EXPLORATION
PROGRAMS OFFICE

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July 1992

First Lunar Outpost Phase One Report

May, 1992

Exploration Programs Office
Johnson Space Center
Houston, Texas

National Aeronautics and Space Administration

First Lunar Outpost
Phase One Report

Scope

This report has been assembled to document the technical work by members of the First Lunar Outpost (FLO) team between October 1991 and April 1992. The intent of this report is to capture the results of activities which have provided the background for the requirements described in the First Lunar Outpost Requirements and Guidelines (FLORG) document and in the FLO Detailed Assumptions document.

This document contains background material submitted by the FLO Teams, and has received no community review. It is intended for use by team members only and should not receive further distribution.

For information concerning this report, please contact Alida Andrews/XP (713/283-5744) or Deb Neubek/XE (713/283-5755).

First Lunar Outpost
Phase One Report

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Executive Summary
First Lunar Outpost (FLO) Phase One Report

The First Lunar Outpost (FLO) is a study focused on providing the functional capabilities required for human exploration missions that will lead to the achievement of the first major programmatic human exploration milestone for the Space Exploration Initiative (SEI). FLO objectives revolve around performing complementary human and robotic lunar surface activities beginning with a first flight in the year 1999. In order to develop a feasible implementation plan for achieving the FLO objectives, a "Phase One" FLO study was initiated by the Exploration Programs Office (EXPO) in late December 1991.

The design activity for the FLO is an integral part of the requirements development process that will progress through numerous iterations before a final selection of the technical approach will be made. The concepts in the report merely represent one of many possible technical approaches which can provide a framework for the continued development and testing of the requirements. The concepts are the result of the first phase of work and other concepts or approaches will be identified and assessed as the studies progress.

The Phase One Report for the FLO study documents the results of work performed by the FLO teams between October 1991 and April 1992. It is divided into sections by subject, each section containing the results of studies conducted by a FLO study team. The following is a summary of the major topics within each section:

Section	Subject
1	**System Engineering and Program Development** This section contains the First Lunar Outpost Requirements and Guidelines (FLORG) document and the Detailed Assumptions Document (DAD) as of April 2, 1992, and a study describing the system effectiveness plan.
2	**Mission Design and Analysis** Within this section are the conceptual flight profiles for the initial cargo and piloted flights and a summary of the results of ongoing navigation analyses supporting the navigation architecture concept definition.

Section	Subject
3	**Operations** The operations assumptions and operations concept are included here, as well as summaries of the operations function partitioning analysis, the FLO control center staffing analysis, and a cost estimate based on these studies. Also included are presentation packages from various team members summarizing their efforts to date.
4	**Science and Payloads** This section contains information on proposed science activities and payloads for the FLO reference mission. Included are detailed descriptions of strawman payloads for experiments in the areas of astronomy, geology and geophysics, space physics, life sciences, and resource utilization.
5	**Space Transportation and Lander** Within this section are the concepts representing one of the many possible technical approaches which can provide a framework for the continued development and testing of the FLO system requirements for the space transportation segment.
6	**Heavy Lift Launch Vehicle (HLLV)** The HLLV Design and Assessment Preliminary Status Report presents a status of the assessment of both a reference National Launch System (NLS)-derived and a Saturn V-derived HLLV. Driving constraints, such as the limit to the launch vehicle length due to Vehicle Assembly Building (VAB) size, are defined and are accounted for in the analysis.
7	**Habitation** This section documents the conceptual design for a lunar habitat based upon the adaptation of a Space Station Freedom module to a lunar surface habitat that will support a four-man crew for a 45-day mission.
8	**Surface** The Surface section of this document contains the complete set of technical data products from the Surface Segment Team as of April 2, 1992. It fully documents the additional levels of detail and rationale which describe the concepts, evaluations, issues, and other considerations leading to the results summarized here.

1. Sys. Engr. & Program Dev.

Abstract

First Lunar Outpost Requirements and Guidelines (FLORG)

This section of the report contains the First Lunar Outpost Requirements and Guidelines (FLORG) document as of April 2, 1992. The FLORG is a synopsis of study requirements and guidelines based on the current understanding of the mission description for the First Lunar Outpost (FLO) mission. Its purpose is to serve as technical guidance for study and concept design of the elements for the First Lunar Outpost mission.

Questions concerning this section should be directed to XE/Deb Neubek (713/283-5755).

Exploration Programs Office

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First Lunar Outpost Requirements and Guidelines (FLORG)
Fully Annotated Working Draft (Status)

1 April 1992

Johnson Space Center - Houston, TX

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Requirements: 03/25/92

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EXPLORATION PROGRAMS OFFICE

920001EXPO FIRST LUNAR OUTPOST REQUIREMENTS AND GUIDELINES (FLORG)

04/01/92

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SCOPE:

The First Lunar Outpost Requirements and Guidelines (FLORG) contai: a synopsis of study requirements and guidelines based on the curre: understanding of the Mission Description for the First Lunar Outpo. Mission.

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T920001EXPO:5.4-1	Factors of Safety (FS)S* Referenced by 504 on page 94	

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F920001EXPO:5.1-1	First Lunar Outpost System Hierarchy Referenced by D301B on page 20	

SCOPE:

The First Lunar Outpost Requirements and Guidelines (FLORG) contain a synopsis of study requirements and guidelines based on the current understanding of the Mission Description for the First Lunar Outpost Mission.

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T920001EXPO:5.4-1	Factors of Safety (FS)S* Referenced by 504 on page 94

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FIGURE ID	FIGURE TITLE
F920001EXPO:5.1-1	First Lunar Outpost System Hierarchy Referenced by D301B on page 20

	MAFE	
1.0 SCOPE AND PURPOSE OF DOCUMENT	?PG	H100
The First Lunar Outpost Requirements and Guidelines (FLORG) contains a synopsis of study requirements and guidelines based on the current understanding of the Mission Description for the First Lunar Outpost Mission. The purpose of the FLORG is to serve as technical guidance for study and concept design of the elements for the First Lunar Outpost Mission. The controlling authority for the FLORG is the Exploration Programs Office at NASA Johnson Space Center. All correspondence regarding this document should be directed to the Program Manager, EXPO, Code XA, NASA Johnson Space Center. J. Carpenter 01/07/92	?PG	D101
2.0 APPLICABLE DOCUMENTS	?PG	H102
TBD J. Carpenter 12/26/91	?PG	D103
2.1 Higher Level Documents	?PG	H104
Higher level Space Exploration Initiative documents are TBD. J. Carpenter 01/15/92	?PG	105
2.2 Lower Level Documents	?PG	H106
TBD J. Carpenter 12/26/91	?PG	D107
2.3 Other Applicable Documents	?PG	H108
TBD J. Carpenter 12/26/91	?PG	D109
3.0 SEI GOALS	?PG	H110
This section contains the SEI goals which specifically apply to the design of elements for the First Lunar Outpost Mission Study. It is not intended to contain the entire set of goals for the Space Exploration Initiative. J. Carpenter 01/28/92	?PG	D304

	MAFE	
The SEI shall expand human presence and activity beyond Earth orbit into the solar system. J. Carpenter 12/26/91 Rationale: Established by the U.S. National Space Policy, signed into effect on November 2, 1989.	?PG	111
The SEI shall expand knowledge of Earth, its environment, the solar system, and the universe. J. Carpenter 12/26/91 Rationale: Established by the U.S. National Space Policy, signed into effect on November 2, 1989.	?PG	112
The SEI shall research the utilization of space resources. J. Carpenter 12/26/91 Rationale: Derived from the U.S. National Space Policy, signed into effect on November 2, 1989. This requirement captures the intent of the policy to "Create new opportunities for use of the space environment through conduct of the appropriate research and experimentation in advanced technology of systems", while adhering to the NASA's traditional role of performing activities which are experimental or enabling for further research rather than commercial in nature.	?PG	113
The SEI shall establish continual human presence on the Moon. J. Carpenter 12/26/91 Rationale: Derived from the Presidential Decision on the Space Exploration Initiative dated February 21, 1990. This statement addresses the stated policy to send man "...back to the Moon....And this time, back to stay", while retaining the option for man-tended lunar operations rather than a permanently-manned lunar base.	?PG	114
The SEI shall provide manned missions to Mars. J. Carpenter 03/03/92 Rationale: Derived from the Presidential Decision on the Space Exploration Initiative dated February 21, 1990. This statement captures the stated policy for "...a manned mission to Mars." This goal provides the "Mars consciousness" to be kept in mind while deriving concepts for the First Lunar Outpost.	?PG	115A

MAFE

4.0 FIRST LUNAR OUTPOST DESCRIPTION ?PG H116

The Space Exploration Initiative (SEI) encompasses a series of exploratory missions that seek to achieve the goal of expanding human presence to the Moon and Mars. The purposes sought by this goal are to enhance our understanding of the universe, to seek terrestrial benefits from the exploration, and to establish the beginnings of a sustainable spacefaring civilization. The SEI thus involves the simultaneous pursuit of two highly interactive tracks of activity: the exploration of the Moon and the exploration of Mars. The implementation of SEI will be pursued within the strategy of a series of major programmatic milestones, each of which encompasses a series of major decision events that affect the emphasis and scope of the downstream phases of the Initiative. These milestones are used as key program mileposts to delineate the completion of one phase of the Initiative and the beginning of the next phase or step of implementation of the SEI. Each major programmatic milestone, therefore, has its own discrete set of objectives, capabilities, and achievements within an evolving SEI architecture. The first major programmatic milestone is well defined and provides clear direction to the near-term planning activities. Later major programmatic milestones are less defined, but provide the foundation and linkage to predecessor milestones. ?PG D546

First Lunar Outpost is a strategy to provide the functional capabilities required for the human exploration missions that will lead to the achievement of the first major programmatic milestone. A comprehensive discussion of the overall strategy may be found in the Reference Mission Scenario as presented at the EXPO Technical Interchange Meeting, January, 1992.

D. Weaver 03/24/92

4.1 Goals and Objectives ?PG H118

This section addresses the specific goals and objectives to be achieved with the First Lunar Outpost Mission. ?PG D119

K. Joosten/A. DuPont 01/08/92

	MAFE	
4.1.1 General First Lunar Outpost Goals	?PG	H236
Establish a human-tended exploration capability on the lunar surface. K. Joosten/A. Dupont 01/16/92 Rationale: Serves as the first step in achieving President Bush's stated goal of returning to the Moon to stay.	?PG	237
4.1.1.1 General First Lunar Outpost Objectives	?PG	H238
Perform complementary human and robotic lunar surface activities. K. Joosten/A. DuPont 01/27/92 Rationale: This objective is based upon the SEI goals of expanding human presence and activity beyond Earth orbit into the solar system; expanding knowledge of Earth, its environment, the solar system, and the universe; and researching the utilization of space resources. Example activities include geologic traverses, scientific payloads deployment, biological and in situ resource utilization experiments, etc.	?PG	239
Provide for future visits to the same outpost. K. Joosten/A. DuPont 02/04/92 Rationale: This objective provides the capability for human-tended operations while also providing the opportunity for evolution to a continuous human presence capability with a larger crew size.	?PG	242A
Return crews safely to Earth. K. Joosten/A. DuPont 01/23/92 Rationale: This objective is based upon the desire to ensure the safety of the crew to the extent practical for all First Lunar Outpost missions.	?PG	243
4.1.2 Science Research Goals	?PG	H204
Conduct important scientific experiments and prepare for future science experiments. M. Duke 01/03/92 Rationale: Derived from SEI goal: Expand knowledge of Earth, its environment, the solar system and the universe, and from the SEI goal: Establish continual human presence on the Moon.	?PG	205

	MAFE	
4.1.2.1 Science Research Objectives	?PG	H206
Initiate local intensive, long-term scientific studies and environmental characterization.	?PG	207

M. Duke 01/03/92
Rationale: The surface of the Moon records both internal activity and external phenomena (meteoroids and radiation). In particular, detailed studies of the lunar regolith can provide a long-term history of solar radiation. Studies of meteoroid impact craters and debris can provide information on timing and composition of meteoroids through time. Experiments emplaced at a lunar outpost can probe the interior structure of the Moon and measure any effects of human activity on the lunar atmosphere. Many studies will require repeated sampling or long-term monitoring to extract important data.

Conduct regional exploration and science missions.	?PG	208

M. Duke 01/03/92
Rationale: A lunar outpost provides a quantum step in exploration capability to address many outstanding problems of lunar history and to understand lunar geologic processes. Regional access allows the examination of more than one terrain type, a full range of chronological ages, important boundaries between major formations, and unusual features recognized in data sets from robotic orbiter and surface missions. Human access introduces the capability for in situ observation/data collection/experiment planning that is central to the promise of the SEI.

Establish prototype observatory instruments and demonstrate operational characteristics.	?PG	209

M. Duke 01/03/92
Rationale: Many important scientific questions can be addressed through telescopic instruments or arrays on the Moon. However, the fundamental operational feasibility of many current concepts needs to be demonstrated before full scale deployment of large instruments or arrays can be planned. Small instruments, used as test beds for sensors and for gaining operational experience, can demonstrate the future potential for large scale astronomy on the Moon.

	MAFE	
4.1.3 Space Resource Utilization Research Goals	?PG	H210
Demonstrate extraction and use of local materials on the Moon. M. Duke 01/03/92 Rationale: This is the first step in achieving the SEI Goal: Research the utilization of space resources.	?PG	211
4.1.3.1 Space Resource Utilization Research Objectives	?PG	H212
Delineate local resources and their regional extension. M. Duke 01/03/92 Rationale: Resource grade must be characterized in order to optimize processes. Resource availability must be ascertained for long-term operations.	?PG	213
Deploy and operate resource demonstration projects. M. Duke 01/16/92 Rationale: End-to-end demonstrations of resource extraction are required before full-scale deployment of resource processing plants is undertaken. Operating characteristics, maintenance problems and failure modes must be understood in the lunar environment.	?PG	215
Test uses of manufactured products. M. Duke 02/04/92 Rationale: Simple demonstrations of the use of manufactured products (or lunar raw materials) are necessary to confirm the usability of the recovered lunar resources, and are an important step toward decisions to produce materials on the Moon. Demonstration of space resource utilization products should take place in the most effective location, e.g., on Earth, in space, or on the Moon.	?PG	216A
4.1.4 Human Operations and Support Research Goals	?PR	H217
Expand and enhance the capabilities of humans on the Moon. M. Duke 02/04/92 Rationale: The SEI shall eventually establish continual human presence on the Moon.	?PG	218A

	MAFE	
4.1.4.1 Human Operations and Support Research Objectives	?PR	H219
Conduct EVA technology tests on the Moon.	?PG	220

M. Duke 01/17/92
Rationale: The long-term operability of humans on the Moon is directly related to their ability to perform EVA. Experiments conducted early in the program can provide valuable data for design and operation of future systems.

Monitor human performance in 1/6 g.	?PG	221A

M. Duke 02/04/92
Rationale: Human performance capabilities must be determined in order to properly plan future missions and surface operations and to understand the nature of operational hazards. A wide variety of EVA and IVA tasks needs to be monitored under realistic conditions to effectively design future equipment and processes.

Demonstrate teleoperations.	?PG	222

M. Duke 01/06/92
Rationale: Many of the tasks to be carried out on the Moon may benefit from the use of machinery teleoperated from the Moon. Early in the program, it should be established whether the benefits of lunar teleoperation are significant with respect to the same activities teleoperated from Earth.

Test/simulate advanced solar flare prediction and protection.	?PG	223

M. Duke 01/27/92
Rationale: Solar flares can be lethal to unprotected humans on the lunar surface or during missions to Mars. The lunar outpost should test solar flare detection systems along with Earth-based solar flare warning and prediction systems. Operational procedures in case of solar flare must be developed and tested. Shielding construction and radiation mitigation strategies should be demonstrated.

MAFE

Conduct life science research. ?PR 224

M. Duke 01/06/92

Rationale: Long-term human habitation in the Moon's environment will require significantly greater medical and physiological experience than is now available, in order to guarantee crew safety and performance. The capability for plants and animals to grow and reproduce in artificial environments on the Moon is essential for long-term continual lunar activity. Early steps can be included in the initial lunar missions at modest cost.

4.2 Design Guidelines and Assumptions ?PG H123

This section addresses the design guidelines which constrain the First Lunar Outpost Mission Study. Many of the statements in this section are assumptions which were made in order to initiate the design process for the FLO study. When/if these assumptions are substantiated or changed by analysis, they will be incorporated into the appropriate sections as requirements. Other guidelines of a general nature will remain as such. For the purposes of this study, all statements contained in this section are to be adhered to. ?PG D124A

K. Joosten/A. DuPont 03/04/92

To the degree practical, minimize both development cost and life cycle cost. ?PG 253A

K. Joosten/A. DuPont 03/03/92

Rationale: Reduced cost can be obtained by reducing the number of elements to be developed, utilizing enabling technologies to increase performance, reducing the number of people to support the mission through streamlining processes, etc. Since design decisions which reduce life cycle cost often increase development cost, a trade must be made to determine the solution which provides the minimum life cycle cost for a feasible development cost.

	MAFE	
To the degree practical, minimize dependence on other programs in the development or planning stages. K. Joosten/A. DuPont 04/01/92 Rationale: Design solutions which build on hardware or technology produced for other programs may incur schedule and technical risk. Hardware and technologies developed for other programs may be used provided that the incurred schedule and technical risk is acceptable to FLO program management. In general, this means these hardware and technical options should not be in the critical path for FLO mission success.	?PG	305B
To the degree practical, utilize existing hardware components and systems. K. Joosten/A. DuPont 01/16/92 Rationale: This guideline is based upon the desire to reduce development and life cycle costs by utilizing existing resources wherever practical. Determination of practicality shall take into account cost savings, schedule reduction, risk, and performance penalties.	?PG	125
To the degree practical, minimize crew module time in orbit. K. Joosten/A. DuPont 02/28/92 Rationale: Sizeable penalties in required mass to LEO result from consumables and propellant needed to support the crew while they are in orbit, where orbit includes Earth orbit, transit between Earth and the Moon, and lunar orbit.	?PG	376
To the degree practical, minimize orbital and surface operations which do not provide direct science and exploration returns. K. Joosten/A. DuPont 01/17/92 Rationale: This guideline is based upon the desire to minimize the activities required to enable or maintain science/exploration activities and maximize the crew time to be devoted to productive activities (e.g., science/resource studies).	?PG	250

MAFE

To the degree practical, minimize orbital operations such as rendezvous, docking, and assembly for piloted and cargo vehicles. ?PG 377

K. Joosten/A. DuPont 03/04/92
Rationale: Orbital operations such as rendezvous, docking, and assembly can lead to sizeable penalties in required mission complexity, time in orbit, and mass to LEO. Generally speaking, operations are easier and less costly if performed on the ground rather than in orbit.

To the degree practical, minimize operations required to prepare the outpost for occupancy, to operate and maintain the outpost, to transition the outpost from manned to unmanned, and to maintain the outpost in an unmanned/untended mode. ?PG 126A

K. Joosten/A. DuPont 02/04/92
Rationale: This guideline is based upon the desire to reduce the crew support requirements on manned flight elements and allow crew time to be devoted to productive activities (e.g., science/resource studies).

To the degree practical, all FLO systems and subsystems will have logical growth paths to human Mars mission capability. ?PG 547

H. Mandell 03/24/92
Rationale: This approach will minimize total lunar and Mars program costs and development lead times. It is to be applied to the greatest extent feasible.

To the degree practical, maximize lifetimes of all lunar surface elements. ?PG 252C

K. Joosten/A. DuPont 03/24/92
Rationale: This guideline is based upon the desire to reduce cost by maximizing element lifetime within the constraints imposed by available technology as well as development cost and schedule. This guideline also provides direct contribution to the development of systems for human missions to Mars through lessons learned in long term reliability.

	MAFE	
To the degree practical, maximize the commonality of the Earth launch vehicle design for piloted and cargo flights. K. Joosten/A. DuPont 02/04/92 Rationale: Based upon the desire to reduce the cost by reducing the number of elements to be developed.	?PG	247A
To the degree practical, maximize the commonality of the trans-lunar injection stage design for piloted and cargo flights. K. Joosten/A. DuPont 02/04/92 Rationale: Based upon the desire to reduce the cost by reducing the number of elements to be developed.	?PG	152A
To the degree practical, maximize the commonality of the lander design for piloted and cargo flights. K. Joosten/A. DuPont 02/28/92 Rationale: This guideline is based upon the desire to reduce total program development cost by reducing the number of elements to be developed.	?PG	254B
To the degree practical, maximize the reusability of the crew module. K. Joosten/A. DuPont 02/28/92 Rationale: Based upon the assumption that refurbishment and reuse of the crew module is cheaper than production of an additional module. A trade must be performed before this guideline can become firm. Reusable systems could include (but are not limited to) the pressure vessel, electronics, seats, controls, life support, engine systems, etc.	?PG	378

	MAFE	
Assume a single Earth launch vehicle shall be utilized for each flight to the Moon. K. Joosten/A. DuPont 02/28/92 Rationale: The use of a single launch vehicle for each flight minimizes on-orbit operations by eliminating rendezvous and assembly, thus increasing the probability of performing TLI during the first window of opportunity. Once an Earth orbit has been established, the regression of the orbital node dictates that the available window of opportunity for trans-lunar injection will occur approximately every nine days. Maximizing the probability of performing TLI during the first window of opportunity meets the design guidelines which call for minimizing crew time in orbit and avoiding cryogenic propellant storage and management.	?PG	246A
Assume sixty (60) days between cargo and piloted flights. K. Joosten/A. DuPont 02/28/92 Rationale: This design goal minimizes the dormant period of the habitation system on the lunar surface between delivery of this system and the first occupancy by a crew. It also sets a minimum time-spacing guideline for launch support. Further study must be done before a firm number may be established and this design goal becomes a requirement.	?PG	379
Assume one crew module design shall be used for all space transit sub-segment elements. K. Joosten/A. DuPont 02/28/92 Rationale: Based upon the desire to reduce the cost by reducing the number of elements to be developed. If an option exists which has more than one crew cab and costs less, this option will be considered.	?PG	150B

	MAFE	
Assume storable propellant shall be utilized for Earth return. K. Joosten/A. DuPont 03/03/92 Rationale: Based on current technology, it appears that handling and boil-off of cryogenic fuels may lead to a sizeable penalty in mass to LEO capability for extended duration stays in orbit or on the lunar surface. Designs which facilitate planned product improvements for the use of high-energy propellants without substantial cost increase are desireable.	?PG	245A
For the first FLO nominal mission, site selection and characterization assumptions are: 1) the site will be on the lunar front-side; 2) the site latitude will be within an 80 degree band centered at the lunar equator; 3) surface properties for highland and mare sites are consistent with the lunar surface model listed in the Natural Environments section 5.5.1, requirement 405 and 4) detailed site characteristics will be confirmed by data returned from precursor missions and/or previous lunar missions prior to the cargo vehicle landing. Duke/Joosten/DuPont 04/01/92 Rationale: Selection of a front-side location within an 80 degree latitudinal band of the lunar equator reduces operational, communications, and thermal complexities for the First Lunar Outpost without unduly restricting the capability to meet science objectives. The description of the applicable lunar surface properties provides a standardized set of assumptions for mission design. Site characterization data may be provided in a variety of ways, but will be managed separately from the First Lunar Outpost missions. Detailed site requirements remain TBD.	?*G	244C

MAFE

?*G 614

For sortie missions, site selection and characterization assumptions are: 1) global access includes polar and farside sites; 2) surface properties are consistent with the lunar surface model listed in the Natural Environments section 5.5.1, requirement 405; and 3) detailed site characteristics will be confirmed to levels equivalent to those used in Apollo site certification.

Duke/Joosten/DuPont 04/01/92

Rationale: Global access for sortie missions is required to meet First Lunar Outpost science objectives and may be provided in the evolutionary phases of the program rather than as an initial capability. The description of the applicable lunar surface properties provides a standardized set of assumptions for mission design. Finally, the description of how detailed site characteristics will be confirmed limits risk to a level similar to that associated with Apollo.

?PG 313A

Assume the flight segment shall provide the capability to deliver 27.5 mt (E) of cargo, inclusive of mission margins, from the Earth's surface to the lunar surface.

K. Joosten/A. DuPont 02/28/92

Rationale: The mass for an outpost fully outfitted to support the crew for a nominal lunar surface stay as described in the DRM was initially estimated to be 25 mt. The additional 2.5 mt is reserved as a mission margin.

?PG 495

Assume piloted landings on the lunar surface will occur within 4 days following lunar sunrise.

K. Joosten/A. Dupont 03/04/92

Rationale: During lunar noon and lunar midnight lunar surface temperatures reach extremes. In addition, the sun angle at lunar noon washes out lunar surface features and causes EVA to become complex. The assumption that piloted landings will occur within four days of lunar sunrise is based upon the desire to maximize the time available for science and exploration activities, many of which require crew EVA.

	MAFE	
Assume that flight elements shall land at least 1 km (E) away from critical elements on the lunar surface to minimize blast ejecta damage. K. Joosten/A. DuPont 03/04/92 Rationale: Preliminary studies indicate that 1 km limits launch/landing blast ejecta damage to previously landed elements to acceptable levels. This avoids the equipment and operational complexity of creating berms or other protection systems. Critical elements on the surface include but are not limited to the habitat, solar arrays, scientific equipment, etc. This design assumption must be proven out with additional trades before it can become a requirement. Note: This is the current design assumption. For the root requirement that drives this assumption, refer to 384.	?PG	380
For nominal missions and revisit missions, assume piloted landers shall provide the capability to land within 100 m (3 sigma) of the designated lunar aim point. K. Joosten/A. DuPont 03/24/92 Rationale: Designated aim points for the piloted flights will be chosen based upon the maximum safe walking distance between the piloted lander and the habitat, and the minimum distance between the piloted lander and the habitat. The piloted lander must have an acceptable landing accuracy to avoid violating these parameters. The current estimate of 100 m (3 sigma) is based upon assumptions that a surface navaid or navsat will be available to assist navigation and that map tie error (inaccuracies in the lunar topographic map knowledge) is minimal as well as estimates of navigation accuracy at lunar orbit insertion and Inertial Measurement Unit drift.	?PG	381A

	MAFE	
Assume cargo landers shall provide the capability to land within 2 km (3 sigma) of the designated lunar aim point. K. Joosten/A. DuPont 03/24/92 Rationale: Designated aim points for the cargo flights will be chosen based upon the ability of the site to meet science and exploration needs, the roughness of the terrain, and the communications infrastructure which is present at the time of the mission (far side landings may not take place until a lunar orbital communications node exists). The current estimate of 2 km (3 sigma) is based upon assumptions that no surface navaid or navsats will be available to assist navigation and that map tie error (inaccuracies in the lunar topographic map knowledge) is minimal as well as estimates of navigation accuracy at lunar orbit insertion and Inertial Measurement Unit drift.	?PG	153C
For sortie missions, assume piloted landers shall provide the capability to land within 2 km (3 sigma) of the designated lunar aim point. K. Joosten/A. DuPont 03/24/92 Rationale: Designated aim points for the piloted sortie flights will be chosen based upon the ability of the site to meet science and exploration needs, the roughness of the terrain, and the communications infrastructure which is present at the time of the mission (far side landings may not take place until a lunar orbital communications node exists). The current estimate of 2 km (3 sigma) is based upon assumptions that no surface navaid or navsats will be available to assist navigation and that map tie error (inaccuracies in the lunar topographic map) is minimal as well as estimates of navigation accuracy at lunar orbit insertion and Inertial Measurement Unit drift.	?PG	382A

	MAFE	
Assume nominal lunar surface EVAs shall not exceed 10 (E) hours, inclusive of EVA preparation and post-EVA preparation prior to entry into the habitat. Ward/Rummel/Brown 03/24/92 Rationale: The maximum duration of EVA shall not degrade crew acute EVA performance or chronic mission performance. Time in suit is influenced by suit flexibility and mobility, crew workload, and in-suit provisions for air revitalization, thermal control, nutrition, and hygiene. This total capability is based on 8 hours actual useful EVA time plus 1 hour preparation prior to airlock depressurization and 1 hour preparation following airlock repressurization prior to entry to the habitat.	?PG	505A
Assume a crewmember is not scheduled for EVA on consecutive days but may perform off-nominal or unscheduled EVA if the total time for both days is not more than 10 hours, and may perform contingency EVA as needed. Ward/Rummel/Brown 03/24/92 Rationale: Fatigue and muscle soreness preclude an individual from scheduled EVA on consecutive days. Moderate length EVA on consecutive days should not be precluded if the gain of an unscheduled EVA is justified.	?PG	506A

MAFE

4.3 - Reserved - ?PG H130

4.4 Design Reference Mission (DRM) ?PG H132

A First Lunar Outpost Design Reference Mission (DRM) has been identified. It will be used to derive the mission-driven requirements during the design study. The analysis of this DRM is mandatory so as to achieve enough insight to generate these requirements. Two design options have also been included. The analysis of these is highly desirable. ?PG D229

The DRM describes the automated delivery of the lunar outpost, the activation/deployment of the outpost, the arrival of the first crew, and the science/technology surface activities. The first option is a revisit to the outpost by a subsequent crew. The second option is a lunar "sortie" where the crew operates out of the lander for a short-duration mission at a location not supported by an outpost.
K. Joosten 01/15/92

4.4.1 Mission Description ?PG H230

The reference mission is accomplished via a split mission profile. Two flights are required: (1) the automated delivery of the lunar outpost, and (2) the human flight. ?PG D231A

The outpost is delivered first. All necessary habitation elements for the lunar outpost are delivered on a single launch. Upon arrival at the moon, an injection into a temporary lunar parking orbit will be performed followed by de-orbit. An automated landing delivers the lunar outpost to the lunar surface. To the degree practical, the outpost will be activated and checked out remotely prior to launch of the crew. The design of the outpost will be such that: (a) it will not be necessary to unload the outpost from the descent stage, (b) the crew will be able to move in and start science and exploration activities as soon as possible after they arrive.

Delivery of a crew of four, the crew ascent vehicle, and any required additional payload is accomplished via a launch vehicle and descent stage elements similar (little or no modification to the design) to those used to deliver the

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outpost. The crew vehicle will perform a landing sequence similar to the cargo flight, resulting in touchdown around lunar dawn within walking distance of the outpost. The crew will power-down the lander, transfer to the outpost, and remain on the surface for up to a lunar day-night-day cycle (45 Earth days inclusive of contingencies).

Surface Mission Description: The first priority for the crew will be to complete any remaining deployment, activation and checkout of the outpost. The program of surface activities will then begin. The science and engineering activities of the crew will be devoted to: (1) characterizing the local environment, (2) deploying astrophysical, space physics and geophysical monitoring packages, (3) conducting ISRU demonstrations, and (4) conducting life sciences research. Initial roving, consistent with the capability of an unpressurized rover, may be limited by crew safety considerations to less than the design rover range. The outpost will provide the EVA support required for all of these activities. Both EVA and roving capability will be available on a daily basis.

After lunar sunset, surface activities will continue when earthshine at the site is sufficient. In addition, the bulk of the laboratory work will be performed during this time. The full set of surface activities will resume at lunar sunrise.

When the second lunar sunset approaches, the crew will configure the outpost for their departure, transfer back to the crew vehicle along with any return cargo, and initiate the return to Earth. At Earth, the crew module will perform a direct reentry and landing.
K. Joosten 02/04/92

?PG H232

4.4.2 Option 1

?PG 233

This option is a revisit by a crew of four (4) to the outpost which was occupied by a previous crew. Any necessary resupply logistics are brought with the new crew to enable a second 45 day stay. Surface activities are similar to those performed in the DRM.
K. Joosten 01/15/92

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?PG H234

4.4.3 Option 2

?PG 235A

This option is a "sortie" by a crew of two (2) to a lunar landing site with no pre-deployed outpost. A similar (little or no modification to the design) crew transfer vehicle to the one used in the primary DRM is employed in this option. The surface mission will be restricted to a maximum of 14 days (daylight only). Surface activities will be similar to those in the primary DRM but limited in scope.
K. Joosten 02/07/92

?PG H142

5.0 TECHNICAL REQUIREMENTS

?PG H143

5.1 System Description

?PG D301B

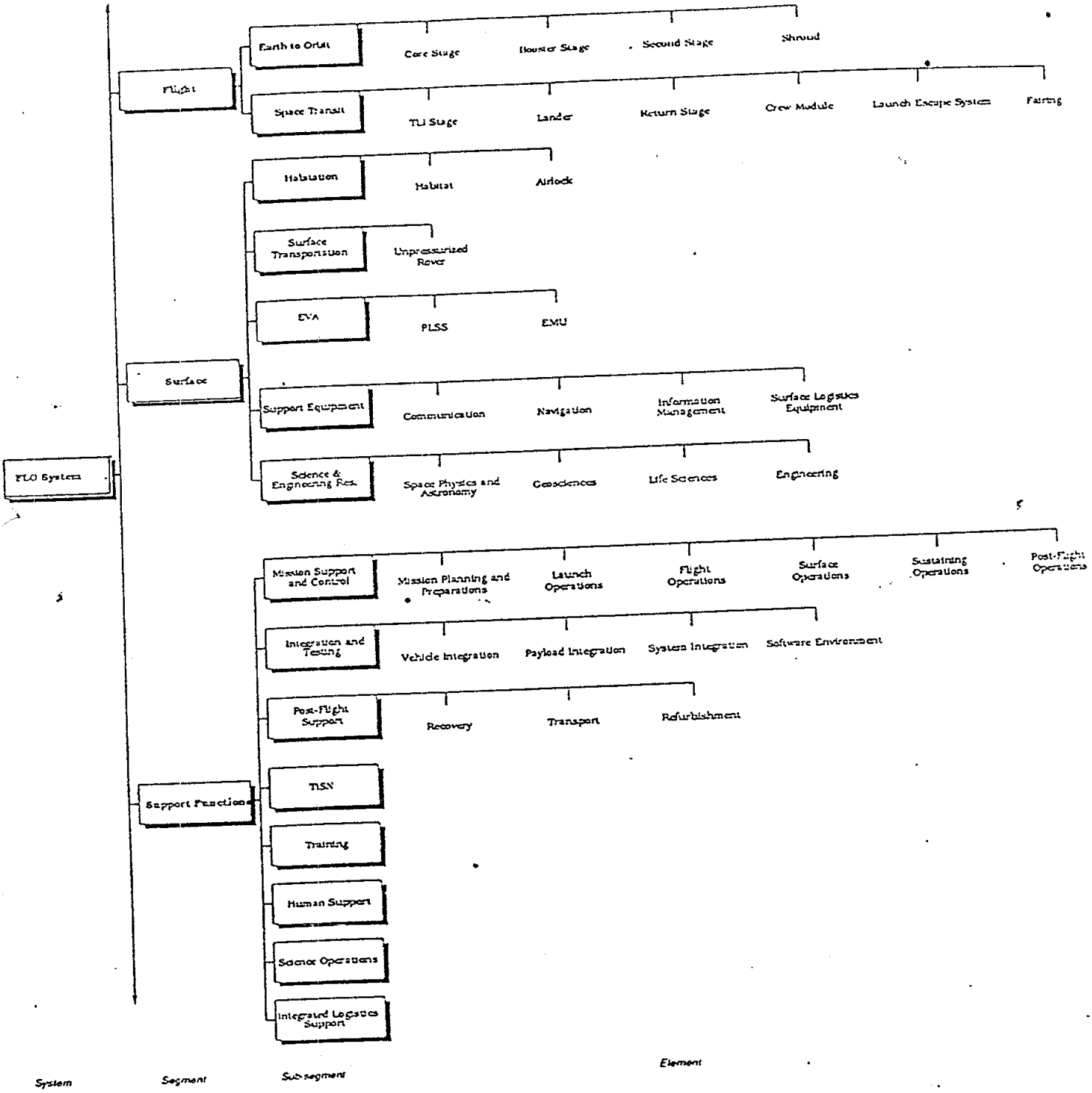
The First Lunar Outpost system has been defined at the segment, sub-segment and element levels. As illustrated in F920001EXPO:5.1-1, "First Lunar Outpost System Hierarchy" on page 21, the Flight segment contains Earth to Orbit and Space Transit sub-segments, each with their own elements. The Surface segment contains Habitation, Surface Transportation, EVA, Support Equipment and Science and Engineering Research sub-segments, each with their own elements. The Support Function segment contains Mission Support and Control, Integration and Testing, Post-Flight Support, TISN, Training, Human Support, Science Operations, and Integration Logistics Support sub-segments, each with their own elements.
J. Carpenter 03/24/92

?PG H145

5.2 System Requirements

?PG D302

This section describes requirements which apply to the entire First Lunar Outpost system.
J. Carpenter 01/28/92


FLO System
Flight
Earth to Orbit
Core Stage
Booster Stage
Second Stage
Shroud
Space Transit
TLI Stage
Lander
Return Stage
Crew Module
Launch Escape System
Fairing
Surface
Habitation
Habitat
Airlock
Surface Transportation
Unpressurized Rover
EVA
PLSS
EMU
Support Equipment
Communication
Navigation
Information Management
Surface Logistics Equipment
Science & Engineering Res.
Space Physics and Astronomy
Geosciences
Life Sciences
Engineering
Support Functions
Mission Support and Control
Mission Planning and Preparations
Launch Operations
Flight Operations
Surface Operations
Sustaining Operations
Post-Flight Operations
Integration and Testing
Vehicle Integration
Payload Integration
System Integration
Software Environment
Post-Flight Support
Recovery
Transport
Refurbishment
TISN
Training
Human Support
Science Operations
Integrated Logistics Support
System
Segment
Sub-segment
Element

	MAFE	
The system shall provide the capability to deliver crew and cargo to the lunar surface and return them safely to Earth. Joosten/Carpenter 03/04/92 Rationale: Establishes a primary capability required to meet overall mission objectives. Return cargo will nominally include samples and return payloads from the Moon. Specific requirements are detailed in the appropriate sections.	?PG	496
The system shall provide the capability to perform manned and robotic activities on the lunar surface. Joosten/Carpenter 03/04/92 Rationale: Establishes a primary capability required to meet overall mission objectives. Specific activities are detailed in the appropriate sections.	?PG	497
The system shall be designed for evolution. K. Joosten/A. DuPont 01/27/92 Rationale: Based upon the desire to allow the expansion of outpost capabilities through planned upgrade and modularity rather than additional design and re-deployment of major elements.	?PG	154
The nominal launch date for the first mission shall be 1999. K. Joosten/A. DuPont 02/07/92 Rationale: Technically, realistic schedules can be discussed from a program start through development, to delivery. 1999 was chosen to drive schedules and is considered to be technically achievable. Actual schedules will depend upon approved funding levels.	?PG	309
The system shall provide the capability to support a minimum of four (4) flights per year. K. Joosten/A. DuPont 04/01/92 Rationale: This is an estimate for the number of launches per year which does not overly stress launch processing for FLO launches as well as other programs. This is the current planning assumption. As the scenario evolves this requirement may change. Evolution to 8 launches per year is currently seen as an upper planning limit as support to the outpost grows.	?PG	307C

	MAFE	
The system shall provide the capability for launch on any day of the lunar cycle. K. Joosten/A. DuPont 03/03/92 Rationale: This requirement ensures that launch windows are not constrained by unfavorable Earth/Moon geometries. Although the variation is slight, there is a delta V variation during the lunar cycle. In order to assure access to a chosen site on the Moon during favorable lighting conditions, the system should be designed to meet the maximum delta V budget.	?PR	411
Nominal crew sizes shall be 4 (E) persons. K. Joosten/A. DuPont 01/15/92 Rationale: Provides for significant exploration capability per mission and falls within the life support and design capabilities which are realistically achievable by the year 2000.	?PG	146
Nominal lunar surface stays shall consist of a lunar day-night-day cycle (45 (E) Earth days inclusive of contingencies). K. Joosten/A. DuPont 01/28/92 Rationale: Provides for significant exploration capability per mission and falls within the life support and design capabilities which are realistically achievable by the year 2000.	?PG	147
The system shall provide the capability to access any lunar latitude/longitude. K. Joosten/A. DuPont 02/28/92 Rationale: This capability is derived from the science research and space resource utilization objectives. Global access is essential to provide a capability to reach areas of highest interest and potential return.	?PG	149A
Sortie option lunar surface stays shall be restricted to lunar daylight. K. Joosten/A. DuPont 02/28/92 Rationale: Based upon the assumption that allowing sortie option surface stays during lunar nighttime would drive lander performance requirements and adversely impact the performance of the systems necessary to achieve the primary DRM.	?PG	387

MAFE

The system shall provide the capability for abort during all mission phases. ?PG 383

K. Joosten/A. DuPont 02/28/92

Rationale: This requirement is derived from crew safety requirements. It does not imply continuous (at any time) abort capability, but rather, a capability to declare and execute aborts during each mission phase. Aborts are provided in addition to reliability and redundancy requirements to achieve an acceptable overall mission risk. Specific abort requirements may be found at the appropriate segment, sub-segment and element levels. Mission phases where abort capability is required include countdown to Earth launch, piloted Earth launch/ascent, piloted Earth orbit, piloted lunar transfer, piloted lunar orbit, piloted lunar descent/landing, lunar surface stay, lunar ascent, piloted Earth transfer, Earth re-entry and recovery.

The system shall provide the capability to protect elements on the lunar surface from blast ejecta damage not greater than TBD magnitude. ?PG 384

K. Joosten/A. DuPont 03/04/92

Rationale: Spacecraft launch and landing from the lunar surface creates considerable blast ejecta which can damage elements on the lunar surface. Potential strategies for mitigating this effect include performing launch/landing at sufficient distance from surface elements, or providing a protection system such as berms or other coverings. The present assumption for the design solution is to land at least 1 km from critical elements and is contained in the design assumptions section (380). Other options will be studied.

