioning equipment on main truss.

G. Assembly
1. Off station assembly.
2. Degree of assembly: Separate tanks connected to primary structure in LEO to form propulsion stage.

H. Habitat
1. ECLSS: Space Station Freedom derived system with similar degree of closure; potable H_2O from cabin condensate; CO_2 reduction/regeneration; Hygiene H_2O from urine processing. CELSS to be evaluated.

Mars SEP Vehicle Technology Requirements (cont.)

2. Structure
   a. Silicon carbide reinforced aluminum matrix, plasma sprayed.
   b. Pressurized to 20 psig on launch for structural integrity.
   c. Insulation & M/D shield external to pressure shell.
   d. No penetrations in end domes.
   e. Radiation storm shelter provided, and configured to utilize equipment & supplies as partial shielding.
   f. External space radiator integral with M/D shield.

3. Cabin repressurizations: 2+ (outbound emergency could use propellant for repress.)
4. Spares: 15% of active equipment - component level.
5. Redundancy: Two complete and separate systems for life critical systems + spares. Component changeout capability.
6. Residence time = 535 days.
7. Science: Transit science as allowed by individual mission.
8. EVA capability: EVA suits provided for all crew; EVA waste fluid recovery for ECLSS.

I. <u>ECCV</u>

1. Apollo size & style as a starting point.
2. Open ECLSS (LiOH, no H_2O recovery).
3. Residence time: 2 - 3 days.
4. Propulsion: RCS only.

Mars SEP Vehicle Technology Requirements (cont.)

II. MEV

A. Cryogenic storage system

1. Thermal protection system: 100 layers of MLI for H_2 and O_2 tanks (2").
2. Tanks: double wall tanks with vacuum annulus;
   low thermal conductivity support system for inner tank.
3. Thermodynamic vent: Simple design for gravity field.
4. Tanks launched dry and filled prior to descent, from MTV tanks, or refrigerated. (no boiloff prior to descent)
5. Stay time from 30 - 600 days on Mars surface.
6. Boiloff level < 20% for surface stay.

B. Propulsion

1. I_{sp} = 460 sec.
2. Thrust = 30 klb / engine.
3. Nozzle area ratio = 200.
4. Throttleability = 15:1.

Mars SEP Vehicle Technology Requirements (cont.)

B. Propulsion (cont.)
  6. Gimbal angle (nominal) = 10°.
  7. No restart capability necessary for nominal case.
  8. Space storage time between burns : NA.
  9. Engine out capability (crossfeed propellant lines).
  10. Expander cycle.
  11. In-space changeout capability.
  12. Off vehicle preflight checks.
  13. Retraction / extension capability.

C. Structure
  1. Vehicle
    a. metal matrix composites / advanced alloys / organic matrix composites.
    b. Micrometeoroid protection for tanks and plumbing.

  2. Aerobrake
    a. L/D = 0.5 to 1.0
    b. Crossrange: 1000 km.
    c. Vhp = 7.07 km/sec^2
    d. Maximum g loading: 6.
    e. Maximum temp: TBD (estimated 3100° F).
    f. Structure material: Carbon Magnesium ribs (σ_{ult} = 200 ksi) bonded to titanium honeycomb shell.
    g. TPS material: Advanced reradiative tiles.
    h. Relative wind angle (reference) = 20°.

Mars SEP Vehicle Technology Requirements (cont.)

D. Avionics
1. Error without beacon = 1 km.
2. Touchdown error = 1 m/s.
3. Obstacle avoidance capability.

E. Power
1. Level: ~ 2.5 kW.
2. System: fuel cells (regenerable).
3. Back-up system: abort to orbit.

F. Assembly
1. Off station assembly.
2. Assembly level (complexity): TBD

G. Habitat
1. ECLSS: open system; stored potable H_2O; LiOH CO_2 adsorption.
2. Structure
   a. Silicon carbide reinforced aluminum matrix, plasma sprayed.
   b. Overpressurized on launch for structural integrity.
   c. Insulation and micrometeoroid protection external to pressure vessel.
   d. No penetrations in end domes.
   e. No radiation shelter provided in MEV.
   f. External space radiator integral with micrometeoroid shield.
3. Repressurizations: 2.
4. Spares: 15% of active equipment mass; component level.
5. Redundancy: EVA suits as backup to cabin repressurization.; no system level ECLSS redundancy required due to low complexity open system.
6. Residence time: ~3 days (surface systems support surface stay).
7. Science: none.
8. EVA capability: provided for all crew; transferred from MTV.

Critical Lunar/Mars Reference Technology Development Concerns

A preliminary set of critical technology development concerns was constructed for the Lunar/Mars reference missions. Its purpose is to show a top level representation of the areas which could prove enabling for the reference Lunar and/or Mars missions, without further concentrated research and development, flight testing, and/or precursor missions. Aerobraking may prove enabling for most Lunar and Mars missions, and significantly enhancing for the rest, primarily due to reduced demands on limited Earth to orbit launch capability and lower launch costs. Aeroheating prediction codes cannot be validated without further experimental data (flight or ground simulation data). The degree of development needed for aerobrake TPS materials will be determined by these predictions. Low gravity human factors, to be evaluated on SSF, may affect vehicle design significantly. Foe example, vehicle designs must accommodate artificial - gravity until a need level can be determined from space station based research. Finally, precise mission design, incorperating advanced tracking, telemetry, and GN&C must be verified to accommodate aerobraking and automated rendezvous & docking requirements.

Critical Lunar/Mars Reference Technology Development Concerns

Technology	Comments
High Energy Aerobraking - Thermal protection - High performance structure - Theoretical code validation - Deep space tracking, telemetry, and communication	- Heating rates greater than seen by AFE for Mars cap. and Mars/Earth return. - High temp reradiative or lightweight ablative materials needed - Precursor missions needed for existing aeroheating/GN&C codes - 17 minute Mars/Earth comm delay will dictate internal GN&C system.
Advanced Space Engine Development - Large engine (150 - 200 klb thrust) - Small engine (15 - 30 klb thrust; throttleable)	- High thrust, high Isp cryogenic engine for TMI stage. - Low thrust, high Isp, throttleable engine for Lunar/Mars descent and ascent.
Low - g Human Factors	- Vehicle designs should accommodate artificial-g configuration until SSF based life sciences research can be carried out.
Autonomous System Health Monitoring	- Reliable autonomous systems with self monitoring, diagnostic, and correcting capability.
Long Term Cryogenic Storage and Management	- Advances in long term low - g cryo fluid storage and management required for Lunar/Mars initiatives. - low - g propellant acquisition and gaging enabling for all cryo missions.
Long Duration, High Degree of Closure ECLSS	- Reliable SSF validated ECLSS equipment critical for early long term missions.
Efficient Radiation Storm Shelter Material & Configuration	- Improved solar flare prediction/detection, with storm shelter designs incorporating effective lightweight materials - Reliable radiation dosimetry techniques also important
In - Space Assembly; AR & D	- Large aerobraked vehicles will require large degree of in - space assembly. - AR&D critical for both Lunar/Mars orbital operations.

6

Preliminary Identified Lunar/Mars Reference High Leverage Technology Issues

A preliminary set of high leverage technologies was assembled for the Lunar/Mars reference missions. These technologies are enhancing for most, and in some cases, all identified mission architectures. Aerobraking will be significantly enhancing for all Lunar and Mars missions where it is not identified as enabling. Other aerobraking issues which could prove enhancing are lightweight reradiative or ablative TPS material, and ECCV vs. aerocapture of MTV at Earth. Low - g propellant handling and low boiloff cryogenic storage are also very enhancing for any missions where it is not enabling. Advanced propulsion options such as NTR, GCR, SEP, and NEP may prove to be high leverage technology options to baseline cryogenic propulsion systems. Finally, developments in advanced materials can be significantly enhancing in a variety of areas.

ADVANCED CIVIL SPACE SYSTEMS

Preliminary Identified Lunar/Mars Reference High Leverage Technology Issues

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Technology	Comments
Aerobraking - Mars Capture (vs. propulsive cap.)	- Aerocapture at Mars can reduce IMLEO >50% over propulsive capture
Aerobraking - Earth Capture (vs. ECCV)	- ECCV reduces IMLEO and thermal protection system (TPS) requirements. - Reusable MTV can reduce life cycle cost.
Aeroshell TPS (reradiative vs. ablative)	- Reusable aeroshell requires rerad. TPS at Mars (or thick lightweight ablator), and ablative at Earth. - Further materials and processes advances or low energy mission may allow Earth/Mars reradiative TPS.
Advanced Long Term Cryogenic Storage Technology	- Cryogenic boiloff reduction technologies such as advanced MLI design and application, VCS, para to ortho H2 conv., and thermal disconnect struts, can reduce IMLEO significantly with low R & D effort - Longer missions offer greater IMLEO savings potential
Low - g Propellant Transfer	- Low - g propellant transfer technology enhancing for all Lunar/Mars mission arch., and enabling for some Lunar missions.
Efficient Cryogenic Refrigeration System	- Cryogenic refrig system can reduce vehicle mass and enhance system reliability at the expense of an increased power level.
O2 - H2 ACS / RCS	- O2 - H2 ACS/RCS (Isp = 400 s) reduces system mass over lower Isp storables
High Isp Advanced Space Engine	- High Isp advanced space engine (Isp = 485 s) enhances all mission phases for all mission arch.
NTR Propulsion System	- NTR propulsion system for the TMI, Lunar transfer, and Mars transfer stages
Advanced In - Space Assembly Techniques	- Launch vehicle capability drives on - orbit assembly level. - Degree of on - orbit assembly capability affects vehicle configuration, ground assembly/processing, and launch manifesting.
Advanced Materials Development	- Advanced materials such as metal and organic matrix composites reduce system inert mass, strength, and/or manufacturing costs. - Some advanced M&P may prove enabling for some mission arch. (ex:Mars/ Earth capture aerobrake)

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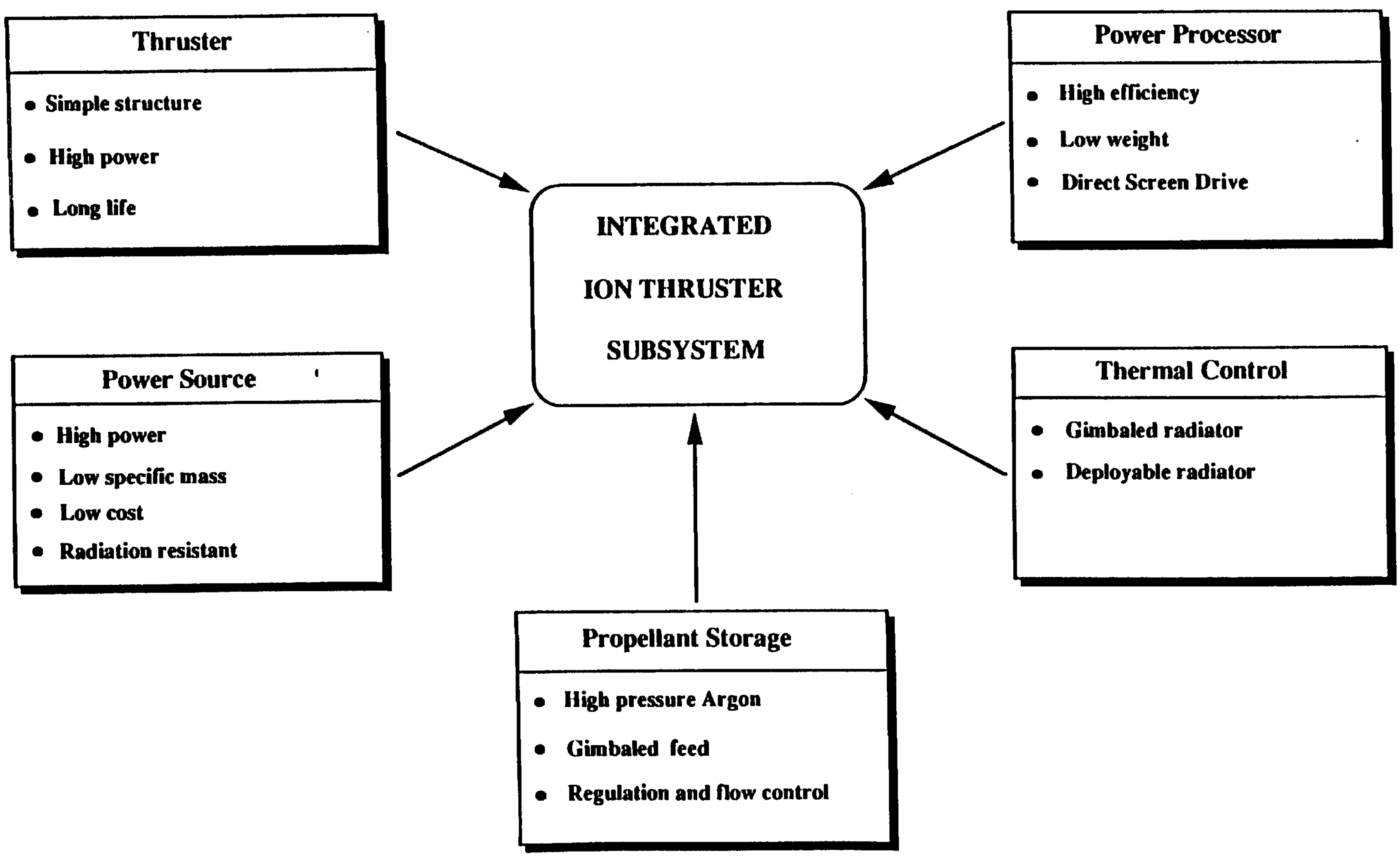
ION Propulsion Subsystem Technology Elements



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SEP Required Technology Development

Arrays

- More efficient lightweight array blankets must be devoloped & tested to implement an array on the megawatt level. Current, near term, & future performace parameters are shown below.

Current				Near term (90's)				Advanced		(~ 2000)			
Safe		APSA		GaAs on Ge		MJ on Ge		THIN GaAs		CLEFT GaAs		CLEFT MJ	
104 w/kg	99 w/m^2	300 w/kg	128 w/m^2	259 w/kg	189 w/m^2	389 w/kg	283 w/m^2	536 w/kg	230 w/m^2	573 w/kg	209 w/m^2	769 w/kg	314 w/m^2

- Based on BOL performance @ 1 AU
- For blankets only (does not include structure)

ION Engines

- Large ION engines would reduce overall system complexity and increase system reliability. Current and projected thruster specific masses are listed below (based on 5000 sec, Isp).

1. 30cm-5.175kw (no DSD) — * 10.9 kg/kw
2. 100cm x 500cm- 625kw (DSD) — * 2.7 kg/kkw
3. 100cm - 175kw (DSD) — * 1.53 kg/kw

* Includes PPU

1. Lewis SEPS
2. Future projection
3. Future projection (optimistic)

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Schedules

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Technology Development Concerns and Schedules - Solar Electric Propulsion (SEP)

Critical technology development issues relating to the reference SEP vehicle are presented in this section. Where applicable, the same charts are also included in the CAB, CAP, NEP, and SEP IP&ED documents. The focus of this section will be to bring out the most important issues relating to the reference NTR vehicle, and to present preliminary technology development schedules for these issues. The issues are presented here in outline form, beginning with the most important, with accompanying schedules wherever possible.

Solar Power System Technology Development

One of the two most important areas of technology and advanced development for this vehicle option is the development of an integrated solar electric power system. The most important area of development for the SEP option is the design, integration, and life testing of a space qualified multi-megawatt solar power system, consisting of high efficiency solar arrays. Major challenges to be overcome in the achievement of a long life efficient system lie in efficient solar array development, and efficient power processing and delivery systems. Long term life testing must be carried out for the power system in order to verify long term system reliability. A related technology development challenge for the program may be test facility design and development. Solar electric propulsion offers a potential performance which may be superior to the any of the other advanced propulsion options, at the expense of a more costly and lengthy technology and advanced development program.

Electric Propulsion PPU/Thruster Technology Development

The second major area of technology development for the SEP is in large scale electric power processing unit (PPU), and thruster design and development. The power system technology development schedule presented in the NEP IP&ED book includes a timeline for electric thruster design. The development of long life PPU/thruster systems on a larger scale than currently available (MW level thrusters needed) is the major area of concern relating to the SEP concept. Thruster lifetimes on the order of a year or more (continuous) will be required for thrusters on the MW level in scale. Test facilities must be developed which are capable of supporting the long term life tests for these high power level thrusters. Finally, high temperature power processing equipment must be developed to increase system efficiency and reliability.

Life Support

A reliable, redundant long term life support system will be enabling for future exploration missions. The degree of closure of, and the reliability of the system are the major technology development concerns. Low-g human factors determination will also be an important technology consideration which will drive vehicle design. An integrated schedule of the major areas of the life support technology development task are presented. It includes radiation shielding and materials, regenerative life support, and EVA systems development. As before, the points where Lunar and Mars full scale development decisions can logically be made in the technology program are highlighted.

Aerobraking (low energy)

Low energy aerobraking will offer mission benefits in the areas of decreased demands on the descent propulsion system, and improved crossrange capability. This area presents a variety of issues for technology development including high strength to mass ratio structural materials, high temperature thermal protection systems (although not as high

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as for high energy aerobraking), avionics, assembly and operations, hypersonic test facilities and computer codes, and Mars atmosphere prediction. High strength structural material options include metal matrix composite, organic matrix composite, and advanced carbon-carbon elements. Other structural considerations include load distribution and attachment of payload for aerocapture, and ETO launch and assembly of large structures. Thermal protection systems issues include low mass ablative and reradiating materials, and structure/TPS integration issues. The aerobrake maneuver will place considerable demands on the vehicle avionics system with the need for real time trajectory analysis, and vehicle guidance and control. The launch and assembly of the large aerobrake structure will present ground and space assembly and ops problems which will require technology and advanced development in both the areas of design and operations. Finally, computational analysis and atmosphere prediction capability will be critical in the development of a man-rated aerobrake for Mars use. A preliminary development schedule for Lunar and Mars aerobrake technology development is presented. It includes the major milestones for both ground and flight testing. The points where a Lunar and Mars full scale development decision can be made are also highlighted on the schedule. It should be noted that this schedule was built with high energy aerobraking in mind, and will possibly be compressed to some degree if only low energy aerobraking is developed.

Vehicle Avionics and Software

Although the technology readiness level of vehicle avionics and software is ahead of many of the other technology areas listed in some respects, the demands on the system in the areas of processing rate, accuracy, autonomous operation, and status/health monitoring will drive technology and advanced development in areas not fully defined at this point. Software requirements cannot be fully determined until the vehicle design is at a more finished stage than the current levels. A preliminary schedule for autonomous systems development is presented. The decision points for full scale development The communications system options can be more fully defined before a final vehicle design is produced, however. A technology development schedule for advanced communications is presented. The SEP vehicle may not place the same level of demand on the avionics system in the area of trajectory analysis, but will likely place more demands on the system in the areas of status and health monitoring, fault diagnosis, and correction.