	MAFE	
For nominal missions, piloted elements shall land within safe walking distance of the habitat, less than or equal to TBD km (E). K. Joosten/A. DuPont 02/28/92 Rationale: Derived from the requirement which calls for maintaining the health, safety, and productivity of the crew (322). The TBD number will be calculated given the allowable EVA traverse time, nominal EVA walking speed, and unmanned to manned mode transition time for the habitat. These calculations will also take into account contingency for return to the piloted lander in the event that the habitat cannot be transitioned to manned mode within one EVA period.	?PG	385
The system shall enable EVA without pre-breathe. Connolly/Carpenter 03/24/92 Rationale: Maximizes use of available crew time to accomplish mission objectives. Elements impacted by this capability include but are not limited to the crew module, the EMU, the PLSS, the habitat, and the airlock.	?PR	548
The system shall provide the capability for direct Earth re-entry of the crew module. K. Joosten/A. DuPont 02/28/92 Rationale: Based upon the desire to minimize propellant required by flight segment elements. Circularization of the crew module at Earth incurs mass to LEO penalties due to the additional propellant which must be delivered to and extracted from the lunar gravity well.	?PG	386
The system shall provide the capability to perform nominal Earth landings on land at TBD locations. Joosten/Carpenter 03/24/92 Rationale: Returning crew and payloads to Earth is a primary capability to meet the mission objectives of returning the crew safely and enabling research. This requirement does not imply landing on all terrain conditions, therefore specific landing locations will be established. Landing on land is the nominal option.	?PG	498A

	MAFE	
The system shall provide the capability to perform Earth water landings. K. Joosten/A. DuPont 03/04/92 Ratonale: Range safety considerations dictate launch trajectories which necessitate water landings in the case of an Earth launch abort.	?PG	373A
The system shall only utilize technologies which have undergone system validation or engineering model demonstration in a relevant environment prior to commitment to development. K. Joosten/A. DuPont 03/04/92 Rationale: Based upon the desire to reduce development cost, risk, and schedule by requiring that all technologies used are well understood.	?PG	155B
5.2.1 Flight Segment	?PG	H161
The flight segment shall provide the capability to deliver crew between the Earth's surface and the lunar surface. K. Joosten/A. DuPont 02/28/92 Rationale: The desire to deliver human crews to the Moon is consistent with the SEI goal of establishing continual human presence on the Moon.	?PG	160C
The flight segment shall provide the capability to deliver cargo between Earth's surface and the lunar surface. K. Joosten/A. DuPont 02/28/92 Rationale: The delivery of cargo to the Moon allows human crews to perform useful science, engineering, and exploration activities.	?PG	314A
The crew shall be launched with the space transfer vehicle crew module. K. Joosten/A. DuPont 02/28/92 Rationale: Based upon the desire to avoid STS launch and rendezvous for crew delivery. This guideline implies that the Earth launch vehicle will be man-rated.	?PG	248B

MAFE

The flight segment shall provide the capability to perform Earth landings under any lighting conditions and under TBD weather conditions. ?PG 256B

K. Joosten/A. DuPont 03/24/92
Rationale: Direct Earth re-entry dictates that landing site lighting will be governed by the geometry of the Earth, Moon, and Sun system at the time of Earth-entry interface. Due to the requirement to provide abort capability during all mission phases, landings can occur during darkness as well as daylight. Weather conditions which are acceptable are under study.

The flight segment shall provide the capability to revisit previous lunar landing sites. ?PG 168A

K. Joosten/A. DuPont 02/10/92
Rationale: Based upon the desire to land piloted flight elements at sites where cargo flight elements have already been delivered. In addition, DRM Option 1 calls for a revisit to a previously emplaced outpost.

The flight segment shall provide the capability to support the crew for a total of 12 (E) days, inclusive of contingencies, during a nominal mission. ?*G 388A

K. Joosten/A. DuPont 04/01/92
Rationale: Based upon the total number of days that the flight segment is required to support the crew on the Earth's surface prior to launch, during transit between Earth and the Moon, on the lunar surface, and on the Earth's surface following landing. Requirements levied on the elements that support the crew are found at the element level.

5.2.1.1 Earth to Orbit Sub-Segment ?PG H268B
Comment: Replaces 5.2.1.1 Earth to Sub-Orbit Sub-Segment

The Earth to orbit sub-segment shall provide the capability to deliver crew from Earth's surface to orbit. ?PG 315A

K. Joosten/A. DuPont 02/28/92
Rationale: Earth to orbit delivery flows down from the requirement to deliver crew between the Earth's surface and the lunar surface (160).

	MAFE	
The Earth to orbit sub-segment shall provide the capability to deliver cargo from the Earth's surface to orbit. K. Joosten/A. DuPont 02/28/92 Rationale: Earth to orbit delivery flows down from the requirement to deliver cargo between the Earth's surface and the lunar surface (314).	?PG	159C
The Earth to orbit sub-segment shall support the capability for crew escape from the time the crew enters the crew module prior to Earth launch until TBD altitude. K. Joosten/A. DuPont 02/28/92 Rationale: Derived from the requirement which calls for maintaining the health, safety, and productivity of the crew (322) and to provide abort during all mission phases. This requirement flows down as a functional allocation to the Earth to orbit sub-segment. An option which meets this requirement is to provide a vehicle health monitoring system which alerts controllers of an abort situation.	?PG	389
5.2.1.1.1 Core Stage Element	?PG	H269
5.2.1.1.2 Booster Stage Element	?PG	H270
5.2.1.1.3 Second Stage Element	?PG	H271
5.2.1.1.4 Shroud Element	?PG	H272
5.2.1.2 Space Transit Sub-Segment	?PG	H273
The space transit sub-segment shall provide the capability to deliver crew from Earth orbit to the lunar surface. K. Joosten/A. DuPont 02/28/92 Rationale: Earth orbit to lunar surface delivery flows down from the requirement to deliver crew between the Earth's surface and the lunar surface (160).	?PG	165B

	MAFE	
The space transit sub-segment shall provide the capability to deliver cargo from Earth orbit to the lunar surface. K. Joosten/A. DuPont 02/28/92 Rationale: Earth orbit to lunar surface delivery flows down from the requirement to deliver cargo between the Earth's surface and the lunar surface (314).	?PG	164B
The space transit sub-segment shall provide the capability to deliver crew from the lunar surface to the Earth's surface. K. Joosten/A. DuPont 02/28/92 Rationale: Lunar surface to Earth's surface delivery flows down from the requirement to deliver crew between the Earth's surface and the lunar surface (160B).	?PG	316A
The space transit sub-segment shall provide the capability to deliver cargo from the lunar surface to the Earth's surface. K. Joosten/A. DuPont 02/28/92 Rationale: Lunar surface to Earth's surface delivery flows down from the requirement to deliver cargo between the EArth's surface and the lunar surface (314).	?PG	317A
The space transit sub-segment shall provide the capability to sustain the crew on the Earth's surface during the pre-launch mission phase. K. Joosten/A. DuPont 02/28/92 Rationale: Derived from the requirement which calls for maintaing the health, safety, and productivity of the crew (322). The pre-launch mission phase extends from the time the crew enters the crew module until Earth launch is performed. Specific times are contained at the element level.	?PG	371A
The space transit sub-segment shall provide the capability to sustain the crew between the Earth's surface and the lunar surface. K. Joosten/A. DuPont 02/28/92 Rationale: Derived from the requirement which calls for maintaining the health, safety, and productivity of the crew (322). This requirement includes launch, lunar transfer, lunar descent, lunar ascent, Earth transfer, and Earth re-entry. Specific times are contained at the element level.	?PG	318A

	MAFE	
The space transit sub-segment shall provide the capability to sustain the crew on the lunar surface during the post-lunar landing phase. K. Joosten/A. DuPont 03/03/92 Rationale: Derived from the requirement which calls for maintaining the health, safety, and productivity of the crew (322). The post-lunar landing mission phase begins at lunar landing and extends through piloted vehicle power down. This phase also includes any time necessary for crew support while the habitat is transitioning from unmanned to manned mode. Specific times are contained at the element level (196).	?PG	320A
The space transit sub-segment shall provide the capability to support the crew during sortie option surface stays. K. Joosten/A. DuPont 03/04/92 Rationale: DRM option 2 calls for sortie missions to a lunar landing site without a supplemental cargo flight. The crew module becomes the living space for the crew during the lunar surface stay for the sortie option.	?PG	171A
The space transit sub-segment shall provide the capability to sustain the crew on the Earth's surface during the post-landing mission phase. K. Joosten/A. DuPont 02/28/92 Rationale: Derived from the requirement which calls for maintaining the health, safety, and productivity of the crew (322). The post landing mission phase extends from the time the crew module lands on Earth until the crew exits the crew module. Specific times are contained at the element level.	?PG	319A
The space transit sub-segment shall provide the capability to sustain the crew on the lunar surface during the pre-lunar launch mission phase. K. Joosten/A. DuPont 02/28/92 Rationale: Derived from the requirement which calls for maintaining the health, safety, and productivity of the crew (322). The pre-lunar launch mission phase extends from the time the crew enters the crew module prior to lunar ascent until lunar ascent begins. Specific times are contained at the element level.	?PG	370A

	MAFE	
The space transit sub-segment shall provide the capability for crew escape from the time the crew enters the crew module prior to Earth launch until TBD altitude. K. Joosten/A. DuPont 02/28/92 Rationale: Derived from the requirement which calls for maintaining the health, safety, and productivity of the crew (322) and to provide abort during all mission phases. This requirement flows down as a functional allocation to the space transit sub-segment. An option which meets this requirement is an Apollo-type launch abort system.	?PG	321A
The space transit sub-segment shall provide the capability for abort at all times during piloted lunar descent. R. Kahl/K. Joosten 03/03/92 Rationale: Derived from the requirement to provide the capability for abort during all mission phases (383). Piloted lunar descent begins with descent burn initiation and ends with lunar surface touchdown of the piloted lunar lander.	?PR	414
The space transit sub-segment shall provide the capability for landing site aimpoint redesignation until TBD seconds prior to lunar touchdown. R. Kahl 03/04/92 Rationale: This capability is required in the event that when the lander performs its pitchover-to-verticle-descent maneuver, spacecraft controllers can see that the primary landing site is obstructed or contains a landing hazard. Landing on obs tacles such as rocks or crater rims of sufficient size could cause the lander to exceed the specified tilt following landing. The current working assumption is that mapping of the lunar surface features will not have the accuracy required to identify all such obstacles and since navigation inaccuracies between the designated and achieved landing site will exist, either hover capability or aimpoint redesignation capability to a site with fewer obstacles is required. Preliminary trades have shown that providing aimpoint redesignation capability is cheaper than providing hover capability. The TBD will be calculated based upon achievable performance and mission risk.	?PR	415

	MAFE	
The space transit sub-segment shall provide the capability for abort at all times following piloted lunar landing within 24 hrs. of abort declaration. K. Joosten/A. DuPont 02/07/92 Rationale: Based upon the desire to provide an abort capability at all times following lunar landing without requiring that the piloted flight system be capable of lift-off at any time following lunar landing.	?PG	148A
The space transit sub-segment shall provide the capability to protect the crew from receiving a TBD radiation dose over TBD period. B. Ward 04/01/92 Rationale: Derived from human support requirements for radiation protection during sortie missions and during piloted flight mission phases. A trade study examining implementation options and their mass penalties must be performed before this requirement can become firm.	?PR	255E
The space transit sub-segment shall provide the capability for crew access between the lunar surface and lander payloads, including the crew module. Joosten/Carpenter 03/04/92 Rationale: Current design concepts indicate that lander payloads (which include both cargo and the crew module) may be 10 meters or more above the lunar surface. Crew access to and from the crew module and the payloads which are on top of the lander must be accommodated.	?PG	499
The space transit sub-segment shall provide the capability for separation of the lander from the TLI stage. R. Kahl 03/03/92 Rationale: The TLI stage will not be utilized for the entire mission. In order to reduce the mass and the delta V required to transfer the spacecraft to the Moon, the TLI stage will be separated from the lander after it is no longer needed.	?PR	412

	MAFE	
The lander, crew module and return stage shall remain functional when the crew compartment is depressurized. R. Kahl 03/03/92 Rationale: The crew module will be depressurized while the crew is living in the surface habitat. This capability is required so that space transit sub-segment systems can be monitored, checked out and maintained during this period and prior to lunar ascent preparations. In addition, trades show that given a limited number of repressurizations , the requirement to provide ingress/egress between the piloted crew module and the lander is best met by depressurizing the entire crew module rathe r than providing an airlock. The current working assumption is that it is cheaper to require the return vehicle to remain functional when depressurized than to repressurize the crew module following egress.	?PR	413
The space transit sub-segment shall provide the capability for separation of the crew module from the return stage. R. Kahl 03/03/92 Rationale: The return stage will not be utilized after Earth-entry. In order to reduce the mass that has to be brought back to Earth, the return stage will be separated from the crew module prior to Earth-entry interface.	?PR	416
5.2.1.2.1 TLI Stage Element	?PG	H186
The TLI stage element shall provide the capability to deliver crew and cargo from Earth orbit to lunar transfer orbit. K. Joosten/A. DuPont 02/28/92 Rationale: Derived from the requirements to provide the capability to deliver crew and cargo from Earth's surface to the lunar surface (160 and 314). This is the primary function of the TLI stage.	?PG	390

	MAFE	
The TLI stage element shall provide the capability for ascent guidance and control of the launch vehicle during launch from Earth. B. Pattison 03/03/92 Rationale: The TLI stage requires hardware to perform the TLI burn maneuvers and orbital functions prior to TLI burn. To minimize hardware, the ascent guidance functions for the launch vehicle can be accomplished by the TLI stage hardware. This approach is the same as used in the Saturn V and SIV-B designs. This functional allocation has been based on preliminary trades. Additional trades must be done before this allocation becomes firm.	?PR	417
The TLI stage element shall provide the capability for attitude correction prior to TLI burn. B. Pattison 03/03/92 Rationale: Upon achieving Earth orbit after launch, the TLI stage will maintain low Earth orbit as needed for checkout prior to TLI burn. Attitude correction capability is required to maintain proper alignment of the vehicle for communications and thermal management. This functional allocation has been based on preliminary trades. Additional trades must be done before this allocation becomes firm.	?PR	418
The capability shall be provided for safe disposal of the TLI stage element. K. Joosten/A. DuPont 02/28/92 Rationale: The TLI stage will be discarded after use, requiring safe disposal. The speciic method of disposal will be determined after further study. Considerations include the desire to avoid creating hazardous space debris in orbit around the Earth or Moon, and to avoid impacting the Earth in a populated area.	?PG	391

	MAFE	
5.2.1.2.2 Lander Element	?PG	H187
The lander shall provide the capability to deliver 5 mt (E) of cargo and a nominal crew from lunar transfer orbit to the lunar surface. K. Joosten/A. DuPont 03/24/92 Rationale: Derived from the flight segment requirement to deliver crew and cargo to the lunar surface. The cargo mass performance measure is an estimate of the resupply and science equipment mass needed to support the nominal mission specified for revisit to an established lunar outpost.	?PG	188D
5.2.1.2.3 Return Stage Element	?PG	H190A
The return stage element shall provide the capability to deliver a nominal crew and 200 (E) kg of return payload from the lunar surface to Earth-entry interface. K. Joosten/A. Dupont 02/28/92 Rationale: Derived from requirements to deliver the crew, crew module, and science return payload from the lunar surface to Earth. Payload return capability is essential for the accomplishment of science objectives.	?PG	392
The return stage element shall provide the capability for abort at all times following lunar landing within 24 hrs. of abort declaration. K. Joosten/A. DuPont 02/28/92 Rationale: Based upon the desire to provide an abort capability at all times following lunar landing without requiring that the piloted flight system be capable of lift-off at any time following lunar landing.	?PG	191B
The capability shall be provided for safe disposal of the return stage element. K. Joosten/A. DuPont 03/03/92 Rationale: The return stage will be discarded after use, requiring safe disposal. The specific method of disposal will be determined after further study. Considerations include the desire to avoid creating hazardous space debris in orbit aro und the Earth or Moon, and to avoid impacting the Earth in a populated area.	?PR	419

	MAFE	
5.2.1.2.4 Crew Module Element	?PG	H194A
The crew module shall provide the capability to support a nominal crew on the Earth's surface for 6 (E) hours prior to Earth launch. K. Joosten/A. DuPont 04/01/92 Rationale: Flow down of the requirement to support the crew prior to launch (371). This capability is an estimate for the nominal time the crew will spend in the crew module during the pre-flight checkout and 4 hour launch window. This period begins when the crew enters the crew module on the pad and ends when the Earth launch vehicle clears the launch tower. A trade must be performed before this estimate can become firm.	?*G	306C
The crew module shall provide the capability to support a nominal crew for 0.25 hours (E) during Earth launch. K. Joosten/A. DuPont 04/01/92 Rationale: Flow down of the requirement to support the crew between the Earth's surface and the lunar surface (318). This capability has been estimated for the nominal time the crew will spend in the crew module during Earth ascent. Earth ascent begins when the Earth launch vehicle clears the tower and ends when the vehicle performs a circularization burn in Low Earth Orbit.	?*G	195D
The crew module shall provide the capability to support a nominal crew for 4.5 hours (E) in LEO prior to TLI. K. Joosten/A. DuPont 02/28/92 Rationale: Flow down of the requirement to support the crew between the Earth's surface and the lunar surface (318). This capability has been estimated for the nominal time the crew will spend in the crew module in LEO prior to TLI. This period begins with Earth orbit circularization and ends with the trans-lunar injection burn. This time allows for on-orbit checkout and three opportunities for TLI.	?PG	393

	MAFE	
The crew module shall provide the capability to support a nominal crew for 96 hours (E) during lunar transfer. K. Joosten/A. DuPont 02/28/92 Rationale: Flow down of the requirement to support the crew between the Earth's surface and the lunar surface (318). This capability has been estimated for the nominal time the crew will spend in the crew module during lunar transfer. Lunar transfer begins with the trans-lunar injection burn and ends with the lunar orbit insertion burn.	?PG	394
The crew module shall provide the capability to support a nominal crew for 12 hours (E) in LLO prior to descent. K. Joosten/A. DuPont 02/28/92 Rationale: Flow down of the requirement to support the crew between the Earth's surface and the lunar surface (318). This capability has been estimated for the nominal time the crew will spend in the crew module during lunar descent. Lunar descent begins with lunar descent burn initiation and ends with touchdown on the lunar surface. This time includes time for final vehicle checkout prior to landing and crew sleep cycle phasing (if necessary).	?PG	395
The crew module shall provide the capability to support a nominal crew for 0.25 hours (E) during lunar descent. K. Joosten/A. DuPont 04/01/92 Rationale: Flow down of the requirement to support the crew between the Earth's surface and the lunar surface (318). This capability has been estimated for the nominal time the crew spends in the crew module during lunar descent. Lunar descent begins with the descent burn and ends with vehicle touchdown on the lunar surface.	?*G	396A

	MAFE	
The crew module shall provide the capability to support a nominal crew on the lunar surface for 48 (E) hours. K. Joosten/A. DuPont 02/28/92 Rationale: Flow down of the requirement to support the crew on the lunar surface (320). This capability has been estimated for the time that the crew spends in the crew module while on the lunar surface. This capability provides support during the following mission phases: from piloted landing until crew egress from the piloted lander, during the transition of the habitat from unmanned to manned mode, and from crew ingress to the piloted lander prior to ascent until ascent initiation.	?PG	196C
The crew module shall provide the capability to support a nominal crew of 0.25 (E) hours during lunar ascent. K. Joosten/A. DuPont 04/01/92 Rationale: Flow down of the requirement to support the crew between the Earth's surface and the lunar surface (318). This capability has been estimated as the maximum time the crew spends in the crew module during lunar ascent. Lunar ascent begins with the ascent burn initiation and ends with the circularization burn which is performed in LLO.	?*G	397A
The crew module shall provide the capability to support a nominal crew for 6 (E) hours in LLO prior to TEI. K. Joosten/A. DuPont 04/01/92 Rationale: Flow down of the requirement to support the crew between the Earth's surface and the lunar surface (318). This capability has been estimated for the nominal time the crew spends in the crew module in LLO. This period begins with the LLO circularization burn and ends with the trans-Earth injection burn.	?*G	398A

	MAFE	
The crew module shall provide the capability to support a nominal crew for 108 hours (E) during Earth transfer. K. Joosten/A. DuPont 03/04/92 Rationale: Flow down of the requirement to support the crew between the Earth's surface and the lunar surface (318). This capability is an estimate for the maximum time the crew spends in the crew module during Earth transfer. This transfer can be reduced by up to 24 hours to ensure longitudinal availability of the preferred Earth landing site. Earth transfer begins with the trans-Earth injection burn and ends at Earth-entry interface.	?PG	399
The crew module shall provide the capability to support a nominal crew for 0.5 hours (E) during Earth re-entry and landing. K. Joosten/A. DuPont 02/28/92 Rationale: Flow down of the requirement to support the crew between the Earth's surface and the lunar surface (318). This capability is an estimate for the nominal time the crew will spend in the crew module during Earth re-entry and landing. This period begins with Earth-entry interface and ends at touchdown on the Earth's surface.	?PG	400
The crew module shall provide the capability to support a nominal crew on the Earth's surface for 4 (E) hours following Earth landing. K. Joosten/A. DuPont 04/01/92 Rationale: Flow down of the requirement to support the crew on the Earth's surface during the post-landing phase (319). This capability is estimated for the maximum reasonable time the crew will spend in the crew module during recovery. The recovery period begins with touchdown on the Earth's surface and ends when the crew exits the crew module. A trade must be performed before this estimate can become firm.	?*G	401A

	MAFE	
The crew module shall provide the capability to support a crew of 2 (E) on the lunar surface for TBD hours. K. Joosten/A. DuPont 02/28/92 Rationale: For DRM option 2 (sortie option), the crew lives out of the crew module for the duration of the mission. The TBD will be calculated based upon the achievable performance derived from the design which meets the primary DRM and a maximum of 14 Earth days (dictated by the restriction of the sortie option to lunar daylight).	?PG	197B
The crew module element shall provide the capability to return a TBD injured crewmember to the Earth's surface. R. Kahl/Joosten 03/03/92 Rationale: Injuries and illnesses are a finite possibility in the remote and hostile environment of the Moon. The return of an ill/injured crewmember will impact the crew module configuration. The TBD is the extent of the injury which is to be used for design purposes.	?PR	420
The crew module element shall provide the capability to return 200 kg (E) of cargo to the Earth's surface. R. Kahl/K. Joosten 03/03/92 Rationale: Derived from requirements to deliver the crew, crew module, and science return payload from the lunar surface to Earth. Payload return capability is essential for the accomplishment of science objectives. The 200 kg estimate has been established based upon Apollo return (87 kg) and the increased area/science objectives of the FLO missions.	?PG	488
The crew module element shall provide the capability to return TBD volume of cargo to the Earth's surface. R. Kahl/Joosten 03/03/92 Rationale: Derived from requirements to deliver the crew, crew module, and science return payload from the lunar surface to Earth. Payload return capability is essential for the accomplishment of science objectives. The TBD volume will be derived from the example payloads defined by science and engineering needs.	?PR	421

	MAFE	
The crew module element shall provide the capability to land on land when returning to Earth. R. Kahl/K. Joosten 03/03/92 Rationale: Flow down from the requirement for the flight segment to provide the capability to perform land landings (257). The crew module is the physical element which will actually make this landing. This is the nominal landing mode.	?PG	489
The crew module element shall provide the capability to land on water when returning to Earth. R. Kahl/Joosten 03/03/92 Rationale: Flow down from the requirement for the flight segment to provide the capability to perform water landings (373). The crew module is the physical element which will actually make this landing. This is primarily an abort mode.	?PR	422
5.2.1.2.5 Launch Escape System Element	?PG	H490
The launch escape system element shall provide the capability to safely propel the crew module away from the launch vehicle during an on-pad or launch abort below TBD km altitude. K. Joosten/A. DuPont 03/03/92 Rationale: Flow down from the requirement to provide a crew escape capability from pre-launch through TBD altitude (321). This element will satisfy this requirement by positioning the crew module for a safe return to Earth following the abort.	?PR	423
The launch escape system element shall provide the capability to assure proper crew module attitude for a safe Earth landing after the abort. K. Joosten/A. DuPont 03/03/92 Rationale: Flow down from the requirement to provide a crew escape capability from pre-launch through TBD altitude (321). Proper crew module attitude is necessary for effective parachute deployment and landing system performance.	?PR	426

	MAFE	
5.2.1.2.6 Fairing Element	?PG	H491
The fairing element shall provide the capability to protect the space transit sub-segment from damage during the lift-off through maximum dynamic pressure phase of the mission.	?PR	427

R. Kahl 03/03/92
Rationale: The spacecraft would incur an excessive mass penalty if it had to be designed to withstand the launch loads and protect the payloads while exposed during this phase of the mission. A fairing element that can be discarded prior to Eart h orbit insertion is a more mass/cost effective solution for reducing the risk of mission failure due to spacecraft and/or payload damage during launch.

5.2.2 Surface Segment	?PG	H173
The surface segment shall provide the capability to support a nominal crew during a nominal lunar surface stay.	?PG	174B

K. Joosten/A. DuPont 03/03/92
Rationale: Provides for significant exploration capability per mission and falls within the life support and design capabilities which are realistically achievable by the year 2000.

The surface segment shall provide the capability to support science research.	?PG	338A

K. Joosten/A. DuPont 03/03/92
Rationale: Flow down from the science research and space resource utilization research goals and objectives. Science research activities may include but are not restricted to science instrument deployment, extravehicular traverses, intra-vehicular laboratory work, instrument monitoring, and telerobotic rover control.

The surface segment shall provide the capability to support engineering tests and demonstrations.	?PR	428

K. Joosten/A. DuPont 03/03/92
Rationale: Flow down from human operations and support goals and objectives. Engineering test and demonstration activities may include but are not limited to site characterization traverses, experiment deployment and operation, and test unit deployment and operation.

	MAFE	
The surface segment shall provide the capability to deploy payloads. K. Joosten/A. DuPont 03/03/92 Rationale: Payload deployment is a critical capability to fulfill mission objectives. Payloads to be deployed may include such elements as telescopes, geophysical stations, and in situ resource utilization demonstration units. Deployment currently consists of transportation of the payload from the lander to the desired site, equipment set-up, and equipment start-up.	?PG	180A
Surface segment elements shall provide the capability to be operated at any lunar latitude/longitude. K. Joosten/A. DuPont 03/03/92 Rationale: Flowdown from the system requirement to access any lunar attitude/longitude (149). Elements may include but are not limited to the habitat, rover, and science/engineering payloads. It is recognized that solar power availability is minimal at the poles, adversely impacting solar generated power system capabilities. Therefore, trades are being performed to assess the impact of outpost emplacement at high latitudes.	?PR	431
The surface segment shall provide the capability to support 34 (E) EVAs per nominal mission inclusive of contingency. K. Joosten/A. DuPont 04/01/92 Rationale: This requirement is derived from mission objectives and the surface DRM. It based upon the estimated number of EVAs to perform science and engineering research, maintenance and logistics. The contingency philosophy is to subtract contingencies from the total number of estimated EVAs. A firm number will be derived from an analysis of the work to be performed, and after study of the implications for evolution have been done (e.g., stay times increasing to 6 months).	?PR	434A

	MAFE	
The surface segment shall provide the capability for the crew to return to the lander crew module at all times following piloted lunar landing. K. Joosten/A. DuPont 03/03/92 Rationale: Flowdown from the requirement to protect the safety of the crew during all mission phases. Mission rules will be developed to establish the capability to return to the piloted lander crew module under conditions which include but are not limited to: a) critical failure of the habitat, b) crew member illness/injury, and c) impending depletion of surface consumables. Surface systems must be designed to provide the capability to meet these mission rules.	?PR	436
The surface segment shall provide the capability for EVA crew members to return to the habitat at all times. K. Joosten/A. DuPont 03/03/92 Rationale: Flowdown from the requirement to protect the safety of the crew during all mission phases. Mission rules will be developed to establish the capability to return to the habitat under conditions which include but are not limited to: a) surface transportation failure, b) crew member illness/injury, c) impending depletion of EVA consumables, and d) crew fatigue. Surface systems must be designed to provide the capability to meet these mission rules.	?PR	437
The surface segment shall provide the capability to communicate with Earth and local lunar elements. K. Joosten/A. DuPont 03/03/92 Rationale: Although the desire is to evolve toward autonomy of the lunar outpost from Earth, communications between the lunar outpost and Earth will always be necessary. Local elements include but are not limited to EVA crewmembers, the habitat , the rover and payloads. Communications may include but are not limited to audio and video links with crewmembers as well as telemetry from systems.	?PR	438

	MAFE	
The surface segment shall provide the capability to store TBD consumables. Joosten/Carpenter 03/04/92 Rationale: Establishes a basic capability required to achieve mission objectives. Consumables may include but are not limited to food, life support gasses such as hydrogen and oxygen, and fuels for vehicles and fuel cells. Specific requirements will be detailed at the element level.	?PG	500
5.2.2.1 Habitation Sub-Segment	?PG	H274
The habitation sub-segment shall provide the capability to sustain a nominal crew during a nominal lunar surface stay. K. Joosten/A. DuPont 03/03/92 Rationale: Flow down from the surface segment requirement to support the crew during a nominal lunar surface stay (174). This is the primary function of the habitation sub-segment. It provides the living and working environment for the crew on a continuous basis throughout the surface stay.	?PR	439
The habitation sub-segment shall provide the capability to protect the crew from receiving a TBD radiation dose over TBD period. B. Ward 03/03/92 Rationale: Derived from crew safety requirements. The allowable dose is presently under study.	?PG	303B
The habitation sub-segment shall provide the capability to support a nominal crew within TBD hours following crew landing. K. Joosten/A. DuPont 03/03/92 Rationale: The habitation system is planned to be transitioned from unmanned to manned mode remotely. However, some crew checkout may be necessary. This TBD will be determined based upon further study. It will not exceed the maximum allowable time the lander can support the crew minus time reserved for contingency.	?PR	440

	MAFE	
The habitation sub-segment shall provide the capability to store TBD consumables. Joosten/Carpenter 03/04/92 Rationale: Flow down from the surface segment requirement. Crew consumables to be stored within the habitation sub-segment may include but are not limited to food and life support gasses. Specific requirements are TBD.	?PG	501
The habitation sub-segment shall provide two (2) independent paths for crew egress. K. Joosten/A. DuPont 03/03/92 Rationale: Flowdown from crew safety requirements. In the event of fire or other emergency which may block crew access to the airlock, an emergency exit (hatch) must be provided for crew egress.	?PR	443
The habitation sub-segment shall provide the capability to support manned intravehicular activities. K. Joosten/A. DuPont 03/03/92 Rationale: Flowdown from science requirements to perform activities which include but are not limited to geoscience and life science tasks. In addition, there are human support and operations requirements to perform activities which include b ut are not limited to IVA habitat maintenance, EVA suit maintenance and crew meal preparation.	?PR	444
The habitation sub-segment shall provide the capability to accommodate ingress/egress of a TBD injured crewmember. J. Carpenter 03/03/92 Rationale: Flowdown from crew safety requirements.The habitat and airlock must be designed to accommodate transfer of an injured crewmember to provide emergency care. Injuries and illnesses are a finite possibility in the remote and hostile envi ronment of the Moon. The care of of an ill/injured crewmember can impact design configurations. The TBD is the extent of injury which is to be used for design purposes.	?PR	445

	MAFE	
The habitation sub-segment shall provide the capability to control interior lunar dust contamination within TBD magnitude. K. Joosten/A. DuPont 03/24/92 Rationale: Lunar dust is both very fine and very abrasive, potentially causing medical problems for the crew and impairing the operations of systems within the habitation sub-segment. The TBD will be determined after further study of actual areas of risk, dust control concepts and their cost/complexity consequences.	?PR	446A
Habitation sub-segment elements shall be capable of being re-outfitted for a nominal lunar surface stay. K. Joosten/A. Dupont 03/03/92 Rationale: Option 2 for the DRM calls for a revisit to an established outpost. Re-outfitting includes all maintenance and replacement which must be performed to support lunar surface activities and support the crew during a nominal lunar surface mission. This requirement also allows for future upgrades on subsequent missions.	?PG	260B
Habitation sub-segment elements shall be capable of being resupplied for a nominal lunar surface stay. K. Joosten/A. DuPont 03/03/92 Rationale: Option 2 for the DRM calls for a revisit to an established outpost. To enable this, consumables (oxygen, water, food, etc.) and equipment (suits, tools, spare parts, etc.) required to support lunar surface activities and the crew during a nominal lunar surface mission must be replenished.	?PG	261B
The habitation sub-segment shall allow future integration of additional pressurized volume. K. Joosten/A. DuPont 03/03/92 Rationale: Provides for outpost expansion. Integrated volume implies shirtsleeve access between the original outpost and future habitable areas. Further trades and design studies must be done before this requirement can be made firm.	?PG	249A

	MAFE	
The habitation sub-segment shall provide the capability to remain dormant for up to TBD months. K. Joosten/A. DuPont 03/03/92 Rationale: The primary DRM calls for revisits to the outpost every six months, however contingency must be built into system design in the event of schedule interruption. Dormant includes minimal thermal protection, power and other support to systems necessary to prevent harm while the outpost is uninhabited. The TBD will be determined after additional study.	?PR	447
The habitation sub-segment shall provide the capability to provide power to the cargo lander element. K. Joosten/A. DuPont 03/03/92 Rationale: Preliminary trades indicate that considerable mass savings can be made by supplying power to the lander from the habitat vs. having a separate power supply for the lander. This results in allowing more payload capability for the cargo lander which may be used by the habitation system. This functional allocation has been based on preliminary trades. Additional trades must be done before this allocation becomes firm.	?PR	448
The habitation sub-segment shall provide the capability to recharge the rover element. K. Joosten/A. DuPont 03/03/92 Rationale: Preliminary trades indicate that recharging the rover from the habitation system is the most efficient approach since the habitation system must have a considerable power capability. This functional allocation has been based on prelim inary trades. Additional trades must be done before this allocation becomes firm.	?PR	449
The habitation sub-segment shall provide the capability to provide power for the ISRU demonstration unit. K. Joosten/A. DuPont 03/03/92 Rationale: Preliminary trades indicate that powering the ISRU demo unit from the habitation system is the most efficient approach since the habitation system must have a considerable power capability. This functional allocation has been based on preliminary trades. Additional trades must be done before this allocation becomes firm.	?PR	450

	MAFE	
5.2.2.1.1 Habitat Element	?PG	H198
The habitat element shall provide the capability to support science activities. J. Carpenter 03/03/92 Rationale: Flow down from science requirements and habitation requirements to support IVA. Laboratory space and elements such as communication with EVA crewmembers are support functions. Potential activities include but are not limited to sample identification and "high grading" of samples for return payload determination and acquisition of life sciences samples.	?PR	451
The habitat element shall provide the capability to support engineering demonstration activities. J. Carpenter 03/03/92 Rationale: Flow down from engineering requirements and habitation requirements to support IVA. Laboratory space and all support elements such as communication with EVA crewmembers are support functions.	?PR	452
The habitat element shall provide the capability to monitor crew health. J. Carpenter 03/03/92 Rationale: Flowdown from the human support requirement to maintain the health, safety, and productivity of the crew. Monitoring crew health includes but is not limited to monitoring for health and physical conditioning.	?PR	453
The habitat element shall provide crew sleeping accommodations. J. Carpenter 03/03/92 Rationale: Flowdown from the human support requirement to maintain the health, safety, and productivity of the crew.	?PR	454

	MAFE	
The habitat element shall provide the capability for the crew to maintain habitat hygiene. J. Carpenter 03/03/92 Rationale: Flowdown from the human support requirement to maintain the health, safety, and productivity of the crew. Maintaining habitat hygiene includes housekeeping activities which include but are not limited to dust removal, clean up follo wing crew meals, and waste disposal/management.	?PR	455
The habitat element shall provide the capability for the crew to maintain personal hygiene. J. Carpenter 03/03/92 Rationale: Flowdown from the human support requirement to maintain the health, safety, and productivity of the crew. Maintaining crew personal hygiene includes but is not limited to cleaning, brushing teeth, and other toiletries.	?PR	456
5.2.2.1.2 Airlock Element	?PG	H275
The airlock element shall provide the capability for crew ingress and egress between the habitat and the lunar environment. J. Carpenter 03/04/92 Rationale: This is the primary function of the airlock. Allows for crew entry into the habitat upon arrival, ingress/egress to support EVA activities, and crew egress prior to lunar departure.	?PG	340A
The airlock element shall provide the capability for cargo ingress and egress between the habitat and the lunar surface. J. Carpenter 03/03/92 Rationale: The airlock must be designed to accommodate cargo. This cargo may be for reoutfitting, resupply, maintenance or repair. Example cargo includes but is not limited to habitat laboratory equipment, consumables (food, air , water), sampl e containers, maintenance equipment, EVA suits and logistic replacement units.	?PR	457

	MAFE	
5.2.2.2 Surface Transportation Sub-Segment Comment: Replaces H201: 5.2.2.2 Surface Mobility Sub-Segment	?PG	H201A
The surface transportation sub-segment shall provide the capability to transport EVA crew on the lunar surface. J. Carpenter 03/24/92 Rationale: Crew transportation is a critical capability to fulfill mission, science and engineering objectives. These objectives require extensive traverses which cannot be accommodated through crew travel on foot.	?PG	342B
The surface transportation sub-segment shall provide the capability to transport cargo on the lunar surface. J. Carpenter 03/24/92 Rationale: Cargo to be transported on the lunar surface may include science payloads, science/engineering traverse return payloads, EVA tools, and the EMU/PLSS augmentation units.	?PG	341B
The surface transportation sub-segment shall provide the capability to support manned extravehicular activities. K. Joosten/A. DuPont 03/24/92 Rationale: A transportation capability is required to perform the manned lunar science and exploration activities which fulfill mission objectives. Although transportation concepts currently include an unpressurized rover, future concepts may include pressurized rovers and manned maneuvering unit-type systems.	?PG	177B
The surface transportation sub-segment shall provide the capability to transport crew from the piloted lander to the habitat upon lunar landing. K. Joosten/A. DuPont 04/01/92 Rationale: Based upon the need for the crew to transfer from the crew module to the habitat following crew landing on the Moon. A mobilized capability reduces crew fatigue and consumables. Implications beyond the first mission are under study.	?PR	172D

	MAFE	
Surface transportation sub-segment elements shall be capable of operating during all parts of the lunar day/night cycle. D. Homan 03/24/92 Rationale: Accomplishment of mission objectives dictate that the rover operate during lunar noon and lunar night. This provides an oportunity to test the rover operational capabilities under extreme conditions. This may require additional subsystems to be placed on the rover (lights, thermal augmentation, etc.).	?PG	549
Surface transportation sub-segment elements shall provide the capability to operate for the nominal mission on the lunar surface without replenishment from Earth. J. Connolly 03/24/92 Rationale: To accomplish the stated mission objectives the rover must have all maintenance, replenishment, etc. available, utilizing only the outpost resources.	?PG	550
5.2.2.2.1 Unpressurized Rover Element Comment: Replaces H276: 5.2.2.2.1 Rover Element	?PG	H276A
The unpressurized rover element shall provide the capability to transport two (2) EVA crew and a cargo of TBD mT a round trip distance of TBD km per use. J. Carpenter 04/01/92 Rationale: The number of crew is the nominal crew for rover use and corresponds to EVA crew safety requirements. The TBD mT of cargo will be derived from further assessment of science, engineering, maintenance and logistics payload requirements. The round trip distance will be based upon giving the crew the maximum possible coverage to fulfill science objectives, while being constrained by achievable rover performance vs. mass trades.	?PG	202E
The unpressurized rover element shall provide the capability to transport four (4) EVA crew a round trip distance of TBD km per use. J. Carpenter . 04/01/92 Rationale: The nominal mission calls for the crew to be transported from the crew module to the habitat and back again.	?PG	615

	MAFE	
The unpressurized rover element shall provide the capability for use on consecutive crew days. J. Connolly 03/24/92 Rationale: Based on crew activity timeline the rover is used on consecutive days.	?PG	551
5.2.2.3 EVA Sub-Segment	?PG	H542
The EVA sub-segment shall provide the capability to sustain the crew during extravehicular activity (EVA) on the lunar surface. J. Carpenter 03/24/92 Rationale: Flow down from surface segment requirements to support science research, engineering test and demonstration, and maintenance activities on the lunar surface. Life support capability during EVA is the primary function of the EMU/PLSS.	?PG	343B
The EVA sub-segment shall provide a capability for simultaneous EVA by all crewmembers. J. Connolly 03/24/92 Rationale: Provides for simultaneous crew transfer (initial crew transfer of operations to habitat, contingency evacuations, etc.).	?PG	554
The EVA system sub-segment shall allow for TBD level of dust removal prior to entry to habitat or crew module. J. Connolly 03/24/92 Rationale: Excessive lunar dust in the habitat or crew module is not desirable. Removal of dust from the EMU is a key step in containing overall dust contamination.	?PG	555

MAFE

5.2.2.3.1 PLSS Element ?PG H277A
Comment: Replaces H277: 5.2.2.2.2 EMU/PLSS Element

The PLSS element shall provide the capability to support 17 (E) EVAs per nominal mission inclusive of contingency. ?PR 460B
J. Carpenter 04/01/92
Rationale: Flow down from the surface segment requirement to provide the capability to support EVA. The performance estimate is based upon preliminary estimates of EVAs required for science/engineering and maintenance/support as well as transportation between the piloted crew module and the habitat. An additional assumption is that a given crewmember is only involved in half the estimated total EVAs. Total number of element lifetime uses is under study.

5.2.2.3.2 EMU Element ?PG H543

The EMU element shall provide the capability for 17 (E) EVAs per nominal mission inclusive of contingency. ?PR 557A
J. Carpenter 04/01/92
Rationale: Flow down from the surface segment requirement to provide the capability to support EVA. The performance estimate is based upon preliminary estimates of EVAs required for science/engineering and maintenance/support as well as transportation between the piloted crew module and the habitat. An additional assumption is that a given crewmember is only involved in half the estimated total EVAs. Total number of element lifetime uses is under study.

The EMU element shall provide the capability for the crew to consume food and drink during EVA. ?PR 461A
J. Carpenter 03/24/92
Rationale: Flowdown from the human support requirement to maintain the health, safety, and productivity of the crew and the science EVA duration requirement.

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The EMU element shall provide the capability for human waste management during EVA. J. Carpenter 03/24/92 Rationale: Flowdown from the human support requirement to maintain the health, safety, and productivity of the crew and the science EVA duration requirement.	?PR	462A
5.2.2.4 Support Equipment Sub-Segment Comment: Replaces H200A: 5.2.2.3 Support Equipment Sub-Segment	?PG	H200B
The support equipment sub-segment shall provide the capability to perform lunar-based support functions. K. Joosten/A. DuPont 03/24/92 Rationale: Lunar based support functions are any equipment or capability needed on the lunar surface to complement and support the major elements. Examples may include but are not limited to communications, lunar surface-based navigation support, information management, and logistics equipment.	?PG	344B
5.2.2.4.1 Lunar Surface Communications Comment: Replaces H278A: 5.2.2.3.1 Communications	?PG	H278B
The capability shall be provided to communicate between elements on the lunar surface. J. Carpenter 03/03/92 Rationale: This includes lunar surface-based monitoring of science and engineering payload elements as needed and crew communications during EVAs.	?PG	345A
The capability shall be provided to communicate between elements on the lunar surface and elements on the Earth's surface. J. Carpenter 03/03/92 Rationale: This includes communications between the habitat, crew, payloads, and Earth.	?PG	347A

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5.2.2.4.2 Navigation ?PG H279B
Comment: Replaces H279A: 5.2.2.3.2 Navigation

The capability shall be provided to aid the navigation of elements in lunar orbit. ?PG 348A
J. Carpenter 03/04/92
Rationale: Improves accuracy for elements in lunar orbit, thus reducing delta V requirements. Mitigates currently incomplete knowledge of the lunar gravitational field. The current planning assumption is for a navigation aid to be emplaced with the initial landing by a cargo lander. This functional allocation has been based on preliminary trades. Additional trades must be done before this allocation becomes firm.

The capability shall be provided to aid the navigation of elements on the lunar surface. ?PG 349A
J. Carpenter 03/04/92
Rationale: EVAs will extend beyond the horizon visible to an astronaut. Surface navigation aids increase the safety of EVAs which are distant from the habitat. The current planning assumption is for a navigation aid to be emplaced with the initial landing by a cargo lander. This functional allocation has been based on preliminary trades. Additional trades must be done before this allocation becomes firm.

The capability shall be provided to aid the navigation of piloted landers during lunar descent during the nominal mission. ?PG 374A
J. Carpenter 03/04/92
Rationale: Improves landing accuracy, thus reducing the walking distance required for crews from the piloted vehicle to the habitat. Mitigates currently incomplete knowledge of the lunar gravitational field. The current planning assumption is for a navigation aid to be emplaced with the initial landing by a cargo lander. This functional allocation has been based on preliminary trades. Additional trades must be done before this allocation becomes firm.

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The capability shall be provided to aid the navigation of ascent vehicles during lunar ascent. J. Carpenter 03/04/92 Rationale: Improves the accuracy for elements ascending to lunar orbit, thus reducing delta V requirements. Mitigates currently incomplete knowledge of the lunar gravitational field. The current planning assumption is for a navigation aid to be emplaced with the initial landing by a cargo lander. This functional allocation has been based on preliminary trades. Additional trades must be done before this allocation becomes firm.	?PG	375A
5.2.2.4.3 Information Management Element Comment: Replaces H280: 5.2.2.3.3 Information Management Element	?PG	H280A
The information management element shall provide the capability to manage information produced by surface systems during surface activities. J. Carpenter 03/03/92 Rationale: Provides for processing and storage of data generated during lunar surface activity.	?PG	350A
5.2.2.4.4 Surface Logistics Equipment	?PG	H544
The logistics equipment element shall provide the capability to preserve and protect cargo against natural and induced environments for a maximum of TBD days. J. Connolly 03/24/92 Rationale: Delivered cargo must be protected against environmental degradation. Potential types of equipment may include a pressurized carrier, unpressurized carrier, and sample return containers.	?PG	545
5.2.2.5 Science and Engineering Research Sub-Segment Comment: Replaces H203A: 5.2.2.4 Science and Engineering Research Sub-Segment	?PG	H203B
The science and engineering sub-segment shall provide the capability to perform research activities on the lunar surface. D. McKay 03/04/92 Rationale: Research activities include space physics and astronomy, geosciences (including space resource utilization), life sciences, and engineering research which will expand our knowledge of Earth, its environment, the solar	?PG	351A

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system, and the universe.

The science and engineering sub-segment shall provide the capability to perform engineering demonstrations on the lunar surface. ?PG 352A

D. McKay 03/04/92
Rationale: Engineering demonstrations will allow demonstration of prototype systems such as small scale interferometry, and the production and utilization of space resources in the lunar environment prior to the delivery of complex and heavy systems to the lunar surface. This increases the probability of success for future systems.

TBD EVAs shall be required to perform the science and engineering research activities for a nominal lunar surface stay. ?PR 463A

D. McKay 04/01/92
Rationale: This requirement is a placeholder for science-driven EVAs and may be reworded as implications of potential evolutionary paths (e.g., expanding stay times to 60 days) are studied. The TBD value will include traverse, set-up and start-up of equipment, and planned equipment adjustments. The number of maintenance and logistics EVAs will be included as a separate supportability requirement.

The science and engineering research sub-segment shall provide the capability to perform robotic extravehicular activities. ?PR 178B

D. McKay/D. Eppler 03/24/92
Rationale: In order to perform the planned science and exploration activities a robotic rover is required to expand the region which may be studied. Current assumptions are for a science rover to be capable of robotically deploying instruments, collecting samples and making in situ measurements up to 100 (E) km from the outpost.

	MAFE	
5.2.2.5.1 Space Physics and Astronomy Function Comment: Replaces H353A: 5.2.2.4.1 Space Physics and Astronomy Function	?PG	H353B
The space physics and astronomy function shall provide the capability to perform space physics research from the lunar surface. D. McKay 03/03/92 Rationale: The data gathered from space physics research will be a major component in environmental characterization, as called for in requirement 207.	?PG	354A
The space physics and astronomy function shall provide the capability to perform astronomy research from the lunar surface. D. McKay 03/03/92 Rationale: Astronomical research from the lunar surface flows from the requirement to establish prototype observatory instruments and demonstrate operational characteristics. (209) It will assist the development of basic scientific data and the understanding of operational and logistics constraints to the operation of telescopes on the lunar surface.	?PG	355A
5.2.2.5.2 Geosciences Function Comment: Replaces H356A: 5.2.2.4.2 Geosciences Function	?PG	H356B
The geosciences function shall provide the capability to perform geosciences research on the lunar surface. D. McKay 03/03/92 Rationale: Conducting geoscience research flows from top-level science objectives to initiate environmental characterization and conduct regional exploration of the lunar surface.	?PG	357A
The geosciences function shall provide the capability to perform space resource utilization research on the lunar surface. D. McKay 03/03/92 Rationale: Space resource utilization research flows from the top-level objectives to delineate lunar resources local to the outpost, deploy and operate resource demonstration projects, and test the use of space resource utilization products.	?PG	358A

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5.2.2.5.3 Life Sciences Function Comment: Replaces H359A: 5.2.2.4.3 Life Sciences Function	?PG	H359B
The life sciences element shall provide the capability to perform life sciences research on the lunar surface. D. McKay 03/03/92 Rationale: Flow down from the top-level objective to conduct life science research (224).	?PG	360A
5.2.2.5.4 Engineering Research Function Comment: Replaces H361A: 5.2.2.4.4 Engineering Research Function	?PG	H361B
The engineering function shall provide the capability to perform engineering tests on the lunar surface. D. McKay 03/03/92 Rationale: Engineering tests may include evaluation of subsystems and prototypes of future equipment, as well as performance testing of existing equipment in the lunar environment. Tests in the lunar environment can increase the probability of success for this equipment.	?PG	362A
The engineering function shall provide the capability to perform engineering demonstrations on the lunar surface. D. McKay 03/03/92 Rationale: Engineering demonstrations allow proof of concept demonstration of prototypes for future lunar surface elements and processes. Demonstration in the lunar environment increases the probability of success for these elements and processes.	?PG	363A
5.2.3 Support Function Segment	?*R	H156
This entire section is in work and under review. It has not been approved by the Exploration Control Board. J. Carpenter 04/01/92	?*R	D616

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The support function segment shall provide and sustain an end-to-end capability to utilize flight and surface elements to accomplish mission objectives. K. Joosten/A. DuPont 03/03/92 Rationale: The intrinsic capabilities of the flight and surface elements must be configured, managed, and maintained to accomplish mission objectives. This statement captures the overall function of the support function segment. The flight and surface segments provide the basic equipment to accomplish the FLO mission. The support function segment identifies the additional capabilities that are needed to perform a mission using this equipment. Thus, the support function segment reflects the additional capability that is needed to use the primary equipment (flight and surface) within a particular operations context. The following statements for the most part capture generic functions that must be provided to use the flight and surface segment equipment to accomplish a mission end-to-end.	?*R	330A
The Support Segment shall be able to support a rate of four FLO missions a year. J. Carpenter 03/03/92 Rationale: The current working requirement for mission rate.	?*R	464
The Support Segment shall provide the capability to support 60-day launch centers. J. Carpenter 03/27/92 Rationale: The current working requirement for time between crew and cargo missions.	?*R	465A
The support function segment shall provide the capability to accommodate any flight/surface mission sequence of cargo and piloted missions. D. Lamar 03/27/92 Rationale: Follow on missions are currently undefined. This requirement provides follow on mission flexibility. The system will potentially be called upon to perform for mission sequences such as cargo/piloted, piloted/piloted, or cargo/cargo. Therefore, this requirement serves to anticipate cost-effectiveness of implementing capability up front rather than incremental expansion.	?*R	558

	MAFE	
5.2.3.1. Mission Support and Control Sub-Segment	?*R	H185
The mission support and control sub-segment shall provide capability for mission support and control for First Lunar Outpost missions. D. Lamar 03/27/92 Rationale: The level of implementation of this requirment will be based upon the need to provide command and monitor functions in support of First Lunar Outpost missions and upon the crew's capability to perform autonomous operations. The allocation of these functions to ground, flight, and surface systems is yet to be determined.	?*R	162C
The mission support and control sub-segment shall provide capability for real-time operations to accommodate routine, operationally complex and unscheduled operations. D. Lamar 03/27/92 Ratoinale: Provides for flexibility and evolution. Routine operations are those that are periodic and well understood. Complex operations are those that are new or difficult. Unscheduled activities are deviations from nominal plans. This capability should allow for near term (e.g., daily) planning and the ability to handle unforeseen situations.	?*R	559
Mission support and control sub-segment shall provide capability for a transition from ground based control to flight and surface based control and autonomy. D. Lamar 03/27/92 Rationale: Operations capability will evolve over the lifetime of the program with operations shifting from primarily Earth-based to vehicle/surface based. This evolution is required to reduce operations overhead. This provides for the further evolution of lunar capability wherein evolution scenarios envision greater local capability on lunar surface. This capability is also likely to support preparation for Mars mission. Long term missions or missions at great distances may very likely demand a significant local autonomy if sophisticated accomplishments are to be gained.	?*R	372B

	MAFE	
The mission support and control sub-segment shall provide the capability to manage discarded hardware. J. Borrer 03/27/92 Rationale: Needed to prevent the uncontrolled buildup of hazardous debris and to maintain efficient operations near the lunar surface outpost. The continuous build-up of space debris near the outpost may be an undesirable result of our main goal to achieve continuous access to space. In order to maintain the efficient and safe operation of the outpost, provisions shall be provided so that non-essential or non-functioning equipment can be relocated into a central area away from essential outpost systems.	?*R	560
The mission support and control sub-segment shall provide capability to warn crew of impending solar events. D. Lamar 03/27/92 Rationale: Early warning of solar events is required to enable the crew to take adequate safety precautions. This implies a capability to monitor solar activity that is not currently allocated. This function must be provided for both flight and surface operations.	?*R	561
The Mission Support and Control Sub-Segment shall maintain a capability during all mission phases to abort the mission and safely return the flight crew to Earth. J. Carpenter 03/24/92 Rationale: Planning and procedures must be such that abort conditions can be identified and suitable recovery options are preserved. The monitoring, and command and control must be able to identify abort conditions and transition to effective recovery modes. For the first FLO mission, abort is tentatively defined as the capability to safely return the crew to Earth.	?*R	466A

	MAFE	
5.2.3.1.1 Mission Planning and Preparation Comment: Replaces H281: 5.2.3.1.1 Pre-Flight Element	?*R	H281A
The Mission Planning and Preparation function shall provide a capability to plan and develop specific missions to accomplish mission objectives. J. Carpenter 03/27/92 Rationale: To accomplish mission objectives, objectives must be translated into specific payloads and mission operations. FLO system configurations that are suitable for accomplishing the mission must be developed. Activities could include (but are not limited to): procedures development and verification; mission data configuration; flight and trajectory planning; and nominal and contingency timeline and schedule definitions; manifesting of spares and mission elements, science operations, maintenance actions, crew operations, and consumables management and resource allocation/scheduling. The capability for real-time replanning will also be required.	?*R	467A
The mission planning and preparation function shall provide capability for the preparation of support systems for First Lunar Outpost operations. D. Lamar 03/27/92 Rationale: Support systems (e.g., consoles, tools) and facilities (e.g., control centers, networks) need to be configured to support each FLO mission.	?*R	562
The mission planning and preparation function shall provide capability to develop mission operations data sets for space, Earth, supporting systems and interfaces. D. Lamar 03/27/92 Rationale: Data are needed to test, diagnose, validate, certify, train and operate space, Earth and supporting systems and interfaces prior to and during the mission.	?*R	563

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The mission planning and preparation function shall provide capability to develop operational flight and surface sequence of events and timelines. ?*R 564

D. Lamar 03/27/92
Rationale: Real-time mission operations will need nominal and contingency event sequences and timelines. This must be prepared in advance to optimize system performance and to identify and to react to off-nominal situations. The allocation of this capability among Earth, flight, surface, and automated systems is TBD.

5.2.3.1.2 Launch Operations ?*R H282A
Comment: Replaces H282: 5.2.3.1.2 Flight Element

The launch operations function shall provide capability for pad operations. ?*R 565

D. Lamar 03/27/92
Rationale: Tanking, system servicing, sequencing from ground support to internal utilities, and critical system statusing are essential to the launch operations function and must be provided to the vehicle in a controlled area under strict security and safety regulations.

The launch operations function shall provide capability for launch control. ?*R 566

D. Lamar 03/27/92
Rationale: Management and control of the launch vehicle sequencing and statusing must be provided.

The launch operations function shall provide capability for operational coordination and interfacing. ?*R 567

D. Lamar 03/27/92
Rationale: Launch operations interfaces with other functions and services in order to insure the timely sequencing of launch events and to insure that launch criteria are maintained. These functions and services include, but may not be limited to, mission operations, meteorology, and range safety.

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5.2.3.1.2.1 Pad Operations ?*R H568

The pad operations function shall provide capability to provide utilities to the integrate launch vehicle via ground support equipment until lift-off. ?*R 569

D. Lamar 03/27/92
Rationale: Because vehicle utilities are provided by flight-critical consumables, it is prudent to provide ground utilities to the vehicle, via ground support equipment, until as late in the countdown as practical. Furthermore, if there is a delay or hold in the count, ground utilities are still available to safe and deservice the launch vehicle.

The pad operations function shall provide capability to access vehicle servicing areas. ?*R 570

D. Lamar 03/27/92
Rationale: Critical areas of the launch vehicle and payload must be provided access in order to close out the vehicle in preparation for launch. While for FLO, the goal is to minimize this access whenever possible, and to provide this access via the MLT swing-arms wherever practical, some access to the vehicle for on-pad servicing will be necessary.

The pad operations function shall provide capability to tank payload fluids. ?*R 571

D. Lamar 03/27/92
Rationale: Hazardous fluids should be loaded into the payload under the secure environment and strict control which the launch pad can accommodate. These should be loaded into the payload as late in the count as practical.

The pad operations function shall provide capability to tank payload fluids. ?*R 572

D. Lamar 03/27/92
Rationale: Hazardous fluids should be loaded into the payload under the secure environment and strict control which the launch pad can accommodate. These should be loaded into the payload as late in the count as practical.

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The pad operations function shall provide capability for ingress/egress of flight crew and support personnel into crew cab. D. Lamar 03/27/92 Rationale: For piloted missions, access to the crew cab is, obviously, essential. For physiological as well as crew safety reasons, the crew should access the crew cab as late in the count as is practical.	?*R	573
The pad operations function shall provide capability for emergency egress of flight crew. D. Lamar 03/27/92 Rationale: In the event of a launch system malfunction that potentially threatens the safety of the crew, this allows the crew to get a safe distance from the vehicle in as short a time as possible.	?*R	574
The pad operations function shall provide capability to refurbish the pad post-launch. D. Lamar 03/27/92 Rationale: The FLO is not a single-launch mission. The pad must be turned around to accommodate the piloted mission once the cargo mission has been launched. This will be an ongoing requirement. This requirement also affects the logistics capability to be provided at the launch site, since the equipment and tools to turn the pad around must be available and in place to maintain flight rates and schedules.	?*R	575
5.2.3.1.2.2 Launch Control	?*R	H576
The launch control function shall provide capability to manage and sequence launch vehicle and ground support assets. D. Lamar 03/27/92 Rationale: The countdown is the process by which the launch vehicle is systematically and sequentially switched from ground-based utilities to internal utilities in preparation for launch. Launch control allows this sequencing to take place, while performing vital step-by-step checks to insure that each launch function is successfully completed before proceeding to the next.	?*R	577

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The launch control function shall provide capability to utilize the launch commit criteria and execute time critical Go/No Go decisions. ?*R 578

D. Lamar 03/27/92
Rationale: Part of the launch control function is to insure that while sequencing through the functional steps proceeding to vehicle launch, the overall status of the vehicle is maintained within defined and measurable safety limits. Periodic vehicle status checks are performed prior to proceeding to succeeding steps in the launch process.

The launch control function shall provide real-time launch system update/interface data to mission operations. ?*R 579

D. Lamar 03/27/92
Rationale: This assumes a split function due to available agency resources. In fact, the function can be split without defining Center roles and missions. The launch function can be defined to continue to an arbitrary point in flight, at which time another function would commence, due to mission phasing.

5.2.3.1.2.3 Launch Coordination and Interfacing ?*R H580

The launch coordination and interfacing function shall provide capability for launch operations to interface with flight operations for launch coordination and hand-off. ?*R 581

D. Lamar 03/27/92
Rationale: Provides data from launch operations to other mission operational functions in order to insure continuity across mission phasing points.

The launch coordination and interfacing function shall provide capability for launch operations to interface with TISN operations for launch site tracking and data acquisition. ?*R 582

D. Lamar 03/27/92
Rationale: Communications with the vehicle is during the launch phase is provided via the agency institutional resources.

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The launch coordination and interfacing function shall provide capability for launch operations to interface with meteorological services for weather forecast and analysis data. D. Lamar 03/27/92 Rationale: Meteorology and forecast is a specialized function that launch control depends upon in order to commit for launch. Although important to launch operations, it remains a separate, specialized function.	?*R	583
The launch coordination and interfacing function shall provide the capability for launch operations to interface with range safety. D. Lamar 03/27/92 Rationale: Range safety is a specialized function that launch control depends upon in order to commit for launch. Although important to launch operations, it remains a separate, specialized function.	?*R	584
5.2.3.1.3 Flight Operations Comment: Replaces H283: 5.2.3.1.3 Surface Element	?*R	H283A
The Flight Operations Function shall provide capability to monitor and control FLO flight systems, elements, and operations during the flight phase of a mission. J. Carpenter 03/03/92 Rationale: The Flight phase is tentatively defined as commencing at liftoff and ending at touch down. This definition is subject to change with further clarification of operations. Considerations during the flight phase would include (but not l imited to) system health, abort conditions, disposal of spent stages, and so forth. The allocation of the control and monitoring functions between earth, the lunar surface crew, and automated systems is TBD.	?*R	469
The flight operations function shall provide capability for Earth-based real-time operations to monitor and control the space transfer elements during First Lunar Outpost missions. D. Lamar 03/27/92 Rationale: Even for highly automated flight elements, the need will exist for ground support as a result of system failures or contingencies.	?*R	585

	MAFE	
The flight operations function shall provide capability for Earth based real-time operations to check-out and verify flight systems operations. D. Lamar 03/27/92 Rationale: Even for highly automated flight elements, the need will exist for ground support as a result of system failures or contingencies.	?*R	586
The flight operations function shall provide capability for crew and Earth-based operations to terminate, initiate, or inhibit critical sequences. D. Lamar 03/27/92 Rationale: Even for highly automated flight elements, the need will exist for human support as a result of system failures or contingencies.	?*R	587
The flight operations function shall provide capability for Earth based real-time operations to support overall vehicle configuration management and provide coordination and conflict resolution. D. Lamar 03/27/92 Rationale: Even for highly automated flight elements, the need will exist for human support as a result of system failures or contingencies.	?*R	588
The flight operations function shall provide capability for Earth-based real-time operations to monitor flight subsystem operations at a level to detect malfunctions, failures, or performance degradation that jeopardize the safety of the crew and overall integrity of the FLO. D. Lamar 03/27/92 Rationale: Even for highly automated flight elements, the need will exist for human support as a result of system failures or contingencies.	?*R	589
The flight operations function shall provide a capability to provide the recovery team with real-time Earth landing site information (including location and weather). D. Lamar 03/27/92 Rationale: Abort conditions may result in a change in landing site. Trajectory constraints limit the Earth landing site capability once TEI has occurred. Accurate landing site weather predictions will be required to plan TEI and mid-course corrections.	?*R	590

	MAFE	
5.2.3.1.4 Surface Operations Comment: Replaces H284: 5.2.3.1.4 Post-Flight Element	?*R	H284A
The surface operations function shall provide a capability to monitor and control FLO systems, elements, and operations on the lunar surface. K. Joosten/A. DuPont 03/03/92 Rationale: The statement is intended not to restrict the capability to a particular period, e.g., only when crew are present. In general, there is a potential for surface activities before crew arrive (e.g., habitat start up) and after crew depart (e.g., science packages). In addition, equipment may be in a dormant or quiescent mode whether crew are present or not (e.g., a rover may be quiescent during crew sleep periods). The allocation of the control and monitoring functions between Earth, the lunar surface crew, and automated systems is TBD.	?*R	328A
The surface operations function shall provide capability for Earth based real-time operations to checkout and verify surface systems operations during the mission. D. Lamar 03/27/92 Rationale: Even for highly automated flight elements, the need will exist for ground support as a result of system failures or contingencies.	?*R	591
The surface operations function shall provide capability for Earth based real-time operations to remotely power and checkout the lunar systems critical to human occupation. D. Lamar 03/27/92 Rationale: The lunar habitat and other surface systems critical to human occupation must be verified operational prior to the crewed mission.	?*R	592
The surface operations function shall provide capability for Earth based real-time operations to support overall surface configuration management and provide coordination and conflict resolution. D. Lamar 03/27/92 Rationale: Even for highly automated flight elements, the need will exist for ground support as a result of system failures or contingencies.	?*R	593

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The surface operations function shall provide capability for Earth based real-time operations to remotely operate the lunar rover. ?*R 594

D. Lamar 03/27/92
Rationale: Many of the tasks to be carried out on the Moon can benefit from the ability to teleoperate the rover. This capability would decrease crew time requirements and increase science yield.

5.2.3.1.5 Sustaining Operations ?*R H331

The Sustaining Operations Function shall provide a capability to monitor and control systems on the lunar surface while these systems are in a dormant state. ?*R 470

J. Carpenter 03/03/92
Rationale: This provides the capability to sustain operational capability of FLO systems that have not been retired as well as to deal with potential hazards that may arise from systems that have been retired. Activities may include monitoring system status, configuration control, and identification and resolution of hazards. The statement is not intended to restrict the capability to a particular period, e.g, only when crew are present. In general, there is a potential for surface activities before crew arrive (e.g.. habitat start up) and after crew depart (e.g.. science packages). In addition, equipment may be in a dormant or quiescent mode whether crew are present or not (e.g., a rover may be quiescent during crew sleep periods). The allocation of the control and monitoring functions between earth, the lunar surface crew, and automated systems is TBD. Depending on the method of disposal of spent flight elements, there may be a similar requirement for monitoring elements that remain in orbit.

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5.2.3.1.6 Post-Flight Operations	?*R	H332
The post-flight operations function shall provide a capability to evaluate mission technical and science data, procedures, etc. and incorporate lessons learned into subsequent mission planning. D. Lamar 03/27/92 Rationale: Data gathered during a mission will be utilized to improve system design, procedures, etc. and will reduce operations complexity.	?*R	329B
The post-flight operations function shall provide for the processing of post-flight products and archived data. D. Lamar 03/27/92 Rationale: Mission and scientific products and data must be processed in order to achieve maximum benefit from the mission and to prepare for subsequent missions.	?*R	595
5.2.3.2 Integration and Testing Sub-Segment	?*R	H285
The integration and testing sub-segment shall provide a capability to prepare, integrate, and verify elements and system configurations for mission operations. K. Joosten/A. DuPont 03/03/92 Rationale: This statement deals with configuring an integrated system of elements and certifying that it is ready to begin mission operations. Various flight, surface, and operations support elements must be integrated and verified as being ready for mission operations. Element preparation would entail processing subsequent to delivery (addition of kits, mission specific data loads, etc.). Verification in this context addresses that elements as installed are in a working condition and properly configured for the mission. Verification that a design or item meets specification is envisioned as being accomplished as part of the procurement or acceptance process and is not intended to be included in this function.	?*R	334A

	MAFE	
5.2.3.2.1 Vehicle Integration Comment: Replaces H286: 5.2.3.2.1 Vehicle Integration Element	?*R	H286A
The vehicle integration element shall provide a capability to integrate and verify FLO launch configurations with ground launch systems. K. Joosten/A. DuPont 03/03/92 Rationale: This statment deals with configuring the items to be launched and the various systems. Activities may include (but are not limited to) transport of launch items to the launch site, check-out, and integration with ground launch systems. Depending on design and the flow of ground processing, payloads may have been loaded onto the flight vehicle stack and integration may require interface the vehicle with ground support systems that support the payloads.	?*R	157B
The vehicle integration function shall provide the capability for launch vehicle integration. D. Lamar 03/27/92 Rationale: This statement deals with configuring flight segment elements and assembling them into a launch stack. Envisioned activities include integration, fueling, and so forth. Depending on implementation, this may include configuring the physical and software features of individual elements. Payloads may have been integrated into the appropriate flight element before the element is integrated into the launch stack.	?*R	327B
The vehicle integration function shall provide the capability for integrated vehicle integration and transport. D. Lamar 03/27/92 Rationale: Provides for the mating and verification checks of the payload to the launch vehicle, and the transport of the integrated stack to the launch pad.	?*R	596

	MAFE	
5.2.3.2.1.1 Launch Vehicle Processing	?*R	H597
The launch vehicle processing function shall provide capability to transport individual core and booster elements of the launch vehicle from the point of arrival to the vehicle processing location. D. Lamar 03/27/92 Rationale: It is essential for the capability to move and handle the different elements of the launch vehicle to exist at the launch site.	?*R	598
The launch vehicle processing function shall provide capability to stack, mate, integrate, and test FLO launch vehicles. D. Lamar 03/27/92 Rationale: These functions constitute launch vehicle processing.	?*R	599
The launch vehicle processing function shall provide capability to handle hazardous materials. D. Lamar 03/27/92 Rationale: Processing launch vehicles may include the handling of hazardous materials; such as toxic and/or corrosive fluids, solid rocket motors, and ordnance.	?*R	600
5.2.3.2.1.2 Integrated Vehicle Processing	?*R	H601
The integrated vehicle processing and transport function shall provide capability to mate payload configuration to launch vehicle. D. Lamar 03/27/92 Rationale: An essential process in preparation for launch.	?*R	602
The integrated vehicle processing and transport function shall provide capability to perform integrated checkout of launch stack. D. Lamar 03/27/92 Rationale: This insures that the launch vehicle is properly configured for launch, and that the payload and launch vehicle properly communicate.	?*R	603

	MAFE	
The integrated vehicle processing and transport function shall provide capability to transport integrated vehicle to the launch pad. D. Lamar 03/27/92 Rationale: Once the vehicle has been integrated and checked out, the launch stack must be transported to the pad area in order to proceed through the sequenced steps leading to launch within a controlled and secure environment.	?*R	604
5.2.3.2.2 Payload Integration	?*R	H287A
The payload integration function shall provide the capability for payload integration. D. Lamar 03/27/92 Rationale: Consists of assembly and integration of "major" payload elements. Payload processing to be performed apart from the launch site. No deferred work or modifications will be performed at the launch site. Consistent with "ship and shoot" philosophy, while allowing launch site rework or repair. The term "Payload" is meant to include all items carried by the launch vehicle. This includes the crew cab, etc. on piloted missions.	?*R	158C
The payload integration function shall provide capability to receive individual payload elements of the flight manifest at the payload integration location. D. Lamar 03/27/92 Rationale: Essential to the integration function and the processes at the launch site leading to launch. Impact is that payloads are received directly at the integration facility; no processing at the launch site to be accommodated.	?*R	605
The payload integration function shall provide capability for storage of mission spares, tools, and other mission support equipment. D. Lamar 03/27/92 Rationale: Necessary payload logistical capability at the launch site; consistent with payload rework or repair.	?*R	606

	MAFE	
The payload integration function shall provide capability to integrate major mission elements into an assembled payload. D. Lamar 03/27/92 Rationale: Definition of payload integration, as contrasted with payload processing.	?*R	607
The payload integration function shall provide capability to handle hazardous material. D. Lamar 03/27/92 Rationale: This includes installation of ordinance. Payload integration may include the handling of hazardous materials; such as toxic and/or corrosive fluids, solid rocket separation motors, ordnance, and possible radioactive materials.	?*R	608
The payload integration function shall provide capability to encapsulate FLO payloads. D. Lamar 03/27/92 Rationale: It is essential to the "ship and shoot" philosophy to close out payload integration prior to mating and interface verification with the launch vehicle. Thus, the final payload integration process is the installation of the flight shroud prior to transport to the vehicle integration facility.	?*R	609
The payload integration function shall provide capability to transport payload configuration from the cargo integration location to the integrated vehicle processing location. D. Lamar 03/27/92 Rationale: This constitutes the release of the payload, assembled, checked out, and ready for launch, to the next step in the processing flow at the launch site.	?*R	610

	MAFE	
The Payload Integration Function shall provide a capability to prepare and to verify surface elements for mission operations.	?*R	472

J. Carpenter 03/03/92
Rationale: Surface elements are considered to include science payloads. Note that no flight science has been identified. This statement deals with configuring payloads and certifying that they are ready for launch. Preparation would entail pro cessing subsequent to delivery (addition of kits, mission specific data loads, etc). Verification in this context addresses whether payloads are in a working condition and properly configured for the mission. Verification that a design or item meets specification is envisioned as being accomplished as part of the procurement or acceptance process and is not intended to be included in this function. This statement is not intended to specify the sequence of ground processing. Payload preparation and verification activities could conceivably occur at various steps in the flow of launch preparation: before, during, or after integration with the flight vehicle; before and during countdown.

5.2.3.2.3 System Integration	?*R	H288A
The Integration and Testing Function shall provide a capability for the overall integration and verification of elements as parts of an interoperating system.	?*R	473

J. Carpenter 03/03/92
Rationale: Element verification per se is envisioned as being accomplished as part of element procurement. This statement deals with configuring and testing the integrated system elements as being ready to begin mission operations. (Analogous t o Shuttle SAIL)

5.2.3.2.4 Software Environment	?*R	H289A
The Integration and Testing Function shall provide a capability to develop and maintain FLO system software.	?*R	474

J. Carpenter 03/03/92
Rationale: Provides general overall environment (standards, tools, etc) for developing and maintaining software. (Analogous to SSF Software Support Environment)

	MAFE	
5.2.3.3 Post Flight Support Sub-Segment	?*R	H225
The post-flight support sub-segment shall provide a capability for post-flight processing of system components and mission products. K. Joosten/A. DuPont 03/03/92 Rationale: Need to retrieve and handle data, crew, samples, and returned flight hardware at the end of the respective mission activity. Activities likely to include retrieve, capture, configure, archive, debrief, distribute, etc. The allocation and interfaces of post-flight activities among the post-flight support sub-segment, the post-flight operation function of the mission support and control sub-segment, and the science operations sub-segment require further definition.	?*R	333A
The post-flight support sub-segment shall be able to support Earth landings that occur under any lighting conditions. K. Joosten/A. DuPont 03/03/92 Rationale: Based upon the desire to provide abort capability at all times during the mission. Earth-Moon phasing restricts the control of landing target sun angle that can be obtained within the capability of the return stage.	?*R	267B
The post-flight support sub-segment shall provide the capability to support Earth landings at TBD locations. P. Campbell 03/27/92 Rationale: The range of possible landing sites must be specified.	?*R	611

MAFE

5.2.3.3.1 Recovery ?*R H290A

The recovery function shall provide a capability to recover crew, flight elements and payload that return to Earth. ?*R 226B

K. Joosten/A. DuPont 03/03/92
Rationale: Ensures capability to recover crew and hardware. This capability would include search and rescue. Currently only the crew module is identified for recovery during nominal missions. Further definition, aborts, anomalous situations may entail the location and recovery of additional items. NOTE: Since they depend upon the particular design of an item, individual recovery equipment should be identified in conjunction with individual item.

The recovery function shall provide a capability to deservice flight elements that have been returned to Earth. ?*R 227B

K. Joosten/A. DuPont 03/03/92
Rationale: Equipment must be made safe before subsequent post-flight processing. Deservicing activities may include (but are not limited to) safing pyrotechnic devices, clearing toxic materials, and so forth. This may also entail a capability to handle anomalous or abort situations.

5.2.3.3.2 Transport ?*R H291A

The transport function shall provide a capability to transport crew, flight elements and payload from the landing area to the post-flight processing facilities. ?*R 228B

K. Joosten/A. DuPont 03/04/92
Rationale: Transportation needs will depend upon the landing recovery mode and the location and hand-over of post-flight processing to other functions and sub-segments. NOTE: Since they depend upon the particular design of an item, individual transportability requirements for equipment should be identified in conjunction with the item.

	MAFE	
5.2.3.3.3 Refurbishment	?*R	H292A
The Refurbishment Function shall provide a capability to refurbish recovered elements and equipment that are designed for reuse.	?*R	475

J. Carpenter 03/03/92
Rationale: NOTE: Since they depend upon the particular design of an item, individual refurbishment facilities for equipment should be identified in conjunction with the item.

5.2.3.4 TISN Sub-Segment	?*R	H293
The TISN sub-segment shall provide an overall integrated capability for communication, information management, and navigation for the terrestrial, lunar, and in-flight systems involved in FLO missions.	?*R	335A

K. Joosten/A. DuPont 03/03/92
Rationale: Establishes capability for integrated and coordinated communication, navigation and information management. The statement is inexact as written since the scope of links and services is not clarified. It is unlikely that this sub-segment will be responsible for all communication, navigation, and information management, e.g., commercial phone lines used by mission support personnel, avionics internal to an element, etc.

The TISN Sub-Segment shall facilitate fault detection, fault isolation, and recovery control for the terrestrial, lunar, and in-flight systems involved in FLO missions.	?*R	476

J. Carpenter 03/04/92
Rationale: Establishes a capability for integrated and coordinated fault detection, fault isolation, and recovery control in transport and planetary elements during test, delivery, and operation. The statement is inexact as written since the op erations/support concepts are not clarified. It is unlikely that this sub-segment will be responsible for all fault management and recovery, e.g., some science may not utilize any common services and certain fault management and recovery may be accom modated entirely within the habitat, etc.

	MAFE	
5.2.3.4.1 Communication	?*R	H492
The Communication Function shall a provide an overall integrated capability for communication among the terrestrial, lunar, and in-flight systems during all mission phases.	?*R	477

J. Carpenter 03/03/92
Rationale: Establishes capability to meet overall integrated communication needs for FLO missions. The function is intended to integrate various local communication capabilities into an integrated architecture that meets mission needs. This may involve providing additional capabilities. Specific links to be supported are not yet defined. It is unlikely that this function will be responsible for all communication links, e.g., commercial phone lines used by mission support personnel, communications internal to an element, etc..

MAFE

5.2.3.4.2 Navigation ?*R H493

The Navigation Function shall provide an overall integrated navigation capability sufficient to meet mission objectives during all mission phases. ?*R 478

J. Carpenter 03/04/92
Rationale: Establishes capability to meet overall integrated navigation needs for FLO missions. It is intended that this function would integrate various navigation capabilities into an overall architecture that meets mission goals (e.g., precision landing at a particular target site). The navigation system will support a multitude of guidance and control functions (e.g., planetary transfers, orbital operations, descent and ascent flight phases, instrument pointing and tracking, surface roving, etc.) as well as mission planning, scientific experiments, and post-flight reconstruction and data analysis. This may involve providing additional capabilities (e.g., lunar beacons or satellites). The mission phases to be supported would include ETO, TLI, LOI, pre-deorbit, Descent/landing, surface operations, lu nar ascent, TEI, Earth approach, and Earth descent. The measurement system may include both ground (Earth) based tracking, onboard sensors, and emplaced navigation aids using necessary estimation algorithms and appropriate displays and controls required for human interaction. Currently required navigation needs are being analyzed and particular allocations are TBD.

The Navigation Function shall provide a capability for state vector updates to flight and surface vehicles. ?*R 479

J. Carpenter 03/04/92
Rationale: Ensures capability to provide vehicles with adequate navigation information. Navigation is the determination of the current and future translational and rotational vehicle state vector (position, velocity, attitude, attitude rate) for space flight, atmospheric flight, and mobile surface systems. This state vector may be inertial, relative to another spacecraft or celestial body, or relative to a surface centered coordinate frame. Potential updates might support ETO, TLI, LOI, pre-deorbit, Descent/landing, lunar ascent, TEI, Earth approach, and Earth descent.

	MAFE	
The Navigation Function shall provide data for post-flight trajectory analysis and reconstruction.	?*R	480

J. Carpenter 03/04/92
Rationale: Ensures capability to provide post-flight analysis and trajectory reconstruction. Post-flight reconstruction products may potentially support system performance evaluations (navigation, sensors, vehicle performance, etc), anomaly investig ation, and science.

5.2.3.4.3 Information Management	?*R	H494
The Information Management Function shall provide a capability for an overall integrated information management capability for FLO mission and support activities.	?*R	481

J. Carpenter 03/04/92
Rationale: Establishes capability to meet overall integrated information management needs for FLO missions. The function is intended to integrate various local information capabilities into an integrated compatible architecture that meets missi on and support needs. This may involve providing additional capabilities. This architecture is intended to be integrated in the sense that information can be accessed and exchanged efficiently. The is no intent to specify the structure of the archit ecture (e.g., distributed or centralized). Specific needs to be supported are not yet defined. It is unlikely that this function will be responsible for all information management, e.g., avionics internal to an element.

The information management function shall provide capability to archive mission and science data.	?*R	612

D. Lamar 03/27/92
Raitonale: Data will be required for post-mission evaluation and science analysis.

	MAFE	
5.2.3.5 Training Sub-Segment	?*R	H295A
The training sub-segment shall provide capability for simulation and training for First Lunar Outpost missions. D. Lamar 03/27/92 Rationale: The FLO mission will involve highly complex hardware and software systems. The crew and mission operators will need to be trained to operate, trouble-shoot, and maintain this hardware and software. The level of implementation of this requirement will be based upon the need to provide command and monitor functions in support of First Lunar Outpost missions and upon the crew's capability to perform autonomous operations.	?*R	613
The training sub-segment shall provide a capability to train crew for First Lunar Outpost missions. K. Joosten/A. DuPont 03/03/92 Rationale: Ensures crew have skills to accomplish mission needs. Training may address equipment maintenance, science and payloads, survival, skills retention, anomalies, and so forth. Particular training facilities and functions could occur at ground facilities (e.g., WETF) or in space (e.g., inside the habitat). The allocation of capabilities to particular equipment and facilities is TBD.	?*R	163C
The Training Sub-Segment shall provide a capability to train mission support operators for First Lunar Outpost missions. J. Carpenter 03/03/92 Rationale: Ensures mission support operators have skills to accomplish mission needs, both nominal and contingency. The allocation of capabilities to particular equipment and facilities is TBD.	?*R	482
The Training Sub-Segment shall provide a capability to train ground support operators for First Lunar Outpost missions. J. Carpenter 03/03/92 Rationale: Ensures mission support operators have skills to accomplish mission needs, both nominal and contingency. The allocation of capabilities to particular equipment and facilities is TBD.	?*R	483

	MAFE	
5.2.3.6 Human Support Sub-Segment	?*R	H325
Note: Not used. See Section 5.3. J. Carpenter 03/04/92	?*R	D502
5.2.3.7 Science Operations Sub-Segment	?*R	H326
The science operations sub-segment shall provide a capability to monitor and control science operations. K. Joosten/A. DuPont 03/03/92 Rationale: Envisioned operations may include (but are not restricted to) science experiments, deployed payloads, teleoperated rover operations, and sample collection and documentation. The allocation of activities between this sub-segment and the surface operation function of the mission support and control sub-segment are TBD. Potentially, there may be ongoing equipment monitoring (e.g., telescopes and seismic stations) and remote science operations control centers. For the purposes of this statement, science includes various demonstrations such as in situ resources.	?*R	336A
The Science Operations Sub-Segment shall provide a capability to curate the products of science operations. J. Carpenter 03/03/92 Rationale: Ensures the capability to store and access products of science activities. The curation function is intended to include the distribution of products to users. For the purposes of this statement, science includes various demonstrations such as in situ resources.	?*R	484

	MAFE	
5.2.3.8 Integrated Logistics Support Sub-Segment	?*R	H294A
The integrated logistics sub-segment shall provide an integrated capability for maintenance and supply of equipment and materials to support FLO systems.	?*R	337A

K. Joosten/A. DuPont 03/03/92
Rationale: Ensures an overall integrated capability for the support of the system. The types of activities that are likely to be required include spares, resupply, storage, parts and materials procurement, inventory control, repair, maintenance, and refurbishment. The allocation of capabilities to particular equipment and facilities, both ground and in space, is TBD. Logistics carriers and supplies are considered as surface elements.