In-Space Assembly and Processing

The in-space assembly and processing of large space transfer vehicles will present a variety of technology advanced development challenges, particularly for the large LTV and MEV aerobrakes, and SEP vehicle. The large solar array structure, along with the large amount of wiring and electrical connections will present a variety of challenges in technology development (e.g. in-space welding), and assembly operations (e.g. robotics). As shown on the accompanying schedule, extensive ground tests must occur before any orbital work can be initiated. The vehicle designs will be driven to a large degree by the assembly facilities and technologies seen as being available during the vehicle buildup sequence. It should be noted that the schedule was not developed specifically for an NEP vehicle. Advances derived from this development process along with flight experience in earlier missions leading up to this evolutionary scenario could possibly accelerate the development plan considerably.

Cryogenic Fluid Management

The level of concern for technology development in the areas of cryogenic fluid management and storage will not be as for electric propulsion vehicles as for the high thrust systems, although many of the areas still remain important for the SEP vehicle. The Argon (or Zenon) propellant utilized for the electric propulsion system will be in a cryogenic liquid state, and will require long term storage and management technology levels similar to those for liquid oxygen storage for the chemical vehicles. Cryogenic storage issues relating to ECLSS fluids and lander/ascent vehicle propellants will remain as well. A preliminary technology schedule is presented for cryogenic fluid system development for Mars mission applications. The cryogenic fluid systems schedule includes Earth-based thermal control and selected component fluid management (tank pressure control, liquid acquisition device effectiveness, etc.) tests, as well as planned flight experiments to carry out system and subsystem development (selected components) and verification/validation tests. Many of the technology issues will be answered during the technology/advanced development work to be carried out for a Lunar program. The major technology obstacles to be overcome by an NEP storage system are in the areas of high reliability long term thermal control systems (particularily for the lander/ascent tanks), and orbital/flight operations (fluid transfer, acquisition, etc.).

Summary

As noted before, some of the identified critical and high leverage technology development issues are common across all of the major vehicle options. Common critical technology issues include low-g human factors, autonomous system health monitoring, long term cryogenic storage and management (H_2, and possibly O_2 for ECLSS), long duration ECLSS, radiation shelter material and configuration, and in-space assembly. Unique SEP technology issues center around efficient solar power systems and electric thruster/PPU development. Common enhancing technologies include cryogenic refrigeration (lander tanks), O_2-H_2 RCS, advanced in-space assembly techniques, higher Isp cryogenic engines, and advanced structural materials development.

Advanced Propulsion Technology Development Schedule - SEP

A proposed development schedule is presented for a solar electric propulsion (SEP) system. The schedule is for a representative multi-Megawatt solar system (10 MW). The schedule includes both technology and advanced development tasks necessary to produce an initial flight article for the flight program in year 15. The years are listed sequentially, so the schedule can be inserted into the appropriate initial year of a given program schedule. This schedule was used to derive funding spread estimates for the life cycle model, where the SEP option was traded against the reference cases for the full science and settlement/industrialization scenarios. Timelines for the development of requirements, system designs, test facilities, PMAD equipment, and integrated systems are included in the schedule. Also included in the schedule is a proposed development schedule for the electric thruster and power processing unit development, the array manufacturing facility development, and the flight array production. Unlike the other advanced propulsion options, the life qualification tests are not the "tent pole" in the DDT&E program. The array production, which will take more than 5 years for a 10 MW system, will to a large degree drive the readiness date. Any changes in the nations solar cell production capability could greatly affect the readiness date.

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Preliminary SEI Technology Development Schedules

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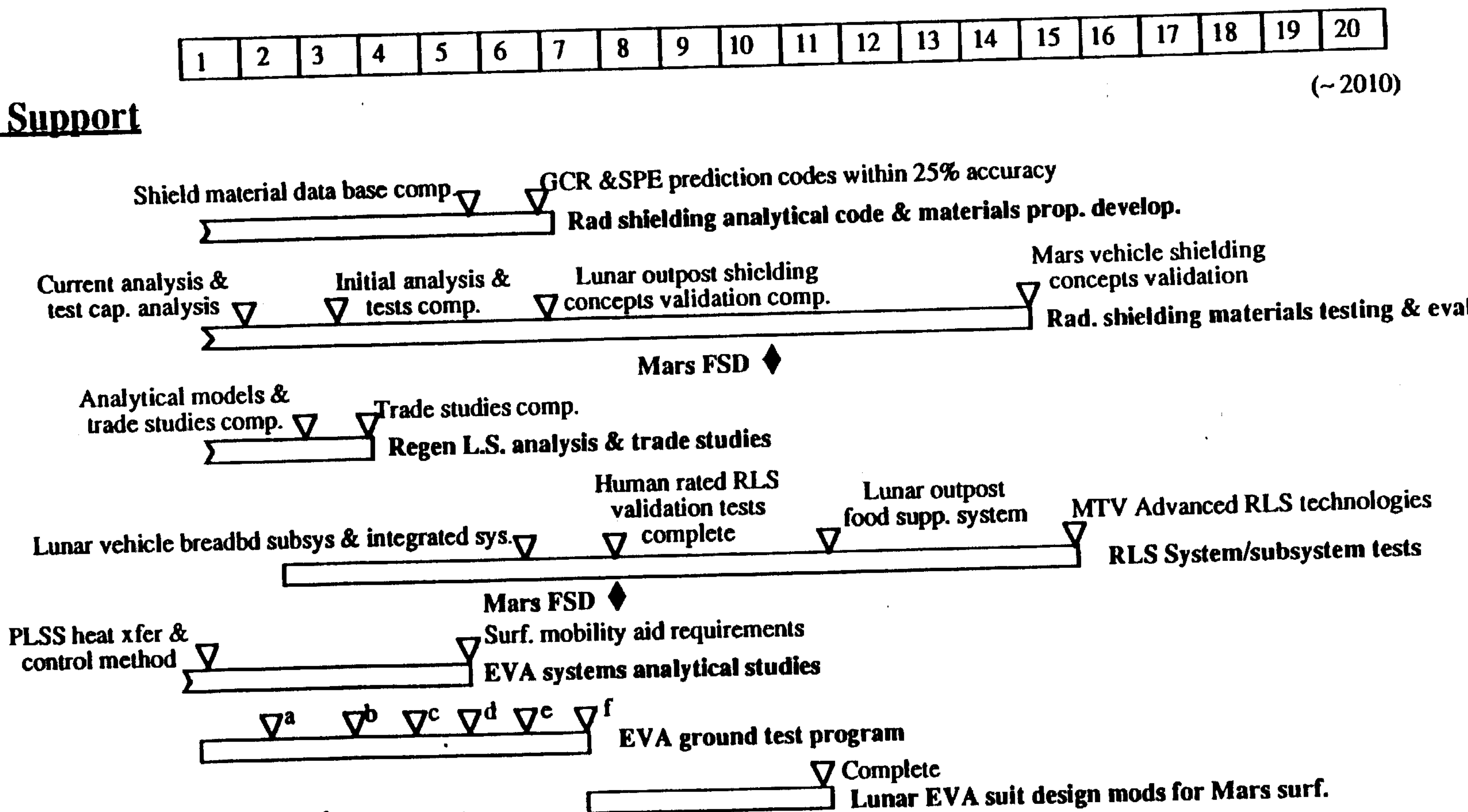
Life Support



a - Helmet/duit display & control concepts tests
b - Lunar surf suit breadbd test
c - Gloves & displays in simul. SSF environment
d - Regen. PLSS breadbd for lunar surf.
e - Verif tests of adv. dexterious gloves & disp.
f - Complete breadbd lunar EVA suit / simulated surf. cond.

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Preliminary SEI Technology Development Schedules (Cont.)

ADVANCED CIVIL SPACE SYSTEMS

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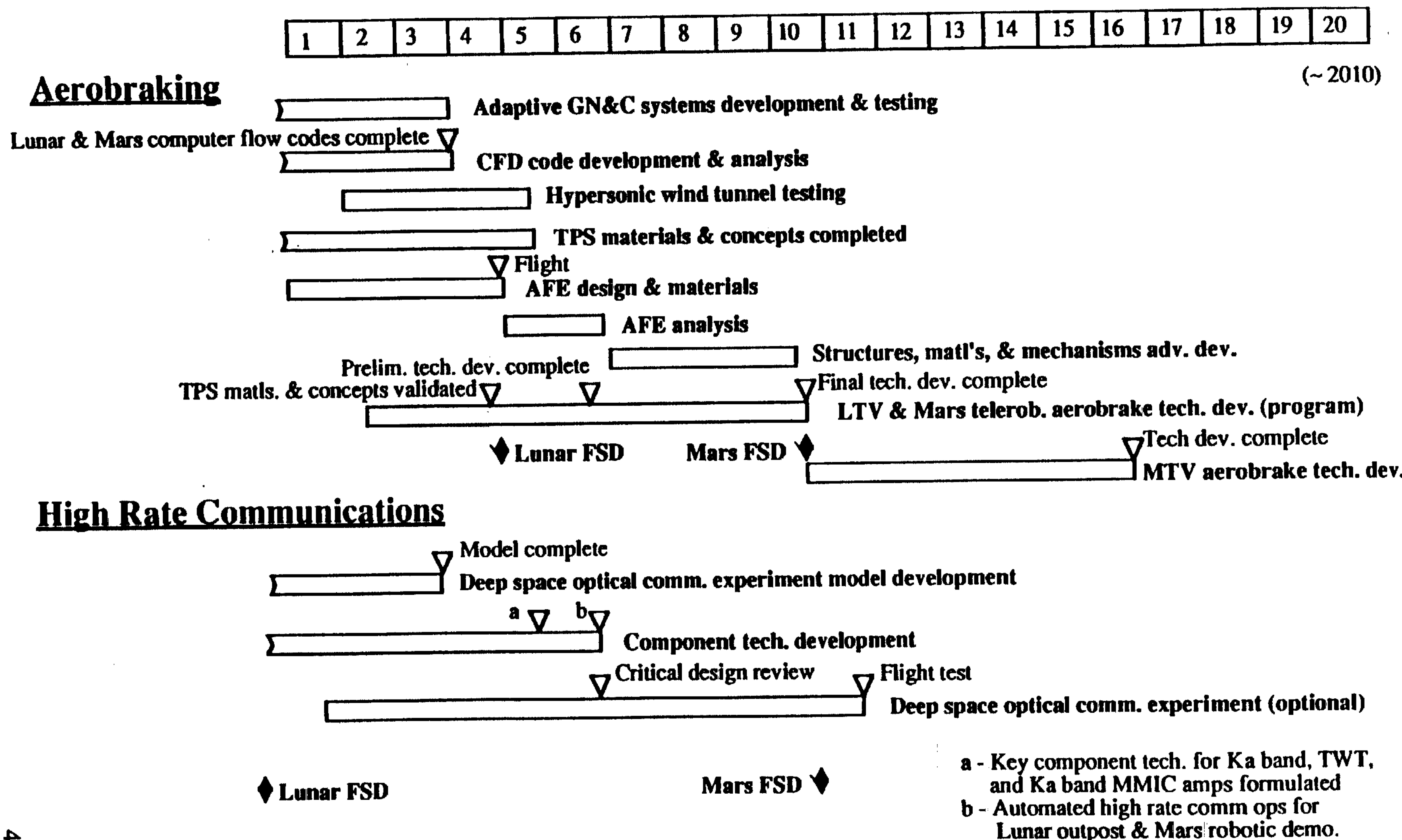
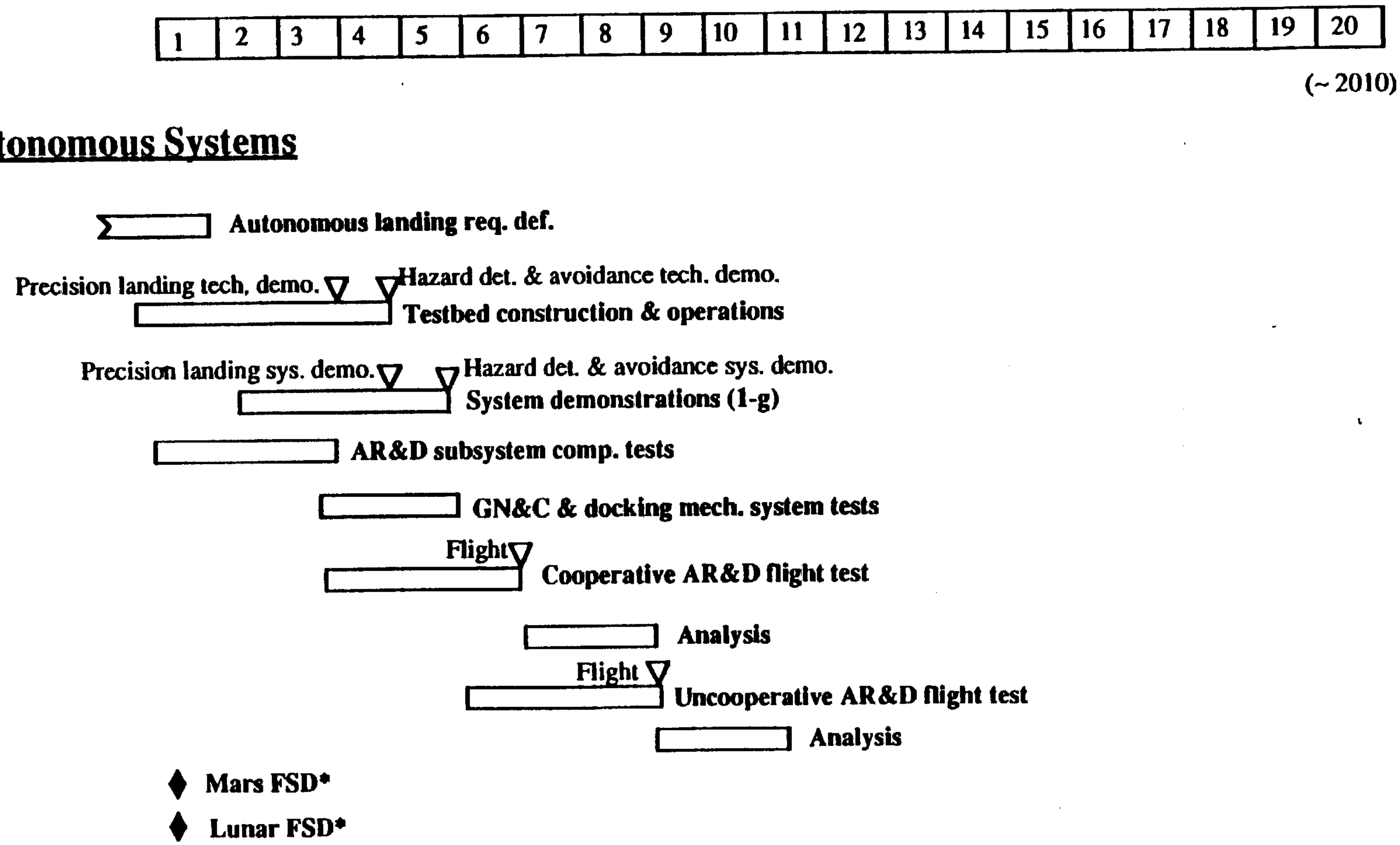


Preliminary SEI Technology Development Schedules (Cont.)

Autonomous Systems



* Technology should not present FSD threatening problems; current technologies adequate for minimum mission.

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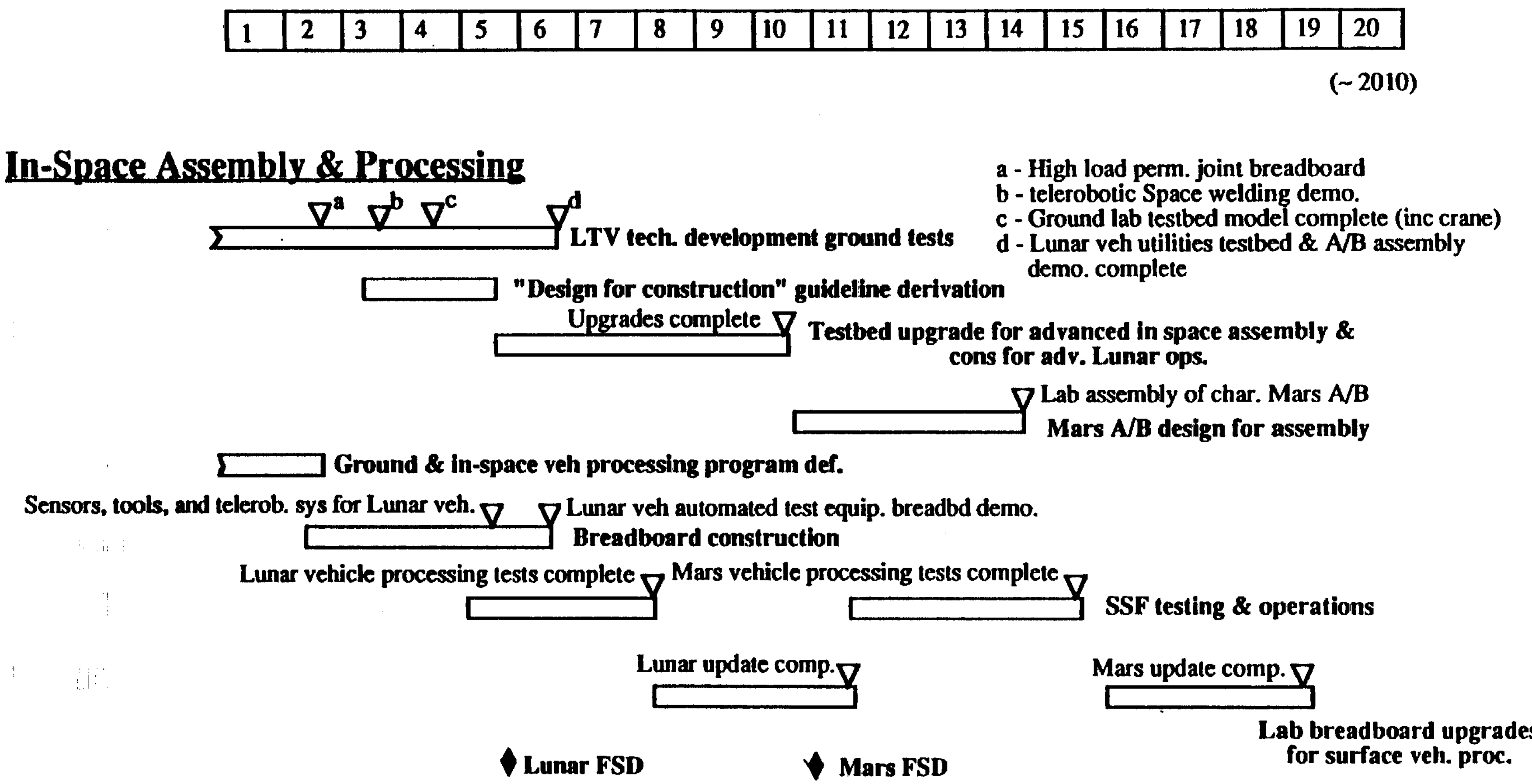
Preliminary SEI Technology Development Schedules (Cont.)



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ADVANCED CIVIL SPACE SYSTEMS

Preliminary SEI Technology Development Schedules (Cont.)

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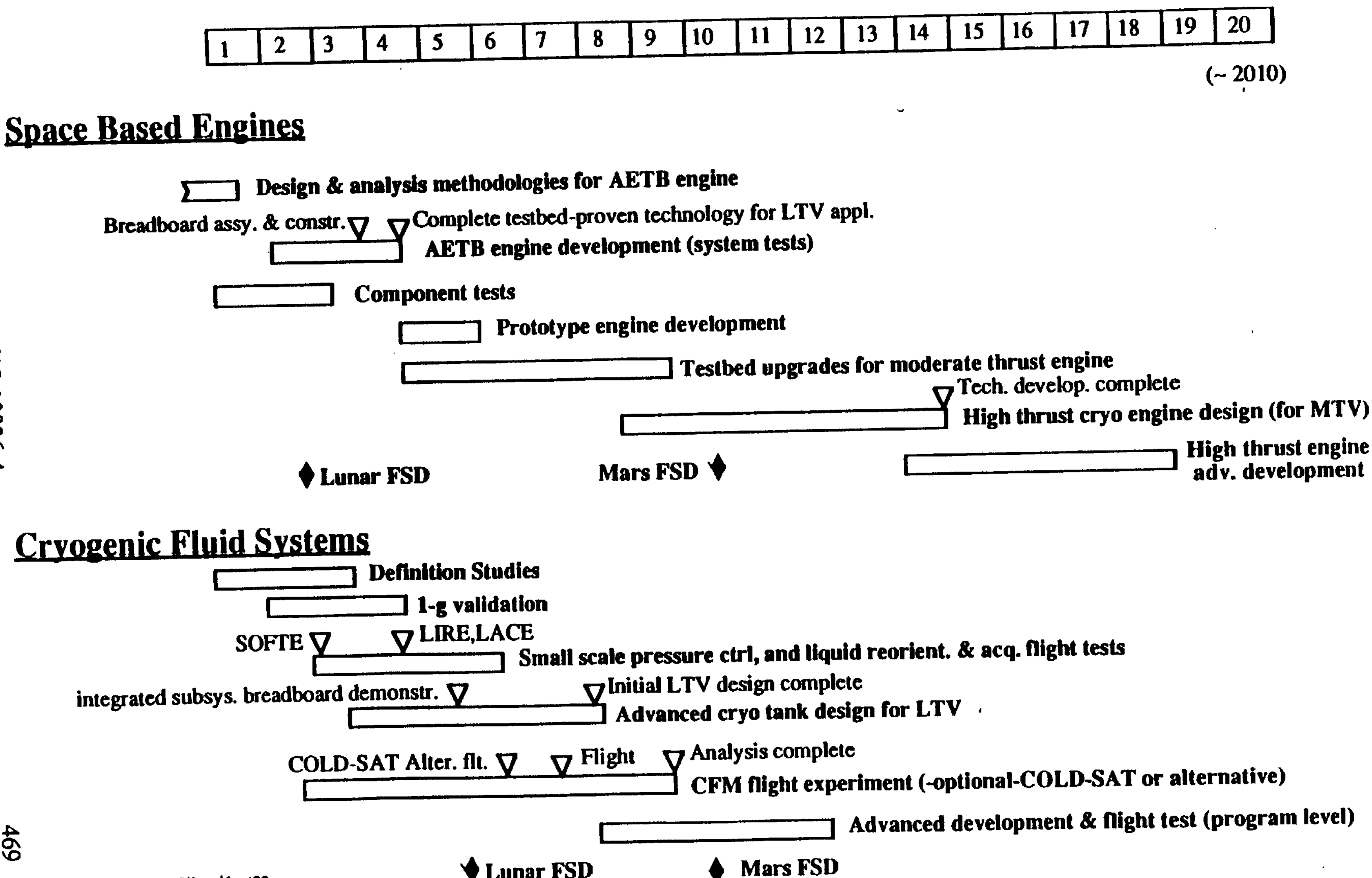

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Facilities

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Facilities

The facility needs have only been identified in this study; the extent of the impact is yet to be determined. A "bona fide" facility development plan has not been done as some of the requirements are only at a top-level needs evaluation. Therefore, the exact nature of the subsystems and their support facilities are undetermined. When these determinations have been made for the final NASA selected vehicle, the results must be integrated with the vehicle development schedule.

In addition to the information here, additional facility and equipment detail is shown in Ground subsection of the Support Systems section of this text. A current listing of the additional required facilities and equipment is shown in the "Special Ground and On-Orbit Processing Facility and Equipment Requirements" chart for processing the advanced vehicles. These requirements will impact the volumes shown for assembly, storage, and launch processing in the "Facilities Requirements" chart as well as the processing time shown in the "Assembly Time per Mission" chart. The information there is for the baseline Cryo/Aerobrake vehicle. All impacts will be to increase the processing time and working volumes required. Any facility requirements must be viewed in the light of and incorporated into the National Launch Facility Plan.

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Special Ground and On-Orbit Processing Facility and Equipment Requirements

Facilities/Equipment	NTR	NEP	SEP
<u>Ground</u>			
• Reactor/engine mating and processing facility	X	X	
• Nuclear fuel loading facility	X	X	
• Contaminated materials storage and disposal facility	X	X	
• Solar array/radiator packing and storage facility	X	X	X
• Alkali metals materials and transferring facility		X	
• Radiation/hazardous materials contamination treatment facility	X	X	
• Robotics to handle radioactive fuels and hazardous chemicals/materials and components	X	X	
• Vehicle truss processing and packaging facility	X	X	X
<u>On-Orbit</u>			
• On-orbit robotic welding and certification equipment	X	X	
• On-orbit alkali metal heating capability		X	
• On-orbit robotic repair/maintenance equipment	X	X	X

Facility Requirements

	Assembly Volume	Storage Volume	Launch Processing
1	20694.13	0	0
2	20694.13	0	0
3	42233.11	0	0
4	56989.01	0	0
5	69879.77	10129.05	0
6	54623.87	10129.05	0
7	39222.88	25031.66	4626.85
8	39222.88	25031.66	0
9	49351.93	14902.61	0
10	20694.13	25031.66	18528.75
11	20694.13	34296.04	0
12	20694.13	34296.04	0
13	20694.13	25031.66	9264.38
14	39481.26	25031.66	0
15	39481.26	25031.66	0
16	0	25031.66	16912.13
17	18528.75	25031.66	0
18	18528.75	10129.05	0
19	0	25031.66	18528.75
20	0	34296.04	0
21	0	34296.04	0
22	0	25031.66	9264.38
23	0	25031.66	0
24	0	25031.66	0
25	0	10129.05	14902.61
26	21207.95	10129.05	0
27	21207.95	30387.15	0
28	0	30387.15	21207.95
29	0	30387.15	10129.05
30	0	30387.15	10129.05
31	0	20258.1	10129.05
32	0	20258.1	10129.05
33	0	20258.1	10129.05
34	0	20258.1	10129.05
35	0	10129.05	10129.05
36	0	10129.05	10129.05

Assembly Time per Mission

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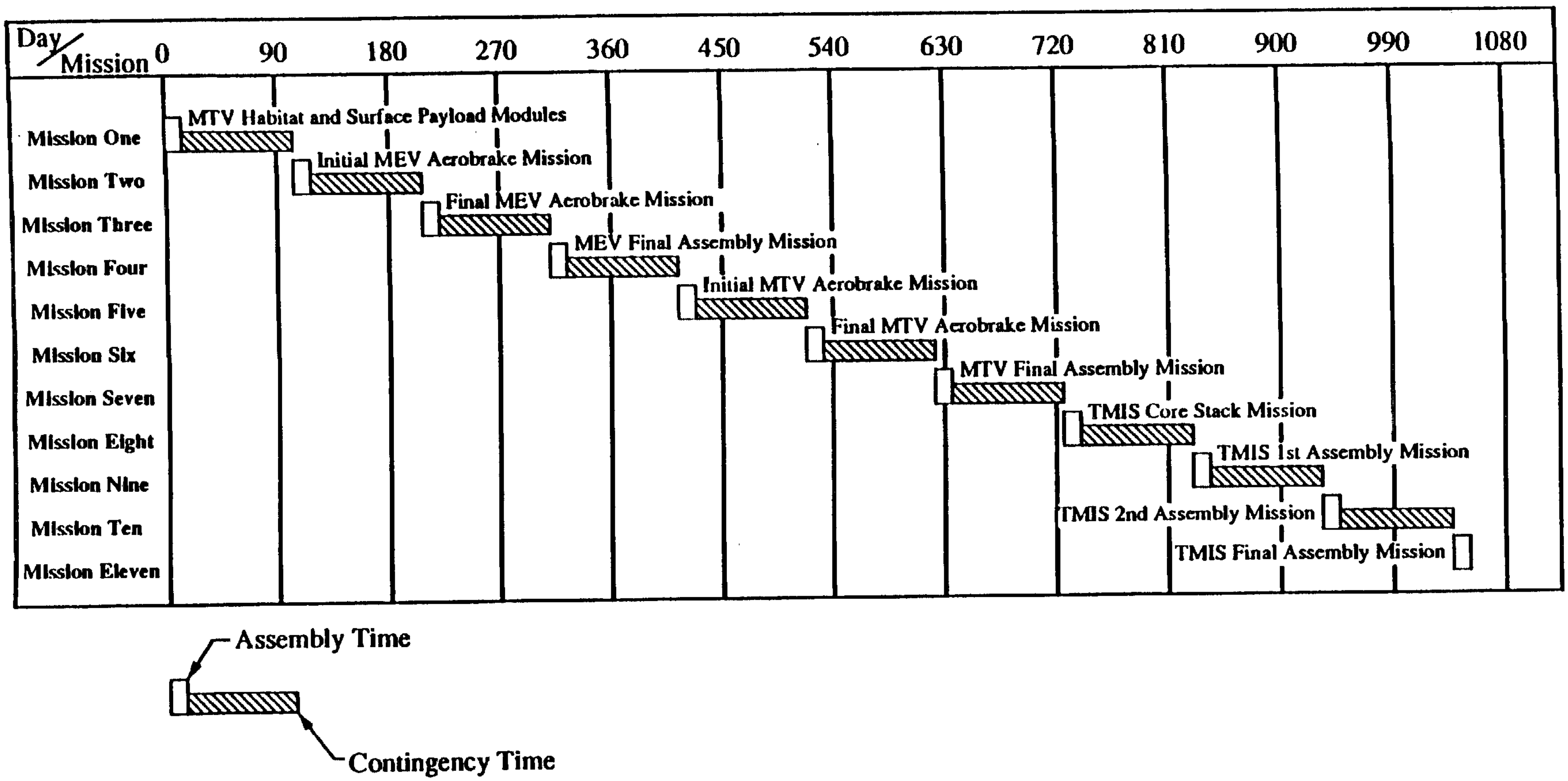

Task 5 Assembly Time - 1

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Costs

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Programmatics

The objectives of the Programmatics task during the current phase of the study were: (1) realistic initial schedules that include initial critical path program elements; (2) initial descriptions of new or unique facilities requirements; (3) development of a stable, clear, responsive work breakdown structure (WBS) and WBS dictionary; (4) initial realistic estimates of vehicle, mission and program costs, cost uncertainties, and funding profile requirements; (5) initial risk analysis, and (6) early and continuing infusion of programmatics data into other study tasks to drive requirements/design/trade decisions.

The issues addressed during the study to date included: (1) capturing all potential long-lead program items such as precursor missions, technology advancement and advanced development, related infrastructure development, support systems and new or modified facility construction, since these are as important as cost and funding in assessing goal achievability; (2) incorporating sufficient operating margin in schedules to obtain high probability of making the relatively brief Mars launch windows; (3) the work breakdown structure must support key study goals such as commonality and (4) cost estimating accuracy and uncertainty are recurring issues in concept definition studies.

Introduction

The study flow, as required by MSFC's statement of work, began with a set of strawman concepts, introduced others as appropriate, conducted "neckdowns", and concluded with a resulting set of concepts and associated recommendations.

As the study progressed, much discussion among the SEI community centered on "architectures". In this study, architectures were more or less synonymous with concepts, since the statement of work required that each concept be fully developed including operations, support, technology, and so forth.

We started with ten concepts as shown in the "Overall Study Flow" chart. After the "neckdown" was completed, significant effort was put into programmatics.

As was indicated earlier, we established three levels of activity to evaluate in-space transportation options. The minimum was 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 median (full science) program aimed at satisfying most of the published science objectives for Lunar and Mars exploration. The maximum program aimed for industrialization of the Moon, for return of practical benefits to Earth, and for the beginnings of colonization of Mars. The range of activity levels, as measured by people and materiel delivered to planetary surfaces, was 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. The high level schedules developed for these three levels of activity are shown in the "Minimum Program", "Full-Science Program" and " Industrialization and Settlement Program" and a comparison of them for both Lunar and Mars is shown in the "Lunar Program Comparison" and "Mars Program Comparison" charts.

Schedule/Network Development Methodology

A PC system called Open Plan by WST Corporation was used, which allows direct control and lower cost over a larger (mainframe) system. The network was purposely kept simple. Summary activities were used in development of the networks. When detailed to a lower level, some activities will require a different calendar than we used. One calendar with a five day work week - no holiday was used. Utilizing multicalendars on a summary network could confuse the development. The Preliminary WBS Structure Level 7 was followed for selection of work to be detailed. An example of Level 7 is: MEV Ascent Vehicle Structure/Mechanisms. We then developed a generic logic string of activities with standard durations for like activities. This logic was then applied against each WBS Level 7 element. To establish interface ties between logic strings and determination of major events, we used the Upper Level Summary Schedule and Summary Level Technology Schedule.

Goals/Purpose

There were two goals for the schedule/network development. These were:

a. Guidelines for Future Development. The schedules are a preliminary road map to follow in the development program.

b. Layout Basis Framework for Network. The networks can be used for future detail network development. This development can be in phases retaining unattended logic for areas which can be be detailed.

Status

Six preliminary networks have been developed. They are:

- Lunar minimum
- Lunar full science
- Lunar industrialization
- Mars missions
- Mars full science
- Mars settlement

These networks will be further developed as information becomes available The technology development plan schedules are shown in the Schedules section of this text ; an example of the standard 6 year program phase C/D schedule is shown in the " Reference 6 yr. Full Scale Development Schedule" chart. The network schedules developed during the study are available in the Final Report Costs Data Book.

Facilities

The facility requirements and approaches are discussed in the Facilities section of this text.

Development Implementation

The integrated technology advancement and full-scale development schedules for the NEP is shown in the "NEP Development Program". The MEV is developed according to the above mentioned standard 6-year FSD schedule. The Man-rating schedules for critical systems, that must be accomplished before first flight, are given in the next several man-rating charts. The long-duration Mars Tansit Habitat, and its critical subsystems, will require operational testing in space to qualify for the Mars mission. How all development and testing is actually done depends on program interrelationships between lunar and Mars missions.

Work Breakdown Structure

The approach to developing a WBS tree and dictionary was to use the Space Station Freedom Work Package One WBS as a point of departure to capture commonality, modularity and evolution potentials. We worked with MSFC to evolve the WBS illustrated in the six WBS charts presented in this section. The WBS dictionary details are provided with the WBS tree in a separate deliverable document.

Cost Data

Overall Approach

Space transfer concept cost estimates were developed through parametric and detail estimating techniques using program/scenario plans and hardware and software descriptions combined with NASA and subcontractor data. Our estimating approach simulates the aerospace development and production environment. It also reflects program options not typical of aerospace programs. This flexibility allows assessment of innovative program planning concepts.

Several tools were employed in this analysis. For developing estimates the Boeing Parametric Cost Model (PCM) designed specifically for advanced system estimating was used. It utilizes a company-wide, uniform computerized data base containing historical data compiled since 1969. The second major tool is a Boeing developed Life Cycle Cost Model. The third tool is the Boeing developed Return on Investment (ROI) Analyses.

The approach to cost estimating was to use the PCM to establish DDT&E and manufacturing cost of major hardware components or to use other estimates, (e.g. Nuclear Working Group estimator) if they were considered superior and then feed them to the LCC model. Variations on equipment hardware or mission alternatives can be run through the LCC and then compared for a return on investment. This flow is illustrated in the "Costing Methodology Flow" chart. We were able to investigate alternative concepts quickly, giving system designers more data for evolving scenario/mission responsive concepts. Transportation concepts, trade studies, and "neckdown" efforts were supported by this approach.

Parametric Cost Model

PCM develops cost from the subsystem level and builds upward to obtain total program cost. Costs are estimated from physical hardware descriptions (e.g., weights and complexities) and program parameters (e.g., quantities, learning curves, and integration levels). Known costs are input directly into the estimate when available; the model assesses the necessary system engineering and system test efforts needed for integration into the program. The PCM working unit is man-hours, which allows relationships that tie physical hardware descriptions first to design engineering or basic factory labor, and then through the organizational structure to pick up functional areas such as systems engineering, test, and development shop. Using man-hours instead of dollars for estimating relationships enables more reliable estimates. The PCM features, main inputs, and results are shown in the "Boeing Parametric Cost Model (PCM)" chart. The applicable PCM results, in constant 1990 dollars, are then put into the Life Cycle Cost Model to obtain cost spreads for the various missions/programs. The various hardware components costed for the three different missions/programs are shown in the "LCCM Hardware Assignments" chart .

The development of space hardware and components needed to accomplish the three different Lunar/Mars missions were identified. These components are grouped into three different categories defined below.

HLLV(Heavy Lift Launch Vehicle) is the booster required to lift personnel, cargo and fuels into LEO and support the LEO node operations.

Propulsion Includes the space propulsion system required to transfer people, cargo and equipment out of LEO and into space. Space means Lunar, Mars and Earth destinations. Propulsion Systems also include an all-propulsive cryogenic Trans Mars Injection System (TMIS) for the Minimum Mission, the Nuclear Electric Propulsion Stage for the Settlement/Industrial Missions.