5.3 Human Support Requirements	?*R	H296
This entire section is in work and under review. It has not been approved by the Exploration Control Board.	?*R	D617

J. Carpenter 04/01/92

Systems and procedures shall be provided which maintain the health, well-being, safety, and productivity of the crew.	?*R	322A

Ward/Rummel/Brown 03/12/92
Rationale: Program and mission success depends upon active, productive, reliable, and continuous crew participation.

5.3.1 Crew Health Maintenance	?*R	H511
Crew health maintenance and medical care shall be provided for all crewmembers during all mission phases.	?*R	512A

Ward/Rummel/Brown 03/24/92
Rationale: In order for crewmembers to be productive, their health must be established prior to and maintained throughout the mission. Levels of health care are being developed at this time and will be incorporated as defined.

	MAFE	
5.3.1.1 Operational Health Care	?*R	H513
Operational health care consistent with mission risk levels shall be provided.	?*R	514A

Ward/Rummel/Brown 03/24/92
Rationale: Health care appropriate to mission parameters including duration, location, and crew activities should be available for preventing, diagnosing, and treating illness or injury. Operational health care is the umbrella function which encompasses environmental health, crew health monitoring, countermeasures, medical intervention and care, and medical management.

5.3.1.1.1 Environmental Health	?*R	H515
The capability shall be provided for the monitoring, analysis, and warning of hazards in the physical environment.	?*R	516A

Ward/Rummel/Brown 03/24/92
Rationale: Crew health will depend on an adequate assessment of the physical environment. The range of hazards to be considered will be identified during conceptual design along with alternative methods for amelioration. General guidelines for caution and warning may be found in NASA-STD-3000, Vol. 1, until the hazards specific to FLO are identified and approved.

The capability shall be provided for the monitoring of microbiology in water, air, food, and on crew access surfaces.	?*R	517A

Ward/Rummel/Brown 03/24/92
Rationale: Understanding the microbiological environment is essential for crew health in a cramped, closed, and remote environment. Monitoring shall be done to the extent reasonable to maintain a healthy, safe and sanitary environment for the crew.

	MAFE	
Protective measures shall be taken to limit total crew monthly, annual, and career doses to blood-forming organs, eyes, and skin, to TBD limits. Ward/Rummel/Brown 03/12/92 Rationale: Human exposure to excessive radiation causes radiation sickness which could jeopardize mission and crew safety. Present guidelines exist only for low Earth orbit (LEO) operations; new levels are being considered for this case. Exposure limits for lunar missions have yet to be established by the NCRP Committee. The possible new limits for LEO operations may have to be used as a general guideline to plan the lunar missions.	?*R	323A
The capability shall be provided for the monitoring, analysis, and warning of radiation levels. Ward/Rummel/Brown 03/13/92 Rationale: For exploratory mission operations which place crewmembers in high-risk radiation environments, monitoring, analyzing, and predicting radiation levels on a real-time basis is essential. There are regulatory requirements to monitor and document exposure histories for occupational hazards of radiation.	?*R	518
The capability shall be provided for solar flare event monitoring and notification. Ward/Rummel/Brown 03/24/92 Rationale: This capability helps to ensure that crews will not receive large radiation doses from solar particle events that could incapacitate crews' performance.	?*R	324B
5.3.1.1.2 Crew Health Monitoring	?*R	H519
The capability shall be provided to monitor crew health during all mission phases. Ward/Rummel/Brown 03/12/92 Rationale: Mission success will depend on an ability to know crew health status during all mission phases. This monitoring may consist of physical examinations, functional/stress testing of crewmembers, EVA monitoring, etc.	?*R	520

	MAFE	
The capability shall be provided for biomedical monitoring of physical and biological parameters of biomedical significance with critical information available in the surface EVA suit, in the habitat, and downlinked to Earth.	?*R	521A

Ward/Rummel/Brown 03/24/92
Rationale: Extensive planned EVA will require assessment of the capability of the EVA crew to continue their mission or identification of the source of inability to complete the mission. The current assumption is that this monitoring will be done from Earth, however, this capability may evolve to include local lunar monitoring.

5.3.1.1.3 Countermeasure Implementation	?*R	H522
Certified countermeasures to partial gravity shall be provided to maintain crew health and performance.	?*R	523

Ward/Rummel/Brown 03/12/92
Rationale: Maladaptation to space flight must be mitigated by appropriate countermeasures. Various types of countermeasures may include pharmacological measures, exercise, dietary measures, etc.

5.3.1.1.4 Medical Intervention and Care Capability	?*R	H524
Medical intervention capability consistent with mission risk levels shall be provided.	?*R	525

Ward/Rummel/Brown 03/12/92
Rationale: Ability to provide medical intervention when there is crew illness or injury is critical to mission success. Capability for medical intervention and care consists of equipment, supplies, and expertise to provide care. Intervention and care is required to minimize mission time lost due to minor injuries, illnesses and complaints, and to prevent or minimize mission disruption, permanent disability or loss of life due to common, significant medical problems that are known to affect otherwise healthy humans, primarily trauma and infectious diseases, and those anticipated due to hazards unique to the space environment (e.g., hypobaric exposure, radiation exposure, and possible complications resulting from absence of gravity).

	MAFE	
5.3.1.1.5 Medical Management	?*R	H526
The capability shall be provided for overall mission medical systems management and integration. Ward/Rummel/Brown 03/24/92 Rationale: Operational health care system has several components and must be fully integrated. This may include such things as bioengineering and development, crew activity planning, test beds, ground operations, etc.	?*R	527A
The capability shall be provided to handle medical contingencies. Ward/Rummel/Brown 03/13/92 Rationale: The successful handling of medical contingencies requires detailed advanced planning. Medical intervention and care capability is required for diagnosis and treatment of minor injuries and illnesses, dental complaints, infectious diseases, trauma and major injury, diseases and complications of hypobaric exposure (decompression sickness, gas embolism, ebullism) and radiation exposure, and medical problems related to stresses of space flight and/or degraded health. Capabilities are required to be consistent with general health of the crew (before, during, and after the mission), number of crewmembers, mission tasks and associated hazards, and U.S. standards of medical care.	?*R	528
The capability shall be provided to protect the crew in the event of TBD contingencies while on the lunar surface during the nominal mission. Ward/Rummel/Brown 03/24/92 Rationale: This capability provides a greater possibility of safe crew return in the event of major system failures or accidents. Specific contingencies are under study, and may include fire, depressurization, etc.	?*R	529A
5.3.1.2 Biomedical Investigations	?*R	H530
The capability shall be provided to evaluate and mitigate human harmful adaptation to space flight. Ward/Rummel/Brown 03/12/92 Rationale: It is essential to understand and mitigate when necessary human adaptation to the space flight environment.	?*R	531

MAFE

5.3.2 Living and Working Conditions ?*R H532

Provisions shall be available during all mission phases to sustain life, to protect the crew, to enable the crew to live and work effectively and efficiently in performing mission activities in the natural and induced environments. ?*R 533

Ward/Rummel/Brown 03/12/92
Rationale: The health and productivity of space flight crews is dependent upon the habitability of their living and working environment. Program success, mission success, and crew life will depend significantly on the degree to which crew productivity is maximized and systems are designed for human interaction.

5.3.2.1 Human Factors ?*R H534

Human interfaces with other humans, systems, provisions, and operations will be structured and designed to account for human capabilities and limitations. ?*R 535A

Ward/Rummel/Brown 03/24/92
Rationale: Sound human factors engineering principles are mandatory to ensure human integration with exploration missions, to optimize commonality and standardization of crew interfaces across all provisions and operations throughout all mission phases, to optimize resource utilization and crew efficiency, to promote effective human-human and human-system interactions, and to ensure crew productivity and safety. The generic NASA standard for man-system integration, NASA-STD-3000, may be used as a general guideline until a separate, dedicated volume which will be controlled by SEI is developed and approved.

	MAFE	
5.3.2.2 Habitability	?*R	H536
The capability shall be provided to support human life and performance, and to ensure habitability of this support for all mission phases and activities.	?*R	537

Ward/Rummel/Brown 03/13/92
Rationale: The space flight environment does not naturally provide essential habitation systems (e.g., food/water, personal hygiene, clothing, recreation, sleeping accommodations, work environments, housekeeping, etc.). These systems are critical to human support. Crew perception of the quality of life (i.e., habitability) is critical to maintaining the health, well-being, safety, and productivity of individuals and groups performing mission activities with the given systems and provisions in the natural and induced hostile, remote, and confined environments.

5.4 System Design Safety and Reliability	?*R	H297B

Comment: Replaces H297A: 5.4 Safety, Reliability and Quality Assurance (SR&QA) Requirements

This entire section is in work and under review. It has not been approved by the Exploration Control Board.	?*R	D618

J. Carpenter 04/01/92

MAFE

?*R 504

Factors of safety shall be provided per T920001EXPO:5.4-1, "Factors of Safety (FS)S*".
E. Graham/G. Grush 03/04/92
Rationale: To achieve a margin in the design which will result in an acceptable level of reliability and safety of the design. The values represented in the table apply to the surface systems. Values for flight vehicles are TBD. The source for the current estimated values are from SSP 31000 rev. G. It is anticipated that the flight vehicle values will correlate to the Shuttle program. Further analysis is required before these values become firm.

T920001EXPO:5.4-1 Factors of Safety (FS)S*

Factors of Safety (FS)S*

	Yield	Ult.
A. General FSs for Metallic Flight Structures		
Manned and Unmanned	1.25	2.00
Verified by analysis only	1.10	1.50
Verified by analysis and static test		
B. General FSs for Nonmetallic Flight Structures		
Verified by analysis and static test	----	1.50
Nondiscontinuity areas	----	2.00
Discontinuity areas and joints		
C. General FSs for Pressure		
(1)a Windows, internal pressure only		
Proof pressure = 2.00 X MOP		
Ultimate pressure = 3.00 X MOP		
(2) Engine structures and components		
b Proof pressure = 1.20 X MOP		
Ultimate pressure = 1.50 X MOP		
(3) Hydraulic and pneumatic systems		
(a) Lines and fittings, < 1.5 inches (38 mm) diameter		
b Proof pressure = 2.00 X MOP		
Ultimate pressure = 4.00 X MOP		
(b) Lines and fittings, 1.5 in. (38 mm) dia. or greater		
b Proof pressure = 1.20 X MOP		
Ultimate pressure = 1.50 X MOP		

. . . table continued on next page . . .

T920001EXPO:5.4-1 Factors of Safety (FS)S*, continued

(c) Reservoirs/pressure vessels
b Proof pressure = 1.50 X MOP
Ultimate pressure = 2.00 X MOP

(d) Actuating cylinders, valves, filters, switches
b Proof pressure = 1.50 X MOP
Ultimate pressure = 2.00 X MOP

(4) Doors, hatches, personnel compartments, internal pressure only
Proof pressure = 1.50 X MOP
Yield pressure = 1.65 X MOP
Ultimate pressure = 2.00 X MOP

a. All windows and glass structures that are subjected to tensile stresses shall be designed utilizing fracture mechanics. A proof test shall be utilized to screen manufacturing flaws and shall be conducted in an environment that will not facilitate flaw growth.

b. Proof factor determined from fracture mechanics service life analysis must be used if greater than minimum factor. "Leak before burst" criteria may supersede these factors. No detrimental deformation at noted FS.

c. Maximum Operating Pressure (MOP)

* Values are estimates for surface systems.
Values are TBD for flight vehicles.

All systems shall be "fail safe". ?*R 485
E. Graham/G. Grush 03/04/92
Rationale: Fail safe is defined in the glossary.

Redundant systems shall provide a capability to verify their operational status to the fail safe level during ground turnaround and during the mission. ?*R 486
E. Graham/G. Grush 03/03/92
Rationale: Ensure ability to check status of system and detect abort conditions.

	MAFE	
The systems for the first lunar outpost mission shall not preclude a piloted mission from restoring full operational capability after a single functional failure. E. Graham/G. Grush 03/04/92 Rationale: Ensures a minimal level of fault tolerance for the initial mission and prevents major loss of outpost infrastructure for single failures.	?*R	487
Neither EVA nor IVA maintenance shall be considered legs of redundancy when dealing with system failures. J. Borrer 03/24/92 Rationale: The first failure of a system should not place the crew's lives in jeopardy such that an immediate IVA or EVA response is required for their survival. IVA and EVA maintenance will be used to repair the failed system but sufficient redundant systems should be operating to insure the crew's survival during the time between the initial failure and its repair or to allow the crew the appropriate time to seek appropriate shelter or abort the mission as required.	?*R	556
5.5 Environments	?PG	H402
This section contains the natural and induced environments to which the FLO system will be exposed and for which it must be designed. J. Carpenter 02/28/92	?PG	D403

	MAFE	
5.5.1 Natural Environments	?PG	H404
The system shall provide the capability to operate in the lunar environment as specified in Lunar Engineering Models: General and Site-Specific Data.	?PG	405

J. Carpenter 02/28/92
Rationale: This document contains the current best data to be used in designing systems for deployment in and around the surface of the Moon. It was assembled by the JSC Solar System Exploration Division (SN) for LMEPO. It presents descriptive information about a "design to" location on the Moon that represents a class of potential outpost locations. The description is divided into two parts. The first part tabulates data that are global and independent of any particular site on the Moon - the meteoroid flux, for example. The second part presents data that will vary somewhat from place to place on the Moon and are therefore pertinent to the "design to" site. This document is currently in Review Draft form, dated January, 1992. The contact for this document is David Kaplan, Mail Code XE, telephone 713/283-5416, NMail ID: DKAPLAN.

The system shall provide the capability to operate in the Earth orbital environment as specified in Orbital Debris Environment for Spacecraft Designed to Operate in Low Earth Orbit.	?PG	406

J. Carpenter 03/04/92
Rationale: This model describes the orbital debris environment for Low Earth Orbit. The model is compatible with current measurements, accounts for time dependent flux variations, and assumes a growth rate for orbital debris consistent with current operations (4-5 explosions per year). NASA TM 100471, Don Kessler et al, April 1989. The contact for this document is Eileen Stansbery, Mail Code XE, telephone 713/283-5393, NMail ID: ESTANSBERY.

MAFE

The system shall provide the capability to operate in the near Earth to lunar surface environment as specified in Meteoroid Environment Model - 1969 [Near Earth to Lunar Surface]. ?PG 407

J. Carpenter 02/28/92

Rationale: This model consists of a subset of the information contained in NASA SP 8038 "Meteoroid Environment Model - 1970 [Interplanetary and Planetary]" and contains the knowledge of the environment known to date. NASA SP 8013, B.G. Cour-Palais 1969. The contact for this document is Eileen Stansbery, Mail Code XE, telephone 713/283-5393, NMail ID: ESTANSBERY.

5.5.2 Induced Environments ?PG H408

TBD ?PG D409

J. Carpenter 02/28/92

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GLOSSARY

DRM	Design Reference Mission
EMU	Extravehicular Mobility Unit
ETO	Earth-to-Orbit
EVA	Extra-Vehicular Activity
EXPO	Exploration Programs Office
Fail Safe	A single functional failure can be sustained and still retain the capability to terminate the mission and return the crew safely to Earth. Failures will not cause a hazardous condition to exist.
FLO	First Lunar Outpost
FLORG	First Lunar Outpost Requirements and Guidelines
ISRU	In situ Resource Utilization
IVA	Intra-Vehicular Activity
JSC	Johnson Space Center
LEO	Low Earth Orbit
LLO	Low Lunar Orbit
LOI	Lunar Orbit Injection
NASA	National Aeronautics and Space Administration
PLSS	Portable Life Support System
SEI	Space Exploration Initiative
STS	Space Transportation System
TBD	To Be Determined
TEI	Trans-Earth Injection
TISN	Telecommunications, Navigation, and Information Systems
TLI	Trans-Lunar Injection

Requirements in 920001EXPO changed since 03/25/92

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First Lunar Outpost

Detailed Assumptions

Working Drafts

1 April 1992

This package contains working drafts of the First Lunar Outpost Detailed Assumptions being carried by sub-teams of FLO. They have been entered into the TRACER format, however, the content of each section is dictated by the Team Lead who has created it. The intent of this package is to capture the driving assumptions for the elements which have been designed. Assumptions as to function, performance and design concept have been made, at various levels of detail. As a working document, discrepancies are expected to occur and will be resolved as additional work is performed. Sections 3-1 to 3-6 have been reviewed by the entire team, with the exception of changes since 3/20/92. Section 3-7 is new and has not been reviewed.

Questions should be directed to Joyce Carpenter, Deputy, Mission Analysis and System Engineering, Exploration Programs Office, Mail Code XE, NASA Johnson Space Center. 713/283-5404.

Change Bar Dates: 03/21/92

Abstract
Detailed Assumptions Document (DAD)

This section of the report contains working drafts of the First Lunar Outpost Detailed Assumptions being carried by the sub-teams of FLO. They have been entered into the TRACER format, however, the content of each section is dictated by the Team Lead who has created it. The intent of this section is to capture the driving assumptions for the elements which have been designed. Assumptions as to function, performance, and design concept have been made, at various levels of detail. As a working document, discrepancies are expected to occur and will be resolved as additional work is performed.

Questions concerning this section should be directed to XE/Deb Neubek (713/283-5755).

	MAFE	
Space Transit Sub-Segment	?*G	H414
The space transit sub-segment shall provide the capability to deliver up to 27.5 mt (E) of cargo to the lunar surface. R. Kahl 03/06/92 Rationale: Derived from FLORG Design Assumption 313.	?*G	70
The space transit sub-segment elements shall include a 20% dry mass margin as part of their mass estimates, excluding the TLI element which includes only a 10 dry mass margin. R. Kahl 03/11/92 Rationale: Use of dry mass contingency is to account for mass unknowns at this phase of the design. The 20% factor is based on historical trends and represents a minimum percentage based upon a space transit sub-segment design that is complex and has more uncertainty in that its elements result from new designs. How are margins being handled in general? See assump. 51 for TLI.	?*G	73
All space transit sub-segment elements shall include a 1% total propellant flight performance reserve as part of their mass estimates. R. Kahl 03/11/92 Rationale: FPR allocation is based upon actual engine performance data and practices. How are FPRs being handled in general? See assumption 50 for TLI.	?*G	74
The space transit sub-segment will not be required to provide an on-board training capability for in-flight crew training. R. Kahl 03/06/92 Rationale: Based on the desire to reduce the additional mass burden that an in-flight training system would impose on the crew module element.	?*G	78

	MAFE	
The space transit sub-segment shall provide the capability for the lander element to remain in lunar orbit only for that time required for the navigation state vector update to achieve the required landing accuracy. R. Kahl 03/12/92 Rationale: Derived from FLORG requirement 376, minimizing crew time on orbit and FLORG requirement 250, the desire to minimize orbital operations which do not provide direct science and exploration returns.	?*G	415
Launch Operations	?*G	H8
KSC shall provide the capability for emergency escape of the crew from the launch pad from the time the crew approaches the launch pad until lift-off. R. Kahl 03/11/92 Rationale: Derived from FLORG requirement 383, the capability to abort during all mission phases. This requirement addresses the pre-launch mission phase abort/crew escape contingency.	?*G	411
Lunar Lander	?*G	H11
The lander element shall provide the capability to tolerate a landing hazard of less than or equal to 0.5 meters in height. R. Kahl 03/11/92 Rationale: Derived from FLORG requirement 149A, the desire to access any lunar latitude/longitude and allowing for lander engine nozzle clearance required for engine operation.	?*G	408
The lander element shall provide the capability to operate with a total lander effective slope of less than or equal to 12 degrees. R. Kahl 03/19/92 Rationale: Derived from FLORG requirement 149A, the desire to access any lunar latitude/longitude and allowing for maximum allowable tilt for crew comfort and safe operation of lander, crew module, return stage and cargo equipment.	?*G	409

	MAFE	
The lander element shall provide the capability to automatically land on the lunar surface with a vertical velocity of 7 feet per second at a horizontal velocity of 4 feet per second or equivalent. R. Kahl 03/19/92 Rationale: Automated landing will be the only way to land the cargo-lander. The same technology can be used to land the crew-lander, thereby reducing the complexity and training for the human mode. A fundamental premise is to utilize crew to explore not as test pilots. Landing velocities are limited to accommodate a safe landing for the crew and to avoid damage to payloads and spacecraft systems.	?*G	62
The lander element shall provide the capability to attach wheel mechanisms to its structure. R. Kahl 03/06/92 Rationale: Based upon the desire to have the capability to move a surface habitat without offloading it from the lander element. Derived from FLORG requirements 339, 154 and goal 242A.	?*G	29
The lander element shall provide an inherent "engine-out" capability due to multiple engines. R. Kahl 03/06/92 Rationale: The lander element will not be specifically designed for "engine-out" but will be able to recover if an engine fails due to the incorporation of multiple engines in its design.	?*G	75
Surface Operations	?*G	H15
The lander element shall provide the capability to powerdown to necessary housekeeping levels within one (E) hour after landing on the lunar surface. R. Kahl 03/06/92 Rationale: Derived from FLORG guidelines 250 and 126A, minimizing operations which do not provide direct science and exploration returns and minimizing operations required to prepare for outpost occupancy.	?*G	18

	MAFE	
The lander element shall provide the capability to be activated for launch from the lunar surface within two (2) (E) hours of the decision to leave the lunar surface. R. Kahl 03/06/92 Rationale: Derived from FLORG requirement 148A, to provide the capability for abort at all times following lunar landing within 24 hours of abort declaration. Also derived from FLORG guidelines 250 and 126A, minimizing operations which do not provide direct science and exploration returns and minimizing operations required to transition outpost from manned to unmanned.	?*G	19
Precursor Mission Interface	?*G	H20
Surface Systems Interface	?*G	H22
The surface segment shall provide any packaging or special handling provisions required by surface system cargo items that will be carried by the space transit sub-segment. R. Kahl 03/06/92 Rationale: This is a project assumption agreed to by each of the element managers to always account for the interface structure masses between elements and to be accounted for by flight element considered to be the payload element at that interface.	?*G	27
The surface segment shall provide the capability to handle (off-load, deploy, relocate, set-up, etc.) all surface segment payloads after the lander element has landed on the lunar surface. R. Kahl 03/06/92 Rationale: Based upon the DRM and science objectives; the desire to deploy payloads on the lunar surface.	?*G	28
The lander element shall provide the capability for the habitation element to be attached to the lander element. R. Kahl 03/11/92 Rationale: Derived from the DRM and FLORG requirement 313; the desire to carry cargo to the lunar surface.	?*G	416

	MAFE	
The lander element shall provide the capability for surface segment cargo to be attached to the lander element. R. Kahl 03/06/92 Rationale: Derived from the DRM and FLORG requirement 313; based upon the desire to carry cargo to the lunar surface.	?*G	30
The surface segment shall provide the capability for the environmental protection or special handling, if needed, on any cargo that is to be returned to Earth by the space transit sub-segment. R. Kahl 03/06/92 Rationale: Derived from the above rqmt. number ? in section 5.2.1.2.4? and the DRM, the desire to return usable, undamaged science payload from the lunar surface to Earth. This is a project assumption agreed to by each of the element managers to always account for the interface structure masses between elements and to be accounted for by the flight element considered to be the payload element at that interface.	?*G	66
The habitat/airlock shall provide the capability to recharge the transfer EVA EMU/PLSS element's PLSS system. R. Kahl 03/19/92 Rationale: After being stored in the habitat during the 45 day surface mission, the transfer EVA suit's PLSS will require recharging prior to the crew returning to the crew module element for return to Earth.	?*G	480
Science and Payloads Interface	?*G	H31
Assume that no science requirements will be placed on the space transit sub-segment. R. Kahl 03/11/92 Rationale: Based on the DRM and to minimize the crew module element's mass.	?*G	32
Assume that no science requirements will be placed on the space transit sub-segment during transfer to and from the Moon, or while in Earth or lunar orbit. R. Kahl 03/12/92 Rationale: Based on the DRM and the desire to minimize space transit sub-segment mass.	?*G	417

	MAFE	
TLI Stage Interface	?*G	H41
The mass of the TLI adapter structure between the TLI stage and the lander shall be carried within the mass considerations for the lander. R. Kahl 03/06/92 Rationale: This is a project assumption agreed to by each of the element managers to always account for the interface structure masses between elements and to be accounted for by flight element considered to be the payload element at that interface.	?*G	42
TLI Stage Design	?*G	H45
The TLI stage element shall provide the capability of a suborbital burn to achieve a launch insertion orbit of 100 nmi and maintain the orbit up to a maximum stay time of 8 hours prior to TLI burn. R. Kahl 03/11/92 Rationale: Derived from studies that 100 nmi is the optimum orbit altitude to maintain the TLI stage and payload for systems checkout prior to proceeding to TLI burn for lunar transfer orbit.	?*G	46
The reference azimuth is 72 to 108 degrees from KSC. R. Kahl 03/06/92 Rationale: This azimuth range provides a 4-hour launch window during any given day for lunar missions. Also recognizes tracking and piloted mission abort constraints.	?*G	47
The TLI stage shall be a stand-alone, load-carrying structure. R. Kahl 03/06/92 Rationale: The TLI stage must be able to withstand the launch loads during liftoff incurred while carrying the piloted and cargo payloads.	?*G	48

	MAFE	
HLLV Interface	?*G	H33
The space transit sub-segment shall be limited to the use of the volume represented by a 33 feet (E) by 60 feet (E) payload shroud provided by the Earth-to-orbit sub-segment for packaging and stacking the space transit sub-segment elements for launch. R. Kahl 03/06/92 Rationale: Based on the Earth-to-orbit sub-segment maximum allowable shroud dimensions. Derived from 35 below.	?*G	34
The Earth-to-orbit sub-segment shall provide a payload shroud that is 33 feet (E) by 60 feet (E) in dimension for the protection of Earth-to-orbit sub-segment payloads during the mission launch phase. R. Kahl 03/06/92 Rationale: Check with Gene Austin for this one if you do not already have it.	?*G	35
Flight segment elements shall be designed to withstand peak loadings during the launch phase of 4 Gs of acceleration and up to 900 psf of dynamic pressure. R. Kahl 03/06/92 Rationale: Check with Gene Austin for this one if you do not already have it.	?*G	37
The mass of the interface support structure between the second stage of the launch vehicle (first stage for NLS Derived) and the TLI stage shall be carried within the mass considerations for the TLI stage. R. Kahl 03/06/92 Rationale: This is a project assumption agreed to by each of the element managers to always account for the interface structure masses between elements and to be accounted for by flight element considered to be the payload element at that interface.	?*G	38

	MAFE	
The TLI stage shall be a LOX/LH2 propellant stage utilizing engine options of J-2S, RL-10s, SSME, or STME engines with no engine out and no throttling. R. Kahl 03/19/92 Rationale: To maintain system and equipment commonality between other flight elements, the engine options and propellant are based upon preliminary trades and actual performance in space environment. The STME engine is baselined by the NLS Program and is also being considered. Additional trades must be done before this allocation becomes firm.	?*G	49
The TLI stage shall use a Flight Performance Reserve (FPR) of 1% of the propellant and a mission allocation for delta V of 3200 meters/sec. R. Kahl 03/06/92 Rationale: FPR allocation is based upon actual performance data and practices. The delta V is based upon mission analysis of the maximum case needed to meet the mission requirements with performance margin. Also see Assumption 54.	?*G	50
The TLI stage shall use a dry mass contingency of 10%. R. Kahl 03/06/92 Rationale: Use of dry mass contingency is to account for mass unknowns at this phase of the design. The 10% factor represents a minimum percentage based upon a TLI stage design that is of a less complex design than other flight elements and of less uncertainty with known mass weights such as engines and other components from previous design and performance data. Also see assumption 73.	?*G	51
The capability shall be provided for the safe disposal of the TLI stage element. R. Kahl 03/19/92 Rationale: The TLI stage element will be discarded after use, just after TLI burn is completed and separation from the descent/ascent stage, requiring safe disposal. The specific method of disposal will be determined after further study. Considerations include the desire to avoid creating hazardous space debris in orbit around the Earth or Moon, and to avoid impacting the Earth in a populated area.	?*G	481

	MAFE	
Crew Module Design	?*G	H53
The crew module element shall be designed with an internal pressure of five (5) psi. R. Kahl 03/06/92 Rationale: A low internal pressure environment minimizes EVA preparation time and for a crew module with no airlock, the consumables needed for repressurization of the module following de-pressurization are also minimized.	?*G	54
The crew module element shall be capable of performing all necessary EVA functions without requiring an airlock. R. Kahl 03/06/92 Rationale: Eliminating the need for an airlock on the crew module element imparts a large weight savings on the space transit sub-segment. The entire crew compartment will be depressurized prior to crew egress. This requirement is particularly important for a sortie mission which will require numerous EVAs from the crew module element.	?*G	55
The crew module element shall provide the capability to repressurize after each of two (2) depressurizations of the crew compartment. R. Kahl 03/06/92 Rationale: Derived from FLORG requirements 319, 370, 320 and 170. Based on the desire to allow for at least 2 EVAs from the crew module element post lunar landing for the crew to prepare the surface habitat.	?*G	56
The crew module element shall provide the capability to repressurize after each of TBD depressurizations of the crew compartment during a sortie mission. R. Kahl 03/06/92 Rationale: Derived from the DRM, Option 2, the sortie mission (FLORG requirement 235A).	?*R	57

	MAFE	
The crew module element shall remain depressurized when it is unoccupied by the crew. R. Kahl 03/06/92 Rationale: Based on the desire to reduce the amount of extra mass required to maintain crew module element pressurization while unoccupied and account for leakage for a 45-day period on the lunar surface.	?*G	58
The space transit sub-segment shall be designed so that no EVA will be required at any time during transit to and from the Moon. R. Kahl 03/06/92 Rationale: Based on the desire to reduce the amount of extra mass required to provide for additional crew module element repressurizations following additional EVAs.	?*G	59
The crew module element shall provide the equivalent of 10 g/cm**2 of water for protection to the crew from exposure to Solar Particle Event (SPE) radiation. R. Kahl 03/11/92 Rationale: The number of lunar missions over time and the duration of the sortie missions increases the chances of crew exposure to an SPE. Derived from FLORG Rqmt 255B.	?*G	64
Assume revision of shielding levels if nuclear power sources are present. Ward/Rümmel/Brown 04/01/92 Rationale: Nuclear sources in proximity to the crew may affect radiation levels present in habitable volumes.	?*G	603
The crew module element shall provide for "first aid kit-level" medical care of the crew while in transit and while living out of the crew module on the lunar surface. R. Kahl 03/06/92 Rationale: Derived from FLORG goal 243, to return crews safely to Earth and FLORG Human Support requirement 322, the health, safety and productivity of the crew shall be maintained.	?*G	68

	MAFE	
The crew module element shall provide the capability to limit the acceleration loads imposed on the crew during the Earth re-entry phase of the mission to 8 gs (E) for up to 2.5 minutes (E) if no movement from the crew is required. R. Kahl 03/11/92 Rationale: Derived from FLORG goal 243, to return crews safely to Earth, and FLORG human support requirement 322, the health, safety and productivity of the crew shall be maintained. A re-entry g-load of 8 g for 2.5 minutes came from the Human Spacecraft Transportation Project Task Report 1.2.2, Human Acceleration Requirements in Long Duration Missions, December 1991.	?*G	69
The crew module element shall provide the capability for a minimum rotation rate of one revolution per hour while in transit between the Earth and the Moon. R. Kahl 03/19/92 Rationale: This capability is required to accommodate the passive thermal control system's constraints during Earth-Moon-Earth coast.	?*G	482
The crew module element shall provide the capability to support the transfer EVA EMU/PLSS element's oxygen, power, electrical and communications systems during operation in the crew module. R. Kahl 03/19/92 Rationale: The transfer suit will be used in the crew module during the Earth launch and entry phases.	?*G	483
The crew module element shall provide the capability to recharge the transfer EVA EMU/PLSS element's PLSS system. R. Kahl 03/19/92 Rationale: EVAs from the crew module when the habitat's resources are not available necessitate that the crew module be able to recharge a transfer suit's PLSS.	?*G	484

	MAFE	
The capability to store food shall be provided during each mission phase to maintain integrity of foodstuffs. Ward/Rummel/Brown 03/25/92 Rationale: Food will require environmental control appropriate to the type of food provided. Alternative types and combinations of dehydrated, thermostabilized, fresh, and frozen food will be traded to determine the optimum complement for each mission phase, including space transit, surface habitation, EVA, and roving.	?*G	494
Transfer EVA EMU/PLSS Element	?*G	H485
The transfer EVA EMU/PLSS element shall provide the capability to sustain the crew during EVA to and from the lunar habitat, during Earth launch and entry phases, and during a surface or in-transit contingency. R. Kahl 03/19/92 Rationale: The crew module element cannot accommodate the 4 large and heavy surface EVA suits without increasing the crew module volume. The transfer EVA suits are designed for dual purpose (pressure suit during Earth launch and entry and EVA suit for crew transfer between the lander and the habitat), are lightweight, can be comfortably worn in the crew module, and provide an easily storable backup suit to the surface suits.	?*G	486
The transfer EVA EMU/PLSS element's PLSS shall provide the capability to be recharged at the surface habitat and by the crew module element. R. Kahl 03/19/92 Rationale: After being stored in the habitat during the 45 day surface mission, the transfer EVA suit's PLSS will require recharging prior to the crew returning to the crew module element for return to Earth. EVAs from the crew module when the habitat's resources are not available necessitate that the crew module be able to recharge a transfer suit's PLSS.	?*G	487

	MAFE	
Return Stage	?*G	H418
The return stage element shall provide an inherent "engine-out" capability due to multiple engines. R. Kahl 03/06/92 Rationale: The return stage element will not be specifically designed for "engine-out" but will be able to recover if an engine fails due to the incorporation of multiple engines in its design.	?*G	76
The capability shall be provided for the safe disposal of the return stage element. R. Kahl 03/11/92 Rationale: The return stage element will be discarded after use, just prior to Earth-entry interface, requiring safe disposal. The specific method of disposal will be determined after further study. Considerations include the desire to avoid creating hazardous space debris in orbit around the Earth or Moon, and to avoid impacting the Earth in a populated area.	?*G	77
Recovery Operations	?*G	H412
KSC shall provide the capability to recover the crew module in the event of an abort or at the completion of a nominal mission. R. Kahl 03/11/92 Rationale: Derived from FLORG requirement 226B, the recovery function will be provided. This requirement identifies WHO will provide this capability. Launch aborts will most likely result in the crew module landing offshore, near KSC. They are the most reasonable choice of NASA Centers to accommodate the need to recover the nearby crew module in this scenario. Since they will have then already developed the procedures to recover the crew module after a launch abort, these procedures can also be applied to a nominal return scenario.	?*G	413

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	MAFE	
5.2.1.1 Earth to Orbit Sub-Segment	?*G	H82
The HLLV shall provide the capability for designed growth to 250 metric tons to a 220 nautical mile orbit. G. Austin 03/06/92 Rationale: Mars missions will require initial mass to LEO (IMLEO) on the order of 1000 metric tons for piloted flights, necessitating multiple launches. To keep the number of launches as low as possible, the launch vehicle must be as large as practical to minimize complex space operations. Current data point to the 250 metric ton range as the practical limit.	?*G	83
The HLLV shall be designed for no engine out on the core, boosters, or upper stage. G. Austin 03/06/92 Rationale: The "no engine out" design permits performance comparisons to be made based upon maximum total system capabilities.	?*G	84
The HLLV shall be sized to provide launch capability any day during the lunar cycle. G. Austin 03/06/92 Rationale: This capability will provide unrestricted ability to land at a given site on the Moon with proper lighting conditions.	?*G	85
The HLLV design shall incorporate a dry weight contingency of 10%. G. Austin 03/06/92 Rationale: This is consistent with the early stages of design maturity and with NLS program ground rules.	?*G	86
The HLLV design shall incorporate an ascent flight performance reserve of 1% of delta V. G. Austin 03/06/92 Rationale: This is consistent with the early stages of design maturity and anticipated dispersions during flight.	?*G	87

	MAFE	
The following orbits shall be used: 100 nm circular for lunar direct; 220 nm circular for Mars missions. G. Austin 03/10/92 Rationale: Lunar flights pass through 100 nm with up to two orbits for targeting. Mars vehicles require on-orbit assembly; 220 nm is a rendezvous-compatible orbit.	?*G	405
All launch phase primary avionics (e.g., GN&C equipment, computers, telemetry equipment, etc.) shall be located on the TLI stage and shall be independent of the LTS avionics. Each stage shall be capable of sending stage sub-system status to the launch phase primary avionics. G. Austin 03/06/92 Rationale: This allows status of systems information to be available for launch abort capability. Independent avionics provides redundancy and availability for other uses. This is consistent with Apollo Saturn V methodology.	?*G	88
The launch vehicle/TLI stage shall be capable of placing the TLI stage, lander, and cargo directly into an orbit with an altitude of 100 nautical miles (circular). G. Austin 03/10/92 Rationale: This is the minimum altitude to provide proper stage disposal, and is consistent with Apollo Saturn V methodology.	?*G	89
The maximum acceleration during boost phase is limited to 4 g (goal). G. Austin 03/06/92 Rationale: This acceleration limitation provides the capability to utilize existing design concepts, such as space station habitation module, airlocks, etc., and to minimize structural loads on other vehicle elements.	?*G	90
The primary propulsion options include F-1A, J-2S, SSME, STME, and RL10-4A. G. Austin 03/10/92 Rationale: Primary propulsion system options utilize existing and planned designs.	?*G	91

	MAFE	
The vehicle shall be capable of launch from 72 degrees to 108 degrees azimuth. G. Austin 03/06/92 Rationale: This azimuth range provides a 4-hour launch window during any given day for lunar missions. Also recognizes tracking and piloted mission abort constraints.	?*G	92
The maximum q=900 psf during ascent. G. Austin 03/06/92 Rationale: Structural loading consideration on shroud and HLLV.	?*G	93
5.2.1.1.1 Core Stage Element	?*G	H94
The vehicle shall provide for on-pad holddown during engine start: this will be on the S-1C stage for Saturn V derivatives. G. Austin 03/10/92 Rationale: This is consistent with this candidate HLLV option and provides provisions for verification of proper engine function before liftoff, and thrust buildup.	?*G	95
5.2.1.1.2 Booster Stage Element	?*G	H96
The vehicle shall provide for on-pad holddown during engine start: this will be on the boosters for NLS derivatives. G. Austin 03/10/92 Rationale: This is consistent with this candidate HLLV option and provides provisions for verification of proper engine function before liftoff, and thrust buildup.	?*G	406
5.2.1.1.3 Second Stage Element	?*G	H97
5.2.1.1.4 Shroud Element	?*G	H98
The usable shroud size for lunar flights shall be greater than or equal to 33 feet by 60 feet (goal). G. Austin 03/10/92 Rationale: This design goal corresponds to the minimum clear space required to accommodate the lunar lander with the habitat module, and is limited by core stage diameter.	?*G	99

	MAFE	
The shroud and nosecap shall be jettisoned at 400,000 ft altitude. G. Austin 03/06/92 Rationale: This is standard launch vehicle operating procedure, and the altitude is outside the area of the atmosphere where aerodynamic drag and heating would affect the vehicle. The shroud serves no purpose beyond this point.	?*G	100

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	MAFE	
Habitat	?*G	H101
The crew will visit the habitat as soon as possible after landing to verify its integrity and initiate any necessary corrective procedures. D. Perkinson 03/06/92 Rationale: Even though electronic remote checkout is performed prior to crew launch, it is still advisable to visit the hab and verify operations of major subsystems.	?*G	102
The habitat will be landed with the solar arrays aligned within +15 degrees of north/south. D. Perkinson 03/12/92 Rationale: Allows tracking of sun by arrays with very little cosine loss.	?*G	103
The requirement for a "safe haven" is TBD. D. Perkinson 03/06/92 Rationale: A "safe haven" is desirable into which to retreat in case of a fire; however, the impacts may be significant. The possibility of using the airlock has not been fully evaluated. If the outside airlock door must be left open when two crew are on EVA, then it cannot serve as a safe haven. However, the proximity of the lander to which the EVA crew may retreat may allow the outside airlock door to be closed during EVAs.	?*G	104
All critical subsystems will be fail safe as far as crew safety and habitat loss is involved, and may be fail op/fail safe in specific areas (such as C&DMS) in order to allow the habitat to be placed in a dormant mode after the first failure. D. Perkinson 03/25/92 Rationale: Beyond only crew safety, it is desirable that the hab itself require minimum repair in case of a critical failure. The term "feasible" was used since the hab pressure shell itself probably cannot be fail-op, and other components need to be considered relative to their reliability (lander legs etc.).	?*G	105

	MAFE	
Pressure shell accessibility from inside the habitat is TBD. D. Perkinson 03/06/92 Rationale: Near infinite life suggests access to the pressure shell is desirable; however, weight restrictions may make it desirable to dispense with removable racks. Entire racks probably cannot be exchanged due to their weight.	?*G	106
Nutritionally robust and palatable foods shall be provided to maintain human health and performance of the crew throughout the mission. Ward/Rummel/Brown 03/25/92 Rationale: Humans require sufficient nutrition to conduct each phase of mission activities. Nutritional insufficiency affects crew performance and health, resulting in decrease in vigilance, mood, and morale; decrease in aerobic capacity, isokinetic muscle strength and endurance; visual, motor, and cognitive disturbances; increased stress levels; decreases in self-initiative; constipation; physiological imbalances; and weight loss.	?*G	492
The capability to store food shall be provided during each mission phase to maintain integrity of foodstuffs. Ward/Rummel/Brown 03/25/92 Rationale: Food will require environmental control appropriate to the type of food provided. Alternative types and combinations of dehydrated, thermostabilized, fresh, and frozen food will be traded to determine the optimum complement for each mission phase, including space transit, surface habitation, EVA, and roving.	?*G	493
A refrigerator and a freezer may be required for food storage beyond 30 days. Delivery of frozen food by the piloted lander and transport of this food to the habitat is also necessary. D. Perkinson 03/12/92 Rationale: TBD.	?*G	107

	MAFE	
A method of limiting dust entry into the airlock and into the habitat shall be provided. A method of removing dust from the airlock and from the habitat shall also be provided. D. Perkinson 03/06/92 Rationale: Lunar dust is both very fine and very abrasive, and may cause problems both medically for the astronauts and impair the operation of the habitat mechanisms and seals.	?*G	108
A method of replacing seals (on the hatches, etc.) shall be provided. D. Perkinson 03/06/92 Rationale: Seals become less effective both due to normal usage and due to lunar dust.	?*G	109
A method of logistics resupply of the habitat shall be provided. This shall include food, possibly refrigerated/frozen food, water as necessary, oxygen, nitrogen, filters, spares, tools, scientific equipment, and seals. D. Perkinson 03/12/92 Rationale: Necessary to maintain habitability.	?*G	110
A method of disposing of waste and trash shall be provided. D. Perkinson 03/06/92 Rationale: Necessary to maintain habitability.	?*G	111
Suitable stowage for consumables sufficient for 45 days for 4 persons shall be provided. D. Perkinson 03/06/92 Rationale: From top-level requirements.	?*G	112
The habitat shall be left between missions with sufficient consumables to support a crew of 4 for TBD days, and such consumables shall not require refrigeration and shall be capable of withstanding a hard vacuum. D. Perkinson 03/06/92 Rationale: Provides recovery window in case of problems with moving new consumables, etc. Major problem may be oxygen.	?*G	113

	MAFE	
It shall be possible for the habitat to be placed in a stand-by (as between missions) mode by one crew person in TBD hours. D. Perkinson 03/06/92 Rationale: It is desirable for one person to be able to perform this function in the abort case in which two crew persons are transporting an injured person to the lander.	?*G	114
It shall be possible for one person to operate the habitat. D. Perkinson 03/06/92 Rationale: Necessary if two crew persons are on EVA and one IVA person become disabled.	?*G	115
The habitat shall provide communications with the lander, rover, EVA crew, and nearby scientific experiments. D. Perkinson 03/25/92 Rationale: Eliminates Moon-Earth-Moon time delay.	?*G	116
Methods of maintaining the habitat shall be identified, and necessary scarring of habitat shall be done to make maintenance as simple as possible. D. Perkinson 03/06/92 Rationale: Necessary for long life of habitat. Maintenance should include refilling or recharging fuel cell reactants tankage.	?*G	117
The possibility of utilizing the fuel cell regeneration system to provide metabolic oxygen to the habitat shall be investigated. This also requires the capability of regular addition of reactant grade water to the fuel cell system. D. Perkinson 03/06/92 Rationale: Eliminates high pressure gas storage of cryogen oxygen boiloff. Allows metabolic oxygen to be kept at habitat for indefinite periods. Allows use of piloted lander fuel cell water to be used for metabolic oxygen.	?*G	118

	MAFE	
A trade shall be performed to identify the optimum atmospheric pressure inside the habitat, taking into consideration suit pressure, airlock effects, hyperbaric treatment, safe haven concerns, habitat repress, and equipment design and certification. D. Perkinson 03/06/92 Rationale: An extensive study of these interrelationships is needed.	?*G	119
The habitat power system shall be capable of supplying power to the cargo lander that is transporting it to the Moon, and supplying this power during the Earth-Moon transit and descent. D. Perkinson 03/25/92 Rationale: The piloted lander does not have a power supply and the use of the habitat power supply will reduce the overall weight of the habitat/lander system, thus allowing more payload capability for the habitat.	?*G	120
The movement of a second habitat (with arrays deployed) shall be slow and gentle enough to prevent damage to these arrays. D. Perkinson 03/06/92 Rationale: Necessary to allow two habitats to be joined.	?*G	121
The habitat shall be landed between lunar dawn and lunar dawn plus 72 hours. D. Perkinson 03/06/92 Rationale: Allows arrays to be deployed before extreme temperatures cause problems and allows fuel cell systems to begin recharge cycle.	?*G	122
The potable water system shall be capable of preventing growth of biological materials in the water during and between crew visits. D. Perkinson 03/11/92 Rationale: Necessary for crew health.	?*G	123
The habitat shall be able to recycle water. D. Perkinson 03/06/92 Rationale: Pay for itself in one visit by reducing resupply costs.	?*G	124

	MAFE	
The photovoltaic array shall be designed by end-of-life considerations. Lifetime shall be TBD years. D. Perkinson 03/06/92 Rationale: Provides long life for habitat.	?*G	125
The radiator shall be designed by end-of-life considerations, including dust problems. Trades shall be performed on methods whereby the radiator can protect itself from dust due to landings at 1 km distance. Radiation life shall be TBD years. D. Perkinson 03/12/92 Rationale: Provides long life via protection from contaminates.	?*G	126
The habitat shall provide a storm shelter with 15 gm/cm2 water equivalent of shielding and with food, control and waste facilities. This storm shelter shall be capable of supporting a crew of four for 3 to 7 days (TBD). D. Perkinson 03/06/92 Rationale: Necessary to protect crew from solar particle events. For contingency habitation, a protection of 60 gm/cm2 is necessary (driven by GCR), and may be required for the entire hab, not just the storm shelter.	?*G	127
Assume revision of shielding levels for the habitable volumes if nuclear power sources are used for power or if nuclear payloads are present. Ward/Rummel/Brown 04/01/92 Rationale: Nuclear sources in proximity to the crew may affect radiation levels present in habitable volumes.	?*G	491
The piloted lander shall bring cryogenic metabolic oxygen on all visits, including the first visit. D. Perkinson 03/06/92 Rationale: It is impractical to keep cryogenic oxygen from habitat delivery to crew arrival. Unless the fuel cell regeneration system can be used to supply oxygen, it must be brought on the piloted flight.	?*G	128
The habitat must be able to supply sufficient data to ground control to provide confidence in its health and operation prior to crew launch. D. Perkinson 03/06/92 Rationale: Insures mission success.	?*G	130

	MAFE	
The cargo lander shall be able to correct any tilt of the lander/habitat such that the habitat is within +/-3 degrees of horizontal. D. Perkinson 03/12/92 Rationale: Excessive tilt increases radiator size (15 degrees results in 10% thermal radiator growth) and reduces habitability of habitat.	?*G	131
More than one method of exiting the habitat is required. The exit minimum hatch dimension is TBD. Loss of habitat shall not be incurred through use of either exit. D. Perkinson 03/12/92 Rationale: Safety in case one door is blocked.	?*G	132
Two EVA suits must remain in habitat at all times that it is occupied. D. Perkinson 03/06/92 Rationale: If two crew are on EVA and inner airlock door won't open, IVA crew (2) must don suits and exit second hatch.	?*G	133
The habitat shall provide a free volume of TBD cubic meters per person for a crew of four. D. Perkinson 03/06/92 Rationale: Habitability concerns.	?*G	134
The concepts for the habitat structure shall not be limited to currently available or designed hardware, and should include options utilizing pressure stabilized structure. D. Perkinson 03/12/92 Rationale: Innovative approaches should be studied, such as inflatable structures.	?*G	420
Concepts for utilizing the lander descent fuel tankage as growth habitable volume shall be developed to supplement the habitat in place of or in addition to the joining of multiple habitats. D. Perkinson 03/12/92 Rationale: The hydrogen tanks alone represent approximately 3/4 of the volume of the habitat, and could serve as work areas, ward rooms, a large storm shelter, etc.	?*G	421

	MAFE	
No nuclear sources exist at the outpost that could result in contamination of crew or equipment requiring a decontamination kit in the habitat. D. Perkinson 03/25/92 Rationale: No sources have been identified, and the decontamination kit has been deleted from the required medical supplies. RTGs containing Pu238 do not represent a contamination source as currently understood..	?*G	495
AIRLOCK	?*G	H135
The airlock allows the transfer of the EVA crew between the lunar surface and the habitat. D. Kissinger 03/06/92 Rationale: Crew will be performing EVA and IVA simultaneously. An airlock will be required to isolate the two environments. The airlock shall provide the transitional volume between the pressurized and unpressurized environments. The airlock also provides a space for dust and contamination clean-up before entering the habitat.	?*G	136
External and internal dust control shall be provided to protect the crew and subsystems. D. Kissinger 03/06/92 Rationale: From FLORG. Lunar dust was shown to have a physiological detriment in Apollo. It also increases abrasion wear in systems hardware. Also, without direct dust control, the ECLSS would have to be sized to accommodate the extra burden.	?*G	137
Stowage and activity volume of TBD m**3 must be provided for EVA preparation, egress, ingress, maintenance and servicing, including dust removal. D. Kissinger 03/12/92 Rationale: Mobility in a suit requires extra consideration when establishing the activity space requirement for EVA preparation, ingress and egress. Storage in accessible areas to permit easy, fast access is required for crew efficiency and emergency actions.	?*G	138
Provisions for the recovery of at least 90% of the airlock gas shall be provided. D.Kissinger 03/06/92 Rationale: Airlock gas becomes a large expendable if frequent EVA is required.	?*G	139

	MAFE	
The 2 IVA crew must be able to perform an EVA when there are 2 crew already performing an EVA to accommodate contingencies. D.Kissinger 03/11/92 Rationale: The second set of crew may need to go EVA for habitat contingencies, lander contingencies, or to assist the first set of crew.	?*G	141
At least 2 crew will perform nominal EVAs. D.Kissinger 03/11/92 Rationale: The buddy system has been required by the crew since the Gemini program.	?*G	142
The capability for consecutive airlock operations shall be provided to accommodate the arrival and departure of 4 crew. The capability for either access to the habitat or the return to the lander shall be assured. D. Kissinger 03/06/92 Rationale: Consecutive airlock operations will be required to effectively move crew and equipment into the habitat. Long periods to wait for the habitat/airlock systems to recover from a depress/repress cycle would be unacceptable.	?*G	143
The EVA system is a critical system and EVA hardware redundancy shall be accommodated that reflects a fail-operational capability. D. Kissinger 03/31/92 Rationale: The EVA system (includes all types of suits that are used) is necessary for the crew to return to the lander. A single credible failure must not require an abort of the mission. Two credible failures must not strand any crew person. A structural failure of the suit or PLSS vent system is not considered a credible failure.	?*G	144
The airlock shall accommodate the transfer of an incapacitated crewperson or cargo of TBD size with crew or cargo of TBD size without crew. D. Kissinger 03/31/92 Rationale: An EVA crewperson can become incapacitated while EVA. The transfer of an incapacitated crew adds additional design constraints for the airlock and EVA system. The transfer of cargo along with the crew adds significant design constraints to the airlock.	?*G	145

	MAFE	
The airlock shall accommodate up to 34 EVAs per 45 days (includes contingency EVAs) plus the initial ingress and final egress. D. Kissinger 03/31/92 Rationale: The number of EVAs define consumable and system sizing as well as provides design objectives.	?*G	146
The minimum hatch dimension for the airlock is TBD. D. Kissinger 03/12/92 Rationale: Hatch size will be determined by the transfer of EVA crew with one possibly incapacitated, and by the transfer of the cargo delivered to the habitat.	?*G	147
The ingress/egress hardware shall be capable of being operated by one crew person to accommodate contingencies. D. Kissinger 03/06/92 Rationale: Contingencies can occur where only one crew person is available to operate the equipment. For example, the airlock pressure control should not require two people to configure the valves properly.	?*G	148
Communications shall be continuous between the EVA crew and the IVA crew. D. Kissinger 03/31/92 Rationale: When the EVA crew is suited, a radio is required for communications. Communications must be continuous to allow for a quick response by the other crew to provide support in an emergency.	?*G	149
The airlock should contain the systems hardware necessary to provide connections to the EMU Service and Performance Checkout Unit. D. Kissinger 03/06/92 Rationale: Contingencies can occur that require connections of the umbilicals at vacuum. The early return of a crewperson to the airlock, while waiting for their buddy, is a desirable practice. Replenishment of expendables in case an airlock failure prudent for crew safety.	?*G	150

	MAFE	
The airlock/habitat should have sufficient room for two crew persons to don/doff the suits by themselves or aided by a third crewperson. Enough space shall be provided for dust removal during doffing. D. Kissinger 03/31/92 Rationale: For crew efficiency, both crew must perform EVA prep simultaneously. It is undesirable to perform don/doff functions in series. A third crew person to aid the first two has been an Astronaut Office requirement for Shuttle and Space Station. See 138.	?*G	151
The airlock shall be operable from outside, inside, and from the habitat side of the airlock. D. Kissinger 03/06/92 Rationale: This requirement affects the location, number, and design of pressure control valves. They must be of a type that does not cause a lock-out situation.	?*G	152
The EVA crew shall be able to return to the habitat and to the lander from any point while EVA. D. Kissinger 03/06/92 Rationale: From FLORG 436 and 437. The habitat may need to be abandoned by the IV crew while a team is EVA. It is essential that the EVA crew be able to reach the lander safely.	?*G	153
EVA may need to be terminated immediately for contingencies. Any surface activity being performed by EVA must be able to be abandoned immediately and still be safe. D. Kissinger 03/06/92 Rationale: A life threatening failure can occur in the EMU, requiring an immediate abort of the EVA. The hardware should not be designed to allow a critical failure if the repair procedures are interrupted. This affects the design of all external systems.	?*G	154

	MAFE	
MEDICAL REQUIREMENTS THAT WILL AFFECT THE AIRLOCK/ HABITAT	?*G	H155
The hyperbaric chamber must be capable of TBD hyperbaric pressurizations per TBD treatment profile. D. Perkinson 04/01/92 Rationale: Determines size and number of high pressure oxygen and nitrogen bottles.	?*G	129
Hyperbaric treatment capability shall be provided. One worst case hyperbaric treatment profile shall be accommodated within the 45 days. D. Kissinger 03/06/92 Rationale: Risk of injury due to hypobaric exposure and depressurization effects, i.e., decompression sickness, gas embolism, and ebublism, can be reduced by providing hyperbaric treatment capability. Current Space Station Freedom includes hyperbaric treatment capability.	?*G	156
Hyperbaric operations (gas pressure, schedule, pass thru) shall be controlled in an adjacent element to the hyperbaric chamber. Direct viewing of the occupants and treatment equipment by the chamber controller shall be provided. D. Kissinger 03/31/92 Rationale: Hyperbaric operations must be controlled by an operator outside the hyperbaric chamber, but with direct view and voice communication with the medical attendant and, if possible, the patient receiving treatment within the chamber. The attendant and/or patient may not be able to respond in event of an accident within the chamber or to effects of pressure and composition transitions that can occur during hyperbaric treatment procedures. Failure to respond properly may result in further injury or death.	?*G	157

	MAFE	
Assume a capability to provide hyperbaric (2.8 atmospheres) treatment to crewmembers on the lunar surface. Ward/Rummel/Brown 03/25/92 Rationale: Over the long term, crewmembers occupying pressurized habitable volumes surrounded by a vacuum may be exposed to accidental but survivable depressurization. Systems reliability and operational safety will be factors in the probability of this depressurization. Hyperbaric capability is required to counter the effects of pressure loss (including decompression sickness, air embolism, and ebullism). For this study, this capability has been allocated to the habitat/airlock.	?*G	490
Communications shall be continuously available between the hyperbaric treatment attendant and the hyperbaric chamber operator. D. Kissinger 03/06/92 Rationale: Since the two are in separate chambers, a communications link is required to pass information on the medical condition and the hyperbaric chamber pressure control. NASA STD 3000.	?*G	158
The capability to pass equipment, supplies, and other material between the hyperbaric facility and the adjacent element during hyperbaric operations shall be provided. D. Kissinger 03/06/92 Rationale: Hyperbaric treatment requires a rapid response. There is no time to move medical equipment into the hyperbaric chamber before pressurization begins. The pass-through is also used for the efficient transfer of EVA tools, ancillary equipment, etc. between IVA/EVA crews.	?*G	159

	MAFE	
The hyperbaric treatment chamber shall have sufficient volume for 3 persons to enter and to accommodate 2 persons for several hours to a few days. A location for the crew member receiving treatment is required with sufficient room around the crew member-patient to permit access by the medical attendant. D. Kissinger 03/31/92 Rationale: Volume sufficient for 3 persons is required to permit transfer and positioning of a crewmember, possibly EVA-suited and by another EVA-suited crewmember assisted by a third attendant, into the hyperbaric treatment chamber. A hyperbaric treatment profile typically requires 5-6 hours, and could require repetition to treat the same patient or a second crewmember. Treatment involving ventilatory support and life saving maneuvers administered by the medical attendant could require rapid, unrestricted access to the crewmember-patient.	?*G	496

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Level_3 docs
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    Linked peers (#)      69 Page width
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    Unlinked children      X Document page headings
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    Reqs. w/target C
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  x Requirements (O)         All requirements
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	MAFE	
PRELAUNCH	?*G	H160
Crew to include at least two trained scientists/mission specialists. D. McKay/D. Eppler 03/13/92 Rationale: Science return will be enhanced by having primary crew members whose specialty is co-incident with the science goals of the mission. A medical doctor will count as a scientist/mission specialist.	?*G	161
Precursor data includes stereo image of [the landing site] area at 15 m or better resolution. D. McKay/D. Eppler 03/06/92 Rationale: Stereo images will allow topographic maps to be made of the landing site area, which will become the primary surface mission planning tool; without stereo images, such variables as slope, height of obstacles and interesting terrain, and depth of craters cannot be determined to significant accuracy. The 15 m resolution is necessary for definition of small objects; assuming a near terminator photography set with a sun angle of 5-10 degrees, objects as small as 1-2 meters high can be detected.	?*G	162
The landing site shall be chosen using precursor data and based on science criteria, resource criteria and operational requirements. D. McKay/D. Eppler 03/13/92 Rationale: All available information must be considered before deciding on an optimum landing site which will have high science interest, high resource potential and allow for effective operations; the order does not imply priority of criteria.	?*G	163
Site maps shall be produced on a photographic data base. D. McKay/D. Eppler 03/06/92 Rationale: Site maps should be tied to a common photographic data base so that subsequent investigations and planning exercises start from a common map scale and photographic resolution, with images that can be easily recognized by crew members.	?*G	164

	MAFE	
Preplanned traverses will be indicated on site maps. D. McKay/D. Eppler 03/06/92 Rationale: Traverse planning will serve as a guide for operational plans on the surface, allowing planners to determine daily schedules, needed rover capabilities and science objectives for a particular site. Indicating these traverses on a photographic data base allows the crew to see the terrain they will be travelling over, and establishes a physiographic context for each traverse.	?*G	165
Site topography and engineering data to be indicated on site maps. D. McKay/D. Eppler 03/06/92 Rationale: These maps will assist the crew in recognizing chemical and mineralogical areas, and in sampling and collecting ISRU feedstock.	?*G	166
Precursor chemical and mineralogical composition to be indicated on site maps. D. McKay/D. Eppler 03/06/92 Rationale: These maps will assist the crew in recognizing chemical and mineralogical areas, and in sampling and collecting ISRU feedstock.	?*G	167
LANDING	?*G	H168
Landing accuracy for delivery of [the] campsite will be +/-2 km (3 sigma). D. McKay/D. Eppler 03/13/92 Rationale: Premission planning will likely be based on a specific target site. Greater variation from a preplanned target will invalidate some of the premission planning for EVA traverses and locations for experiments, and may cause unacceptable delays as traverses and operations are replanned.	?*G	169
[The] capability for revision of site maps and traverses after landing shall be preserved. D. McKay/D. Eppler 03/06/92 Rationale: As early geologic and geochemical data are acquired, premission maps may have to be revised to improve EVA effectiveness. This will be especially important in the event of unexpected discoveries.	?*G	170

	MAFE	
HLLV, TLI STAGE AND POST TLI INTERFACES	?*G	H171
HLLV system shall provide for science and payloads mass of (2.9) mt. D. McKay/D. Eppler 03/06/92 Rationale: The current estimated mass of the science and ISRU payloads is 2.9 metric tons.	?*G	172
TLI Stage Interface shall provide for science and payloads mass of (2.9) mt. D. McKay/D. Eppler 03/11/92 Rationale: The current estimated mass of the science and ISRU payloads is 2.9 metric tons.	?*G	173
Post TLI Interface shall provide for science and payloads mass of (2.9) mt. D. McKay/D. Eppler 03/06/92 Rationale: The current estimated mass of the science and ISRU payloads is 2.9 metric tons.	?*G	174
HABITAT INTERFACE	?*G	H175
The habitat will contain a laboratory area for science activities. D. McKay/D. Eppler 03/06/92 Rationale: Sample identification and sorting of samples for return payload determination will be an important activity for geosciences. Further, acquisition of samples for life sciences experiments will be a primary IVA activity. Each of these activities will require laboratory space to be completed.	?*G	176
This laboratory will provide for geoscience instruments and a work space. D. McKay/D. Eppler 03/06/92 Rationale: Sample identification and sorting will allow the estimated 2 metric tons of collected samples to be reduced to 200 Kg for return to Earth.	?*G	177

	MAFE	
This laboratory will provide for life science instruments and a work space. D. McKay/D. Eppler 03/06/92 Rationale: IVA science activities relating to human parameters will be carried out in the habitat and provision must be made to collect and preserve, and to monitor human performance and biomedical parameters. After screening, key samples will be returned to Earth for further analysis.	?*G	178
Provision will be made to package 200 kg of samples (including containers) for transfer to the ascent stage and return to Earth. D. McKay/D. Eppler 03/06/92 Rationale: Current estimates are that 200 Kg of geologic and biomedical samples (including containers) will be returned to Earth on the first mission. Provision to package and pack these samples must be provided.	?*G	179
The habitat will provide 2000 W intermittent (up to 4 hours at a time) power for the ISRU Demonstration [through] an exterior electrical connector. D. McKay/D. Eppler 03/06/92 Rationale: 2000 W and 4 hours are best estimates for the conceptual ISRU experiment package.	?*G	180
A total of 10 runs of 4 hours each will be required. D. McKay/D. Eppler 03/11/92 Rationale: These are current design concepts for the ISRU experiment package.	?*G	181
All runs can be during the lunar daytime and at times of low power demand. D. McKay/D. Eppler 03/06/92 Rationale: ISRU samples will not be "perishable", and can be run at whatever times that power is available.	?*G	182
A maintenance level of 200 W will be provided for nightime and between-run power. D. McKay/D. Eppler 03/06/92 Rationale: These are current design values necessary to maintain systems in operating conditions in the ISRU experiment package, and to maintain reasonable temperature levels.	?*G	183

	MAFE	
SURFACE SCIENCE OPERATIONS AND INTERFACE	?*G	H184
Provision will be made to unload the rovers, and science payloads from their transport cargo positions to positions for lunar surface operations. D. McKay/D. Eppler 03/13/92 Rationale: Unloading is necessary before this equipment can be used on the lunar surface.	?*G	185
EVA capability using unpressurized manned rover shall be 8 hours long, including time for preparation and housekeeping. D. McKay/D. Eppler 03/13/92 Rationale: Needed to maximize science return relative to the driving overhead for locations >10 km away from the habitat. Extended time can be provided by having replenishable consumables on the manned rover and the ability to transfer to PLSS.	?*G	186
Provision shall be made to allow the crew to dig trenches up to 30 cm, to scoop up soils in a minimum of 1 kg sample sizes, to photograph samples and sampling locations, to scoop or rake up walnut-sized fragments from the regolith, and to drill holes and take cores up to 3 m in depth. D. Eppler/D. McKay 03/31/92 Rationale: Samples cannot be collected and properly documented without the proper tools.	?*G	498
Provisions shall be made to collect up to two metric tons of geoscience samples over cumulative EVA missions and return them to habitat for sorting and packaging of 200 kgm for return to Earth. D. McKay/D. Eppler 03/13/92 Rationale: Will provide the capability to sort and carefully choose important samples frm a much larger population in a shirt sleeve environment rather than use highly valuable EVA traverse time and less certain EVA methods for this purpose.	?*G	187

	MAFE	
Provisions shall be made to operate a manned rover and an unmanned robotic science rover, either one at a time or simultaneously. D. McKay/D. Eppler 03/06/92 Rationale: The benefits of being able to simultaneously operate an unmanned science rover in complement to a human crew are immense. These benefits include: 1) instrument deployment and sample collection beyond walkback distance constraints; 2) collection of routine samples from different locations, such as ISRU samples; 3) routine science activities, such as "mowing the lawn" to obtain statistical composition data on the lunar surface, which leaves the human crew available to concentrate on unusual locations, or in integrating the whole science picture.	?*G	188
Uplink and downlink Earth-to-habitat capability for digital images and maps shall be provided. D. McKay/D. Eppler 03/13/92 Rationale: The crew and ground support personnel need to be able to modify mission planning documents in real time, based on the science discoveries made by the lunar surface crew. In order to do that efficiently, digital data, maps and photographs need to be able to be passed back and forth in real-time to aid replanning efforts.	?*G	189
ISRU SURFACE OPERATIONS	?*G	H190
The ISRU Demo will be a deployment and setup by a two person crew to a position near the side of the habitat within power cord reach of a power connection to the habitat. D. McKay/D. Eppler 03/06/92 Rationale: The ISRU demo will be an important research goal of FLO DRM-1 and, different from other science payloads, will require active participation by the crew to load samples and recover the evolved materials. Placing the demo package near the habitat makes the running and recovery of materials more convenient to the crew and eliminates the need for an additional power supply.	?*G	191

	MAFE	
The ISRU demo will be capable of unattended operation after loading and will have communication to the ground. D. McKay/D. Eppler 03/13/92 Rationale: As above, allowing the payload to work unattended makes running any number of different samples easy and convenient for the crew. Communication with the ground allows crew members to work on other activities while the demo package is processing samples, increasing the efficiency of crew time.	?*G	192
ASTRONOMY, SPACE PHYSICS PAYLOAD SURFACE OPERATIONS	?*G	H193
Surface system shall have the capability to transport and deploy astronomy payloads to at least 10 km from the outpost. D. McKay/D. Eppler 03/13/92 Rationale: Deployment of these payloads at >=10 km from the outpost is levied to ensure that detectors, optical surfaces and electrical grids do not become coated with dust, thereby requiring further maintenance visits from the crew.	?*G	194
Surface system shall have the capability to transport and deploy space physics payloads to at least 10 km from the outpost. D. McKay/D. Eppler 03/13/92 Rationale: Deployment of these payloads at >=10 km from the outpost is levied to ensure that detectors, optical surfaces and electrical grids do not become coated with dust, thereby requiring further maintenance visits from the crew.	?*G	195

	MAFE	
GEOPHYSICAL TRAVERSE SURFACE OPERATIONS	?*G	H196
Surface systems shall provide unpressurized rover systems capable of transporting geophysical equipment on traverses as far as it is possible to drive away from the outpost [NOTE: this is a changed requirement from that submitted on 2/13/92].	?*G	197

D. McKay/D. Eppler 03/06/92
Rationale: Given that the Moon is an essentially unexplored planet, the science community would like to go as far as is possible away from the outpost to conduct exploration and science investigations. Any traverse is going to be subject to distance limitations imposed by safety requirements, rover design and the limitations imposed by the required wear of an often cumbersome EVA suit. However, limitations on the distance that may be traversed in a given 8 hour EVA will be imposed on the science program by these other areas of the program, and not vice versa.

GEOSCIENCE TRAVERSE SURFACE OPERATIONS	?*G	H198
Surface systems shall provide the capability to carry two crew members on geologic/ geophysical traverses as far as it is possible to drive away from the outpost [NOTE: this is a changed requirement from that submitted on 2/13/92].	?*G	199

D. McKay/D. Eppler 03/06/92
Rationale: As above, given that the Moon is an essentially unexplored planet, the science community would like to go as far as is possible away from the outpost to conduct exploration and science investigations. Any traverse is going to be subject to distance limitations imposed by safety requirements, rover design and the limitations imposed by the required wear of an often cumbersome EVA suit. However, limitations on the distance that may be traversed in a given 8 hour EVA will be imposed on the science program by these other areas of the program, and not vice versa.

	MAFE	
ROBOTIC SCIENCE ROVER SURFACE OPERATIONS	?*G	H200
The robotic science rover will have range sufficient to reach areas at least 100 km away from the outpost and return. D. McKay/D. Eppler 03/13/92 Rationale: This will allow a significant increase in science capability beyond manned range, and will allow coverage of an area 25 times larger than is accessible to a manned unpressurized rover, and will allow visiting and sampling of many geologic features beyond the vicinity of the outpost.	?*G	201
The robotic science rovers shall have the capability for daytime operation, and shall have the provision for maintaining systems in a powered down state at night. D. McKay/D. Eppler 03/13/92 Rationale: A requirement to return to the habitat for recharging the power system would greatly limit the range and versatility of a robotic system.	?*G	202
The robotic science rover shall have the capability to operate between missions when the habitat is unoccupied using either Earth-based teleoperation or autonomous robotic guidance. D. McKay/D. Eppler 03/13/92 Rationale: The ability of a teleoperated rover to complement crew science activities is significant. Extending this capability to periods of time when the habitat is unoccupied will greatly increase the range from which science data can be returned, and will increase the diversity of samples that can be returned to the outpost and studied when the human crew returns. In addition, the remote sensing data downlinked by the rover during traverses can influence the mission planning process for future occupation of the outpost.	?*G	203
The robotic science rover shall have sufficient direct communication to the habitat for teleoperations demonstrations from the outpost. D. McKay/D. Eppler 03/13/92 Rationale: Local control of the robotic rover will provide a test of the feasibility of using teleoperations from the habitat to augment EVA activities.	?*G	204

	MAFE	
The robotic science rover shall have communications capability to Earth for teleoperations, command updating, and video. D. McKay/D. Eppler 03/13/92 Rationale: Ground control and data linkage will be necessary for normal operation of the robotic science rover.	?*G	205
The robotic science rover shall have attachment points for digging and trenching tools, specialized sampling tools, and other accessory equipment which might be added on later missions. D. McKay/D. Eppler 03/06/92 Rationale: Scarring the basic rover design as appropriate to engage in these additional activities will reduce the need for new rover designs as additional tasks are assigned to the robotic rover on later missions.	?*G	206
GROUND SUPPORT	?*G	H207
All payloads, except the habitat laboratory instruments, will communicate directly with the ground for data return and control. D. McKay/D. Eppler 03/13/92 Rationale: Each of these instruments is planned to operate as a separate, principle investigator/science team-controlled instrument. Once the equipment is up and operating, it is expected that crew interaction with the instrument will cease, except for scheduled or unscheduled maintenance. Consequently, the requirement exists for direct to Earth communication for each instrument.	?*G	208
Earth ground support will be required, but can be distributed among principal investigator institutions. Provision will be made for post-mission communication, control and data analysis. D. McKay/D. Eppler 03/06/92 Rationale: The advent of communications satellite relayed data to home institutions makes this method of instrument control attractive to this mission, particularly in that it reduces that operational overhead that NASA must provide for science instrument control and monitoring. Consequently, the requirement exists for direct-to-Earth communication for each instrument.	?*G	209

	MAFE	
Data rates will range from 1 kilobit per second for some instruments up to 1.7 megabits per second for the solar telescope and the sounder portion of the traverse geophysical package. D. McKay/D. Eppler 03/06/92 Rationale: Follows from the preliminary designs of each instrument package.	?*G	210
Except for the ISRU demo [package], all payloads will provide their own communication. D. McKay/D. Eppler 03/06/92 Rationale: This will allow for distributed ground control and reduce the need for a central data network and a central ground communication and control facility.	?*G	211
Provision will be made to provide ISRU demo data to Earth through the habitat system (1 kbps). D. McKay/D. Eppler 03/13/92 Rationale: Being able to link the ISRU demo package through the habitat to Earth may allow for future operation during periods when the outpost is unmanned, and will remove the necessity for a separate ground communication system on the ISRU demo package.	?*G	212
MANNED ROVER ELEMENT DESIGN	?*G	H213
Provide the capability to carry two crew members with additional cargo of 400 Kg, or 1 crew member and 600 Kg cargo. D. McKay/D. Eppler 03/13/92 Rationale: Many of the science instrument packages weigh up to 400 kg and need to be deployed at distances of up to 10 km, or carried on traverses. Normally, two crew members are required to deploy payloads, but occasions may arise when fewer or more crew members must be transported at the same time.	?*G	214
The manned rover will have the capability to land, transport, offload and deploy payloads of up to 400 kg at distances up to 10 km away from the habitat. D. McKay/D. Eppler 03/13/92 Rationale: Many of the science instrument packages weigh up to 400 kg and need to be deployed at distances of up to 10 km, or carried on traverses.	?*G	215

	MAFE	
The manned rover will have the capability to carry crew members at average speeds of 7.3 km/hr. D. McKay/D. Eppler 03/13/92 Rationale: This is a current design speed based on Apollo roving guidelines. This design speed is necessary to meet the science objectives requiring traverses out to 20 km within a single 8 hour period.	?*G	216
The manned rover will carry an 8 hour store of consumables for each EVA crew member during each EVA. D. McKay/D. Eppler 03/13/92 Rationale: The addition of such a buddy store increases the safety margin for each crew member in the event of a rover failure, and will allow a walkback from 20 km at an average walking speed of 2.5 km/hr.	?*G	217
The manned rover will have a capability for direct Earth link communications. D. McKay/D. Eppler 03/13/92 Rationale: Terrain features may place the rover out of direct communication links with the outpost, requiring the rover to be able to talk directly to Earth, and where possible, relay through an Earth or Earth-orbiting communications station.	?*G	218
The manned rover will be scarred with attachment points for addition of digging tools, drilling unit, or other useful attachments on future missions. D. McKay/D. Eppler 03/13/92 Rationale: A number of detailed scientific investigations will require ability to see a vertical dimension in the regolith and the underlying rock units. Scarring the rover to be able to accommodate such attachments as digging tools and drilling rigs eliminates the need to initially design separate vehicles or systems for such activities.	?*G	219

	MAFE	
ROBOTIC SCIENCE ROVER ELEMENT DESIGN	?*G	H220
The robotic science rover will have the capability for operation during periods between missions when the habitat is not occupied. D. McKay/D. Eppler 03/13/92 Rationale: The ability of a teleoperated rover to complement crew science activities is significant. Extending this capability to periods of time when the habitat is unoccupied will greatly increase the range from which science data can be returned, and will increase the diversity of samples that can be returned to the outpost and studied when the human crew returns. In addition, the remote sensing data downlinked by the rover during traverses can influence the mission planning process for future occupation of the outpost.	?*G	222
The robotic science rover shall have the capability for both teleoperation and autonomous operation. D. McKay/D. Eppler 03/13/92 Rationale: The ability for autonomous or teleoperated use is key to utilizing the rover during periods when the outpost is unoccupied.	?*G	223
Teleoperation of the robotic science rover shall be possible from both the habitat and from Earth. D. McKay/D. Eppler 03/13/92 Rationale: The ability to do teleoperated science from both locations will greatly enhance the science return from the teleoperated rover.	?*G	224
Robotic science rover shall have stereo video image capability and capability for chemical analysis of geologic materials. D. McKay/D. Eppler 03/06/92 Rationale: Stereo video imaging will be key to safe teleoperations. In situ analysis capability will allow the teleoperator to make real-time decisions on sample collection and traverse planning and execution.	?*G	225
Provisions shall be made to provide, deploy and operate instruments for astronomy investigations. D. Eppler/D. McKay 03/31/92 Rationale: Necessary to implement FLORG requirement #209.	?*G	500

	MAFE	
ASTRONOMY PAYLOAD ELEMENT DESIGN	?*G	H226
Instruments shall have 10 year operational lifetime (experiments, telemetry and power). D. McKay/D. Eppler 03/06/92 Rationale: Each of these instruments should be able to operate for periods of time when the outpost is not permanently occupied. Further, the requirement to keep the instrument area free of dust and disturbance requires that each of these instruments be able to operate without regular visits from a human or robotic crew.	?*G	227
Payloads shall have 2-way communication to Earth. D. McKay/D. Eppler 03/06/92 Rationale: Each of these instruments is planned to operate as a separate, principle investigator/science team-controlled instrument. Once the equipment is up and operating, it is expected that crew interaction with the instrument will cease, except for scheduled or unscheduled maintenance. Consequently, the requirement exists for direct to Earth communication for each instrument.	?*G	228
Payloads shall require no monitoring from the outpost. D. McKay/D. Eppler 03/06/92 Rationale: Each of these instruments is planned to operate as a separate, principle investigator/science team-controlled instrument. Once the equipment is up and operating, it is expected that crew interaction with the instrument will cease, except for scheduled or unscheduled maintenance. Consequently, the requirement exists for direct to Earth communication for each instrument.	?*G	229
SPACE PHYSICS/ENVIRONMENTAL PAYLOAD ELEMENT DESIGN	?*G	H230
Provision shall be made to provide, deploy and operate instruments for geophysical and space physics investigations. D. Eppler/D. McKay 03/31/92 Rationale: Necessary to implement FLORG requirement #207.	?*G	499

	MAFE	
Instruments shall have a 10 year operational lifetime (experiments, telemetry and power). D. McKay/D. Eppler 03/06/92 Rationale: Each of these instruments should be able to operate for periods of time when the outpost is not permanently occupied. Further, the requirement to keep the instrument area free of dust and disturbance requires that each of these instruments be able to operate without regular visits from a human or robotic crew.	?*G	231
Payloads shall have 2-way communication to Earth. D. McKay/D. Eppler 03/06/92 Rationale: Each of these instruments is planned to operate as a separate, principle investigator/science team-controlled instrument. Once the equipment is up and operating, it is expected that crew interaction with the instrument will cease, except for scheduled or unscheduled maintenance. Consequently, the requirement exists for direct to Earth communication for each instrument.	?*G	232
Provision shall be made to provide, deploy, and operate experiments in human physiology, exobiology, and gravitational biology. D. Eppler/D. McKay 03/31/92 Rationale: Necessary to meet FLORG requirements #221 and #224.	?*G	497

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	MAFE	
FLO Mission Assumptions	?*G	H604
The following assumptions were extracted from an early version of the FLORG to provide a basis for the construction of a design reference mission. They are currently being reviewed in the context of the current FLORG. D. Petri 04/01/92	?*G	D576
The first element launch is in late 1999. D. Petri 04/01/92 Rationale: This date is considered to be technically achievable. Validation in FLORG requirement (309).	?*G	577
The Outpost location is unconstrained but will be preselected and certified. D. Petri 04/01/92 Rationale: By preselecting and certifying the outpost site, the surface systems can better be designed for any unique constraints presented by the location. Validation in FLORG requirement (149,244).	?*G	578
FLO Surface segment validation requires operation over at least a lunar day/night cycle. D. Petri 04/01/92 Rationale: Because the FLO mission is for 42 days which covers a day/night cycle validation should occur during both extremes because surface systems are influenced by the different lighting condition. Validation in FLORG requirement (D231,147,174)	?*G	579
Surface systems can be remotely placed in habitat mode and subsequently into storage mode or unmanned mission mode after departure. D. Petri 04/01/92 Rationale: Because the surface elements will have to be functional before the crew arrives, as well as supporting them until they leave, remote operations will be needed. The Validation in FLORG requirement (250,126,251,D231).	?*G	580

	MAFE	
The Surface Segment shall select the second and subsequent landing site aim points at the First Lunar Outpost based upon validated Space transportation landing performance. D. Petri 04/01/92 Rationale: Emplaced equipment from the first visit to the FLO will place certain restrictions on where subsequent landings might be made. The influencing factors include blast ejecta effects and safe flight path profiles. Validation in FLORG requirement (114,237,242,172,260).	?*G	581
During the first mission, space transportation vehicle accommodations will be provided for arrivals and departures only. D. Petri 04/01/92 Rationale: The flight vehicle requirements for the initial mission while on the surface will not include any vehicle servicing. Validation in FLORG requirement (250,126,245).	?*G	582
During the first mission surface operations, space transportation cargo vehicles will be inert. D. Petri 04/01/92 Rationale: Minimizing mass and operations on the initial FLO mission will not allow for movement of cargo vehicles on the surface. Validation in FLORG requirement (250,126).	?*G	583
During the first mission surface operations, space transportation return vehicle support functions will be performed autonomously or remotely from Earth. D. Petri 04/01/92 Rationale: Because of crew size and available crew time, return vehicle support allows the opportunity to handle certain operations remotely and free the crew to do other tasks. Validation in FLORG requirement (250,126,245,162).	?*G	584
The piloted vehicle can carry limited cargo to and from the surface. D. Petri 04/01/92 Rationale: The initial piloted vehicle will only have the capability to carry 4 crew and minimal cargo for logistics. Validation in FLORG requirement (188).	?*G	585

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The Outpost may be in storage mode for up to 12 months between crew visits. D. Petri 04/01/92 Rationale: Validation in FLORG requirement (237,242,233,260).	?*G	586
The ground segment, surface segment and space transportation segment shall collaborate on the accomplishment of all first mission objectives. D. Petri 04/01/92 Rationale: This is to minimize mass and optimize operations required during first FLO mission. Validation in FLORG requirement (162,163).	?*G	587
While on the lunar surface, the piloted vehicle in any mode other than flight mode is an element of the Surface Segment. D. Petri 04/01/92 Rationale: While any flight element is stationary on the surface the responsibility for the well-being of that element is with the surface segment.	?*G	588
There is a large landing zone with provisions for multiple landing sites satisfying mutual separation requirements. D. Petri 04/01/92 Rationale: Because there will be additional visits to the FLO, discarded descent stages from any previous landers will have to be accommodated in the landing zone.	?*G	589
In the habitat zone, non-habitat affixed utility generation and distribution assets will become part of a central utility system. D. Petri 04/01/92 Rationale: This allows for evolution of delivered initial power elements into a system that will grow and accommodate possible future users.	?*G	590
The modes listed below were conceived to describe different operating regimes and identify the state transitions required of the space transportation vehicle and outpost. Different operating modes will demand relatively large differences in resource needs and generation capabilities, such as crew utilization and power generation. D. Petri 04/01/92	?*G	D591

	MAFE	
The space transportation vehicles will have at least 5 modes of operation: Flight Mode - the vehicle is capable of flight Habitat Mode - the vehicle is capable of habitation and surface mission support, but not flight Surface Mission Mode - vehicle is capable of surface mission support, but not habitation or flight Storage Mode - vehicle is not capable of habitation, surface mission support or flight, but is capable of being configured for the other modes Expended Mode - vehicle is not capable of performing any functions and is available to be recycled by the surface segment. D. Petri 04/01/92 Rationale: The designation of modes is needed because of the different functional and operational needs placed upon the space transportation vehicle during different phases of the FLO mission.	?*G	605
The outpost elements will have at least 5 modes of operation: Transport Mode - outpost elements are configured for space transportation Habitat Mode - outpost elements are configured for habitation but not for the pursuit of other surface mission objectives Manned Mission Mode - outpost elements are configured for habitation and other surface manned mission operations Unmanned Mission Mode - outpost elements are configured for unmanned mission operations Storage Mode - outpost elements are configured for survival but not habitation or other surface mission operations D. Petri 04/01/92 Rationale: The designation of modes is needed because of the different functional and operational needs placed upon the outpost elements during different phases of the FLO mission.	?*G	592

	MAFE	
Surface operations begin when the space transportation vehicle is placed in habitat mode and conclude when the surface systems are placed in habitat mode prior to crew departure. D. Petri 04/01/92 Rationale: This helps define a clear point at which responsibility for the space transportation vehicle transfer from the flight segment to the surface segment and then back to the flight segment.	?*G	593
Crew Support Assumptions	?*R	H307
The following crew support assumptions are based upon currently adopted shuttle mission rules (crew procedures management plan, JSC-22359) and proposed space station guidelines. Additional review with operations, crew and life science representatives is required to establish crew constraints unique to FLO. D. Petri 04/01/92	?*R	602
A crew-day timeline used in planning shuttle operations is attached: this timeline is adopted for use in FLO planning. D. Petri 03/09/92	?*R	308
Crew follows a regular Earth circadian rhythm, i.e. 24 hour work-rest cycle, following one work shift. D. Petri 03/09/92 Rationale: Close quarters and integrated activities preclude multiple shifts. Productivity is enhanced if sleep periods are regular and of uniform duration.	?*R	235
Three one-hour meal periods each day with exceptions. D. Petri 03/09/92 Rationale: Meal preparation and consumption takes at least an hour. Exceptions include noon meals taken by EVA crewmembers. These are limited to in-suit solids and liquids.	?*R	236

	MAFE	
Crew sleep periods last 8 hours. Pre-sleep and post-sleep activities last one hour each. D. Petri 03/09/92 Rationale: Productivity is enhanced if sleep periods are regular and of uniform duration. Crewmembers need time to wind down, spin up, perform hygiene tasks, etc., before and after work.	?*R	237
EVA periods (egress to ingress) last no more than ten hours. D. Petri 03/09/92 Rationale: Science objectives presently dictate EVAs as long as 10 hours. A large fraction of the time is spent riding from place to place. This mode of activity does not utilize excessive consumables nor is it particularly fatiguing. An EMU consumables augmentation kit is carried aboard the rover to extend the EVA duration to 10 hours.	?*R	238
Each crewmember receives one day off out of seven mission days. D. Petri 03/09/92	?*R	239
All crew have their days off at the same time. D. Petri 03/09/92	?*R	240
Crew receives a day off one day prior to lunar orbit insertion (assuming minimal loiter in lunar orbit) and one day prior to ascent from the lunar surface. This is so that the crew is fresh (in the best possible physical and mental condition) for powered flight contingencies. These off days are completely free: no duties are scheduled. D. Petri 03/11/92	?*R	241
Other off days (during the surface stay) are light-duty days (e.g. routine housekeeping). D. Petri 03/09/92	?*R	242
Work cycles begin upon landing: no off-day is needed since the crew just had one in space (prior to LOI). D. Petri 03/09/92	?*R	243
The four crewmembers are paired into two two-man teams: all four are EVA qualified. D. Petri 03/09/92	?*R	244

MAFE

After a team performs a full EVA, they are not permitted without waiver to go EV again for two days. This leads to the following basic template for a typical crew week: ?*R 245

TEAM	Mon	Tue	Wed	Thu	Fri	Sat	Sun	Mon	Tue	Wed	Thu	Fri	Sat	Sun
A	E	I	I	E	I	I	R	E	I	I	E	I	I	R
B	I	I	E	I	I	E	R	I	I	E	I	I	E	R

where: E = EVA day
I = IVA day
R = Rest/light duty day

D. Petri 03/09/92

Typical habitat-based EVA operations have no less than two EVA personnel at a time, with one IVA in a support role. ?*R 246
D. Petri 03/09/92
Rationale: Safety-based standard practice.