Modules Include the space systems that are required to transfer people, cargo and equipment from LEO to Lunar and Mars orbit; to de-orbit and sustain life and operations on the Lunar and Mars Surface; and, finally, to return personnel and equipment to LEO.

Cost Buildups

The PCM cost Model can be used directly to obtain complete DDT&E cost, including production of major test articles, by entering into the manufacturing section the equivalent numbers of units for each item, including the first flight article. However, when operated in this way, PCM does not give the first unit cost. To save time, we operated PCM so as to give first unit cost, which we needed for life cycle cost analyses, and used the first unit cost to manually estimate the test hardware content of the DDT&E program. The "wrap factors" shown in the cost buildup sheets were derived from the PCM runs as the factor that is applied to design engineering cost to obtain complete design and development costs, e.g. including non-recurring items such as systems engineering and tooling development.

Life Cycle Cost Model

The LCCM cost data is a composite of HLLV costs, launch base facilities cost estimate based on $/sq. ft. and parametric estimates derived from the Parametric Cost Model. The principal source of information is from the PCM. All hardware cost estimates, with the exception of HLLV, have been developed with this model.

The LCCM consists of three individual models. One model is for the Minimum Program Scale; the second is for the Full Science Program Scale; while the third model is for the Settlement/Industrialization Program Scale. The Minimum Program meets the President's Space Exploration Initiative (SEI) objectives. These capabilities include permanent Lunar facilities but not permanent human presence and three missions to Mars. The Full Science program not only meets the President's SEI objectives but also provides for long term bases for far-ranging surface exploration. The Settlement/Industrialization program accomplishes the objectives of the Minimum and Full Science program scales and additionally returns practical benefits to Earth. These models were developed using the three architecture levels described in the Boeing manifest worksheets. Total cost for each system are tabulated by year and each year's totals feed into a summary sheet that calculates the total program cost for each level. Since the LCCM results are mission related, not just vehicle related, they are not provided here but are available in the Final Report Cost Data Book. The LCCM was developed using Microsoft Excel version 2.2 for the Macintosh computer. Any Macintosh equipped with Excel 2.2 can be used to execute the model.

Return On Investment

One of the principal uses of the LCCM is to develop trades and return on investment for technology options. As shown in the "Costing Methodology Flow" chart , two separate life cycle cost models (which include DDT&E and production cost data derived from the parametric cost models) must be developed for each ROI case; a reference, and a case utilizing a technology option. The two life cycle cost streams are separately entered, and the ROI model is executed. The flow also illustrates that not all of the data entered into the life cycle cost model is derived from available costing software. Technical analysis must accompany this data. For example, the number of units which must be produced for the DDT&E program must be determined. This is done at the subsystem level based on knowledge of past programs, and proposed system/subsystem tests. Since the ROI analysis is mission related, not just vehicle related, the data is not presented here but is available in the Final Report Cost Data Book.

Results

It should be noted that the solar array and ion thruster costs for the NEP are not included in the PCM results but are included in the the cost build ups. A summary of the cost data produced by the PCM for the vehicle are given in the "Mars SEP Preliminary PCM Summary" and "Mars SEP Preliminary PCM Summary - continued" charts. The PCM program was used to produce DDT&E and production cost estimates for each of our reference Mars and lunar vehicles to the subsystem level. The DDT&E costs generated by the PCM do not include all of the necessary hardware for the first mission vehicle. Hence all necessary additional units (prototypes,test units, lab units, etc.) were added into the vehicle cost buildups as shown in the three separate "SEP Cost Buildup" chart tables. These three figures represent costs for solar array efficiencies of $100/watt, $500/watt and $1000/watt respectively. As shown the total DDT&E includes additional costs(e.g.. additional units in the DDT&E program), contractor fees and the engineering wrap factor. The total DDT&E from the cost buildup and the unit cost from the PCM are the primary vehicle cost inputs to the LCC model

Risk Analyses

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

Development Risk

All of the architectures and technologies investigated 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 involve a great many layers of multi-layer insulation (MLI), and one or more vapor-cooled shields. Analyses and experiments have indicated the efficacy of these, but demonstration that such insulation systems can be fabricated at light weight, capable of surviving launch g and acoustics loads, remains to be accomplished. In addition, there are issues associated with propellant transfer and zero-g gauging. These, however, can be avoided for early lunar systems by proper choice of configuration and operations, e.g. the tandem-direct system recommended elsewhere in this report. This presents the opportunity to evolve these technologies with operations of initial flight systems.

Engines - There is little risk of being able to provide some sort of cryogenic engine for lunar and Mars missions. The RL- 10 could be modified to serve with little risk; deep throttling of this engine has already been demonstrated on the test stand. The risk of developing more advanced engines is also 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. The "Development Risk Assessment for Aerobraking by Function" chart provides a qualitative development risk comparison for these six functions.

Aerocapture of vehicles requires large aerobrakes. For these to be efficient, low mass per unit area is required, demanding efficient structures made from very high performance materials as well as efficient, low mass thermal protection materials. By comparison, the crew capsules benefit much less from high performance structures and TPS.

Launch packaging and on-orbit assembly of large aerobrakes presents a significant development risk that has not yet been solved even in a conceptual design sense. Existing concepts package poorly or are difficult to assemble or both. While the design challenge can probably be met, aerobrake assembly is a difficult design and development challenge, representing an important area of risk.

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 are recommending 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.

Reactor and engine tests during the 1960s jetted hot, slightly radioactive hydrogen directly into the atmosphere. Stricter environmental controls since that time prohibit discharge of nuclear engine effluent into the atmosphere. Design and development of full containment test facilities presents a greater development risk than obtaining the needed performance from nuclear reactors and engines. Full- containment facilities will be required to contain all the hydrogen effluent, presumably oxidize it to water, and remove the radioactivity.

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. Minimizing mass of power distribution leads to high distribution voltage and potential problems with plasma losses, arcing, and EMI. Thus while power management is a mature technology, the unique requirements of electric propulsion introduce a number of development risks beyond those usually experienced in space power systems.

Electric thruster technology has been under development since the beginning of the space program. Small thrusters are now operational, such as the resistance-heat-augmented hydrazine thrusters on certain communications spacecraft. Small arc and ion thrusters are nearing operational use for satellite stationkeeping.

Space transfer demands on electric propulsion performance place a premium on high power in the jet per unit mass of electric propulsion system. This in turn places a premium on thruster efficiency; power in the jet, not electrical power, propels spaceships. Space transfer electric propulsion also requires specific impulse in the range 5000 to 10,000 seconds. Only ion thrusters and magnetoplasmadynamic (MPD) arc thrusters can deliver this performance. Ion thrusters have acceptable efficiency but relatively low power per unit of ion beam emitting area. MPD thruster technology can deliver the needed Isp with high power per thruster, but has not yet reached efficiencies of interest. Circular ion thrusters have been built up to 50 cm diameter, with spherical segment ion beam grids. These can absorb on the order of 50 kWe each. A 10 MWe system would need 200 operating thrusters. 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 - Space power reactor technology now under development (SP-100) may be adequate; needed advances are modest. Advanced power conversion systems are required to obtain power-to-mass ratios of interest. The SP-100 baseline is thermoelectric, which has no hope of meeting propulsion system performance needs. The most likely candidates are the closed Brayton (gas) cycle and the potassium Rankine (liquid/vapor) cycle. (Potassium provides the best match of liquid/vapor fluid properties to desired cycle temperatures.) Stirling cycle, thermionics, and a high-temperature thermally-driven fuel cell are possibilities. The basic technology for Brayton and Rankine cycles are mature; both are in widespread industrial use. Prototype space power Brayton and Rankine turbines have run successfully for thousands of hours in laboratories. The development risk here is that these are very 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, at power levels of interest for SEI space transfer application, are large enough to require in-space assembly and checkout. Space welding will be required for fluid systems assembly.

Solar power for space transfer propulsion - Solar power systems for space propulsion must attain much higher power-to-mass ratios than heretofore achieved. This implies a combination of advanced solar cells, probably multi-band-gap, and lightweight structural support systems. 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. Where the nuclear electric systems are high development risk because of complexity and the lack of experience base at relevant power levels and with the space power conversion technologies, 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 techniques such as expert and neural systems are likely to play an important role.

An important factor in avionics and software development is that several vehicle elements having similar requirements will be developed, some concurrently. A major reduction in cost and integration risk for avionics can be achieved by advanced development of a "standard" avionics and software suite, from which all vehicle elements would depart.

Further significant cost savings are expected from advancements in software development methods and environments.

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. The Mars transfer requirement is for a highly closed physio-chemical system capable of 3 years' safe and dependable operation without resupply from Earth. The development risk arises from the necessity to demonstrate long life operation with high confidence; this may be expensive in cost and development schedule.

Man-Rating Approach

Man-rating includes three elements: (1) Design of systems to manned 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. Several briefing charts follow: the first summarizes a recommended approach and lists the subsystems and elements for which man-rating is needed; subsequent charts present recommended man-rating plans.

Mission and Operations Risk

These risk categories include Earth launch, space assembly and orbital launch, launch windows, mission risk, and mitigation of ionizing radiation and zero-g risks.

Earth launch - The Earth launch risk to in-space transportation is the risk of losing a payload because of a launch failure. Assembly sequences are arranged to minimize the impact of a loss, and schedules include allowances for one make-up launch each mission opportunity.

Assembly and Orbital Launch Operations - Four sub-areas are covered: assembly, test and on-orbit checkout, debris, and inadvertent re-entry.

Assembly operations risk is reduced by verifying interfaces on the ground prior to launch of elements. Assembly operations equipment such as robot arms and manipulators will undergo space testing at the node to qualify critical capabilities and performance prior to initiating assembly operations on an actual vehicle.

Assembly risk varies widely with space transfer technology. Nuclear thermal rocket vehicles appear to pose minimum assembly risk; cryo/aerobraking are intermediate, and nuclear and solar electric systems pose the highest risk.

Test and on-orbit checkout must deal with consequences of test failures and equipment failures. This risk is difficult to quantify with the present state of knowledge. Indications are: (1) large space transfer systems will experience several failures or anomalies per day. Dealing with failures and anomalies must be a routine, not exceptional, part of the operations or the operations will not be able to launch space transfer systems from orbit; (2) vehicles must have highly capable self-test systems and must be designed for repair, remove and replace by robotics where possible and for ease of repair by people where robotics cannot do the job; (3) test and on-orbit checkout will run concurrently with propellant loading and launch countdowns. These cannot take place on Space Station Freedom. Since the most difficult part of the assembly, test and checkout job must take place off Space Station Freedom the rest of the job probably should also.

Orbital debris presents risk to on-orbit operations. Probabilities of collision are large for SEI-class space transfer systems in low Earth orbit for typical durations of a year or more. Shielding is mandatory. The shielding should be designed to be removed before orbital launch and used again on the next assembly project.

Creation of debris must also be dealt with. This means that (1) debris shielding should be designed to minimize creation of additional debris, especially particles of dangerous size, and (2) operations need to be rigorously controlled to prevent an inadvertent loss of tools and equipment that will become a debris hazard.

Inadvertent re-entry is a low but possible risk. Some of the systems, especially electric propulsion systems, can have very low ballistic coefficient and therefore rapid orbital decay rate. Any of the SEI space transfer systems will have moderately low ballistic coefficient when not loaded with propellant. While design details are not far enough along to make a quantitative assessment, parts of these vehicles would probably survive reentry to become ground impact hazards in case of inadvertent reentry. For nuclear systems, it will be necessary to provide special support systems and infrastructure to drive the probability of inadvertent reentry to extremely low levels.

Launch Windows - Launch windows for single-burn high-thrust departures from low Earth orbit are no more than a few days because regression of the parking orbit line of nodes causes relatively rapid misalignment of the orbit plane and departure vector. For lunar missions, windows recur at about 9-day intervals.

For Mars, the recurrence is less frequent, and the interplanetary window only lasts 30 to 60 days. It is important to enable Mars launch from orbit during the entire interplanetary window. Three-impulse Mars departures make this possible; a plane change at apogee of the intermediate parking orbit provides alignment with the departure vector. Further analysis of the three-burn scheme is needed to assess penalties and identify circumstances where it does not work.

Launch window problems are generally minimal for low-thrust (electric propulsion) systems.

Mission Risk - Comparative mission risk was analyzed by building risk trees and performing semi-quantitative analysis. The next chart presents a comparison of several mission modes; after that are the risk trees for these modes.

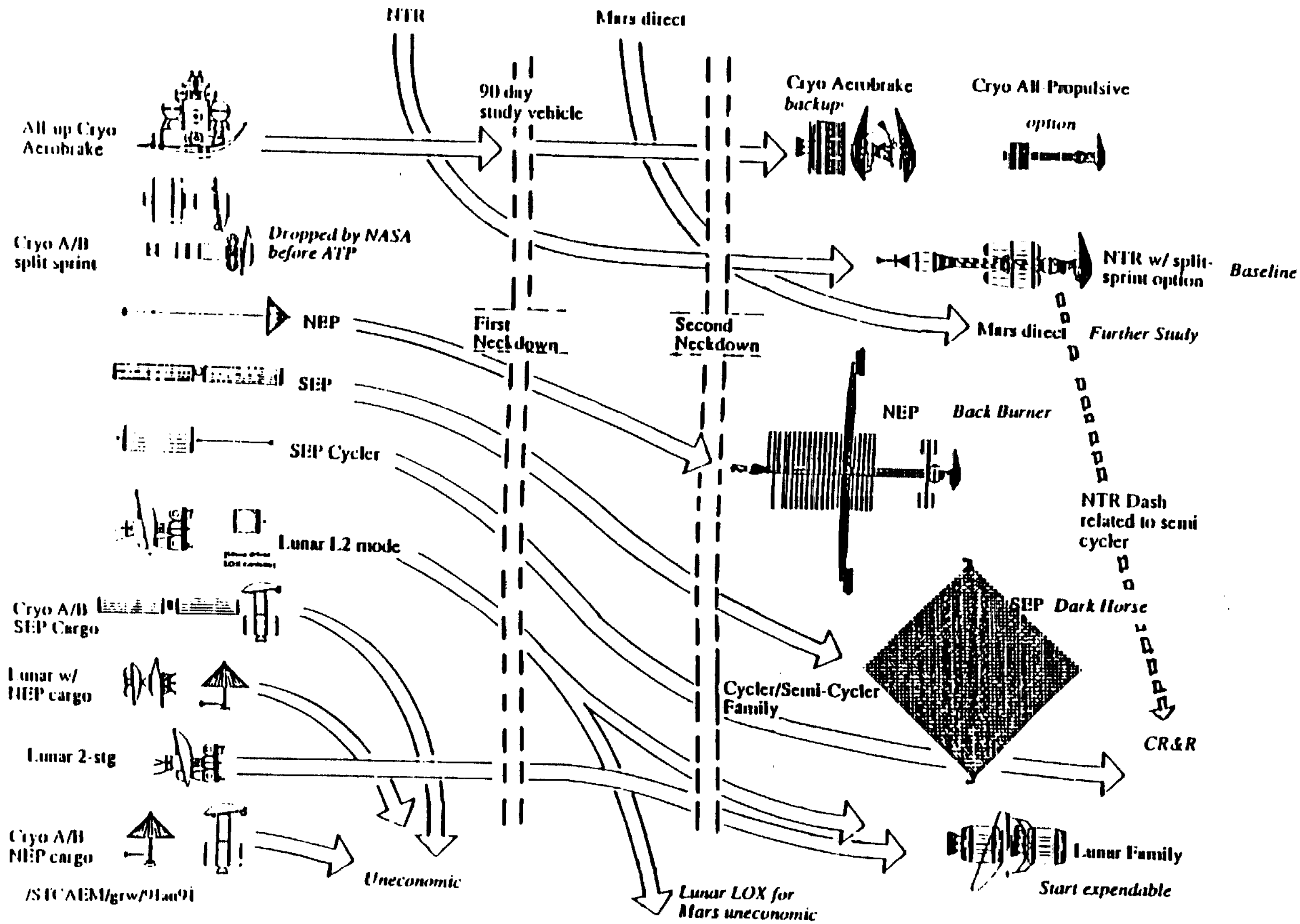
Ionizing Radiations and Zero G - The threat from ionizing radiations is presented elsewhere in this document. Presented here are the mitigating strategies for ionizing radiations and zero g.

Nuclear systems operations present little risk to flight crews. Studies by University of Texas at Austin showed that radiation dose to a space station crew from departing nuclear vehicles is very small provided that sensible launch and flight strategies are used. On-board crews are protected by suitable shielding and by arrangement of the vehicle, i.e. hardware and propellant between reactors and the crew and adequate separation distances. After nuclear engines are shut off, radiation levels drop rapidly so that maneuvers such as departure or return of a Mars excursion vehicle are not a problem. On-orbit operations around a returned nuclear vehicle are deferred until a month or two after shutdown, by which time radioactivity of the engine is greatly reduced.

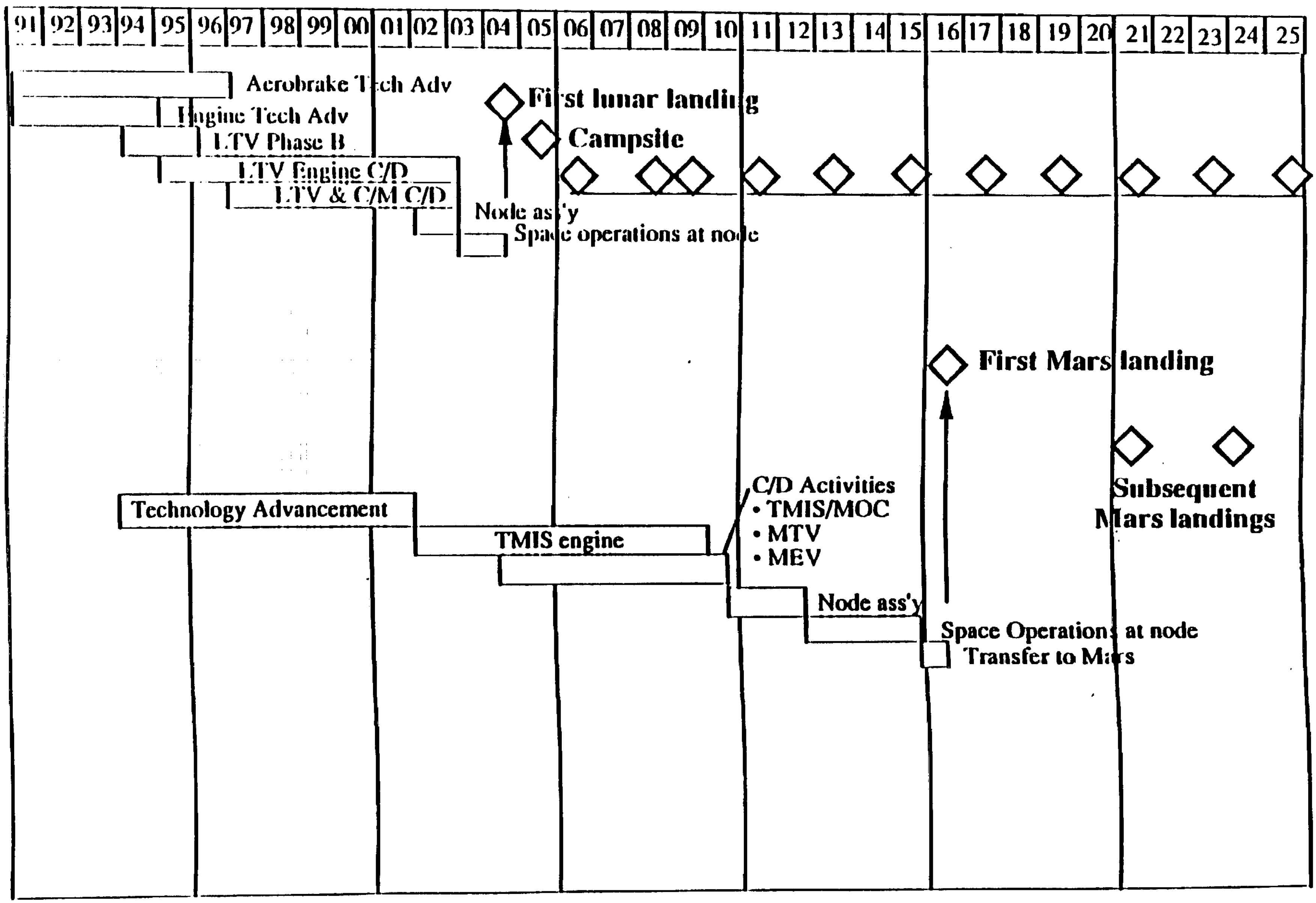
Reactor disposal has not been completely studied. Options include solar system escape and parking in stable heliocentric orbits between Earth and Venus.

Crew radiation dose abatement employs "storm shelters" for solar flares, and either added shielding of the entire vehicle or fast transfers (or both) to reduce galactic cosmic ray exposure. Assessments are in progress; tradeoffs of shielding versus fast trips have yet to be completed. Expected impact for lunar missions is negligible and for Mars missions, modest.

Overall Study Flow



Minimum Program



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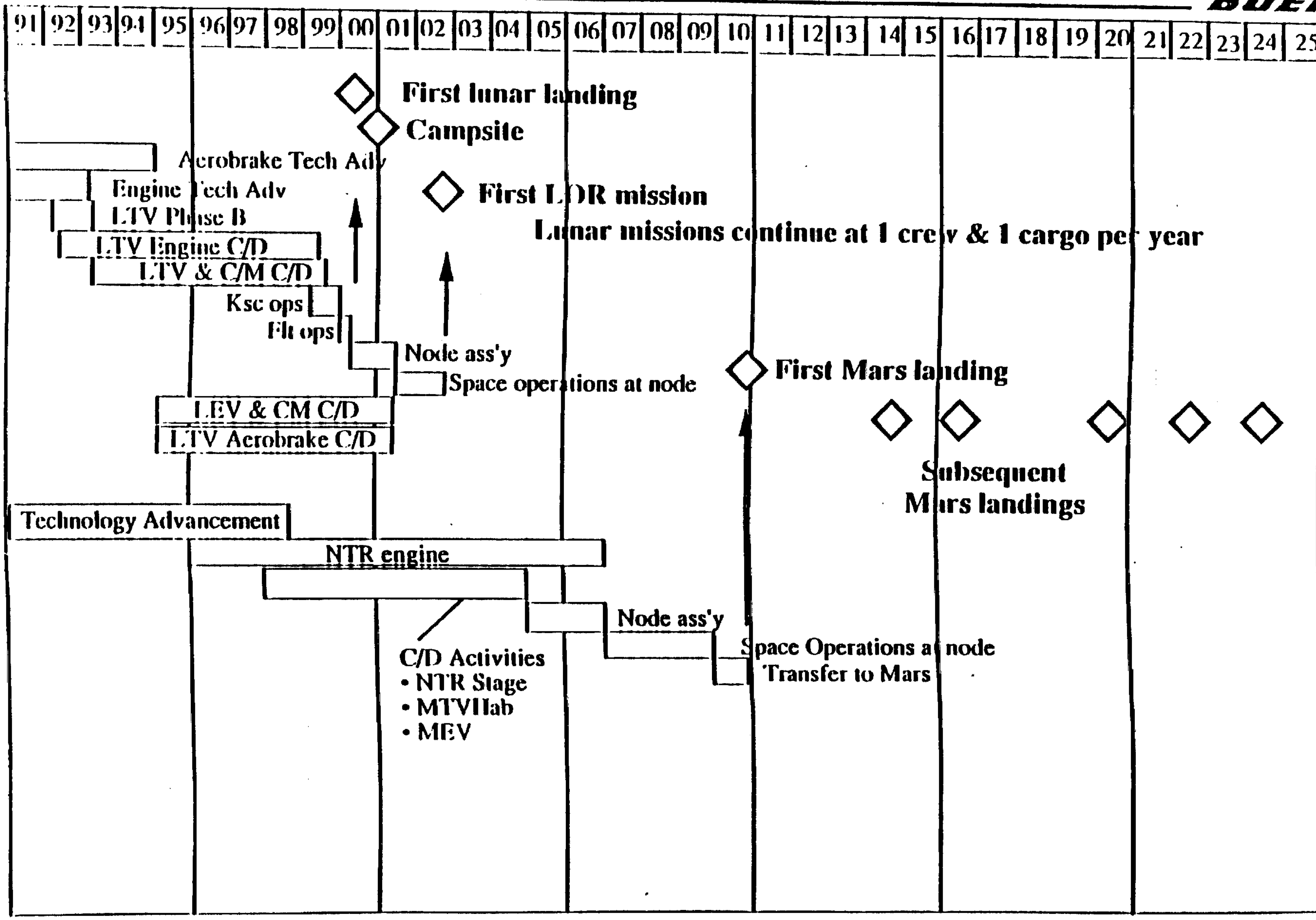
Full Science Program



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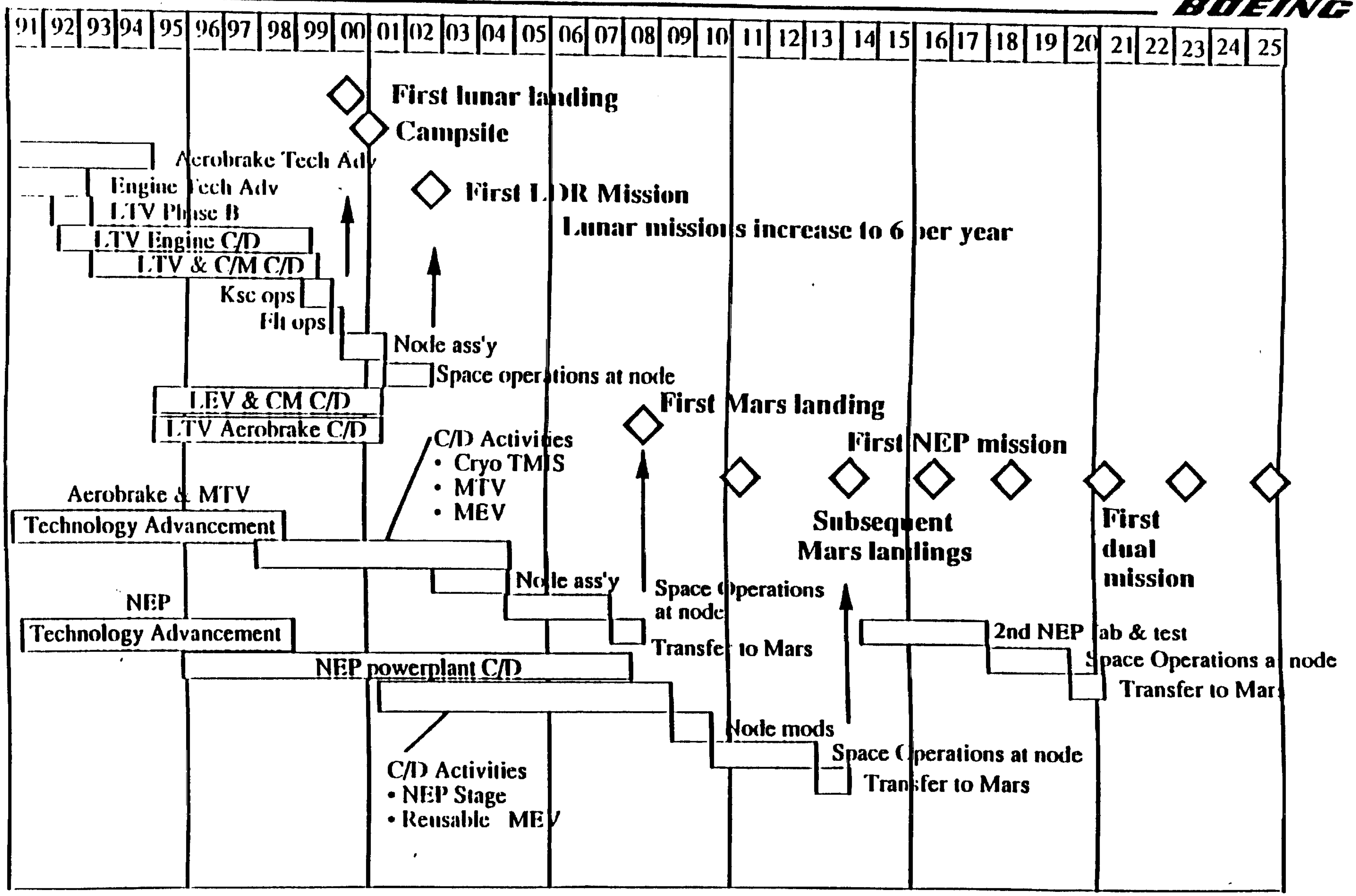
Industrialization and Settlement Program



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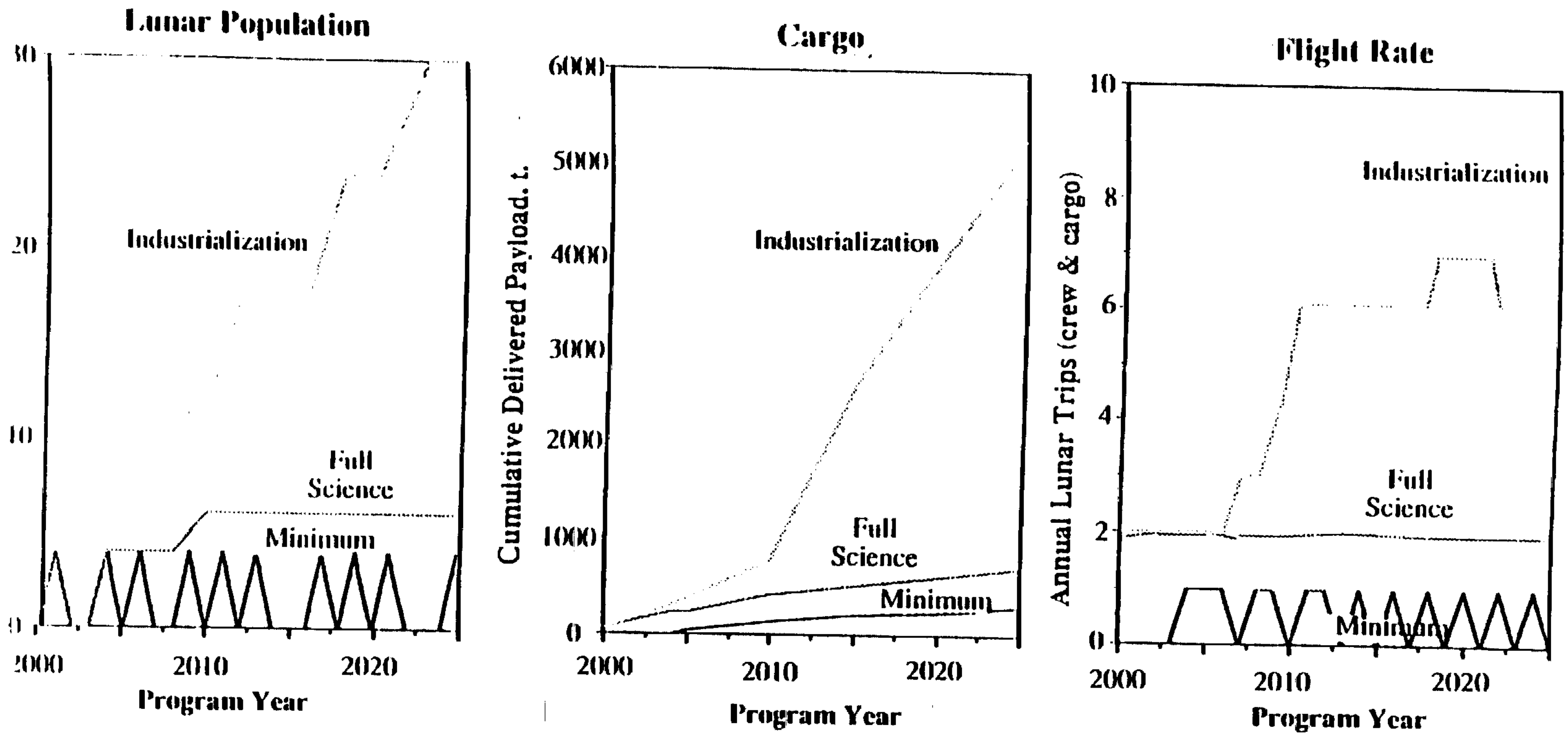
Lunar Program Comparison



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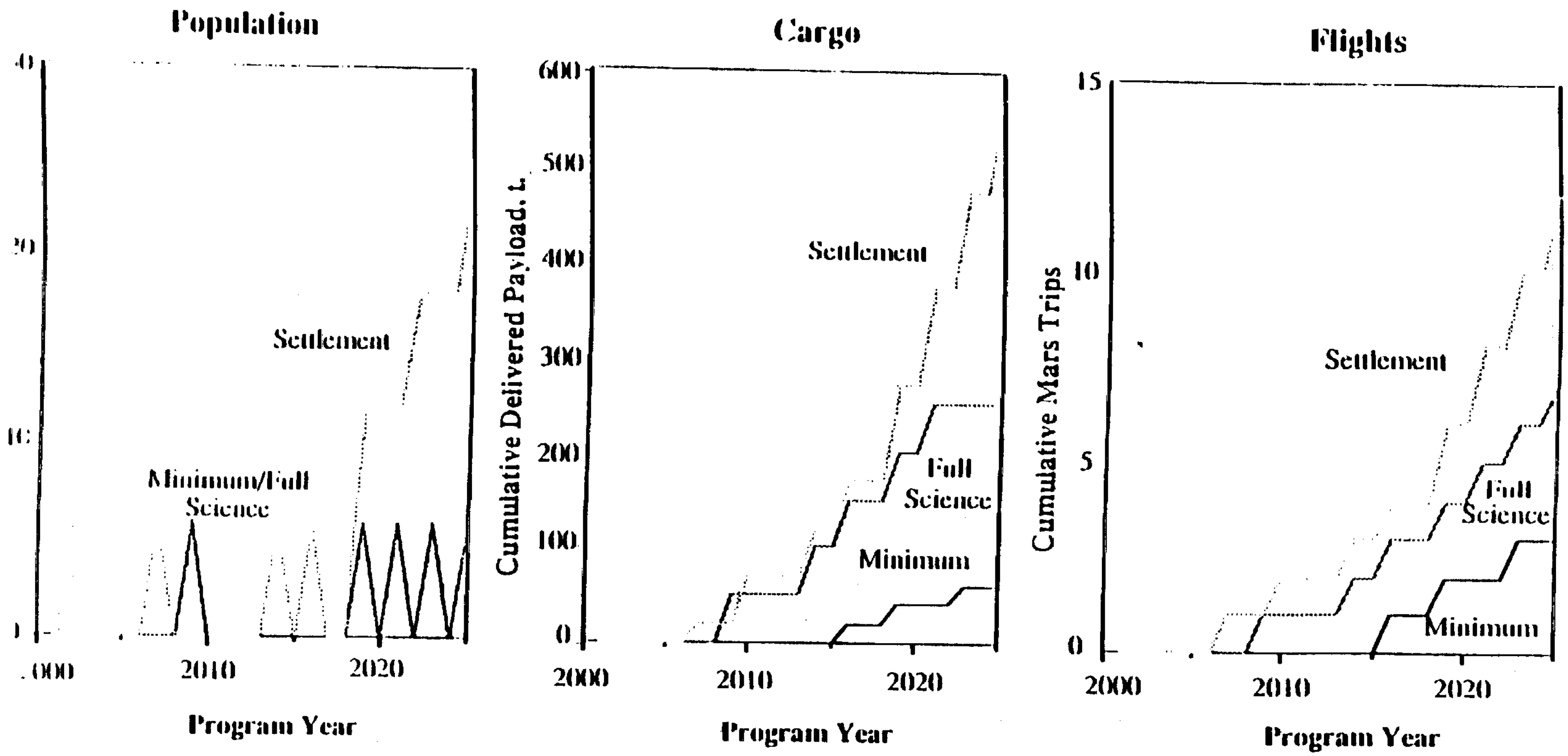