EVA preparation and post-EVA procedures last no more than 1 hour. ?*R 248
D. Petri 03/09/92
Rationale: Necessary to preserve a 24 hour timeline.

EVAs may be performed on consecutive mission days utilizing alternate pairs of crewmen. ?*R 249
D. Petri 03/13/92
Rationale: Necessary to accomplish mission objectives.

The potential for decompression sickness (bends) for normal operations will be reduced to an insignificant level. ?*R 250
D. Petri 03/09/92
Rationale: The isolated, hostile environment of the Moon demands a level of protection greater than that provided in LEO.

An R-factor of 1.2 or less will be selected. ?*R 251
D. Petri 04/01/92
Rationale: This is consistent with no pre-breathe and the need to reduce decompression sickness to a minimum.

	MAFE	
Spartan hygiene accommodations (e.g. toilet and lavatory, but no shower) are provided. D. Petri 03/09/92 Rationale: For 45 days, 4 crewmembers can well tolerate the hygiene accommodations found on the Space Shuttle Orbiter, for example. More elaborate provisions will increase cost and weight unnecessarily.	?*G	252
High productivity in inventory management, training, nominal and contingency planning and scheduling is achieved with a blend of techniques, procedures, assets, automation and ground support available from Space Station Freedom experience. D. Petri 03/13/92 Rationale: For FLO, the precedent in Space Station Freedom operations will be followed to avoid excessive technology and advanced development costs.	?*G	253
Sufficient monitoring and countermeasures are provided for (TBD) partial-g deconditioning and radiation dosage. D. Petri 03/09/92 Rationale: Necessary to successfully execute and mitigate the long-term effects of a 45-day mission on the lunar surface.	?*G	254
No surgical health care facilities are provided (i.e. medicate but not operate). D. Petri 03/09/92 Rationale: This is an acceptable risk. Healthy individuals are selected for the endeavor. They are likely to return in better condition than Skylab personnel. More extensive facilities are postponed until later in the SEI program.	?*G	255
Stabilization for major trauma is provided. D. Petri 03/09/92 Rationale: Although major trauma is a remote possibility, stabilization is required prior to surface or space transport.	?*G	256

	MAFE	
No heroic treatment for acute illness is provided. D. Petri 03/09/92 Rationale: This is an acceptable risk. Healthy individuals are selected for the endeavor. More extensive facilities are postponed until later in the SEI program. Precedent for this assumption is the Space Station Freedom/ACRV functional requirements.	?*G	257
Invasive procedures for crew health monitoring and life science research occurs only on a crewmembers' IVA days. Otherwise this activity would stretch out an already heavily loaded EVA day. D. Petri 03/09/92	?*G	258
Mental health is preserved by providing: appropriate design, appropriate crew selection, training and psychological support. D. Petri 03/09/92 Rationale: Necessary to overcome the stresses induced by the mission environment.	?*G	259
Time and safety critical functions will fail-operational/fail safe. D. Petri 03/11/92 Rationale: Time and safety critical functions must first fail-operational or potential mission objectives are lost. They must subsequently fail safe such that the crew is not endangered.	?*R	260
All systems must fail safe. D. Petri 03/09/92 Rationale: Systems must fail safe such that the crew is not endangered.	?*R	261
Planet surface training facilities are provided for maintenance of safety-critical skills. D. Petri 03/13/92 Rationale: Training for maintenance of safety-critical skills must be provided such that the crew is not endangered.	?*G	262

	MAFE	
Sufficient skills, tools and materials exist at the outpost to cope with unexpected events, including corrective maintenance. D. Petri 03/09/92 Rationale: The isolated, hostile environment of the Moon demands sufficient skills, tools and materials be must be provided at the outpost such that the crew is not endangered.	?*G	263
The ground will have full capability for failure detection, isolation (FOI) and reconfiguration. The ground segment will be primarily responsible for FDI of all systems and will recommend reconfiguration to the crew. D. Petri 03/13/92 Rationale: For FLO, the precedent in Space Station Freedom operations will be followed to avoid excessive technology and advanced development costs.	?*G	264
During the first several days of the surface mission, every crew transit from the lander to the habitat involves cargo transfer, with the most critical items being carried first. D. Petri 03/09/92	?*G	265
Rover Assumptions	?*G	H277
EVA augmentation package that is carried by the rover, if required, is part of the EVA system. It is only considered as active cargo for the rover. D. Petri 03/13/92	?*G	278
Science cargo requires no active support (power, comm., etc.) from the rover. D. Petri 03/13/92	?*G	279
The rover is manifested on at least the first piloted vehicle. D. Petri 03/09/92	?*G	280
Rovers will have the capability to evolve to full autonomy. D. Petri 03/09/92	?*G	281

	MAFE	
Manned Rover Element Requirements	?*G	H422
The manned rover element shall be capable of operating without consumables replenishment until crew operations are transferred from the lander to the habitat. J. Connolly 03/18/92 Rationale: Initial capability can sustain the required rover operations until the habitat can supply services to the rover.	?*G	423
The manned rover element shall be fully operational upon initial deployment. J. Connolly 03/18/92 Rationale: No capability exists to supply consumables to the rover during the first few days of the mission, until the habitat is functional.	?*G	424
The manned rover element shall provide the capability for teleoperation from either the lunar surface or from Earth. J. Connolly 04/01/92 Rationale: Teleoperation of the rover was deemed desireable for performing performing scientific investigations and maintenance inspections when an EVA crew is not present, for delivering the rover to the vicinity of the piloted lander following crew descent, and to remove the rover from the vicinity of the piloted lander prior to crew ascent. Teleoperation in the sense of this requirement is defined as involving little to no automation, with the rover following commands sent from a manual control s tation. This requirement also assumes some mechanism for the rover to be connected to the rover recharge station.	?*G	595
The manned rover element shall have the capability to transport 1-4 crew and/or cargo, together not to exceed 1000 kg. D. Homan 03/18/92 Rationale: Current payloads and crew mass require 1000 kg for the accomplishment of mission objectives. The extended traverse for the emplacement of astrophysics equipment requires 2 EVA crew (500 kg [E]) and a science payload (500 kg [E]).	?*G.	425

	MAFE	
The manned rover element shall be capable of operating for at least 50 km (E) at 15 km/hr (E) on smooth, level mare. D. Homan 03/18/92 Rationale: This is a worst case scenario for a one day use, as identified by the science team. 15 km/hr is based on Apollo LRV data. This capability is assumed in order to size the power supply.	?*G	426
The manned rover element shall be capable of a climbing 25 km (E), at 4 km/hr (E) on a 30 degree slope. D. Homan/J. Connolly 03/18/92 Rationale: Based on Apollo LRV data. This capability is assumed in order to size the power supply.	?*G	427
The manned rover element shall be capable of a continuous forward and reverse speed range of 0-15 km/hr (E) over smooth, level mare. D. Homan 03/18/92 Rationale: Based on Apollo LRV data. Reverse is required for efficient mobility of the rover.	?*G	428
The manned rover element shall be designed to operate in a lunar dust environment. D. Homan 03/18/92 Rationale: The lunar dust environment caused significant problems during the Apollo missions. Proper dust control will lower the maintenance requirements on the rover, and will not place a burden on available crew time.	?*G	429
The manned rover element shall provide crew and payload protection from lofted dust. D. Homan 03/18/92 Rationale: The lunar dust environment caused significant problems during the Apollo missions. Proper dust control will lower the maintenance requirements on the payloads carried by the rover and the crew.	?*G	430

	MAFE	
The manned rover element shall have the capability to provide navigation information to the operator at all times. D. Homan/J. Connolly 03/18/92 Raitonale: The sorties required by the mission will place the crew at various locations on the surface. To insure that the crew is always within the safe walk back distance to the habitat as required, they will need, at a minimum, bearing and distance information to the habitat.	?*G	431
The manned rover element shall be maintainable on sorties. D. Homan/C. Callaway 03/18/92 Rationale: The rover should allow the crew to perform minor maintenance. Small tool kit will be provided.	?*G	432
The manned rover element shall be maintainable at the outpost. D. Homan/C. Callaway 03/18/92 Rationale: The rover should allow the crew to replace critical components. Critical spares are kept at the outpost. This requires further trade studies.	?*G	433
The manned rover element shall be deployable by one crewmember. D. Homan 03/18/92 Rationale: This is consistent with Apollo LRV requirements.	?*G	434
The manned rover element shall be serviceable by one crewmember. D. Homan 03/18/92 Rationale: The rover servicing should place minimum burden on available EVA crew time.	?*G	435
The manned rover element shall provide crewmember restraints. D. Homan 03/18/92 Rationale: Crew safety dictates that restraints be provided so a crew member is not inadvertently thrown from the rover.	?*G	436

	MAFE	
The manned rover element shall preclude overturning within its operating regime, or shall prevent injury of the crew due to overturning. D. Homan 03/18/92 Rationale: Crew safety dictates that the rover be designed so it cannot overturn on a crewmember.	?*G	437
The manned rover element shall provide displays and warning indications to the operator of critical subsystems. D. Homan 03/18/92 Rationale: This allows the crew to manage the rover's susbsystems wisely. Displays should include speed, power status, etc.	?*G	438
The manned rover element shall have braking capacity sufficient to provide a minimum stopping distance of TBD meters at an intial velocity of 15 km/hr (E) down a 35 degree (E) slope. D. Homan 03/18/92 Raitonale: The rover needs a basic braking capability for operational and safety considerations. This is consistent with Apollo LRV requirements.	?*G	439
The manned rover element shall have a braking capability sufficient to reduce the rover speed from 15 km/hr (E) to a steady state of 4 km/hr (E) down a 35 degree (E) slope and sustain this condition for a distance of TBD meters. D. Homan 03/18/92 Rationale: The rover needs a basic braking capability for operational and safety considerations. This is consistent with Apollo LRV requirements.	?*G	440
The manned rover element shall have the capability to negotiate variable lunar surface with slopes of +/- 30 degrees (E). D. Homan 03/18/92 Rationale: This is consistent with Apollo LRV requirements.	?*G	441

	MAFE	
The manned rover element shall have the capability to negotiate obstacles of at least 30 cm (E) in height. D. Homan 03/18/92 Rationale: As shown during Apollo, the lunar surface does have obstacles such as craters and rock debris.	?*G	442
The manned rover element shall have a ground clearance of at least 40 cm (E). D. Homan 03/18/92 Rationale: Sufficient ground clearance is required so the rover does not get stuck while travelling over rough terrain.	?*G	443
The manned rover element shall be statically stable in pitch and roll for angles of at least 45 degrees (E). D. Homan 03/18/92 Rationale: The lunar surface will present situations where the rover will be on slopes. Good stability is required so the rover does not get in unsafe positions while travelling on traverses.	?*G	444
The manned rover element shall be capable of supporting the loading, unloading, and restraint of all cargo. D. Homan/J. Connolly 03/18/92 Rationale: Because cargo items (particularly science equipment) will have to be moved to various locations on the surface, the rover will have to support the movement of this cargo.	?*G	445
The manned rover element shall be capable of being operated by one crewmember. G. Grush 03/18/92 Rationale: This is consistent with Apollo LRV requirements.	?*G	446
The manned rover element shall have the capability of transporting a payload with a footprint of 2 m (E) x 2 m (E). G. Grush 03/18/92 Rationale: Assumed largest payload footprint delivered by lander(s).	?*G	447

	MAFE	
The manned rover element shall be capable of negotiating voids 70 cm (E) wide. D. Homan 03/18/92 Rationale: This is consistent with Apollo LRV requirements.	?*G	448
EVA Systems Assumptions	?*G	H290
The EVA system shall provide the capability to perform EVAs initiated at either the crew module or the habitat. J. Connolly 04/01/92 Rationale: Provides the capability for space and surface EVAs, and for surface transfer EVAs between the crew module and habitat. The current design assumes the use of 2 EVA suits - a surface optimized suit for robust surface exploration and a fligh t suit which is worn during powered flight, space EVA and short-duration surface transfer EVAs. The 2 EVA suit configuration is due to high volume and mass of the surface exploration suit which makes it incompatible with the volume and mass constraints o f the current crew module design.	?*G	594
The active EMUs and a suite of spare parts will be stowed in the servicing and recharge station adjacent to or within the airlock. The remainder of EMUs and spare parts inventory will be stored in other locations. J. Connolly 03/13/92	?*G	291
The EVA System operations concept for 10 hour EVA assumes that the EV crewmembers will be riding the rover for the majority of the EVA time. Therefore, crew fatigue is within acceptable limits. The EMU consumables augmentation kit provides adequate consumables plus margin for the 10 hour EVA. J. Connolly 03/11/92	?*G	294
Limited EVAs are planned on the days surrounding local noon. EVAs occuring then will be foreshortened to accommodate the increased demand on consumables. J. Connolly 03/13/92	?*G	295

	MAFE	
EMU Element Requirements	?*G	H451
The EMU element shall be capable of being donned and doffed by a single crew person. J. Connolly 03/18/92 Rationale: Nominal operations require that first and/or last crewmember to don/doff suit will have no assistance.	?*G	452
The EMU element shall provide the capability to transmit and receive voice and data communication. G. Grush 03/18/92 Rationale: Communications with habitat, crew module, Earth ground and other EVA crewmembers are required for safe EVA. The network for accomplishing this requirement is not defined at this time, but must be capable of relaying voice and data to the habitat, crew module, Earth ground and other EVA crewmembers.	?*G	453
The EMU element shall require no more than 1.0 (E) hrs maintenance for every 10 hours of nominal operation. J. Kosmo 04/01/92 Rationale: Surface segment timeline allows 2 hours of EMU/PLSS maintenance (assumed to be 1.0 hr PLSS, 1.0 hr EMU maintenance), for every EVA. This maintains a reasonably productive crew schedule.	?*G	454
The EMU element shall allow a crewmember to walk 20 km in 8 hr. J. Connolly 04/01/92 Rationale: This is the maximum walkback distance needed in the event of a rover element failure, assuming a 20 km maximum sortie distance and sortie planning which takes the crew to the furthest point from the outpost first and then works back towards the outpost, keeping the remaining walkback distance within the capability of the PLSS consumables.	?*G	455

	MAFE	
A minimum of N+1 EMU elements shall be provided for a crew size of N. J. Connolly 03/18/92 Rationale: At least one spare EMU must be carried as a contingency for a failed primary EMU. This is a crew safety item. The current design utilizes 2 suits per crewmember to satisfy this requirement.	?*G	456
The EMU element shall provide the capability for use for up to 45 EVAs, including contingency, during a FLO mission. J. Connolly 04/01/92 Rationale: Provides the capability to re-use EVA suits for the duration of the worst case 45 day mission. This requirement is for hardware design only, and does not reflect nominal mission requirements.	?*G	596
The EMU element shall provide the capability for use on subsequent FLO missions. J. Connolly 04/01/92 Rationale: Provides the capability for re-use of EVA equipment on subsequent FLO missions. This requirement is intended to preclude disposal of EVA equipment following a mission and to encourage whole or partial re-use of equipment.	?*G	597
PLSS Element Requirements	?*G	H457
The PLSS element shall provide a capability for recharging for up to 45 EVAs, including contingency, during a FLO mission. J. Connolly 04/01/92 Rationale: Provides capability to reuse EVA equipment for duration of the worst case 45 day FLO mission. This requirement is for hardware design only, and does not reflect nominal mission requirements.	?*G	458
The PLSS element shall be capable of being recharged within TBD hours. J. Connolly 03/18/92 Rationale: Provides capability to perform EVAs at desired frequency.	?*G	459

	MAFE	
The PLSS element shall require no more than 1.0 (E) hours maintenance for every 10 hours of nominal operation. J. Kosmo 04/01/92 Rationale: Surface segment timeline allows 2 hours of EMU/PLSS maintenance (assumed to be 1.0 hr PLSS, 1.0 hr EMU maintenance) for every EVA. This maintains a reasonably productive crew schedule.	?*G	460
A minimum of N+1 PLSS elements shall be provided for a crew of size N. J. Connolly 03/18/92 Rationale: At least on spare PLSS must be carried as a contingency for a failed primary PLSS. This is a crew safety item. The current design utilizes 2 suits per crewmember to satisfy this requirement.	?*G	461
The PLSS element shall provide the capability for use on subsequent FLO missions. J. Connolly 04/01/92 Rationale: Provides the capability for re-use of EVA equipment on subsequent FLO missions. This requirement is intended to preclude disposal of EVA equipment following a mission and to encourage whole or partial re-use of equipment.	?*G	598
The EVA augmentation kit shall provide the capability to recharge 8 hours (E) of PLSS consumables to 2 (E) EVA crewmembers. J. Connolly 04/01/92 Rationale: This requirement attempts to define the EVA augmentation kit to be carried on long rover sorties. The augmentation kit is envisioned to be carried as a rover payload and will give the crew the capability to "top off" their PLSS consumabl es to begin a walk back to the habitat in the event of a rover failure. The augmentation kit may also allow the crew to extend their sortie duration by drawing consumables from it while the rover is in motion.	?*G	599

	MAFE	
The EVA augmentation kit carried by the rover shall be servicable by the same EVA servicing hardware used to service and recharge EMUs and PLSSs worn by crewmembers. J. Connolly 04/01/92 Rationale: No special provisions shall be made to recharge and servide the EVA augmentation kit carried on the rover during long-distance sorties. The augmentation kit shall be designed to connect to the same service and recharge hardware used to se rvice crewmember EMUs and PLSSs.	?*G	600
The EVA augmentation kit is delivered as cargo on a piloted flight. J. Connolly 04/01/92 Rationale: To minimize delivery mass of the habitat, only the minimum required EVA elements (EMU suits and PLSSs for four crew) are delivered with the habitat. Other EVA elements such as the rover EVA augmentation kit, maintenance kit and spare par ts may be delivered as cargo on the piloted flight.	?*G	601
Logistics Assumptions	?*G	H298
The First Lunar Outpost (FLO) may eventually evolve into a larger base, requiring an advanced logistics carrier to support the increasing demands for resupply. J. Connolly 03/13/92	?*G	299
The primary state of gas resupply is cryogenic. J. Connolly 03/09/92	?*G	300
Lunar outpost systems will be eventually upgraded such that trash, waste and obsolete components will eventually become 'feedstock' resources. J. Connolly 03/09/92	?*G	301
Trash and waste materials can be stored in the pressurized logistics carriers indefinitely. J. Connolly 03/09/92	?*G	302
Obsolete components can be stored in unpressurized logistics carriers indefinitely. J. Connolly 03/09/92	?*G	303

	MAFE	
Both pressurized and unpressurized logistics carriers will be provided with passive services while in-transit on the lander and at their surface storage location. J. Connolly 03/09/92	?*G	304
Both pressurized and unpressurized logistics carriers will provide some active services while in-transit on the lander and at their surface storage location. These services may be autonomous or distributed. J. Connolly 03/09/92	?*G	305
Both pressurized and unpressurized logistics carriers require the capability to store their logistics for extended durations on the planet surface, and may need to do so between crew missions. J. Connolly 03/09/92	?*G	306
Logistics Element Requirements	?*G	H462
The logistics element shall provide pressurized transport carriers for cargo requiring pressurized transport. J. Connolly 03/18/92 Rationale: Cargo, such as crew food, crew beverages, crew clothing, medical supplies, biological supplies may require pressurized transport.	?*G	463
The logistics element shall provide transport carriers for unpressurized cargo. J. Connolly 03/18/92 Rationale: Cargo, such as cryogenic O2, cryogenic N2, spare components, tools, equipment, science packages require only unpressurized transport.	?*G	464
The logistics element shall provide monitoring of critical pressurized and unpressurized cargo. J. Connolly 03/18/92 Rationale: Critical cargo needs to be monitored with respect to condition of the cargo, the cargo environment, and the quantity/levels of the cargo.	?*G	465

	MAFE	
The logistics element shall provide thermal control in order to maintain set temperatures within a temperature range -34 degrees C to 20 degrees C (E). J. Connolly 03/18/92 Rationale: Protect the cargo from thermal degradation.	?*G	466
The logistics element shall provide pressure control for pressurized cargo in order to maintain set pressures within a pressure range of TBD psia. J. Connolly 03/18/92 Rationale: Pressurized items require different pressures to prevent degradation due to vaporization or off-gassing.	?*G	467
The logistics element shall provide the capability for IVA unloading of pressurized carriers. J. Connolly 03/18/92 Rationale: This allows protection of pressurized items and reduces EVA operations. This requirement is meant to provide pressurized containers a means of being externally connected to the habitat or airlock, and then being unloaded from inside the habitat without the necessity of bringing the pressurized carrier through the airlock.	?*G	468

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	MAFE	
The objective of the navigation design guidelines and assumptions is to identify the design drivers for defining the navigation architecture concepts. Each design G & A should provide definition for the navigation architecture supporting the First Lunar Outpost study. Appendix A lists, as reference, the FLO requirements and guidelines (4 March 1992) that were considered drivers for the navigation architecture. The numbers in parenthesis are the requirement tracking numbers. If a navigation G & A is related or derived, then the nav G & A includes the requirement tracking number. Appendix B lists the navigation related detail design guidelines and assumptions from the March 13, 1992 draft to show the inter-relationships between each segment and the nav. G. Barton 03/20/92	?*G	D309
Navigation Architecture Concepts	?*G	M310
The navigation architecture will include the Earth-based infrastructure, the lunar-based infrastructure, and the onboard (spacecraft and surface vehicles) sensors. G. Barton 03/09/92 Rationale: The purpose of this guideline is to define the components that comprise the navigation architecture.	?*G	31
All lunar-based infrastructure elements must address any emplacement, maintenance, health monitoring and life-time issues. G. Barton 03/09/92 Rationale: The above are major factors that must be considered before selecting an infrastructure element as a navigation option. FLORG mission guideline #252B states that the lifetime of the lunar surface elements should be maximized.	?*G	313

	MAFE	
The integrated navigation architecture will be designed with synergism in the support of the flight segment (ETO to lunar landing & lunar ascent to Earth landing), the lunar surface segment, and any lunar-based infrastructure survey, ephemeris maintenance or health monitoring. G. Barton 03/19/92 Rationale: FLORG #478 states that the navigation function shall provide an overall integrated navigation capability sufficient to meet mission objectives during all mission phases. The purpose of this guideline is to define the integrated mission areas of support. The navigation architecture should be designed as an integrated capability where, for example, elements selected to support a particular mission phase may also provide a useful hierarchical function during other phases.	?*G	314
The navigation architecture will place strong consideration on utilizing existing navigation technology and systems. G. Barton 03/19/92 Rationale: This guideline is based on the desire to utilize existing hardware components and systems (FLORG #125), to minimize dependence on developing new technology (especially for technology sake) and to minimize dependence on other program's schedule and technical risk that is in the critical path for FLO mission success (FLORG #305A).	?*G	315
The navigation architecture will be designed for evolution. G. Barton 03/09/92 Rationale: This guideline is a flowdown from the FLORG system requirement #154 and based on the desire to allow evolutionary-expansion of the navigation infrastructure. FLORG #253A states that both development cost and life cycle cost should be minimized to the degree practical. As a guideline on the extent of infrastructure utilization, assume a minimum of 4 flights per year. (Reference FLORG 307A & 464)	?*G	316

	MAFE	
The navigation architecture will have the appropriate compatibility and synergism with telecommunications. G. Barton 03/09/92 Rationale: FLORG #335A states that an overall integrated communications, information management, and navigation for the terrestrial, lunar and in-flight system shall exist. The navigation concepts and options will be navigation driven but the integrated navigation architecture will factor in the mission communication objectives.	?*G	317
The navigation architecture will be required to support abort during all mission phases. G. Barton 03/19/92 Rationale: This guideline is directly derivable from FLORG #383.	?*G	473
Strong consideration shall be placed on in situ nav. G. Barton 03/11/92 Rationale: This guideline is based on the desire to minimize the required ground support. FLORG #372A states that surface operations should transition from ground-based to autonomous systems.	?*G	320
Precursors	?*G	H321
The navigation architecture will require gravity knowledge sufficient to keep state vector propagation errors under TBD km after one orbital revolution for all orbital inclinations. G. Barton 03/09/92 Rationale: Navigation studies have shown that the driving factor in propagating the state vector is the uncertainty in the gravity knowledge. The ability to land within a 2 km (3 sigma) of the nominal landing point for the cargo missions (FLORG #153B) and the sortie missions (FLORG #382) may depend on a higher degree of gravity knowledge. The above guideline assumes that navigation will place accuracy requirements on gravity knowledge.	?*G	325

	MAFE	
The navigation architecture will require pre-mission terrain maps of TBD accuracy of the landing site and the final TBD km of groundtrack. G. Barton 03/09/92 Rationale: The above guideline assumes that nav will require terrain models for mission planning, simulation and onboard data base in support of the altimeter and terminal landing site redesignation. DG&A #162 assumes stereo images of the landing site will have a 15 m or better resolution. DG&A #164 assumes that site maps should be tied to a common photographic data base.	?*G	326
Assume the gravity and terrain mapping precursor missions have preceded the FLO. G. Barton 03/09/92 Rationale: FLORG #244A states that precursor missions will have been performed to sufficiently characterize the landing site prior to outpost delivery. Any specific knowledge requirements from navigation should be detailed and levied back on the precursor missions.	?*G	323
Precursor missions will be utilized, whenever appropriate, for emplacement and calibration of the lunar-based infrastructure. G. Barton 03/11/92 Rationale: The navigation architecture may require a lunar-based infrastructure of which navigation satellites and/or surface beacons are required to be emplaced and calibrated. Any specific requirements should be detailed and levied back on the precursor missions.	?*G	322
Assume the Artemis lander can deliver and emplace (soft land) the lunar-based navsites that are required in the navigation architecture. G. Barton 03/09/92 Rationale: The Artemis lander is required to provide global access and is a prime candidate for delivering any navigation surface beacons that may be required in the navigation architecture. Any specific emplacement requirements should be detailed and levied back on the precursor mission.	?*G	324

	MAFE	
ETO	?*G	H327
The navigation architecture will be required to support launch escape of the crew from the launch vehicle in case of an abort. G. Barton 03/19/92 Rationale: FLORG #423 states that the crew will be safely propelled away from the launch vehicle during an on-pad or launch abort below TBD km altitude. FLORG #426 states that the crew module will be at a proper attitude for a safe landing after abort.	?*G	474
The design of the FLO mission does not require LEO rendezvous and docking navigation. G. Barton 03/19/92 Rationale: FLORG mission guideline #246A assumes that a single Earth launch vehicle will be utilized for each flight to the Moon. FLORG #248B states that the crew will be launched with the crew module.	?*G	475
The navigation architecture will support a launch insertion orbit of 100 nmi for up to a maximum stay time of 8 hours to TLI. G. Barton 03/09/92 Rationale: This guideline was derived from DG&A #46 and #405. FLORG #417 and DG&A #88 state that the launch avionics will be on the TLI stage. Navigation should call out any minimum stay time required due to Earth-based tracking constraints.	?*G	328
TLI	?*G	H329
The navigation architecture will provide a post-TLI navigation state update to within TBD accuracy (3 sigma) in support of the first mid-course burn. G. Barton 03/09/92 Rationale: The above guideline addresses one of the critical mission phases which will drive the design of the navigation architecture. The assumption now is to perform the first midcourse burn as soon as the navigation state update can support it.	?*G	330

	MAFE	
The navigation architecture will support disposal of the TLI stage after the stage is separated from the transportation system. G. Barton 03/20/92 Rationale: FLORG #412 states that the lander will be separated from the TLI stage. FLORG #391 states that the TLI stage must be safely disposed. The above guideline addresses one of the safety issues which must be considered in the design of the navigation architecture. Consideration includes the desire to avoid creating hazardous space debris in orbit around the Earth or Moon, and to avoid impacting the Earth in a populated area.	?*G	331
Trans-Lunar / trans-Earth Coast	?*G	H332
Lunar Approach / LOI	?*G	H333
The navigation architecture will provide a state precision accuracy at LOI such that the uncertainty in the navigation state (3 sigma) is within the allowable LOI corridor limits. G. Barton 03/09/92 Rationale: The lander must have inserted into an orbit that allows landing within the pre-designated landing site. Large navigation errors can drive the delta-V requirements higher than the allowed fuel budget. The above guideline addresses one of the critical mission phases which will drive the design of the navigation architecture.	?*G	334
The navigation architecture will provide the capability to support any approach orbital inclination. G. Barton 03/09/92 Rationale: FLORG #149A states that the lander will be required to access any lunar latitude / longitude.	?*G	335

	MAFE	
Orbital Operations	?*G	H336
The navigation architecture will define for the transportation segment the minimum number of orbits between LOI and the de-orbit burn required to acquire a navigation state update. G. Barton 03/09/92 Rationale: FLORG #376 states that the crew module time in lunar orbit should be minimized. The transportation segment has identified a desire to reduce the loitering time in lunar orbit to minimize transportation consumables. DG&A #415 states that the lander will remain in orbit only long enough to acquire a state update to achieve the required landing accuracy.	?*G	337
The current implementation of the FLO requirements does not require LLO rendezvous and docking navigation. G. Barton 03/19/92 Rationale: This guideline is based on the current implementation of the FLO mission requirements for global access (FLORG #149A) and any time return (FLORG #383 and 148A). Other implementations that meet those requirements may exist which will require this support.	?*G	476
Descent / Landing	?*G	H338
The navigation architecture will be required to provide navigation accuracy during powered descent such that state updates do not have a significant impact on fuel usage. G. Barton 03/20/92 Rationale: Large navigation corrections late in the powered descent can lead to guidance course corrections which significantly increase fuel usage.	?*G	488

	MAFE	
The navigation architecture will be required to support an autonomous landing of a cargo vehicle and a sortie mission to within 2 km (3 sigma) of the nominal landing point. G. Barton 03/19/92 Rationale: DG&A #62 assumes the lander has the capability to land autonomously. FLORG mission guideline #153B, #382 and DG&A #169 assume that cargo/sortie missions will have the capability to land within 2 km (3 sigma) of the designated landing site. There, however, currently is no requirement for accuracy. Instead, a 2 km capability was given to the mission definition lead as a placeholder for site certification planning. The navigation performance capability guideline should be based on the following: - Assume no surface or orbital navaids. - Balance between 1) Gravity knowledge (call out required knowledge). 2) State update accuracy of TBD km (3 sigma) TBD minutes before de-orbit burn. 3) Map tie errors (call out required knowledge).	?*G	339
The navigation architecture will be required to support an autonomous landing of a piloted vehicle with a precision landing performance capability of 100 (E) meter (3 sigma) of the designated landing point. G. Barton 03/20/92 Rationale: The above nav guideline was derived from the following. DG&A #62 assumes the lander has the capability to land autonomously. FLORG mission guideline #381 assumes that the piloted landers have the capability to land within 100 m (3 sigma) of the designated landing site. However, there currently is no requirement for precision landing. FLORG mission guideline #380 states that the piloted lander should land at least 1 km (E) away from the critical elements on the lunar surface to minimize blast ejecta damage. FLORG #385 states that the piloted lander shall land within safe walking distance of the habitat, less than or equal to TBD km (E).	?*G	340

	MAFE	
The navigation architecture will be required to support landing site target redesignation for cargo and piloted landers. G. Barton 03/19/92 Rationale: FLORG #415 states that the lander will have the capability to redesignate the landing point until TBD seconds prior to touchdown due to landing hazards that could not be identified from orbit or the large landing site that must be certified due to the navigation inaccuracies. DG&A #408 states that the lander can tolerate hazards up to 0.5 meters in height. DG&A #409 states that the lander can tolerate total tilt up to 12 degrees. DG&A #122 assumes that the habitat landing will occur within 3 days of lunar sunrise. FLORG mission guideline #495 assumes that the piloted landings will occur within 4 days following lunar sunrise. Both cargo and piloted landings will have low sun angles to aid in the identification of surface hazards.	?*G	477
The navigation architecture will be required to land the cargo lander with the solar arrays aligned within +/- 15 degrees of north/south. G. Barton 03/19/92 Rationale: DG&A #103 states that the habitat will be landed with the solar arrays aligned within +15 degrees of north/south to allow tracking of the sun by the arrays with very little cosine loss.	?*G	478
Surface	?*G	H343
The navigation architecture will be required to support manned roving distances of TBD km from the outpost with TBD meter accuracy (3 sigma). G. Barton 03/19/92 Rationale: The purpose of this guideline is to give an idea of the range (and accuracy) of the rover in which the navigation architecture must support. EVA period is 8 hours. (Reference FLORG 202C.) DG&A #215 states that a manned roving capability of up to 10 km away from the habitat should exist. DG&A #165 assumes that preplanned traverses will be indicated on site maps. DG&A #218 states that manned rovers will have a capability for direct Earth link communications.	?*G	344

	MAFE	
The navigation architecture will be required to support robotic roving distances of 100 km from the habitat with TBD meter accuracy (3 sigma). G. Barton 03/09/92 Rationale: The purpose of this science design guideline is to give an idea of the range (and accuracy) of the telerobotic rover in which the navigation architecture must support. (Reference DG&A #201)	?*G	345
The navigation architecture support for EVA will be TBD. G. Barton 03/09/92 Rationale: At this point in time, the guideline is unknown. FLORG #441 states there will be 34 (E) EVAs exclusive of contingency.	?*G	346
Ascent	?*G	H347
The navigation architecture will support a launch insertion orbit of 100 km for up to a maximum stay time of TBD hours before TEI. G. Barton 03/09/92 Rationale: The above guideline addresses one of the critical mission phases which will drive the design of the navigation architecture. The navigation design will need to consider the inclination range.	?*G	348
TEI	?*G	H349
The navigation architecture will provide a post-TEI navigation state update to within TBD accuracy (3 sigma) in support of the first mid-course burn. G. Barton 03/09/92 Rationale: The above guideline addresses one of the critical mission phases which will drive the design of the navigation architecture. The assumption now is to perform the first midcourse burn as soon as the navigation state update can support it.	?*G	350

	MAFE	
Earth Approach	?*G	H351
The navigation architecture will provide sufficient state accuracy to place the vehicle within the flyable entry corridor. G. Barton 03/09/92 Rationale: The return module must fly within the entry corridor. If the flight angle is too steep, the vehicle may exceed g-limits or thermal constraints. If the flight angle is too shallow, the vehicle may skip out. The above guideline addresses one of the critical mission phases which will drive the design of the navigation architecture.	?*G	352
The navigation architecture will support disposal of the return stage after the stage is separated from the crew module. G. Barton 03/20/92 Rationale: FLORG #416 states that the crew module will be separated from the return stage. FLORG #419 states that the return stage must be safely disposed. The above guideline addresses one of the safety issues which must be considered in the design of the navigation architecture. Considerations include the desire to avoid creating hazardous space debris in orbit around the Earth or Moon, and to avoid impacting the Earth in a populated area.	?*G	479
Direct Descent	?*G	H353
The navigation architecture must provide accuracy during atmospheric entry such that the vehicle heating, loads, and flight path are within design limits. G. Barton 03/20/92 Rationale: Large uncertainties in navigated state during re-entry can impact vehicle design requirements for thermal protection, loads, and lift/drag ratio.	?*G	489

	MAFE	
The navigation architecture will be required to support direct descent for a landing on water and land of TBD accuracy (3 sigma). G. Barton 03/19/92 Rationale: The above guideline addresses one of the critical mission phases which will drive the design of the navigation architecture and is derived from the following requirements. FLORG #386 states that the crew module will have a direct Earth re-entry. FLORG #498 states that the Earth landings on land will be at TBD locations. FLORG #257A states that the nominal landing point will be on land. FLORG #373A and #422 states that a contingency landing will be on water. FLORG #256A states that landings will be performed under any lighting constraints.	?*G	354

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?*G H469

?*R D470

APPENDIX A

The following are the FLO Requirements and Guidelines (FLORG) that are considered drivers for the navigation architecture. The numbers in parenthesis are the requirement tracking numbers.

4.0 FIRST LUNAR OUTPOST DESCRIPTION

4.2 Design Guidelines and Assumptions

(253A) To the degree practical, minimize both development cost and life cycle cost.