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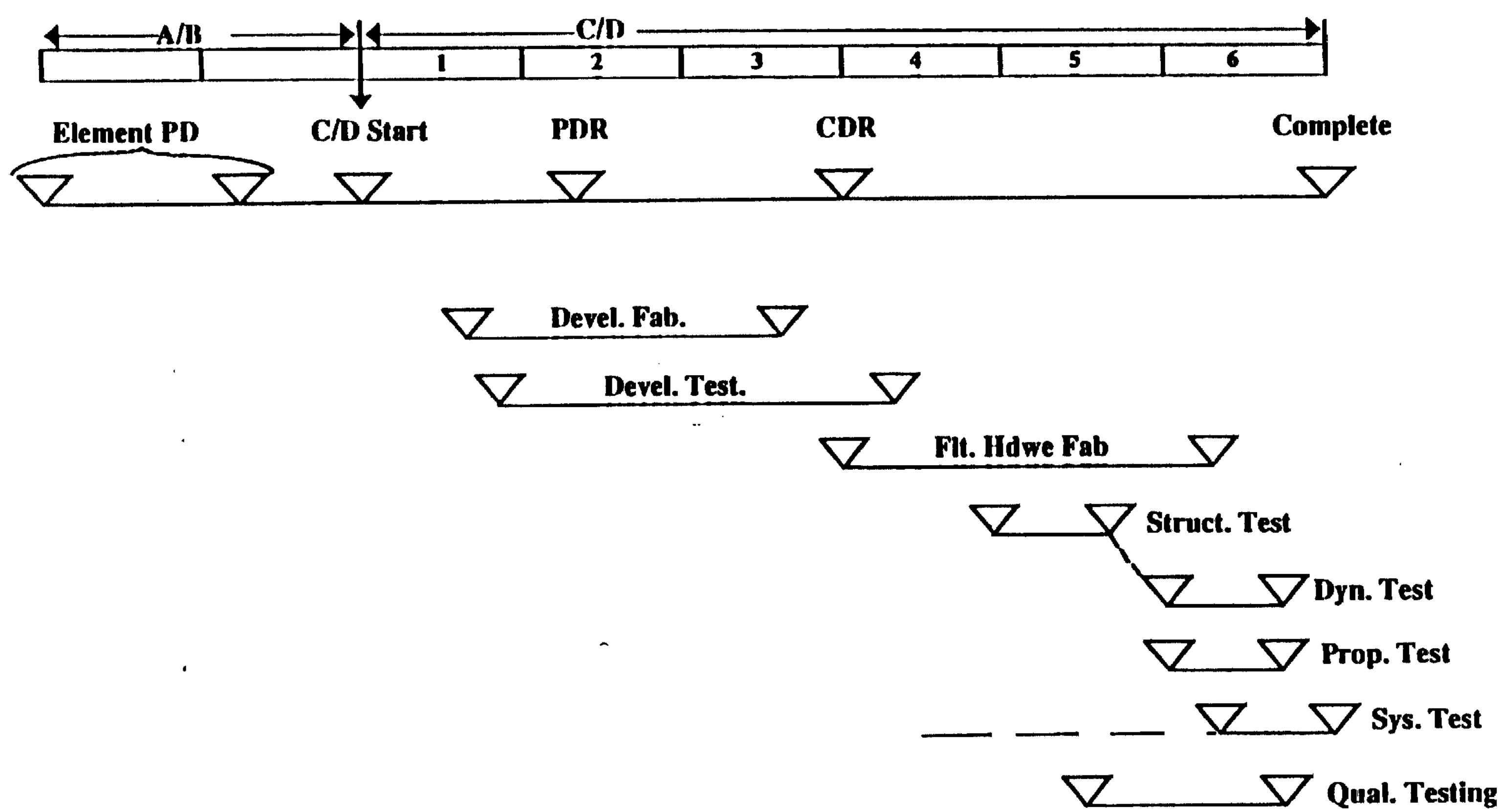
Reference 6 yr Full-Scale Development Schedule

ADVANCED CIVIL SPACE SYSTEMS

BOEING



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ADVANCED CIVIL SPACE SYSTEMS

SEP Development Program

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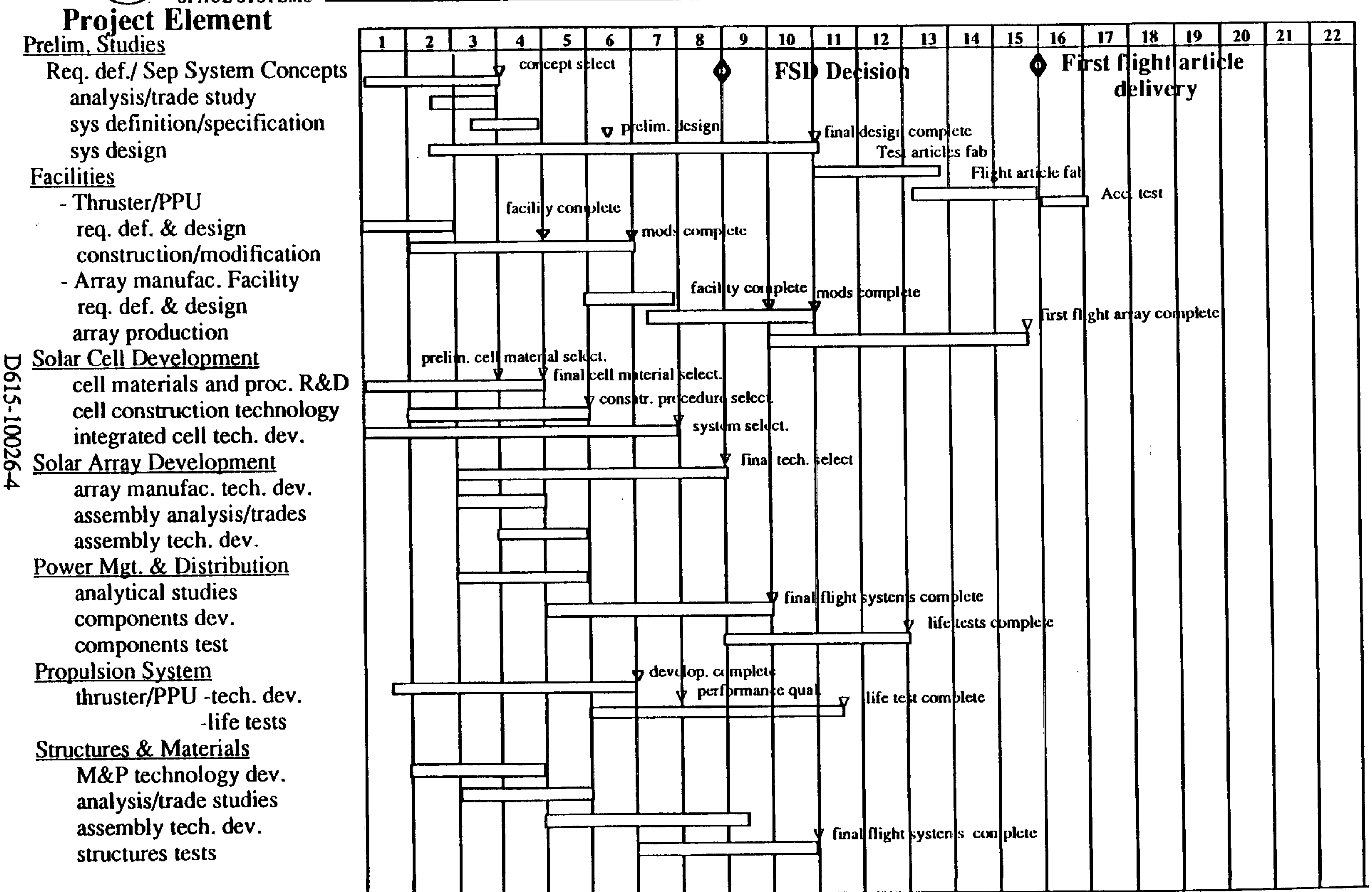
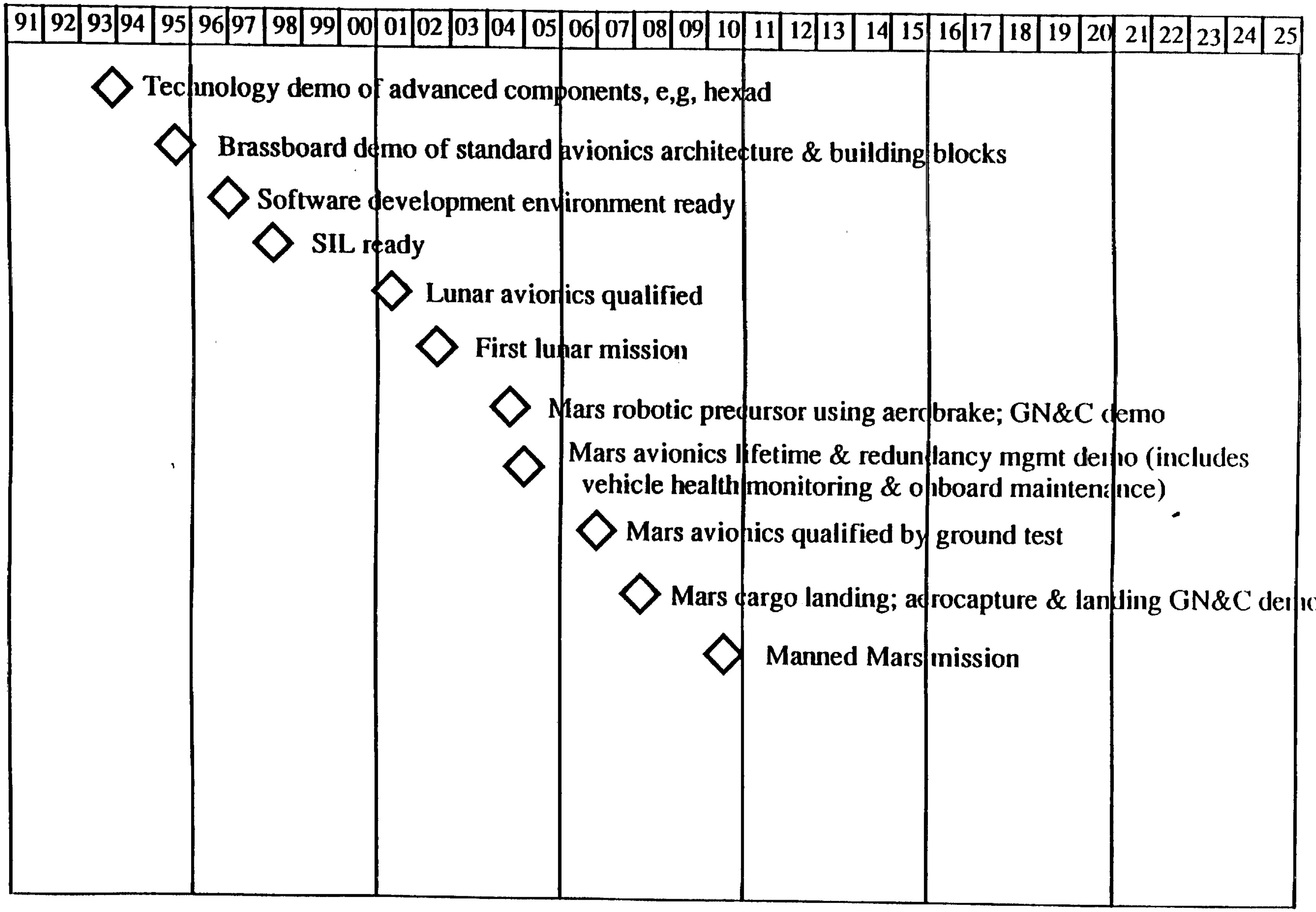
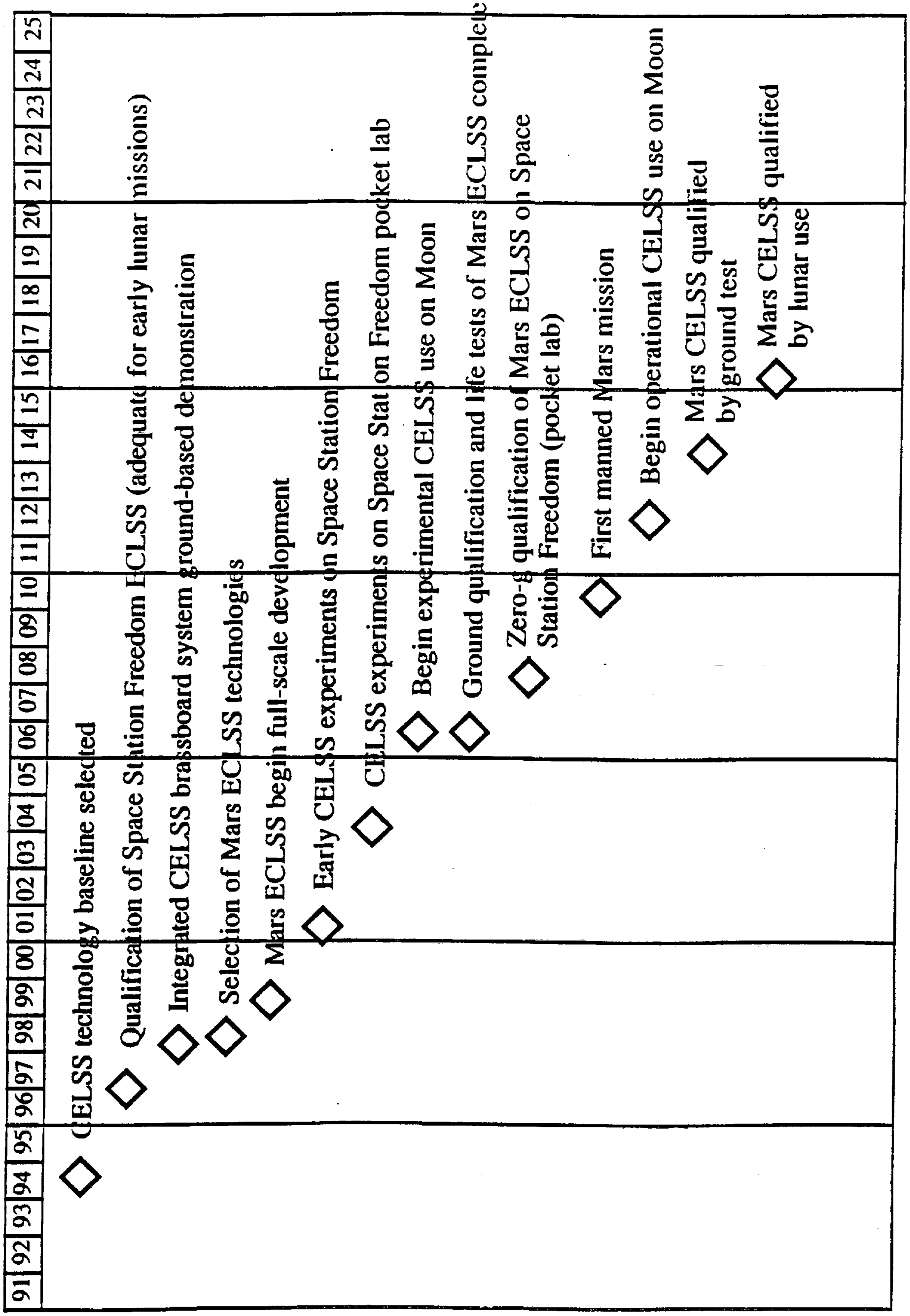


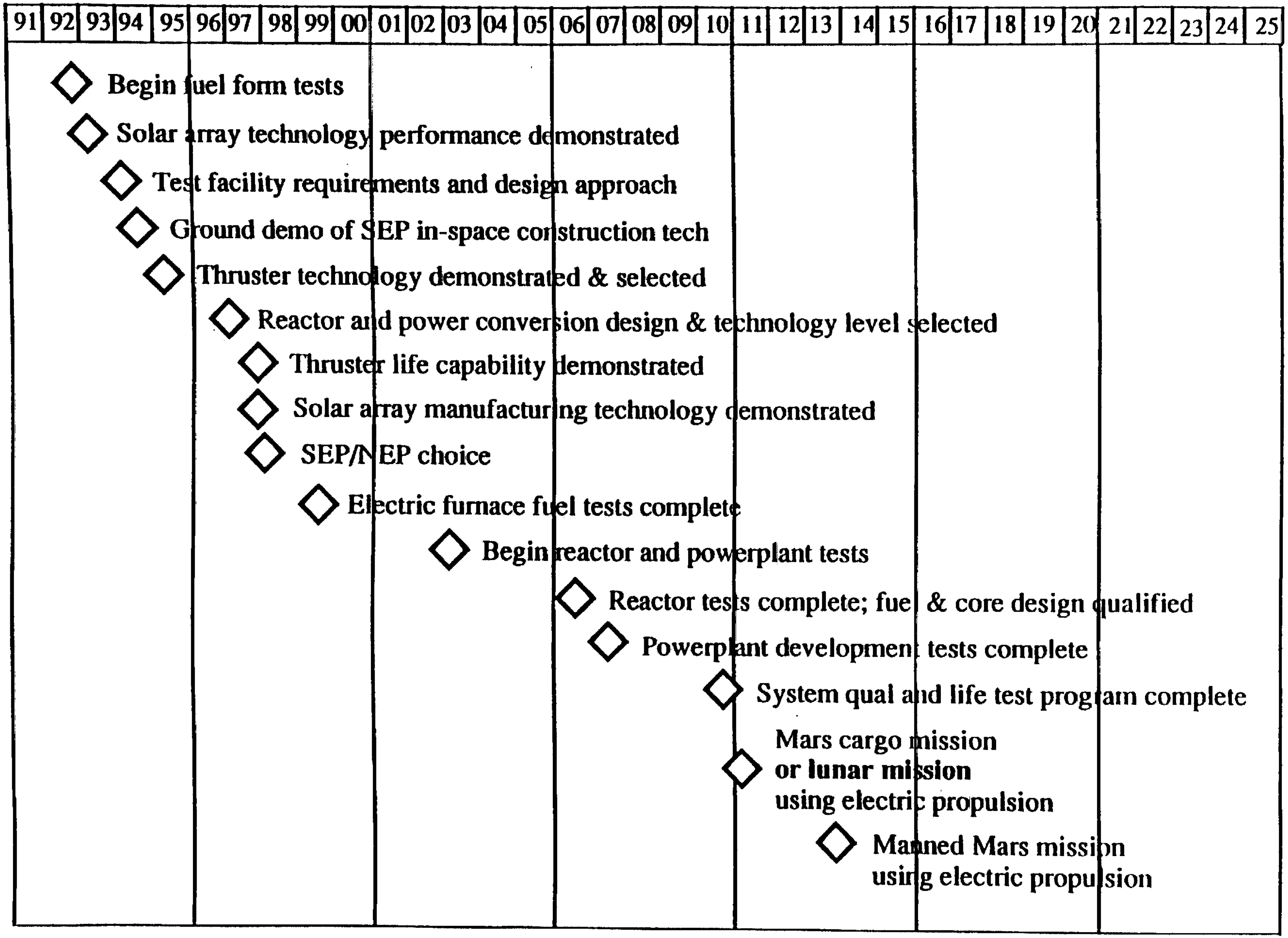
Avionics Major Test/Demo Man-Rating Approach



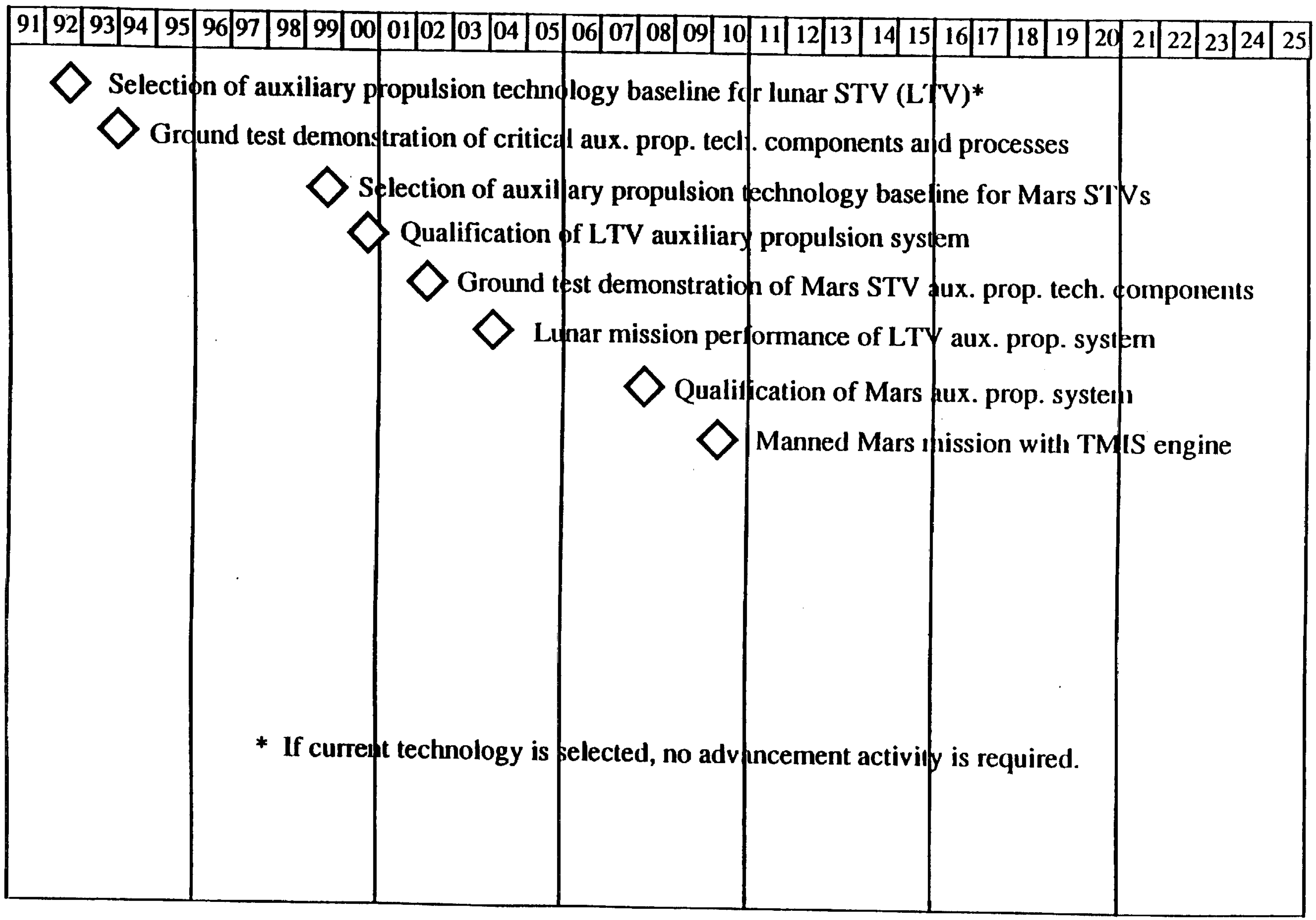
ECLSS Systems Man-Rating Approach



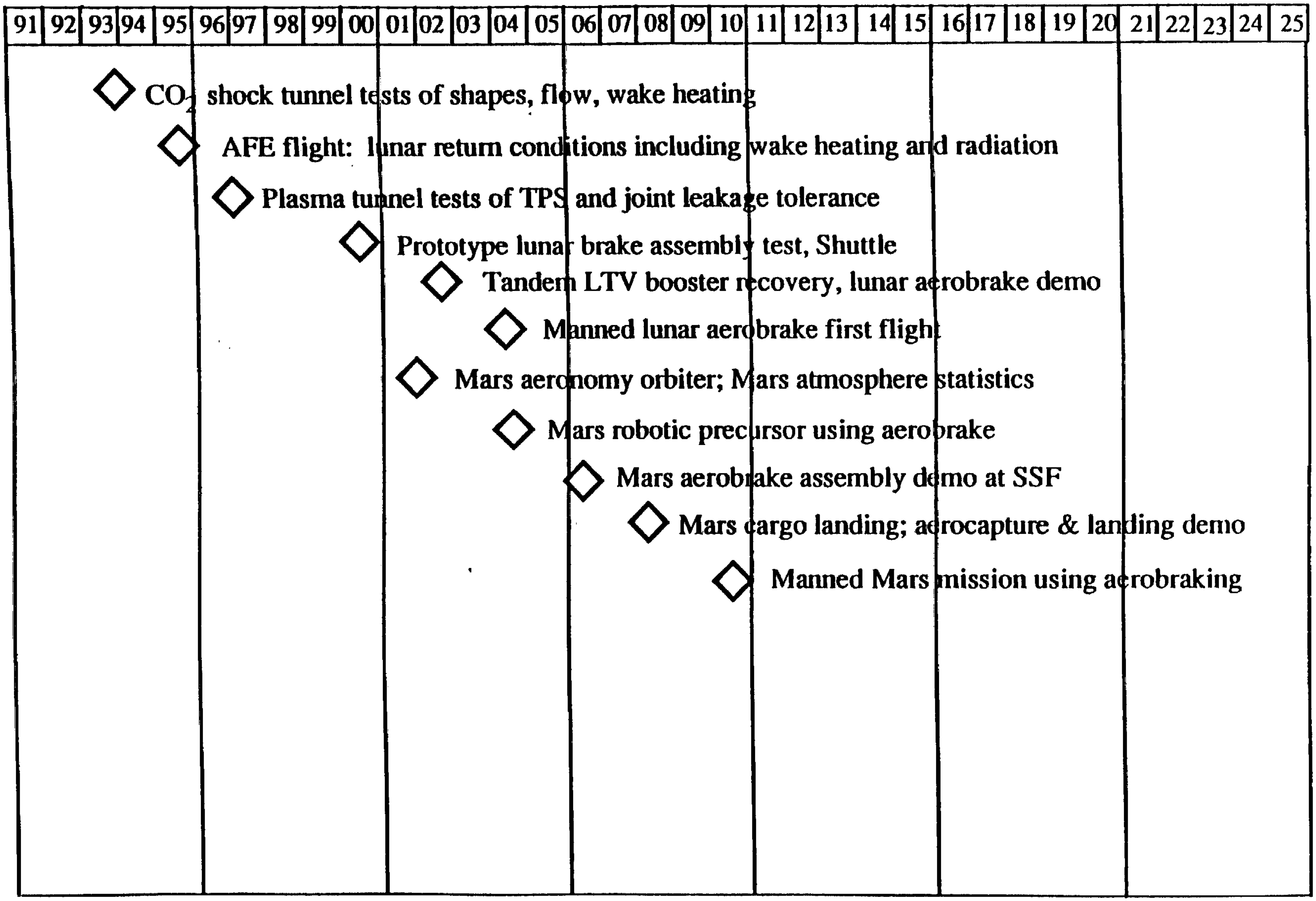
Nuclear and Solar Electric Propulsion Man-Rating Approach



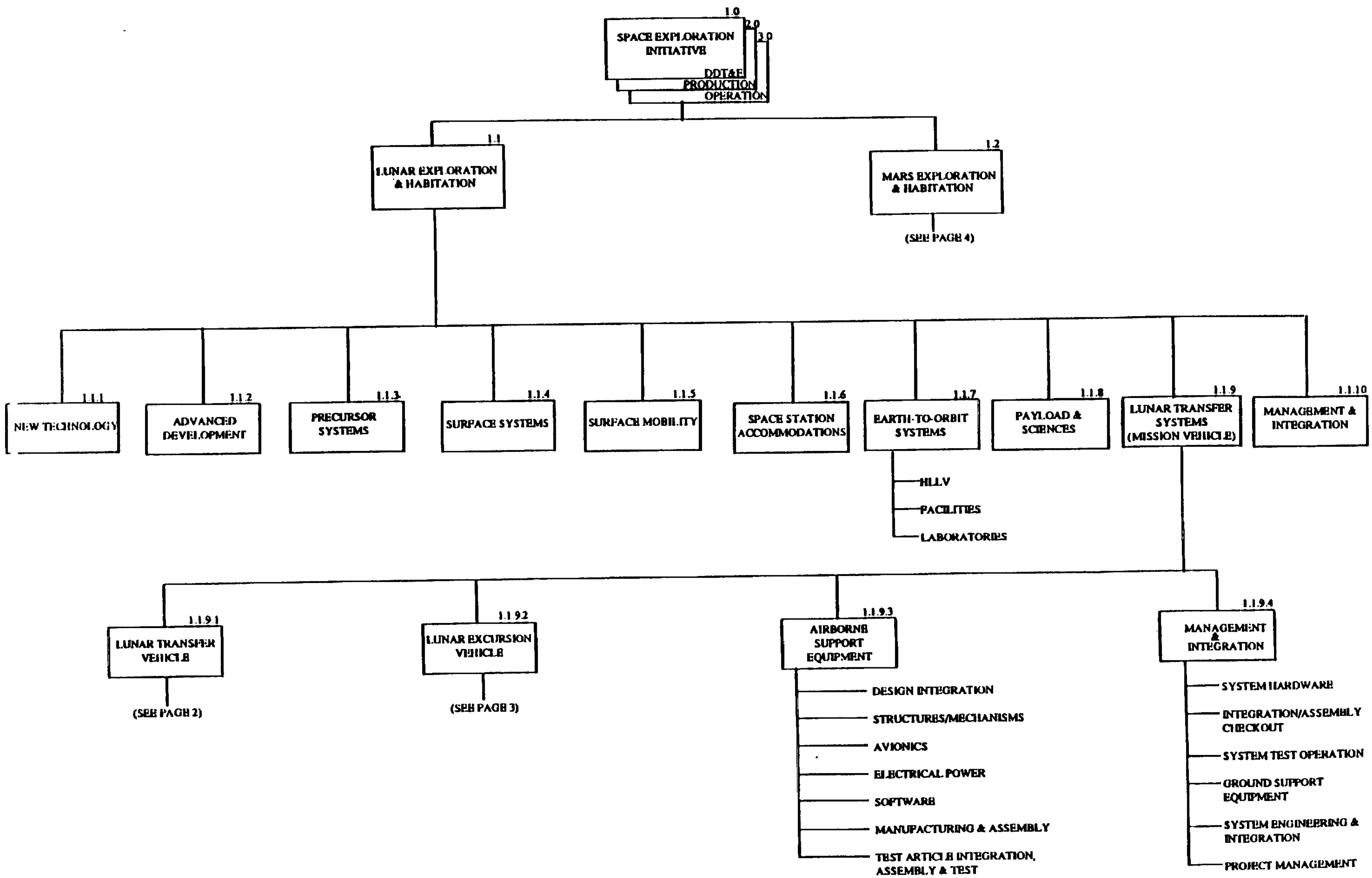
Advanced Auxiliary Propulsion Man-Rating Approach



Aerobraking Major Test/Demo Man-Rating Approach



SPACE TRANSFER CONCEPTS AND ANALYSIS FOR EXPLORATION MISSIONS



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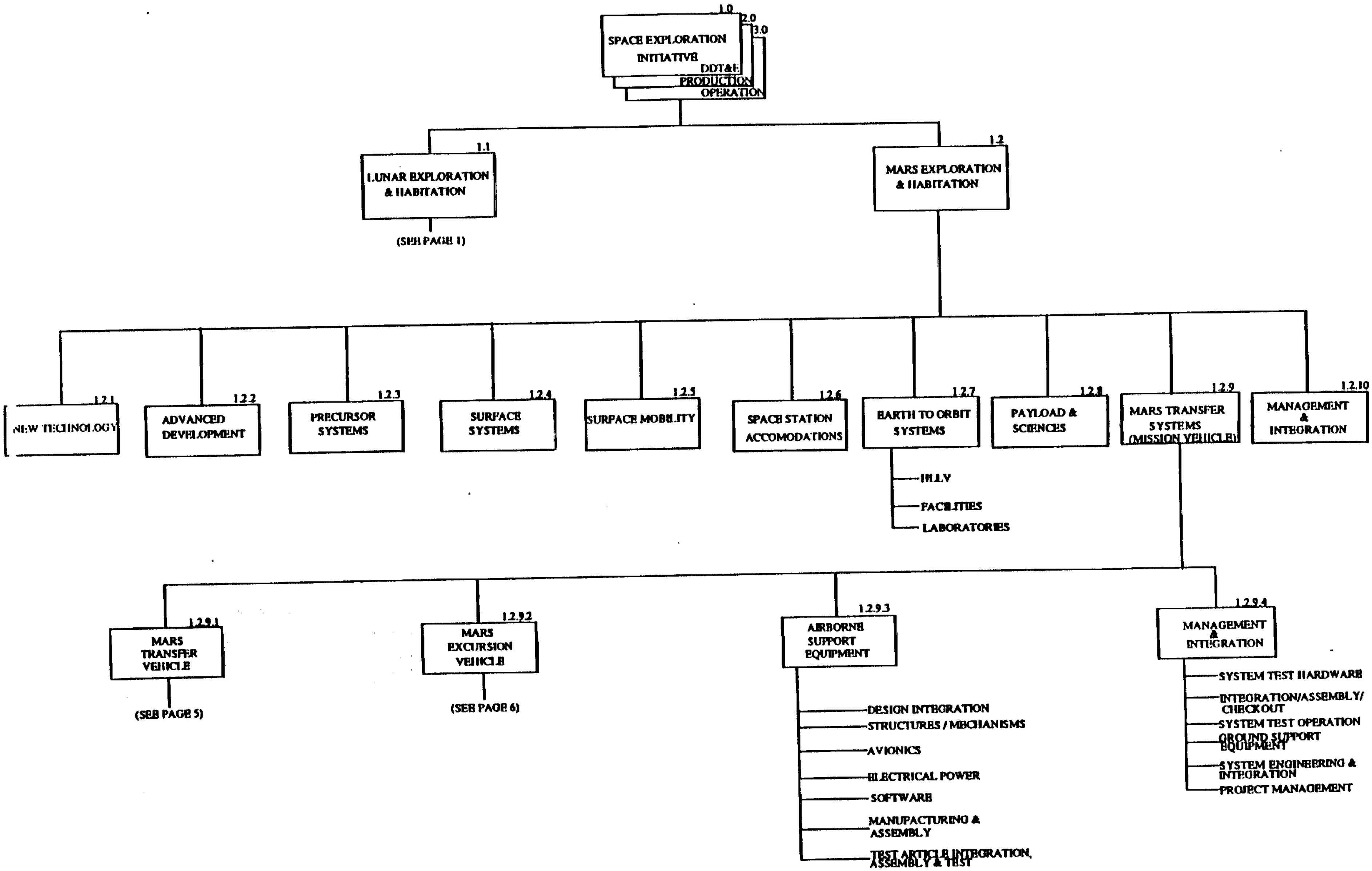
SPACE TRANSFER CONCEPTS AND ANALYSIS FOR EXPLORATION MISSIONS