(305A) To the degree practical, minimize dependence on other programs.

(125) To the degree practical, utilize existing hardware components and systems.

(376) To the degree practical, minimize crew module time in orbit.

(377) To the degree practical, minimize orbital operations such as rendezvous, docking, and assembly for piloted and cargo vehicles.

(252B) To the degree practical, maximize lifetimes of all lunar surface elements.

(246A) Assume a single Earth launch vehicle shall be utilized for each flight to the Moon.

(244A) Assume the necessary precursor missions

will have been performed to sufficiently characterize the landing site prior to outpost delivery.

(495) Assume piloted landings on the lunar surface will occur within 4 days following lunar sunrise.

(380) Assume that flight elements shall land at least 1 km (E) away from critical elements on the lunar surface to minimize blast ejecta damage.

(381) For nominal missions and revisit missions, assume piloted landers shall provide the capability to land within 100 m (3 sigma) of the designated lunar landing site.

(153B) Assume cargo landers shall provide the capability to land within 2 km (3 sigma) of the designated lunar landing site.

(382) For sortie missions, assume piloted landers shall provide the capability to land within 2 km (3 sigma) of the designated landing site.

5.0 TECHNICAL REQUIREMENTS

5.2 System Requirements

(154) The system shall be designed for evolution.

(307A) The system shall provide the capability to support a minimum of four (4) flights per year.

(149A) The system shall provide the capability to access any lunar latitude/longitude.

(387) Sortie option lunar surface stays shall be restricted to lunar daylight.

(383) The system shall provide the capability for abort during all mission phases.

(385) For nominal missions, piloted elements shall land within safe walking distance of the habitat, less than or equal to TBD km (E).

(386) The system shall provide the capability for direct Earth re-entry of the crew module.

(498) The system shall provide the capability to perform Earth landings on land at TBD locations.

(373A) The system shall provide the capability to perform Earth water landings

(257A) Nominal Earth return landings shall take place on land.

(155B) The system shall only utilize technologies which have undergone system validation or engineering model demonstration in a relevant environment prior to commitment to development.

5.2.1 Flight Segment

(248B) The crew shall be launched with the space transfer vehicle crew module.

(256A) The flight segment shall provide the capability to perform Earth landings under any lighting conditions.

(168A) The flight segment shall provide the capability to revisit previous lunar landing sites.

5.2.1.2 Space Transit Sub-Segment

(414) The space transit sub-segment shall provide the capability for abort at all times during piloted lunar descent.

(415) The space transit sub-segment shall provide the capability for landing site aimpoint redesignation until TBD seconds prior to lunar touchdown.

(148A) The space transit sub-segment shall provide the capability for abort at all times following piloted lunar landing within 24 hrs. of abort declaration.

(412) The space transit sub-segment shall provide the capability for separation of the lander from the TLI stage.

(416) The space transit sub-segment shall provide the capability for separation of the crew module from the return stage.

5.2.1.2.1 TLI Stage Element

(417) The TLI stage element shall provide the capability for ascent guidance and control of the launch vehicle during launch from Earth.

(418) The TLI stage element shall provide the capability for attitude correction prior to TLI burn.

(391) The capability shall be provided for safe disposal of the TLI stage element.

5.2.1.2.3 Return Stage Element

(191B) The return stage element shall provide the capability for abort at all times following lunar landing within 24 hrs. of abort declaration.

(419) The capability shall be provided for safe disposal of the return stage element.

5.2.1.2.4 Crew Module Element

(489) The crew module element shall provide the capability to land on land when returning to Earth.

(422) The crew module element shall provide the capability to land on water when returning to Earth

5.2.1.2.5 Launch Escape System Element

(423) The launch escape system element shall provide the capability to safely propel the crew module away from the launch vehicle during an on-pad or launch abort below TBD km altitude.

(426) The launch escape system element shall provide the capability to assure proper crew module attitude for a safe Earth landing after the abort.

5.2.2 Surface Segment

5.2.2.1 Habitation Sub-Segment

(441) The habitation sub-segment shall provide the capability to support 34 (E) EVAs exclusive of contingency.

5.2.2.2 Surface Mobility Sub-Segment

5.2.2.2.1 Rover Element

(202C) The rover element shall provide the capability to transport a crew of 2 (E) and a cargo of TBD mt a distance of TBD km from the outpost.

5.2.2.3.2 Navigation

(348A) The capability shall be provided to aid the navigation of elements in lunar orbit.

(349A) The capability shall be provided to aid the navigation of elements on the lunar surface.

(374A) The capability shall be provided to aid the navigation of piloted landers during lunar descent during the nominal mission.

(375A) The capability shall be provided to aid the navigation of ascent vehicles during lunar ascent.

5.2.3 Support Function Segment

(464) The Support Segment shall be able to support a rate of four FLO missions a year.

5.2.3.1 Mission Support and Control Sub-Segment

(466) The Mission Support and Control Sub-Segment shall maintain a capability during all mission phases to abort the mission and safely return the flight crew to Earth.

5.2.3.1.4 Surface Operations

(372A) The surface operations function shall provide for an operational transition from ground-based control to surface-based control and autonomy of lunar surface systems and elements.

5.2.3.4 TISN Sub-Segment

(335A) The TISN sub-segment shall provide an overall integrated capability for communication, information management, and navigation for the terrestrial, lunar, and in-flight systems involved in FLO missions.

(476) The TISN sub-segment shall facilitate fault detection, fault isolation, and recovery control for the terrestrial, lunar, and in-flight systems

involved in FLO missions.

5.2.3.4.2 Navigation

(478) The navigation function shall provide an overall integrated navigation capability sufficient to meet mission objectives during all mission phases.

(479) The navigation function shall provide a capability for state vector updates to flight and surface vehicles.

(480) The navigation function shall provide data for post-flight trajectory analysis and reconstruction.

5.4 System Design Safety and Reliability

(485) All systems shall be "fail safe".

(486) Redundant systems shall provide a capability to verify their operational status to the fail safe level during ground turnaround and during the mission.

(487) The systems for the first lunar outpost mission shall not preclude a piloted mission from restoring full operational capability after a single functional failure.

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APPENDIX B ?*R H471

The following are the mission/system detailed design guideline and assumptions for the First Lunar Outpost that are considered drivers for the navigation architecture. The numbers in parenthesis are the guideline tracking numbers. ?*R D472

3.1 LUNAR TRANSPORTATION SYSTEM (LTS) DETAILED ASSUMPTIONS

Space Transit Sub-Segment

(415) The space transit sub-segment shall provide the capability for the lander element to remain in lunar orbit only for that time required for the navigation state vector update to achieve the required landing accuracy.

Lunar Lander

(408) The lander element shall provide the capability to tolerate a landing hazard of less than or equal to 0.5 meters in height.

(409) The lander element shall provide the capability to operate with a total lander tilt of less than or equal to 12 degrees.

(62) The lander element shall provide the capability to automatically land on the lunar surface.

TLI Stage Design

(46) The TLI stage element shall provide the capability of a suborbital burn to achieve a launch insertion orbit of 100 nmi and maintain the orbit up to a maximum stay time of 8 hours prior to TLI burn.

Return Stage

(76) The capability shall be provided for the safe disposal of the return stage element.

3.2 HEAVY LIFT LAUNCH VEHICLE (HLLV) DETAILED ASSUMPTIONS

Earth to Orbit Sub-Segment

(405) The following orbits shall be used: 100 nm circular for lunar direct; 220 nm circular for Mars missions.

(88) All launch phase primary avionics (e.g., GN&C equipment, computers, telemetry equipment, etc.) shall be located on the TLI stage and shall be independent of the LTS avionics. Each stage shall be capable of sending stage sub-system status to the launch phase primary avionics.

3.3 HABITATION SUB-SEGMENT DETAILED ASSUMPTIONS

Habitat

(103) The habitat will be landed with the solar arrays aligned within +15 degrees of north/south.

(122) The habitat shall be landed between lunar dawn and lunar dawn plus 72 hours.

3.4 SCIENCE AND PAYLOADS DETAILED ASSUMPTIONS

Prelaunch

(162) Precursor data includes stereo image of [the landing site] area at 15 m or better resolution.

(163) The Landing site shall be chosen using precursor data and based on science criteria, resource criteria, and operational requirements.

(164) Site maps shall be produced on a photographic data base.

(165) Preplanned traverses will be indicated on site maps.

(166) Site topography and engineering data to be indicated on site maps.

Landing

(169) Landing accuracy for delivery of [the] campsite will be +/- 2 km (3 sigma).

Robotic Science Rover Operations

(201) The robotic science rover will have range sufficient to reach areas at least 100 km away from the habitat and return.

Manned Rover Element Design

(215) The manned rover will have the capability to land, transport, offload and deploy payloads of up to 400 kg at distances up to 10 km away from the habitat.

(218) The manned rover will have a capability for direct Earth link communications.

Robotic Science Rover Element Design

(225) Robotic science rover shall have stereo video image capability and capability for chemical analysis of geologic materials.

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General	?*R	H502
The Earth-based operations shall encompass all activities and related services conducted on Earth in direct support of the First Lunar Outpost (FLO) mission and to which resources shall be acquired to meet facility, outfitting, hardware systems, software systems, and manpower requirements. D. Lamar 04/01/92 Rationale: To establish scope and objective for Earth based operations requirements.	?*R	503
Requirements for the acquisition and utilization of mission support services from provider sources external to SEI program shall be in accordance with NASA Management instructions, Memorandum of Agreements, Memorandum of Understandings, or Program Directives as instructed by the Associate Administrator for Exploration. D. Lamar 04/01/92 Rationale: Associate Administrator for Exploration is responsible for policy and external management interfaces.	?*R	504
Each Earth-based operations function, as appropriate, shall develop support concepts (timelines, resources, flows, etc.) in such a manner to isolate the effects of unplanned operations and to eliminate the propagation of schedule impacts for expected contingencies. D. Lamar 04/01/92 Rationale: Providing margin (schedule, resources) for each Earth based operations function is necessary to insure some reserve capacity to accommodate contingencies and still meet the mission schedules.	?*R	505
Earth based operations shall evolve to the capability to support a minimum of eight flights per year and 45-day launch centers through capability augmentation and learning experience by the TBD timeframe. D. Lamar 04/01/92 Rationale: The segment must have adequate capability to support the required flight rate.	?*R	506

	MAFE	
Earth based operations shall provide capabilities to support any specific function subsequently identified and added. D. Lamar 04/01/92 Rationale: The overall mission level operation support requirements need to be established to justify the subsequent implementation.	?*R	507
Earth based operations shall provide the capability to provide facility operations for First Lunar Outpost Missions. D. Lamar 04/01/92 Rationale: The level of implementation of this requirements will be based upon the need to provide command and monitor functions in support of First Lunar Outpost missions and upon the crew's capability to perform autonomous operations.	?*R	508
Earth based operations shall provide capability to maintain operations support systems. D. Lamar 04/01/92 Rationale: Operations capabilities must be maintained in order to support the program. Operations support systems include hardware, software, and facilities.	?*R	509
Where appropriate, hardware components and systems which were developed for other programs but which will be used for the FLO program shall be modified to allow system checkout and maintenance by either the crew or by robotic means. D. Lamar 04/01/92 Rationale: FLO will attempt to use existing proven hardware from ongoing programs (Shuttle, STF) in order to save cost and time. This hardware was designed to perform specific technical functions which mat not be supportive of EVA or robotic mainten ance especially on the lunar surface. Appropriate hardware modification may be required for the outpost to be evolvable and maintainable.	?*R	510

	MAFE	
Mission Operations	?*R	H511
The Earth-based real-time support consists of flight and lunar surface systems and payloads operations support. The functions of real-time operations support are command and control, flight crew well-being, space systems management, training, and data collection and processing. D. Lamar 04/01/92 Rationale: TBD	?*R	512
Earth-based real-time operations shall develop and execute mission plans and ensure effective conduct for FLO in-space objectives. D. Lamar 04/01/92 Rationale: TBD	?*R	513
The payloads operations support function shall ensure the successful deployment and operation of payload equipment to satisfy mission goals. D. Lamar 04/01/92 Rationale: The focus of Earth-based real-time operations is on effective utilization of available FLO resources to establish an initial outpost and accomplish payload objectives.	?*R	514
FLO will function as an integrated facility. D. Lamar 04/01/92 Rationale: TBD	?*R	515
The crew will be selected and trained according to the skills required to operate and maintain the outpost and the payload systems available. D. Lamar 04/01/92 Rationale: TBD	?*R	516
A crew of four will work single shifts. D. Lamar 04/01/92 Rationale: TBD	?*R	517
Crew activities, though flexible, will be coordinated with and monitored by Earth-based real-time personnel. D. Lamar 04/01/92 Rationale: TBD	?*R	518

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The lunar surface operation of FLO includes planning and operations for manned based systems and payloads logistics support, and support to Earth-based real-time activities. D. Lamar 04/01/92 Rationale: TBD	?*R	519
Planning and Analysis	?*R	H520
Mission planning and preparation shall design FLO missions to accommodate a lunar surface stay of up to 45 days. D. Lamar 04/01/92 Rationale: The nominal surface stay is 45 days, however, the capability to abort the mission at anytime is required.	?*R	521
The mission will be designed to allow Earth return within 24 hours of a decision to abort the mission. D. Lamar 04/01/92 Rationale: The nominal surface stay is 45 days, however, the capability to abort the mission at anytime is required.	?*R	522
In transit to and from the lunar surface the Earth-based real-time operations monitors and controls FLO mission operations. D. Lamar 04/01/92 Rationale: TBD	?*R	523
Capability to abort the mission for a safe return of crew will be available during all phases. D. Lamar 04/01/92 Rationale: TBD	?*R	524
Mission planning and analysis shall plan for early mission termination such that abort modes to the Earth have priority over those to the Moon. D. Lamar 04/01/92 Rationale: Abort modes to the moon will require additional action to return the crew to Earth.	?*R	525
No in-flight EVAs will be planned. D. Lamar 04/01/92 Rationale: Provides a balance between crew time and crew health and safety.	?*R	526

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Mission planning and preparation shall design First Lunar Outpost missions such that adequate windows are provided to accommodate delays in the launch preparation process. D. Lamar 04/01/92 Rationale: Allowances must be made to accommodate problems which will occur during the launch preparation process.	?*R	527
Earth return trajectories should be designed; if possible, to allow daylight land landings. D. Lamar 04/01/92 Rationale: TBD	?*R	528
Earth orbit loiter capability prior to TLI of at least 2 (E) orbits shall be provided. D. Lamar 04/01/92 Rationale: Earth orbit loiter will allow time for final systems checkout before committing to lunar transfer.	?*R	529
On the lunar surface, a crew of four shall operate the lunar outpost in accordance with pre-planned activity schedules; however, flexibility to modify schedules and procedures shall be provided. D. Lamar 04/01/92 Rationale: TBD	?*R	5
The level of Earth-based real-time operations support is expected to diminish as future missions evolve. D. Lamar 04/01/92 Rationale: TBD	?*R	531
Mission Operations shall provide the capability to initiate flight vehicle termination from Earth liftoff until main engine cutoff. D. Lamar 04/01/92 Rationale: Derived from the system requirement to provide the capability for flight vehicle termination from Earth liftoff until main engine cutoff due to range safety limit violations (e.g., tumble rates, launch corridor).	?*R	532
The Mission Planning and Preparation Function shall provide capability to support strategic and tactical planning for FLO missions. D. Lamar 04/01/92 Rationale: TBD	?*R	533

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Mission planning and preparation shall support the capability for one planned, two person EVA on the lunar surface of eight hours suit duration daily. D. Lamar 04/01/92 Rationale: Provides a balance between crew time and crew health and safety.	?*R	534
Real-time Operations	?*R	H535
Real-time FLO operations will be supported in an Earth-based real-time facility to accomplish the mission requirements. D. Lamar 04/01/92 Rationale: TBD	?*R	536
A mission control center will monitor flight and surface systems status and conduct activities related to operations and maintenance of the outpost systems. D. Lamar 04/01/92 Rationale: TBD	?*R	537
A payload operations center will provide support for the integration of payloads into the vehicle and lunar outpost and in the conduct of lunar surface activities to accomplish payload objectives. D. Lamar 04/01/92 Rationale: TBD	?*R	538
The FLO system element participants will support the mission control center through staff located at the control center. D. Lamar 04/01/92 Rationale: TBD	?*R	539
The Earth-based real-time operations shall operate as a single integrated facility. D. Lamar 04/01/92 Rationale: TBD	?*R	540
Real-time operations shall be implemented so that crew safety is tolerant to one (E) operational failure. D. Lamar 04/01/92 Rationale: TBD	?*R	541

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The Surface Operations shall provide capability to monitor and record crew radiation dosage. D. Lamar 04/01/92 Rationale: TBD	?*R	542
Real-time operations shall provide access to data on a scheduled basis allowing users to control, monitor and analyze their data. D. Lamar 04/01/92 Rationale: Payload unique access equipment, procedures, and commands are assumed to be supplied by the user.	?*R	543
Real-time operations shall have the capability to provide precise navigation information to elements and equipment on the lunar surface. D. Lamar 04/01/92 Rationale: Precise location on the lunar surface is required for safety and mission planning.	?*R	544
The Flight Operations Function shall provide real-time capability for navigation updates to space transfer vehicles. D. Lamar 04/01/92 Rationale: Navigation and state vector updates will be required to support the onboard systems.	?*R	54
Other	?*R	H546
The participation of flight crew members in test activities for training purposes associated with operating FLO flight equipment shall be accommodated on a noninterference basis with the preceding cargo mission. D. Lamar 04/01/92 Rationale: TBD	?*R	547
Where practical, real-time operations shall be designed to accommodate new technology. D. Lamar 04/01/92 Rationale: TBD	?*R	548
Where practical, real-time operations shall be designed to accommodate changing mission requirements. D. Lamar 04/01/92 Rationale: TBD	?*R	549

	MAFE	
Where practical, real-time operations shall be designed to accommodate increasing autonomy and automation. D. Lamar 04/01/92 Rationale: TBD	?*R	550
Launch Site Operations	?*R	H551
Launch site operations shall consist only of those activities/functions to be performed at the launch site. D. Lamar 04/01/92 Rationale: TBD	?*R	552
The Launch Operations Function shall provide capability for launch site logistics operations. D. Lamar 04/01/92 Rationale: The launch site must have the capability to provide maintenance and repair to the launch vehicle and the payload in order to insure mission success. In order to do this, spares, tools, and storage must be provided to the level that the la unch rate demands.	?*R	553
Payload unique access equipment, procedures, and commands shall be supplied by the user. D. Lamar 04/01/92 Rationale: TBD	?*R	554
The launch site operational processing timeline shall be based on a work schedule of five (5) days/week and up to three (3) shifts/day availability. D. Lamar 04/01/92 Rationale: This has been picked for a reference which is consistent with the recommendations made by KSC. This reference will provide a 25% manpower margin to mitigate schedule slip due to contingencies. The effectiveness of increasing the availabl e manpower versus systems and designs to increase efficiency to meet the required processing flow versus relaxing the processing schedule is subject to analysis.	?*R	555
Payload integration shall be conducted at the launch site. D. Lamar 04/01/92 Rationale: Consistent with "ship and shoot" philosophy.	?*R	556

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No deferred work or modifications shall be performed at the launch site. D. Lamar 04/01/92 Rationale: Consistent with "ship and shoot" philosophy, while allowing launch site rework or repair.	?*R	557
Payload processing shall occur prior to payload arrival at the launch site. D. Lamar 04/01/92 Rationale: This corresponds to a "ship and shoot" philosophy at the launch site; yet, it acknowledges that the work needs to be performed.	?*R	558
All on pad vehicle access and servicing shall be furnished from the Mobile Launch Tower (MLT) surface or via swing arms from the MLT. D. Lamar 04/01/92 Rationale: Established by inference from mission requirements.	?*R	559
Operations at the launch pad shall accommodate flight crew ingress at TBD (2-4 hrs. (E)) prior to liftoff (T-zero). D. Lamar 04/01/92 Rationale: Established by inference from mission requirements.	?*R	56
Logistics operations shall provide on-site logistical support of the FLO launch vehicle, ground support equipment, and launch control center. D. Lamar 04/01/92 Rationale: TBD	?*R	561
Recovery Operations	?*R	H562
Network Operations	?*R	H563
The Information Management Function shall provide users access to real-time data on a scheduled basis. D. Lamar 04/01/92 Users need data to control and monitor their payloads and to analyze their data on a scheduled basis.	?*R	564

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The TISN Sub-Segment shall provide capability to capture science telemetry data and route the data to remote science users. D. Lamar 04/01/92 Rationale: Many scientists will want to operate their experiments from remote science centers. This will require the routing of science data (and mission data) to these centers. This statement does not rule out the possibility for scientists to cap ture data directly from payloads.	?*R	565
The only video data will come from the rover. There is no video from the habitat or crew cab. D. Lamar 04/01/92 Rationale: Interpretation of current requirements and implementations.	?*R	566
Integrated Logistics	?*R	H567
Integrated Logistics shall provide support equipment to support SEI FLO element launch. D. Lamar 04/01/92 Rationale: TBD	?*R	568
Integrated Logistics shall provide manpower and personal availability and status to support SEI FLO element launch. D. Lamar 04/01/92 Rationale: TBD	?*R	569
Integrated Logistics shall provide procurement support to support SEI FLO element launch. D. Lamar 04/01/92 Rationale: TBD	?*R	570
Integrated Logistics shall provide terrestrial packaging, handling, storage, and transportation to support SEI FLO element launch. D. Lamar 04/01/92 Rationale: TBD	?*R	571
Integrated Logistics shall provide off-line maintenance and storage facilities as appropriate to support SEI FLO element launch. D. Lamar 04/01/92 Rationale: TBD	?*R	572

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Integrated Logistics shall provide inventory management to support SEI FLO element launch. D. Lamar 04/01/92 Rationale: TBD	?*R	573
Integrated Logistics shall provide storage and access to technical data and documentation to support SEI FLO element launch. D. Lamar 04/01/92 Rationale: TBD	?*R	574
Special Services	?*R	H575

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506:p1	521:p4	536:p6	551:p8	566:p10
507:p2	522:p4	537:p6	552:p8	567:p10
508:p2	523:p4	538:p6	553:p8	568:p10
509:p2	524:p4	539:p6	554:p8	569:p10
510:p2	525:p4	540:p6	555:p8	570:p10
511:p3	526:p4	541:p6	556:p8	571:p10
512:p3	527:p5	542:p7	557:p9	572:p10
513:p3	528:p5	543:p7	558:p9	573:p11
514:p3	529:p5	544:p7	559:p9	574:p11
515:p3	530:p5	545:p7	560:p9	575:p11
516:p3	531:p5	546:p7	561:p9	

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502:p1	517:p3	532:p5	547:p7	562:p9
503:p1	518:p3	533:p5	548:p7	563:p9
504:p1	519:p4	534:p6	549:p7	564:p9
505:p1	520:p4	535:p6	550:p8	565:p10
506:p1	521:p4	536:p6	551:p8	566:p10
507:p2	522:p4	537:p6	552:p8	567:p10
508:p2	523:p4	538:p6	553:p8	568:p10
509:p2	524:p4	539:p6	554:p8	569:p10
510:p2	525:p4	540:p6	555:p8	570:p10
511:p3	526:p4	541:p6	556:p8	571:p10
512:p3	527:p5	542:p7	557:p9	572:p10
513:p3	528:p5	543:p7	558:p9	573:p11
514:p3	529:p5	544:p7	559:p9	574:p11
515:p3	530:p5	545:p7	560:p9	575:p11
516:p3	531:p5	546:p7	561:p9	

Abstract

First Lunar Outpost System Effectiveness Preliminary Report

This section of the report documents work done to date in system effectiveness for the First Lunar Outpost (FLO) activity. A description of the system effectiveness plan is presented and includes discussion of the overall system effectiveness methodology, preliminary criteria, and necessary parameters to be measured. A detailed example associated with power option assessments was prepared as part of the FLO Consolidated Engineering Review. This example consolidated power system data from each major element of FLO for an assessment of overall mission impact.

Questions concerning this section should be directed to XE/Eileen Stansbery (713/283-5393).

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First Lunar Outpost System Effectiveness

Preliminary Report

Exploration Programs Office
Johnson Space Center
Houston, Texas

NASA

National Aeronautics and Space Administration

First Lunar Outpost System Effectiveness

Preliminary Report

May, 1992

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1 Purpose

The purpose of this report is to document work done to date in system effectiveness for the First Lunar Outpost (FLO) activity. A description of the system effectiveness plan is presented and includes discussion of the overall system effectiveness methodology, preliminary criteria, and necessary parameters to be measured. A detailed example associated with power option assessments was prepared as part of the FLO Consolidated Engineering Review. This example consolidated power system data from each major element of FLO for an assessment of overall mission impact.

2 System Effectiveness Overview

System effectiveness deals with assessing the capability of a system to meet its mission objectives when called upon to do so. In the broadest sense, it includes the capability of the system to meet performance, cost and schedule goals within specified risk. System effectiveness analysis supports a design process in which a balance is achieved between cost, performance and supportability, and schedule. Typically, specialty engineering disciplines relating to supportability, particularly reliability and maintainability, are implemented early in the design process since they can have the greatest impact on overall design decisions.

The goal of the current system effectiveness activity is to provide the tools, database, and analysis necessary to assess the effectiveness of alternative implementation concepts for FLO. The assessments are based on measures of effectiveness (MOEs), also called figures of merit, that provide a quantitative means of comparing alternative system configurations or comparing proposed changes to the baseline configuration.

Early in program definition phases, most concepts have low fidelity, therefore the data used in effectiveness analysis is soft. Parametric sensitivity analysis is therefore an important part of early system effectiveness analysis so that an understanding can be gained of trends associated with and impacts due to varying concept characteristics. This provides insight into the relative importance of particular parameter firmness and any risk associated with confidence in the fidelity of parameter characteristic estimates. Another important aspect of effectiveness analysis in early program phases is element interaction analysis. Defining the interactions between major elements of the system through measurable parameters allows us to understand how changes in the performance of one major system effect the performance of another major system or the overall accomplishment of mission objectives.

Cost effectiveness analysis is performed to aid in making judgements on the relative merits of competing concepts. Currently, cost effectiveness is limited to including development, production, launch, and replacement costs for comparable life cycle assessments. As our understanding of overhead factors such as EVA, IVA, and Earth support man-hour rates mature the relative costs of differences associated with these variables can

be included. Current assessments include these parameters as part of the supportability analysis.

3 Methodology

The components of system effectiveness are shown in figure 3-1. The system utility is defined uniquely for each element, and consists of typical hardware parameters such as mass, volume, power, operational life and performance degradation rate. Supportability is defined in terms of reliability, maintainability, deployability, operability, and resource consumption. Maintainability, deployability, and operability are expressed in terms of EVA, IVA, and Earth support man-hours, equipment necessary, and a qualitative complexity factor associated with the crew skill or training level or the monitoring and control automation necessary to perform the activity. This accounts for cognitive as well as physical attributes associated with function performance.

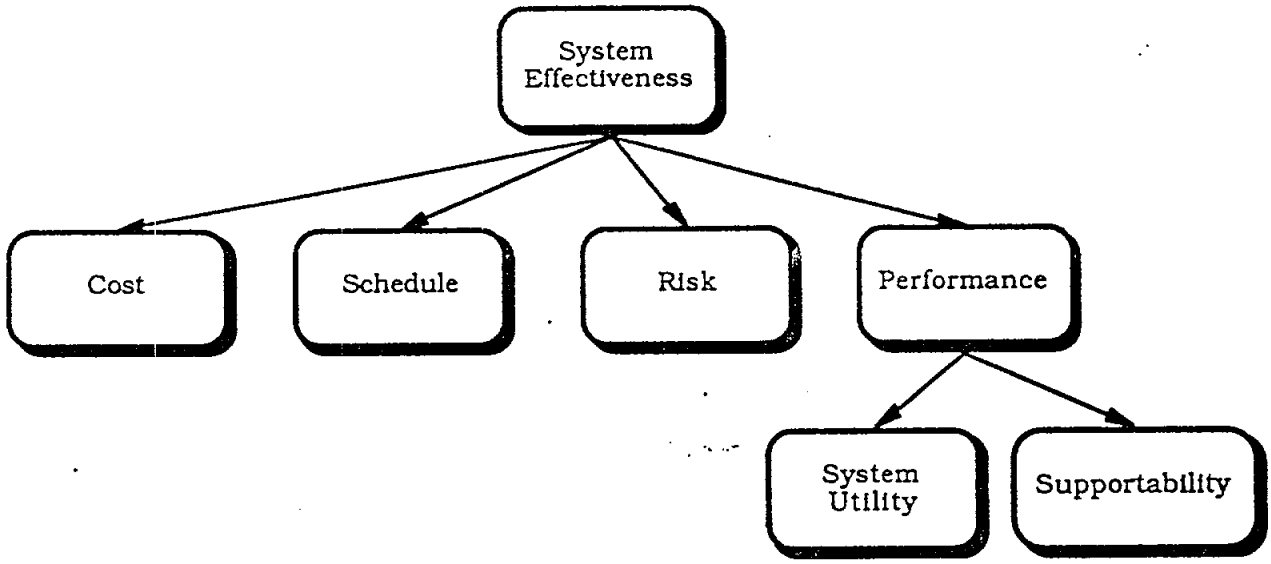

Figure 3-1: System Effectiveness Structure

Models provide a means of evaluating alternate system configurations with respect to effectiveness categories. A multi-attribute criteria structure similar to figure 3-2 is prepared for each major element of FLO. Preliminary criteria lists for each element are provided in appendix A. In the System Utility category, the parameters may also be identified as Technical Performance Measurements (TPM). The criteria structure may be carried to the level of detail necessary to provide visibility into the actual subsystem design parameters which comprise the top-level measurables. Each element can be influenced by the performance of other elements. These interactions are key to the usefulness of effectiveness analysis at the system level. These interactions are represented through algorithms linking individual element modules.

The model under development is based on linked spreadsheet objects. A spreadsheet database is developed for each element (or major subsystem) containing values associated with alternative design concepts. Parametric interdependencies are then described through algorithms linking

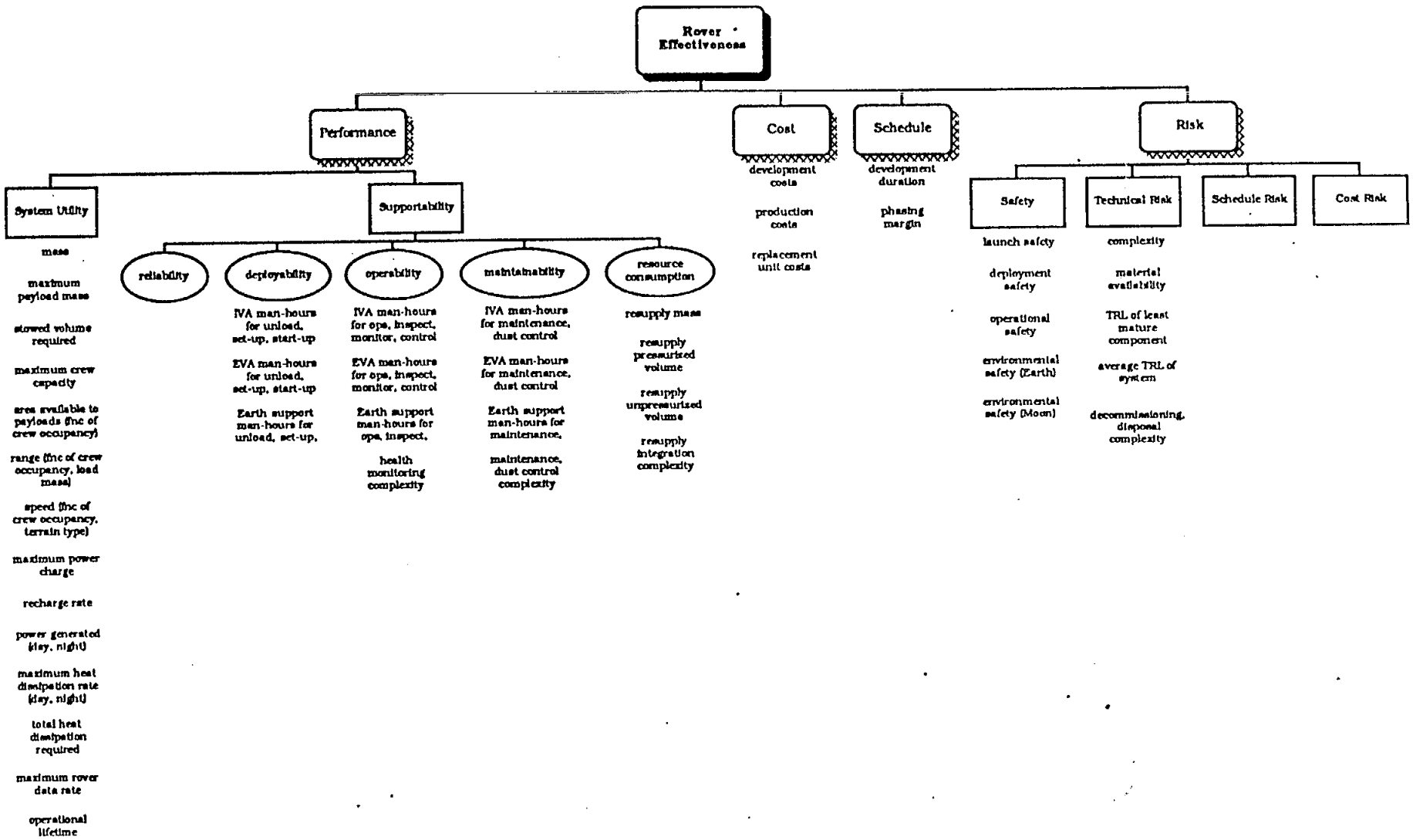

figure 3-2: Multi-Attribute Model for Rover Effectiveness

parameter values in one module to characteristics within another module. Figure 3-3 illustrates this environment by showing the links between modules (e.g., launch vehicle, habitat, rover power) required to support aggregation modules such as the power option effectiveness or mission effectiveness.

3.1 Criteria Selection

In order to provide a logical basis for making judgements on the relative merits of competing concepts, it is necessary to first establish appropriate criteria. The method to choose criteria began with research into work done as part of synthesis group studies, the Augustine report, assessing a questionnaire concerning evaluation criteria from the NRCSpace Studies Task Group on Priorities in Space Research, and examining DoD MOEs to prepare a strawman set of criteria. This example criteria list was then sent out for comments and a request for additional input so that an exhaustive list of criteria could be generated. A 3-stage Delphi approach was then used to examine if a consensus existed within the community on important criteria. The subsequent criteria structure was sent to each functional area lead within ExPO for review. A preliminary list was created for each FLO element based on comments received. One-on-one discussions were then held with each FLO design team lead for comments on the criteria relating to his element. All comments were incorporated to produce the initial criteria set. It is expected that the criteria will evolve as the program and our understanding matures.

3.2 Data Collection

After criteria selection, suitable data associated with measurable parameters must be collected. Early in the design process most data are based on design engineering judgement. Even data with low confidence factors are valuable for performing parametric analyses to provide information on prioritizing future studies. Other avenues of data collection include researching analogous systems, discussions with technology developers, and drawing on specialty engineers to extract data from past and current programs for any relevance they may provide.

As each element team produces preliminary concept designs, engineering estimates and technology choices are made for major subsystems. This data is incorporated into effectiveness analysis as it becomes available. The preliminary designs may have some heritage from previous systems and therefore historical data may be applicable. Information associated with technology choices is obtained from the technology development organizations.

Specialty engineering assessments are performed as concepts are defined to address the issue of equipment/system performance supportability. For example, reliability engineers analyze the concept designs and apply analytical methods and historical statistical data to develop an estimate of the probability for successful system performance (or failure). The analysis identifies the strengths and weaknesses of the design so that improvements can be recommended. Results from this analysis are then be combined with

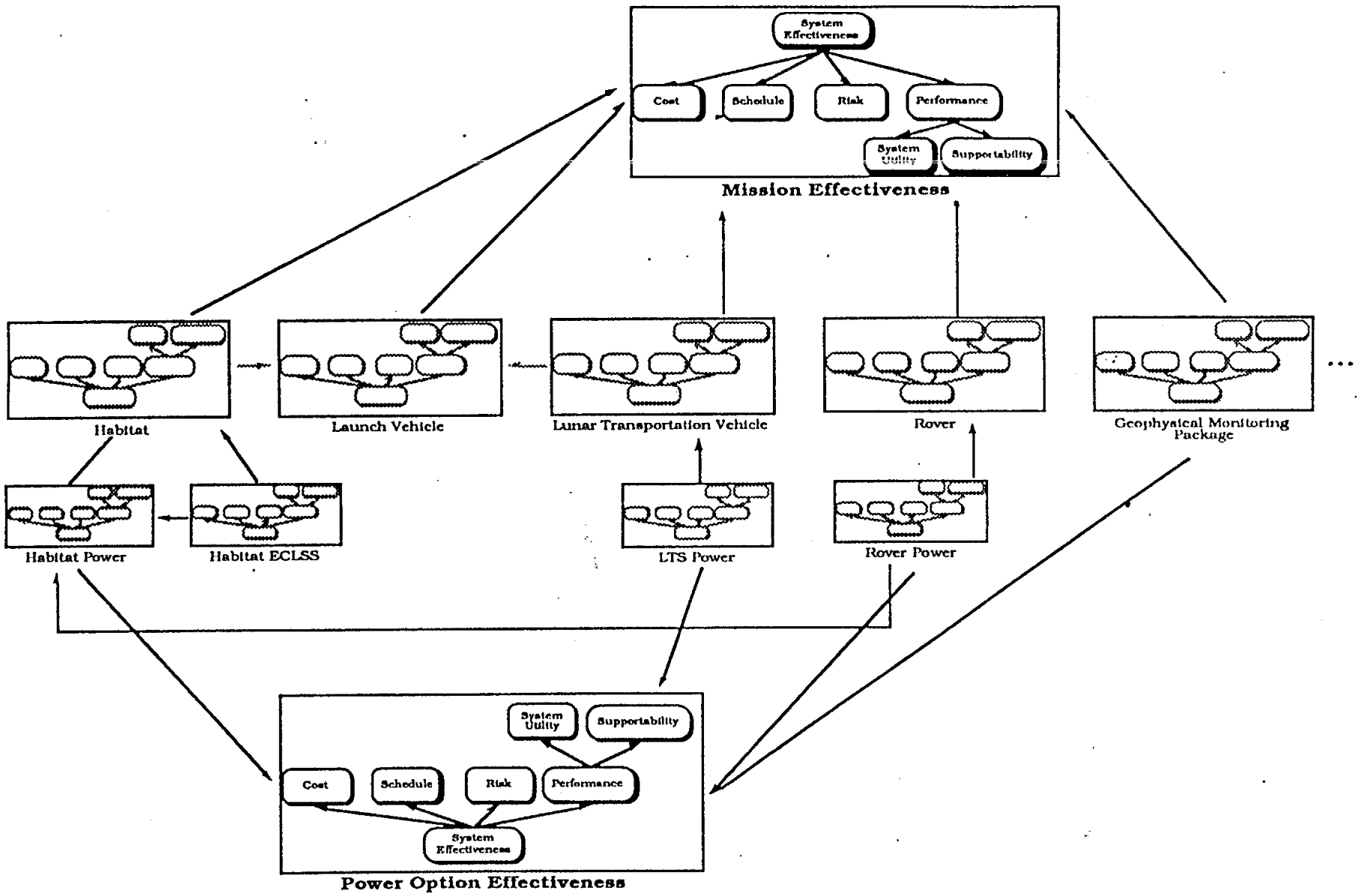

figure 3-3: Effectiveness Model Environment

maintainability assessments from maintainability engineers to provide a synthesized performance supportability assessment.

4 Power Effectiveness Assessment

An assessment of power implementation scenarios was chosen as an illustrative example of effectiveness analysis for the Consolidated Engineering Review. This analysis consisted of describing the power subsystem concepts chosen for each baseline element concept. This data was integrated to form an overall description of the power each scenario's performance, cost, schedule, and associated risk. Example scenarios were prepared to provide relative comparisons of overall system effectiveness. Table 4-1 describes the power implementations assumed for each element for the baseline and scenario options. The baseline implementations are the power choices selected for the baseline design concepts by each design team. The alternate scenarios were chosen through discussions with the FLO System Engineering lead. The particular implementation alternatives for various elements were made through discussions of applicable power technologies with the FLO Power Integration Team.

The criteria used in this assessment was developed through discussions with the FLO Power Integration Team and personnel from DOE as well as deriving from previous research and survey activities as discussed in section 3-1. The multi-attribute structure shown in figure 4-1 was used in this preliminary example. Maturation of this criteria is expected through further discussions with the power technology and design engineering communities.

To assess the overall effectiveness of power alternatives from the system perspective, characteristics from the power subsystems of each FLO element were collected and integrated together. This allowed interactions and synergies among elements to be accounted for.

4.1 Data

Since the purpose of this assessment was as an illustrative example, power subsystem characteristics of each FLO element came from a variety of sources. Table 4-2 is an example of the power system characteristics data sheets created for the FLO elements. For a complete catalog of the data and associated references for each element see Appendix B. The system utility performance data were taken from concept descriptions provided by the design teams (primarily mass, power generation, power consumption, energy storage) and the FLO power integration team (performance degradation and operating life). Cost data was provided primarily by the FLO cost analysts, but additional nuclear costing information such as algorithms for dynamic isotope power system (DIPS) and radioisotope thermoelectric generators (RTGs) were provided by the power integration team. The baseline cost estimates are compatible with the costs presented by the design teams at the Consolidated Engineering Review.