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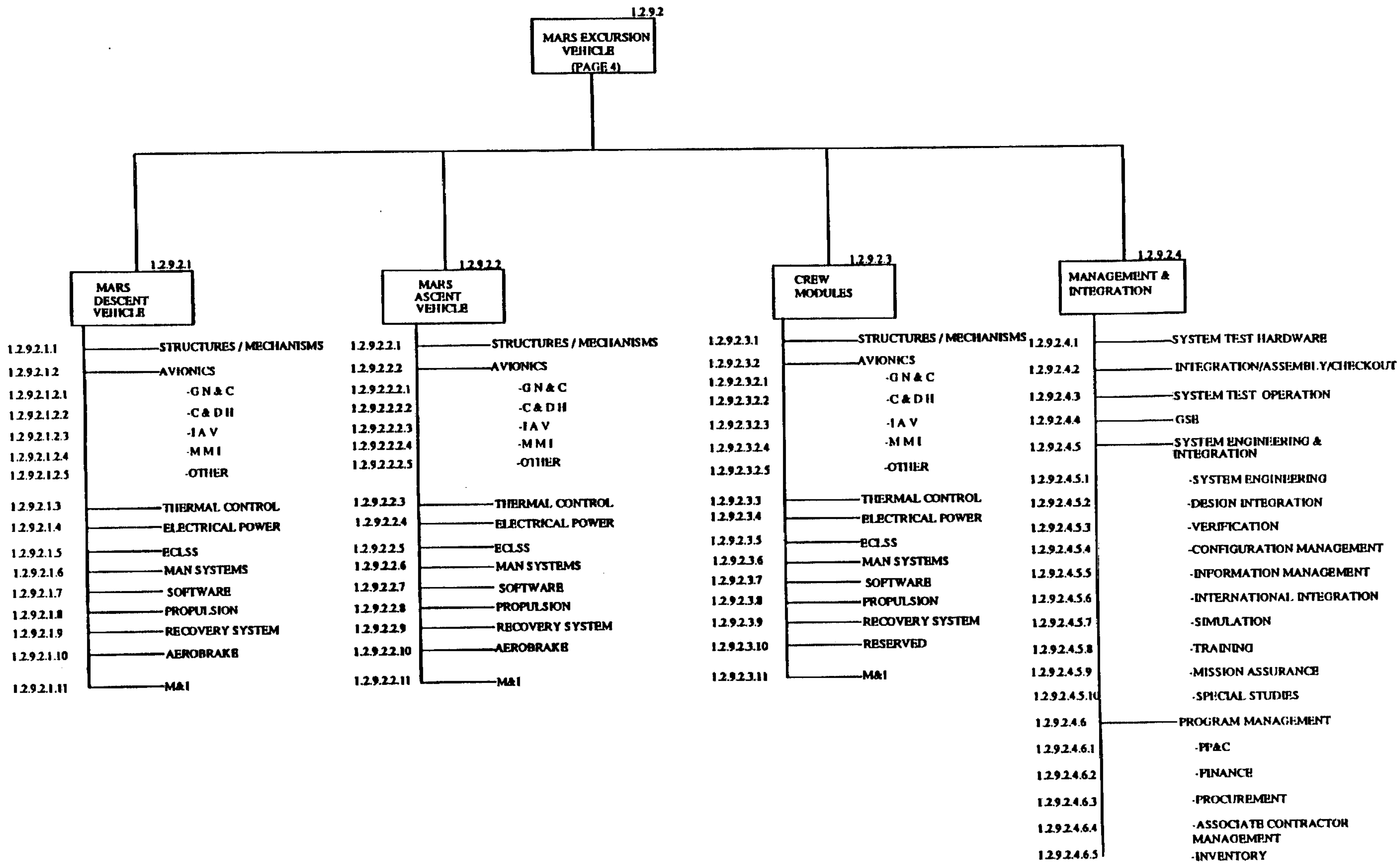
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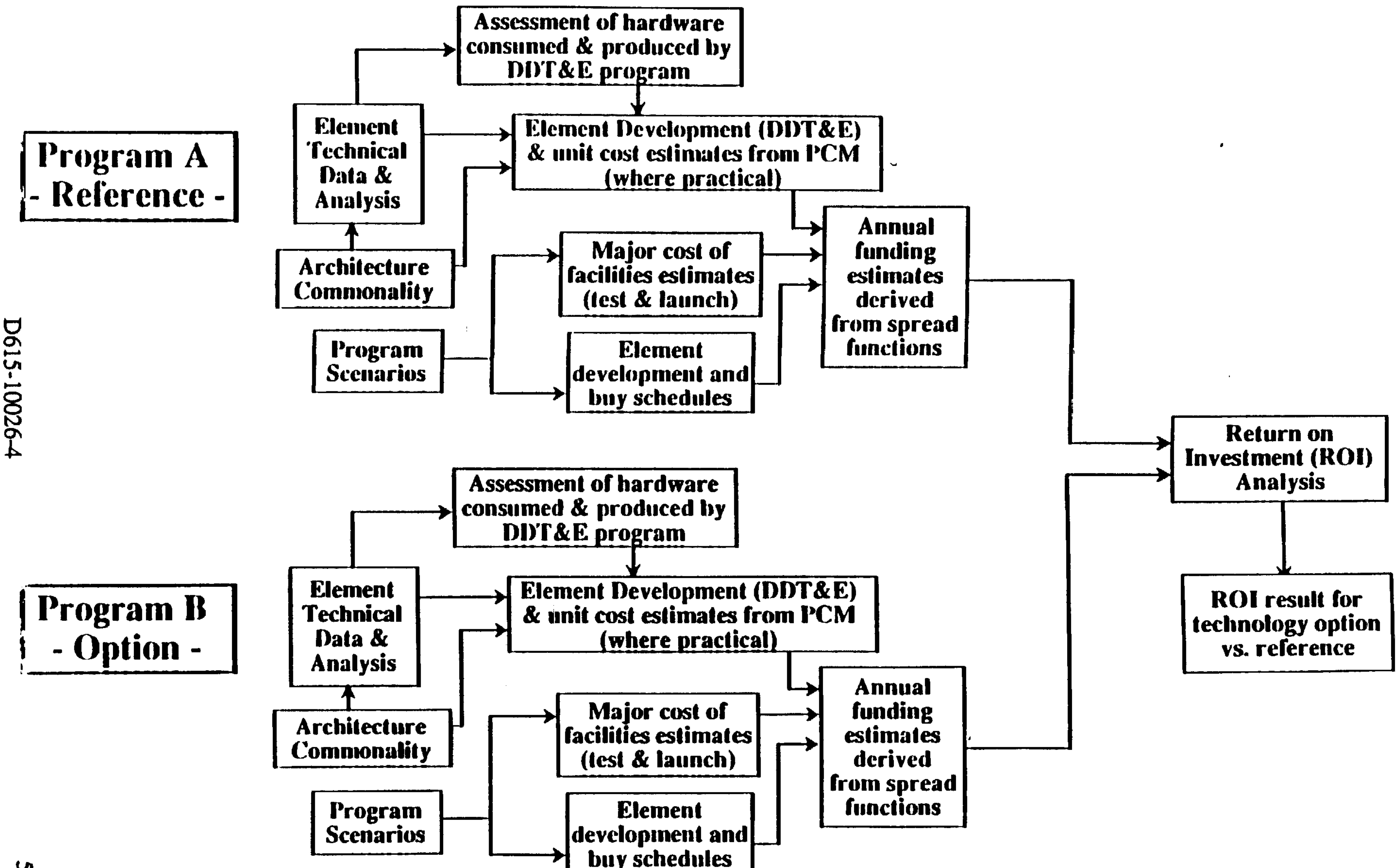
Costing Methodology Flow



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Boeing Parametric Cost Model (PCM)

Features

- Designed specifically for advanced system estimating
- Uses company-wide, uniform computerized data base
- Contains historical data compiled since 1969
- Allows direct input of known costs into the estimate

Main Inputs	Results
• Hardware Characteristics - Category (e.g., primary structure, power conditioning, etc.) - Weight (or Thrust) - Complexity - % Off-the-Shelf - Maturity - Quantity - Manufacturing Learning Curve • Support Cost Factors - Systems Engineering - Management - Operations - Spares	• DDT&E and Manufacturing Estimates - Based on previous Boeing programs - Provides first flight unit costs - Excludes test hardware - Excludes fees • New hardware must be relatable to PCM database to produce reasonable estimate • PCM estimates improve with increasing hardware detail.

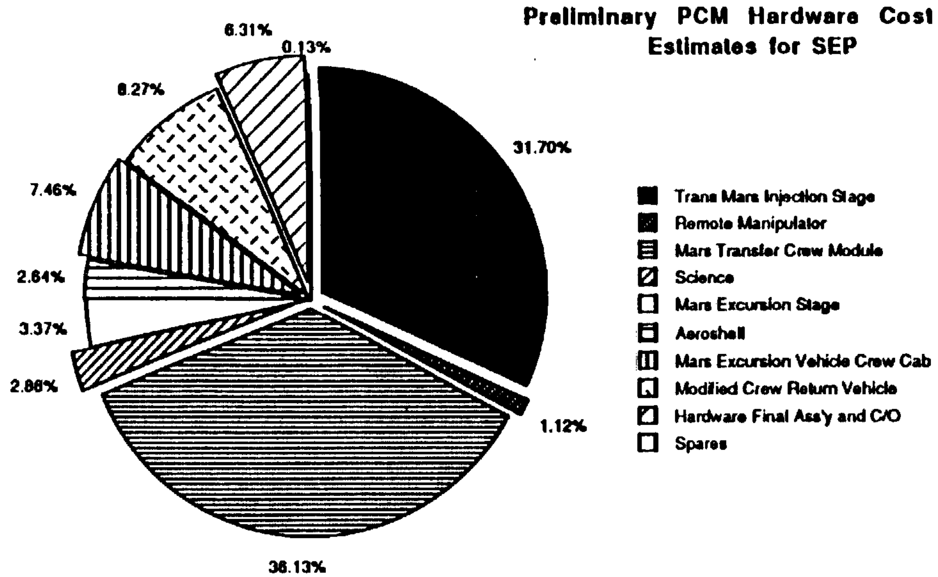
Components		Lunar/Mars		
		Minimum	Full Science	Settle/Ind
HLLV	Cargo Carrier & Core	X	X	X
	STME	X	X	X
	Recov PA Mod	X	X	X
Propulsion	Std Avionics Suite	X	X	X
	Adv Space Engine	X	X	X
	NTR Tanks		X	
	MOC Tank	X		X
	MOC Core	X		X
	NTR Stage		X	
	NTR Engine		X	
	NEP Stage			X
	NEP Engine			X
	TMIS Engine	X		X
	TMIS Tank	X		X
	TMIS Core	X		X
Modules	LEO Tanker	X	X	X
	LTV Hab	X	X	X
	LTV	X	X	X
	LEV	X	X	X
	LEV Crew Module	X	X	X
	MTV	X		X
	MTV Crew Module	X	X	X
	MEV	X	X	X
	RMEV			X
	mini-MEV		X	
	MEV Crew Module	X	X	X
	Lunar Aerobrake	X		
	MTV Aerobrake			
	MEV Aeroshell	X	X	X
	MCRV	X	X	X

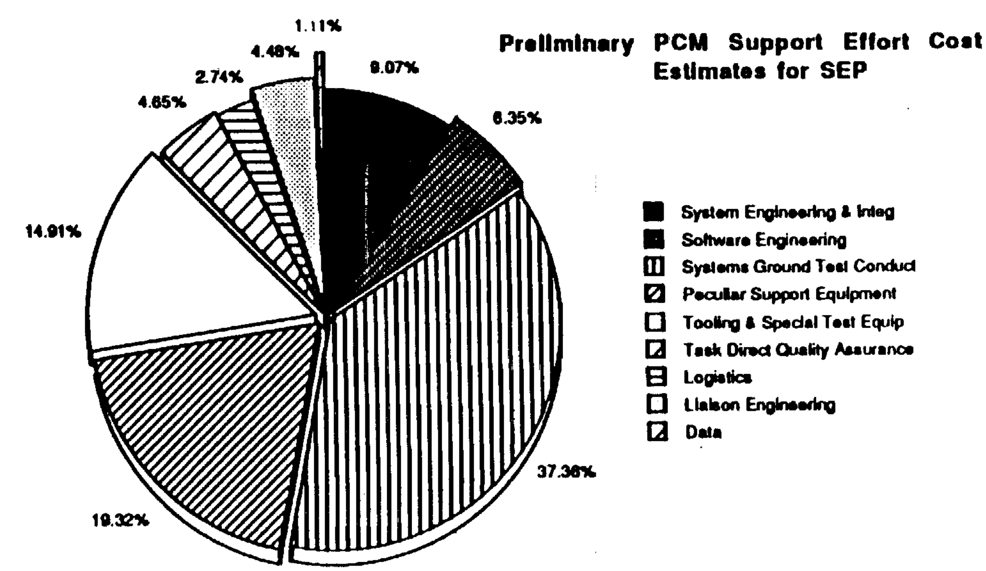
Mars SEP Preliminary PCM Summary

Item	Engineering ($Millions)	Manufacturing ($Millions)	Total ($Millions)
Trans Mars Injection Stage	933.892	877.130	1811.022
Remote Manipulator	19.701	44.303	64.004
Mars Transfer Crew Module	1138.947	925.073	2064.020
Science	100.651	62.517	163.167
Mars Excursion Stage	58.783	133.912	192.695
Aeroshell	99.473	51.556	151.030
Mars Excursion Vehicle Crew Cab	315.766	110.413	426.178
Modified Crew Return Vehicle	273.312	199.326	472.637
Hardware Final Ass'y and C/O	----------	360.634	360.634
Spares	----------	7.213	7.213
Hardware Total Costs	2940.526	2772.075	5712.598
System Engineering & Integration	519.762	----------	519.762
Software Engineering	364.019	----------	364.019
Systems Ground Test Conduct	2141.350	----------	2141.350
Systems Flight Test Conduct	----------	----------	----------
Peculiar Support Equipment	968.801	138.243	1107.044
Tooling & Special Test Equipment	----------	854.528	854.528
Task Direct Quality Assurance	----------	266.779	266.779
Logistics	157.285	----------	157.285
Liaison Engineering	256.811	----------	256.811
Data	63.770	----------	63.770
Training	O/H	----------	----------
Facilities Engineering	O/H	----------	----------
Safety	O/H	----------	----------
Graphics	O/H	----------	----------
Outplant	O/H	----------	----------
Program Management	O/H	----------	----------
Support Effort Total	4471.793	1259.550	5731.336
Total Estimate	**7412.320**	**4031.625**	**11443.945**

O/H = Overhead charge (included in above costs)

Mars SEP Preliminary PCM Summary - continued





	A	B	C	D	E	F	G	H
1		Eng'r Cost	Cost Δ's	Wrap Factor	Total D&D	Unit Cost	# Units in DDT&E	
2	SEP Core Veh	953	0	2.81	2677.93	1059	1.5	
3	SEP Array	2000		1	2000	1000	1.2	
4	SEP Thrusters	300		2.79	837	100	1.3	
5	Crew Module	1113	0	2.79	3105.27	1281	3.1	
6	MEV Stg	66.8		2.79	186.372	107.5	3.5	
7	MEV Engine	0	0	2.79	0	8	5	
8	MEV Aeroshell	112.1		2.79	312.759	64.7	2	
9	Mev CM	142		2.79	396.18	126.5	3.5	
10	ECCV	280		2.79	781.2	214	3.5	
11	RMEV	670	0	2.79	1869.3	1336	3.65	
12								
13								
14								

	I	J	K	L	M	N	O
1	DDT&E no Fe	Fee Factor, %	Total DDT&E	Units/Msn	Unit $/Msn	Msn Cost w/	Fee
2	4266.43	8	4607.7444	0.2	211.8	228.744	
3	3200	8	3456	0.1	100	108	
4	967	8	1044.36	1	100	108	
5	7076.37	8	7642.4796	0.2	256.2	276.696	
6	562.622	8	607.63176	1	107.5	116.1	
7	40	8	43.2	7	56	60.48	
8	442.159	8	477.53172	1	64.7	69.876	
9	838.93	8	906.0444	1	126.5	136.62	
10	1530.2	8	1652.616	1	214	231.12	
11	6745.7	8	7285.356	0.2	267.2	288.576	
12		Grand Total	27722.9639				
13					Cost/msn exp MEV		1335.636
14					Cost/msn reus MEV		1010.016

	A	B	C	D	E	F	G	H
1		Eng'r Cost	Cost Δ's	Wrap Factor	Total D&D	Unit Cost	# Units in DDT&E	
2	SEP Core Veh	953	0	2.81	2677.93	1059	1.5	
3	SEP Array	2000		1	2000	5000	1.2	
4	SEP Thrusters	300		2.79	837	100	1.3	
5	Crew Module	1113	0	2.79	3105.27	1281	3.1	
6	MEV Stg	66.8		2.79	186.372	107.5	3.5	
7	MEV Engine	0	0	2.79	0	8	5	
8	MEV Aeroshell	112.1		2.79	312.759	64.7	2	
9	Mev CM	142		2.79	396.18	126.5	3.5	
10	ECCV	280		2.79	781.2	214	3.5	
11	RMEV	670	0	2.79	1869.3	1336	3.65	
12								
13								
14								

	I	J	K	L	M	N	O
1	DDT&E no Fe	Fee Factor, %	Total DDT&E	Units/Msn	Unit $/Msn	Msn Cost w/	Fee
2	4266.43	8	4607.7444	0.2	211.8	228.744	
3	8000	8	8640	0.1	500	540	
4	967	8	1044.36	1	100	108	
5	7076.37	8	7642.4796	0.2	256.2	276.696	
6	562.622	8	607.63176	1	107.5	116.1	
7	40	8	43.2	7	56	60.48	
8	442.159	8	477.53172	1	64.7	69.876	
9	838.93	8	906.0444	1	126.5	136.62	
10	1530.2	8	1652.616	1	214	231.12	
11	6745.7	8	7285.356	0.2	267.2	288.576	
12		Grand Total	32906.9639				
13					Cost/msn exp MEV		1767.636
14					Cost/msn reus MEV		1442.016

	A	B	C	D	E	F	G	H
1		Eng'r Cost	Cost Δ's	Wrap Factor	Total D&D	Unit Cost	# Units in DDT&E	
2	SEP Core Veh	953	0	2.81	2677.93	1059	1.5	
3	SEP Array	2000		1	2000	10000	1.2	
4	SEP Thrusters	300		2.79	837	100	1.3	
5	Crew Module	1113	0	2.79	3105.27	1281	3.1	
6	MEV Stg	66.8		2.79	186.372	107.5	3.5	
7	MEV Engine	0	0	2.79	0	8	5	
8	MEV Aeroshel	112.1		2.79	312.759	64.7	2	
9	Mev CM	142		2.79	396.18	126.5	3.5	
10	ECCV	280		2.79	781.2	214	3.5	
11	RMEV	670	0	2.79	1869.3	1336	3.65	
12								
13								
14								

	I	J	K	L	M	N	O
1	DDT&E no Fee	Fee Factor, %	Total DDT&E	Units/Msn	Unit $/Msn	Msn Cost w/ Fee	
2	4266.43	8	4607.7444	0.2	211.8	228.744	
3	14000	8	15120	0.1	1000	1080	
4	967	8	1044.36	1	100	108	
5	7076.37	8	7642.4796	0.2	256.2	276.696	
6	562.622	8	607.63176	1	107.5	116.1	
7	40	8	43.2	7	56	60.48	
8	442.159	8	477.53172	1	64.7	69.876	
9	838.93	8	906.0444	1	126.5	136.62	
10	1530.2	8	1652.616	1	214	231.12	
11	6745.7	8	7285.356	0.2	267.2	288.576	
12		Grand Total	39386.9639				
13					Cost/msn exp MEV		2307.636
14					Cost/msn reus MEV		1982.016

Development Risk Assessment For Aerobraking By Function

MISSION FUNCTION	BRAKE SIZE	ATMOSPHERE KNOWLEDGE & UNCERTAINTY	TARGET FOR ENTRY: GN&C PRECISION	HEATING/TPS	AERO PASS GN&C PRECISION REQUIRED
Lunar return Earth landing	Small, no ass'y required	Accurate knowledge, low uncert. effect	Very high	State-of-the-Art	State-of-the-Art
Lunar return Earth landing	Moderate requires assembly	Accurate knowledge, high uncert. effect	Very high	State-of-the-Art	Believed State-of-the-Art
Mars landing from orbit	Large, requires assembly	Poor knowledge, low uncert. effect	Can be high, e.g. done from Mars orbit	State-of-the-Art	Believed State-of-the-Art
Mars return Earth landing	Small, no ass'y required	Accurate knowledge, moderate uncertainty effect	Very high	Very high heating rates, TPS advancement needed	Believed State-of-the-Art
Mars return aerocapture	Large, requires assembly	Accurate knowledge, high uncert. effect	Very high	Very high heating rates, TPS advancement needed	Believed State-of-the-Art
Mars return aerocapture	Large, requires assembly	Poor knowledge, high uncert. effect	Poor, unless nav-aids in Mars orbit	High heating rates, some TPS advancement needed	Advancements required

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Solar Array Cost Estimate

Cost Estimate For Advanced SEP System Photovoltaic Blanket Using 30% Multiple Junction Solar Cells Data From LeRC

- Current Space Solar Array Costs: \$1000/W to \$1500/W (1 kW to 10 kw Arrays)
- Total Annual U.S. Production of Space Solar Cells ~ 100 kW
- Total Annual U.S. Production of Terrestrial Modules ~ 15MW
  - Single crystal Si, amorphous and polycrystalline Si
  - Module costs in range \$10/W to \$20/W
  - DOE goal ≤ \$1/W Module cost
- Blanket production rate suitable for SEP: 2 MW/YR, 5 YR total
- Est. Cost For New Automated Terrestrial Plant: \$60M (Siemans Solar)
  - 5 MW/YR Capacity in 1993 (Space Solar Cells)
- Estimated \$100/W For Space Solar Arrays in Mid 1990's if Needed in Large Quantities, Such as a SEP Vehicle Array Requires