The schedule and risk data for each element implementation is based primarily on the selected technology. This information was generated through discussions with technology developers, the power integration

Power User	Baseline	Option 1 distributed surface science power	Option 2 no nuclear power	Option 3 all DIPS for surface elements
Cargo LTS	EPDC only, uses habitat power in transit	EPDC only, uses habitat power in transit	Primary Fuel Cells	Primary Fuel Cells
Piloted LTS	Ag-Zn Batteries & Alkaline Fuel Cells	Ag-Zn Batteries & Alkaline Fuel Cells	Ag-Zn Batteries & Alkaline Fuel Cells	Ag-Zn Batteries & Alkaline Fuel Cells
Habitat/Airlock	PV/RFC (Si, high press. reactants)	PV/RFC (Si, high press. reactants)	PV/RFC (Si, cryo stored reactants)	DIPS
Unpressurized Rover	RFCs	NaS Battery	RFCs	DIPS
EMU/PLSS	Batteries (no data)	Batteries (no data)	Batteries (no data)	Batteries (no data)
ISRU Demonstration Package	uses Habitat power system	uses Habitat power system	uses Habitat power system	uses Habitat power system
Geophysical Monitoring Package	RTG	RTG	uses science utility power station	uses science utility power station
Solar System Physics Experiment Package	uses science utility power station	RTG	uses science utility power station	uses science utility power station
Lunar Transit Telescope	uses science utility power station	RTG	uses science utility power station	uses science utility power station
Small Research Telescope	uses science utility power station	DIPS	uses science utility power station	uses science utility power station
Small Solar Telescope	uses science utility power station	RTG	uses science utility power station	uses science utility power station
Traverse Geophysical Package	uses unpressurized rover power system	uses unpressurized rover power system	uses unpressurized rover power system	uses unpressurized rover power system
Lunar Geologic Tool Set	uses unpressurized rover power system	uses unpressurized rover power system	uses unpressurized rover power system	uses unpressurized rover power system
Robotic Science Rover	DIPS	RTG	RFC	DIPS
Science Utility Power Station	DIPS	not used in this option	PV/RFC (Si, cryo stored reactants)	DIPS

Table 4 Power Implementation Scenarios

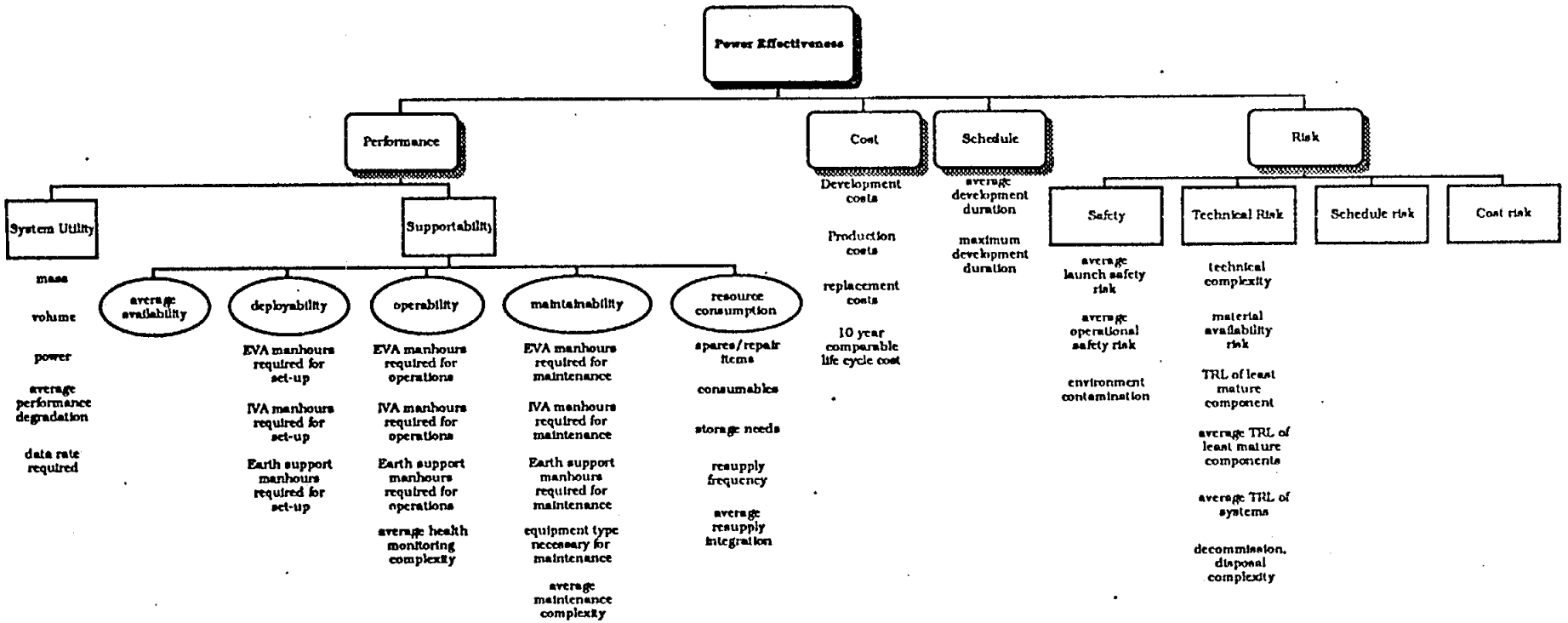

figure 4-2: Multi-Attribute Model for Power Effectiveness

Habitat/Airlock

Power System Data Sheet

Power System Type:		PV/RFC (St, high press. reactants)	PV/RFC (St, cryo stored reactants)	DIPS	
Performance					
System Utility					
mass	kg	5219	3000	1640	includes all necessary support structure burden
volume	cu. m	52	26	11	
day power	kW	10.5	10	10	4 units of 2.5kW DIPS provide 10kW continuous power
night power	kW	9.1	9.1	10	
quiescent power	kW	2	2	2.5	actual quiescent power requirement unknown
performance degradation	%/yr	2	2	5	dust accumulation, thermal cycling, radiation, isotope decay
operating life	yrs	3	3	10	
data rate required	Mbps				no data available
energy	kW-hr	3186	3186	continuous	need calculated from power load for night
Supportability					
availability factor	%	95	95	97	% time meeting load
deployability					deployed prior to crew arrival
EVA manhours required for set-up	hr	0.0	0.0	0.0	
IVA manhours required for set-up	hr	0.0	0.0	0.0	
Earth support manhours required for set-up	hr	12.0	12.0	2.0	
equipment necessary for set-up		none	none	off-loading	no data available
operability					
EVA manhours required for operations	hr/day	0.0	0.0	0.0	
IVA manhours required for operations	hr/day	0.2	0.2	0.3	
Earth support manhours required for operations	hr/day	0.2	0.2	0.3	quiescent period monitoring & control
health monitoring complexity		low/mod	mod/high	moderate	subjective value: fnc(# sensors, monitoring criticality)
maintainability					
EVA manhours required for maintenance	hr/day	0.2	0.2	0.1	
IVA manhours required for maintenance	hr/day	0.3	0.4	0.1	
Earth support manhours required for maintenance	hr/day	0.3	0.4	0.1	
equipment type necessary for maintenance		supporting	supporting	supporting	
maintenance complexity		high	high	low	subjective value: fnc(# items, item type)
resource consumption					
spares/repair items	kg	320.0	320.0	0.0	
consumables	kg	0.0	0.0	0.0	
storage needs	cu. m				no data available
resupply frequency	#/yr	0.3	0.3	0.1	
resupply integration complexity		moderate	low	moderate	subjective value: fnc(item type)
Cost					
Development costs	$M	86	101	120	
Production costs	$M	93	106	365	includes plutonium cost for first DIPS units
replacement unit cost	$M	65	78	20	uses original plutonium

Table 4-[illegible] Example Element Power Data Sheet

Habitat/Airlock

Power System Data Sheet

Power System Type:		PV/RFC (Si, high press. reactants)	PV/RFC (Si, cryo stored reactants)	DHPS	
Schedule					
development duration	yrs	5.5	6	5	time from ATP to available for use
schedule margin	yrs	-0.5	-1	0	time from available for use to need date (assumes '94 start)
Risk					
Safety					
launch safety		low	low	mod/high	high >> high risk relating to launch safety concerns
operational safety		moderate	moderate	low/mod	
environment contamination		low	low	moderate	
Technical Risk					
complexity		moderate	mod/high	moderate	
material availability		none	none	high	
TRL of least mature component		4	3	4	
average system TRL		4.5	4	5	
decommissioning/disposal complexity		none	none	low/mod	
Schedule risk		low/mod	moderate	mod/high	
Cost risk		low/mod	moderate	high	
		limited to <80° latitude for reasonable masses	limited to <80° latitude for reasonable masses	global	

Evolution capability

not currently implemented

Table 4-2: Example Element Power Data Sheet (cont.)

team, and reference to *Conceptual Design of Power Systems*, JSC-24101 December 1989. The supportability performance data is the least mature of all data used in this example. This data was generated through a general understanding of the concept designs and assumptions of operational scenarios associated with them.

4.2 Assumptions

Several assumptions associated with costing and operations philosophies were necessary for the development of meaningful data. Assumptions associated with development cost credits for elements utilizing the same technology are established. A philosophy for element replacement is developed for preliminary equivalent life cycle cost comparisons. Operational philosophies associated with EVA monitoring and maintenance estimates are also presented. The next few paragraphs discuss the specific assumptions.

Development costs for technology used in more than one element in a scenario are costed only once. For ease in bookkeeping this data, I did not divide the cost among the user elements, rather I bookkept the development cost with the element having the greatest power requirement. For example, in the baseline scenario both the robotic science rover and science utility station utilize DIPS. The development cost for DIPS in the baseline scenario is therefore bookkept with the science utility station.

Costing data has not yet been prepared by the cost analysts for PV arrays with cryogenic regenerative fuel cells (RFCs). I have assumed a 20% cost increase over a similar system providing the same power output using gaseous RFCs. The 20% increase is applied to both development and production costs. This increase was developed by discussing the relative complexities of the two RFC technologies with technology developers. Additional work is required to develop more accurate cost data for the PV/cryogenic RFC systems.

Due to the operational lifetime differences between implementation options of surface power equipment, a replacement philosophy was developed so that meaningful comparisons between scenarios can be made. Since the nuclear-based systems have a lifetime between 10 and 12 years, I have chosen to evaluate the replacement needs of other systems for comparable lifetimes. In addition to regular maintenance, the electrolyzer stack and fuel cells must be replaced periodically for RFC systems. The current lifetime estimate for RFCs is 3 to 4 years. In this analysis, I have assumed replacement every 3 years. It is recommended that sensitivity analysis be performed with respect to RFC lifetimes. Although this assessment did not call for the replacement of nuclear devices due to the life cycle comparison of 10 years, the philosophy used to develop replacement unit costs for DIPS and RTGs included plutonium reuse.

Some basic operational philosophies were needed to assess supportability parameters. Several options exist for providing support to EVA activities. These choices range from no IVA or Earth support for any activity to continuous IVA and Earth support for all activities. Although not all EVA

activities require continuous monitoring, there are certain activities that warrant IVA and/or Earth support monitoring. I have assumed that all maintenance based EVA activities will require some IVA/Earth support monitoring. The extent and distribution of responsibility should be specified by the operations team. For this example, I have assumed that both IVA and Earth support monitoring will occur simultaneously for maintenance EVA tasks. Although this is an overly conservative philosophy, this can be used to validate the IVA monitoring capabilities and aid in the evolution towards self-sufficient outposts. This has not been coordinated with the FLO operations team.

The development of EVA maintenance time estimates is based on preliminary data obtained from a database of hardware failure rates and repair times corresponding to analogous elements combined with additional estimates of time necessary for dust control activities. I have assumed one 8-hour dust control activity for the habitat PV/RFC power system types during a 45-day mission period. All stationary DIPS systems are assumed to require one 4-hour dust control activity during a 45-day mission period (all DIPS are identical modules with easy access to radiator panels). Rover assumptions include one 15 minute dust control activity every third traverse for RFC and DIPS systems. I have also assumed that EVA maintenance associated with RTG systems is minimal (none during a 45 day mission) due to human proximity limits expected with these devices.

4.3 Analysis and Results

This power implementation assessment is intended as an illustrative example for effectiveness analysis. Therefore, the data utilized in this example does not have sufficient fidelity to warrant programmatic decisions. However, some basic judgements on priorities for future areas of study can be made. Table 4-3 presents the resulting comparison for the total system implementation scenarios shown in table 4-1.. As expected, utilizing only nuclear technologies for surface power subsytems results in a total launch mass of less than 60% and stowed volume of less than 50% of that necessary for the baseline implementation. Using the more advanced technology of cryogenic RFCs instead of gaseous RFCs results in a total launch mass of just over 75% and stowed volume of less than 70% of that necessary for the baseline implementation. Further definition of the PV/cryogenic RFC system must be done to develop realistics cost, schedule, and risk estimates. An assessment comparing the two PV/RFC systems must be done for a realistic assessment of the impact of utilizing the advanced technology.

In generating the average power data, it became apparent that the PV/RFC systems sized for the habitat in this example are not compatible with continuous occupancy while the DIPS system is. For a realistic comparison the array and structure for the PV/gaseous RFC system must be sized for ~17.5 kWe due to the need to fully recharge RFCs during occupied periods while the array and structure for the PV/cryogenic RFC system must be sized for ~23 kWe (due to additional load for reliquifaction). Additional analysis describing these comparable systems must be performed. This will increase the mass and cost estimates associated with scenarios that utilize PV/RFC systems.

Power Option Summary Sheet

Implementation Scenario	units	Baseline	distributed surface science power Option 1	no nuclear power Option 2	all DIPS for surface elements Option 3	Remarks
Performance						
System Utility						
mass of all power subsystems	mT	8.0	8.0	6.2	4.7	intended to include necessary support structure burden
mass of surface power subsystems	mT	5.9	5.9	4.0	2.5	intended to include necessary support structure burden
volume	cu. m	73.7	71.8	50.6	34.8	
power	kW	17	17	18	19	
average performance degradation	%/yr	4.3	3.7	4.0	5.0	dust accumulation, thermal cycling, radiation, isotope decay
data rate required	Mbps					data unavailable
Supportability						
average availability factor	%	96	95	95	96	% time meeting load
deployability						habitat power deployed prior to crew arrival by Earth support
EVA manhours required for set-up	hr	0	0	0.5	0	
IVA manhours required for set-up	hr	1	0	2.5	2	
Earth support manhours required for set-up	hr	13.2	12	14.9	4.4	
operability						
EVA manhours required for operations	hr/day	0.17	0.17	0.17	0	
IVA manhours required for operations	hr/day	0.17	0.17	0.25	0.33	
Earth support manhours required for operations	hr/day	0.23	0.23	0.35	0.43	
average health monitoring complexity		low/mod	low/mod	moderate	low/mod	subjective value: fnc(# sensors, monitoring criticality)
maintainability						
EVA manhours required for maintenance	hr/mission	23	16	28	16	
IVA manhours required for maintenance	hr/mission	30	22	44	17	
Earth support manhours required for maintenance	hr/mission	28	21	42	16	
equipment type necessary for maintenance						
average maintenance complexity		low/mod	low	low/mod	low	subjective value: fnc(# items, item type)
resource consumption						
spares/repair items	kg/yr	123	149	160	0	
consumables	kg/yr	0	0	0	0	data incomplete
storage needs	cu. m	0	0	0	0	data incomplete
resupply frequency	#/yr	0.3	0.3	0.3	0.1	
average resupply integration complexity		low/mod	moderate	low	low/mod	subjective value: fnc(item type)
Cost						
Development costs	$M	511	483	446	385	includes development cost for DIPS as applicable
Production costs	$M	273	453	252	651	includes plutonium cost for DIPS
replacement costs	$M	148	134	213	0	cost of replacements for 10 year comparable life
total	$M	932	1070	912	1036	launch costs not included
Schedule						
average development duration	yrs	4	3	4	4	time from ATP to available for use
maximum development duration	yrs	6	6	6	5	

Table 4-3: Power Option Summary

Power Option Summary Sheet

Implementation Scenario	units	Baseline	Option 1	Option 2	Option 3	Remarks
			distributed surface science power	no nuclear power	all DIPS for surface elements	
Risk						
Safety						
average launch safety risk		moderate	mod/high	low	moderate	high >> high risk relating to launch safety concerns
average operational safety risk		low/mod	low/mod	low/mod	low/mod	
environment contamination		low/mod	low	low	low/mod	
Technical Risk						
technical complexity		low/mod	low/mod	low/mod	low/mod	
material availability risk		low/mod	moderate	none	moderate	
TRL of least mature component		4	3	3	4	
average TRL of least mature components		5.6	6.7	4.7	5.0	
average TRL of systems		6.3	6.9	5.6	5.8	
decommissioning/disposal complexity		low	low	none	low/mod	
Schedule risk		low/mod	low/mod	low/mod	moderate	
Cost risk		low/mod	low/mod	low/mod	mod/high	

Evolution capability

currently not implemented

Table 4-3: Power Option Summary (cont.)

Although the supportability data presented in this example is internally consistent, as previously discussed it is not of the maturity necessary for programmatic decisions to be made. To perform reasonable supportability assessments, each element must be defined to the subsystem level. Maintenance engineering is necessary to determine the real maintainability of differing implementations. A reliability analysis is also needed for meaningful availability factor descriptions. Currently, the resource consumption category includes only replacement items for the various systems. A logistics analysis must be included in design considerations so that accurate assessments of spares and repair items can be considered as a necessary discriminator between options.

The development costs for the option utilizing only DIPS for the surface power systems appears significantly lower than other options. This is due primarily to the philosophy of developing 2.5 kWe modules that can be integrated together and used for a variety of applications. This enables a greater amount of design flexibility and inherent application redundancy. Additional study of the development schedules and risks associated with nuclear power systems must be done to update the data currently used in this example before recommendations can be made.

Since the major criteria categories are not independent I have used the risk category to describe the risk associated with particular performance, cost, and schedule choices. The basic premise associated with this current work, is the desire to balance these criteria early in program development. Therefore, changes in any of the other major criteria will also change the associated risk. Ideally, the risk parameters are defined in parallel with the program risk management plan. This plan does not currently exist, therefore an initial set of parameters and associated estimates have been developed. The safety risk data was derived from the engineering judgement of technology developers. For optimum results, an unbiased assessment must be provided by SR&QA. The schedule and cost risks are primarily the uncertainties involved in meeting performance goals within the given schedule and cost.

For meaningful judgements on the relative effectiveness of differing power implementation scenario options, other viable scenarios must be assessed. Sensitivity analysis must be performed to determine which characteristics and elements warrant additional study for increased confidence in results.

5 Summary and Recommendations

A methodology for system effectiveness has been developed along with preliminary criteria for each FLO element. In addition, a working prototype effectiveness module has been created. Obviously, more analysis and development is necessary to provide the scope and level of detail needed to perform system effectiveness assessments for FLO, but this work demonstrates the viability of the presented concept.

The supportability parameter values and relationships require the development of an operations philosophy. I recommend that this

philosophy be developed and described to FLO personnel early in this next phase of analysis.

We are all familiar with the adage "Garbage in equals Garbage out." This is especially true for early analysis of this type. A balance must be maintained between the level of detail available and that desired for analysis. The level of detail available for effectiveness analysis will evolve as design concepts mature, but prudent use of sensitivity analysis permits meaningful analysis using soft data. Thorough documentation of assumptions and references associated with any assessment is of primary importance. As data matures, it will be necessary to continue this documentation so that consistent and comparable analyses can be performed.

Future database development is essential to effective and consistent manipulation of data for analysis. I recommend that database development continue for element parameter descriptions and characteristics. An initial database has been prepared for the power subsystem data used in the power effectiveness example (see appendix B). The database(s) must include data from technology developers, cost analysts, design engineers, and operations engineers. As a result it may be necessary to develop various databases for particular purposes. Work is underway to create a maintenance database for FLO elements based on SSF maintenance database MRUs. To effectively utilize this data some MRUs must be modified, others must be developed, and overhead factors must be modified for lunar surface conditions.

I recommend continuation of module development for each FLO element including developing algorithms describing parametric interactions between elements. Defining and performing alternative concept analysis, similar to the power option assessment, for each major element is the next logical step.

As a continuation of the power assessment, I recommend reviewing the criteria and fidelity of the data with the community. In particular, integration with the risk, SR&QA, and operations teams is necessary for realistic analyses. Sensitivity analysis must also be performed on selected parameters to aid in the prioritization of future power studies. Studies associated with the resulting high priority areas can then be scheduled. Possible evolution strategies must also be identified and analyzed to determine the impact of various growth paths on the effectiveness of FLO power scenarios.

Acronyms

DIPS	dynamic isotope power system
DoD	Department of Defense
DOE	Department of Energy
EVA	Extravehicular Activity
ExPO	Exploration Programs Office
FLO	First Lunar Outpost
IVA	Intravehicular Activity
MOEs	Measures of Effectiveness
MRUs	maintenance replacement units
NRC	National Research Council
PV	photovoltaic
RFCs	regenerative fuel cells
RTGs	radioisotope thermoelectric generators
SR&QA	Safety, Reliability, and Quality Assurance
TPM	Technical Performance Measurements

Appendix A: Preliminary Element Criteria

Launch System A-1

Lunar Transportation System A-2

- Crew Module A-2
- Lander A-4
- Launch Escape System A-5
- Return Stage A-6

Habitat/Airlock A-7

Unpressurized Rover A-9

EMU/PLSS A-11

System Effectiveness

Preliminary Measurable Parameters

Launch System (including TLI)

- **Performance**
  - *System Utility*
    - payload mass capability to TLI
    - payload volume available
    - data transmission rate
  - *Supportability*
    - reliability
    - operability
      - manpower required for integration
      - manpower required for launch processing
      - manpower required for launch operations
      - time required in assembly building
      - time required for transport to pad
      - time required on the pad prior to launch
      - time required for pad turnaround
      - launch system health monitoring complexity
      - launch vehicle health monitoring complexity
- **Cost**
  - development costs
  - production costs
  - replacement unit costs
  - facilities dev. costs (test, integ., encaps., pad trans., pad)
  - facilities tooling costs
- **Schedule**
  - average development duration
  - schedule margin
  - environmental impact statement duration
- **Risk**
  - Safety
    - testing safety
    - integration safety
    - launch operations safety
    - operational (flight) safety
    - environmental safety (Earth)
  - Technical Risk
    - complexity
    - TRL of least mature component
    - average TRL of system
    - decommissioning/disposal complexity
  - Schedule Risk
  - Cost Risk

Lunar Transportation System

Crew Module

Performance

System Utility

- mass
- launch volume
- habitability
  - habitable volume
  - *to be defined by Man-Systems Division*
- storage volume (pressurized, unpressurized)
- energy storage
- power required (pad, flight, surface)
- data transmission capability
- data transmission rate required (pad, flight, surface)
- heat dissipation rate capability (flight, surface)
- heat dissipation rate required (pad, flight, surface)
- internal pressure
- radiation protection
- depress/repress capability
- life support duration capability

Supportability

- availability
  - operational lifetime
  - reliability
- operability
  - IVA man-hours for inspection/monitoring/control
  - EVA man-hours for inspection/monitoring/control
  - Earth supportman-hours for inspection/monitoring/control
  - time required for depress, repress
  - health monitoring complexity
- maintainability
  - IVA man-hours for maintenance/hygiene/dust control
  - EVA man-hours for maintenance/hygiene/dust control
  - Earth support man-hours for maintenance/hygiene/dust control
  - maintenance/hygiene/dust control complexity

Cost

- development costs
- production costs
- replacement unit costs
- recovery/refurbishment unit costs
- equivalent life cycle cost

Schedule

- average development duration
- schedule margin

Risk

- Safety
  - launch safety
  - deployment safety
  - operational safety
  - environmental safety (Earth, Moon)
- Technical Risk
  - complexity
  - material availability
  - TRL of least mature component
  - average TRL of least mature components
  - decommissioning/disposal complexity
- Schedule Risk
- Cost Risk

Lander

- **Performance**
  - *System Utility*
    - mass
    - payload mass capability
    - launch volume
    - payload volume capability
    - energy storage
    - power required (pad, flight, surface)
    - data transmission capability
    - data transmission rate required (pad, flight, surface)
    - heat dissipation rate capability (flight, surface)
    - heat dissipation rate required (pad, flight, surface)
    - landing accuracy
  - *Supportability*
    - reliability
    - operability
      - IVA man-hours for inspection/monitoring/control
      - EVA man-hours for inspection/monitoring/control
      - Earth support man-hours inspection/monitoring/control
      - health monitoring complexity
    - maintainability
      - IVA man-hours for maintenance
      - EVA man-hours for maintenance
      - Earth support man-hours for maintenance
      - maintenance complexity
    - resource consumption
      - resupply mass
      - resupply pressurized volume
      - resupply unpressurized volume
      - resupply integration complexity
- **Cost**
  - development costs
  - production costs
  - replacement unit costs
- **Schedule**
  - average development duration
  - schedule margin
- **Risk**
  - Safety
    - launch safety
    - operational safety
    - environmental safety (Earth, Moon)
  - Technical Risk
    - complexity
    - material availability
    - TRL of least mature component
    - average TRL of system
    - decommissioning/disposal complexity
  - Schedule Risk
  - Cost Risk

Launch Escape System

- **Performance**
  - *System Utility*
    - mass
    - stowed volume
    - power required
    - data transmission rate required
    - crew loading factor (g-load vs time)
  - *Supportability*
    - reliability
    - operability
      - versatility (land, water, safe altitude range)
      - health monitoring complexity
- **Cost**
  - development costs
  - production costs
  - replacement unit costs
- **Schedule**
  - average development duration
  - schedule margin
- **Risk**
  - Safety
    - operational safety
    - environmental safety (Earth)
  - Technical Risk
    - complexity
    - material availability
    - TRL of least mature component
    - average TRL of system
    - disposal complexity
  - Schedule Risk
  - Cost Risk

Return Stage

- **Performance**
  - *System Utility*
    - mass
    - payload mass capability
    - stowed volume
    - power required (flight, surface)
    - data transmission rate required (flight, surface)
    - heat dissipation rate required (flight, surface)
  - *Supportability*
    - availability
      - reliability
      - operational lifetime
    - operability
      - IVA man-hours for inspection/monitoring/control
      - EVA man-hours for inspection/monitoring/control
      - Earth support man-hours inspection/monitoring/control
      - health monitoring complexity
    - maintainability
      - IVA man-hours for maintenance
      - EVA man-hours for maintenance
      - Earth suppor man-hourst for maintenance
      - maintenance complexity
- **Cost**
  - development costs
  - production costs
  - replacement unit costs
- **Schedule**
  - average development duration
  - schedule margin
- **Risk**
  - Safety
    - launch safety
    - operational safety
    - environmental safety (Earth, Moon)
  - Technical Risk
    - complexity
    - material availability
    - TRL of least mature component
    - average TRL of system
    - decommissioning/disposal complexity
  - Schedule Risk
  - Cost Risk

Habitat/Airlock

Performance

System Utility

- mass
- launch volume
- storage volume (pressurized, unpressurized)
- power utility
  - power required (day, night)
  - power generation (day, night)
  - energy storage
- thermal utility
  - heat dissipation required (day, night)
  - heat dissipation capability (day, night)
- habitability
  - *to be defined by Man-Systems Division*
  - habitable volume
  - internal walk space
  - medical capability
- communication utility
  - data transmission capability
  - data transmission rate required (day, night)
- life support duration capability
- internal pressure
- depress/repress capability
- radiation protection
- airlock utility
  - airlock crew capacity (suited, non-suited)
  - airlock internal storage volume
  - airlock maximum power/energy storage
  - airlock depress/repress capability

Supportability

- availability
  - operational lifetime
  - reliability
  - average airlock availability
  - maximum continuous airlock downtime
- deployability
  - EVA man-hours for deployment & checkout
  - IVA man-hours for deployment & checkout
  - Earth support man-hours for deployment & checkout
- operability
  - IVA man-hours for inspection/monitoring/control
  - EVA man-hours for inspection/monitoring/control
  - Earth support man-hours for inspection/monitoring/control
  - time required for habitat depress/repress
  - time required for airlock depress/repress
  - habitat health monitoring complexity
  - airlock health monitoring complexity
  - airlock hyperbaric capability

- maintainability
  - IVA man-hours for maintenance/hygiene/dust control
  - EVA man-hours for maintenance/hygiene/dust control
  - Earth support man-hours for maintenance/hygiene/dust control
  - maintenance/hygiene/dust control complexity
- resource consumption
  - spares/repairs
  - consumables
  - resupply pressurized volume
  - resupply unpressurized volume
  - resupply integration complexity

Cost

- development costs
- production costs
- replacement unit costs
- equivalent life cycle cost

Schedule

- average development duration
- schedule margin

Risk

- Safety
  - launch safety
  - deployment safety
  - operational safety
  - environmental safety (Earth, Moon)
- Technical Risk
  - complexity
  - material availability
  - TRL of least mature component
  - average TRL of system
  - decommissioning/disposal complexity
- Schedule Risk
- Cost Risk

Rover

Performance

- *System Utility*
  - mass
  - maximum payload mass
  - stowed volume required
  - maximum crew capacity
  - area available to payloads (fnc of crew occupancy)
  - range (fnc of crew occupancy, load mass)
  - speed (fnc of crew occupancy, terrain type)
  - power generated (day, night)
  - energy storage
  - power recharge rate
  - maximum heat dissipation rate (day, night)
  - total heat dissipation required
  - data rate
- *Supportability*
  - availability
    - operational lifetime
    - reliability
  - deployability
    - IVA man-hours for unload, set-up, start-up
    - EVA man-hours for unload, set-up, start-up
    - Earth support man-hours for unload, set-up, start-up
  - operability
    - IVA man-hours for ops, inspect, monitor, control
    - EVA man-hours for ops, inspect, monitor, control
    - Earth support man-hours for ops, inspect, monitor, control
    - health monitoring complexity
    - IVA man-hours for power recharge
    - EVA man-hours for power recharge
  - maintainability
    - IVA man-hours for maintenance, dust control
    - EVA man-hours for maintenance, dust control
    - Earth support man-hours for maintenance, dust control
    - maintenance, dust control complexity
  - resource consumption
    - resupply mass
    - resupply pressurized volume
    - resupply unpressurized volume
    - resupply integration complexity

Cost

- development costs
- production costs
- replacement unit costs
- equivalent life cycle cost

Schedule

- development duration
- phasing margin

Risk

- Safety
  - launch safety
  - deployment safety
  - operational safety
  - environmental safety (Earth, Moon)
- Technical Risk
  - complexity
  - material availability
  - TRL of least mature component
  - average TRL of system
  - decommissioning, disposal complexity
- Schedule Risk
- Cost Risk

EMU/PLSS

- **Performance**
  - *System Utility*
    - mass
    - suit pressure
    - flight stowed volume
    - habitat stowed volume
    - data transmission capability
    - life support duration without recharge
    - pre-breathe time vs R factor (calculated value)
  - *Supportability*
    - availability
      - operational lifetime
      - reliability
      - EVA duration capability (noon, night, nominal)
      - EVA frequency capability
      - maximum continuous downtime
    - operability
      - IVA man-hours for donning, doffing
      - IVA man-hours for inspection/monitoring/control
      - EVA man-hours for inspection/monitoring/control
      - Earth support man-hours for inspection/monitoring/control
      - IVA man-hours for battery/PLSS recharge
      - EVA man-hours for PLSS recharge
      - dexterity *(to be defined by Man-Systems)*
      - health monitoring complexity
    - maintainability
      - IVA man-hours for maintenance
      - EVA man-hours for maintenance
      - Earth support man-hours for maintenance
      - maintenance complexity
      - IVA man-hours for hygiene/recharge/dust control
      - EVA man-hours for hygiene/recharge/dust control
      - Earth support man-hours for hygiene/recharge/dust control
      - recharge frequency
      - hygiene/recharge/dust control complexity
    - resource consumption
      - resupply mass
      - resupply pressurized volume
      - resupply unpressurized volume
      - resupply integration complexity
- **Cost**
  - development costs
  - production costs
  - replacement unit costs
- **Schedule**
  - average development duration
  - schedule margin

Risk

- Safety
  - operational safety (flight, surface)
  - environmental safety (Earth, Moon)
- Technical Risk
  - complexity
  - TRL of least mature component
  - average TRL of system
- Schedule Risk
- Cost Risk

Appendix B: Element Power Data Sheets with References

Cargo LTS B-1

Piloted LTS B-6

Habitat/Airlock B-10

Unpressurized Rover B-18

Geophysical Monitoring Package B-26

Solar System Physics Experiment Package B-30

Lunar Transit Telescope B-34

Small Research Telescope B-38

Small Solar Telescope B-42

Lunar Geologic Tool Set B-46

Robotic Science Rover B-50

Science Utility Power Station B-59

Cargo LTS **Power System Data Sheet**

Power System Type:		EPDC only, pass habitat power in transit	Primary Fuel Cells	

Performance

System Utility				
mass	kg	125	220	
volume	cu. m			no data available
power	kW		2	
performance degradation	%/yr	5	5	dust accumulation, thermal cycling, radiation, isotope decay
operating life	hrs	N/A	96	96 hours needed
data rate required	Mbps			no data available
energy	kW-hr		192	

Supportability				
availability factor	%	95	95	% time meeting load
deployability				integrated into vehicle
EVA manhours required for set-up	hr	0	0	
IVA manhours required for set-up	hr	0	0	
Earth support manhours required for set-up	hr	0	0	
equipment necessary for set-up		none	none	
operability				
EVA manhours required for operations	hr/day	0	0	
IVA manhours required for operations	hr/day	0	0	
Earth support manhours required for operations	hr/day	0.02	0.05	monitoring and control
health monitoring complexity		low	low	subjective value: fnc(# sensors, monitoring criticality)
maintainability				
EVA manhours required for maintenance	hr/day	0	0	
IVA manhours required for maintenance	hr/day	0	0	
Earth support manhours required for maintenance	hr/day	0	0	rerouting/switching
equipment type necessary for maintenance		none	none	
maintenance complexity		none	none	subjective value: fnc(# items, item type)
resource consumption				
spares/repair items	kg	0	0	
consumables	kg	0	0	
storage needs	cu. m	0	0	
resupply frequency	#/yr	0	0	
resupply integration complexity		none	none	subjective value: fnc(item type)

Cost

Development costs	$M	3	3	development cost of fuel cells bookkept in piloted vehicle
Production costs	$M	7	49	
replacement unit cost	$M			

Cargo LTS

Power System Data Sheet

Power System Type:		EPDC only, uses habitat power in transit	Primary Fuel Cells	
Schedule				
development duration	yrs	3	3	time from ATP to available for use
schedule margin	yrs	2	2	time from available for use to need date (assumes '94 start)
Risk				
Safety				
launch safety		low	low	high >> high risk relating to launch safety concerns
operational safety		low	low	
environment contamination		none	none	
Technical Risk				
complexity		low	low	
material availability		none	none	
TRL of least mature component		8	7	
average TRL		8	7	
decommissioning/disposal complexity		none	none	
Schedule risk		low	low	
Cost risk		low	low	

Evolution capability

currently not implemented

Note: EPDC type: scaled Apollo components

Note: PFC type: Shuttle derived

Note: EPDC mass reference: Jim Masciarelli, 4/2 "Dry Run" presentation

Note: PFC mass reference: LeRC's FLO Sizing description 3/17/92

Note: PFC power reference: LeRC's FLO Sizing description 3/17/92

Note: EPDC performance degradation reference: Lee Mason guesstimate

Note: PFC performance degradation reference: Lee Mason guesstimate

Note: EPDC operating life reference: Fisher-Price database indicates ~200 yr MTBF for most power distribution elements

Note: PFC operating life reference: Lee Mason guesstimate based on 2 kW load for 192 kW-hrs stored (4 day trip)

Note: PFC energy reference: Lee Mason guesstimate based on 2 kW load for 96 hrs (4 day trip)

Note: EPDC availability factor reference: Lee Mason guesstimate

Note: PFC availability factor reference: Lee Mason guesstimate

Note: EPDC EVA deploy reference: EKS guesstimate based on integrated design

Note: PFC EVA deploy reference: EKS guesstimate based on integrated design

Note: EPDC IVA deploy reference: EKS guesstimate based on integrated design

Note: PFC IVA deploy reference: EKS guesstimate based on integrated design

Note: EPDC ground support deploy reference: EKS guesstimate based on integrated design

Note: PFC ground support deploy reference: EKS guesstimate based on integrated design

Note: EPDC deploy equipment type reference: EKS guesstimate based on integrated design

Note: PFC deploy equipment type reference: EKS guesstimate based on integrated design

Note: EPDC EVA operations reference: Lee Mason/EKS guesstimate based on cargo mission - no proximity to personnel (EVA or IVA)

Note: PFC EVA operations reference: Lee Mason/EKS guesstimate based on cargo mission - no proximity to personnel (EVA or IVA)

Note: EPDC IVA operations reference: Lee Mason/EKS guesstimate based on cargo mission - no proximity to personnel (EVA or IVA)

Note: PFC IVA operations reference: Lee Mason/EKS guesstimate based on cargo mission - no proximity to personnel (EVA or IVA)

Note: EPDC ground support operations reference: EKS WAG (~1 minutes per day of monitor & control)

Note: PFC ground support operations reference: EKS WAG (~3 minutes per day of monitor & control)

Note: EPDC health monitoring complexity reference: EKS WAG (issue: additional complexity associated with using habitat power?)

Note: PFC health monitoring complexity reference: EKS WAG

Note: EPDC EVA maintenance reference: Lee Mason/EKS guesstimate based on cargo mission - no proximity to personnel (EVA or IVA)

Note: PFC EVA maintenance reference: Lee Mason/EKS guesstimate based on cargo mission - no proximity to personnel (EVA or IVA)

Note: EPDC IVA maintenance reference: Lee Mason/EKS guesstimate based on cargo mission - no proximity to personnel (EVA or IVA)

Note: PFC IVA maintenance reference: Lee Mason/EKS guesstimate based on cargo mission - no proximity to personnel (EVA or IVA)

Note: EPDC ground support maintenance reference: EKS WAG

Note: PFC ground support maintenance reference: EKS WAG (design for life)

Note: EPDC maintenance equipment type reference: EKS guesstimate based on assumption that necessary equipment is same as that used for monitoring

Note: PFC maintenance equipment type reference: EKS guesstimate based on assumption that necessary equipment is same as that used for monitoring

Note: EPDC maintenance complexity reference: EKS guesstimate (design for life)

Note: PFC maintenance complexity reference: EKS guesstimate (design for life)

Note: EPDC spare reference: EKS assessment of needs based on operational lifetime of components vs. lifetime needed

Note: PFC spare reference: EKS assessment of needs based on design-for-life philosophy

Note: EPDC consumables reference: EKS estimate, no consumables associated with EPDC

Note: PFC consumables reference: EKS estimate, no consumables due to short lifetime needed

Note: EPDC storage reference: EKS assessment of needs based on operational lifetime of components vs. lifetime needed

Note: PFC storage reference: EKS assessment of needs based on operational lifetime of components vs. lifetime needed

Note: EPDC resupply frequency reference: EKS assessment of needs based on operational lifetime of components vs. lifetime needed

Note: PFC resupply frequency reference: EKS assessment of needs based on operational lifetime of components vs. lifetime needed

Note: EPDC resupply complexity reference: EKS assessment of needs based on operational lifetime of components vs. lifetime needed

Note: PFC resupply complexity reference: EKS assessment of needs based on operational lifetime of components vs. lifetime needed

Note: EPDC development cost reference: Howard Ashley cost estimate 3/24/92

Note: PFC development cost reference: bookkept as part of piloted vehicle

Note: EPDC production cost reference: Howard Ashley cost estimate 3/24/92

Note: PFC production cost reference: Howard Ashley cost estimate 3/24/92 from ascent and descent stage of piloted vehicle

Note: EPDC development schedule reference: EKS WAG based on complexities involved with using habitat power, discussed with Phil Dempsey

Note: PFC development schedule reference: JSC-24101, Table 3-VI

Note: EPDC launch safety reference: EKS estimate based on component types (cables, switches, etc.)

Note: PFC launch safety risk reference: EKS estimate based on JSC-24101 pg 4-3

Note: EPDC operational safety reference: EKS estimate based on component types (cables, switches, etc.)

Note: PFC operational safety risk reference: EKS WAG based on component types

Note: EPDC contamination reference: EKS estimate based on component types (cables, switches, etc.)

Note: PFC contamination reference: EKS WAG based on component types

Note: EPDC technical complexity reference: EKS estimate based on component types (cables, switches, etc.)

Note: PFC technical complexity reference: JSC-24101 pg 4-67

Note: EPDC material availability reference: EKS estimate based on component types (cables, switches, etc.)

Note: PFC material availability reference: EKS guesstimate based on technology understanding

Note: EPDC lowest TRL reference: EKS estimate based on JSC-24101 and component types (cables, switches, etc.)

Note: PFC lowest TRL reference: EKS estimate based on technology discussion in JSC-24101 pg 3-25, 3-29

Note: EPDC disposal complexity reference: EKS estimate based on JSC-24101 and component types (cables, switches, etc.)

Note: PFC disposal complexity reference: JSC-24101 pg 4-67

Note: EPDC schedule risk reference: EKS estimate based on component types (cables, switches, etc.) [we've done these before]

Note: PFC schedule risk reference: EKS estimate based on component type [we've done these before]

Note: EPDC cost risk reference: EKS estimate based on component types (cables, switches, etc.) [we've done these before]

Note: PFC cost risk reference: EKS estimate based on component type [we've done these before]

Piloted LTS

Power System Data Sheet

LTS+Crew Module+Ascent Stage+Descent Stage

Power System Type: Ag-Zn Batteries & Alkaline Fuel Cells

Performance

System Utility			
mass	kg	1975	
volume	cu. m	18	
transit power	kW	2	
quiescent power - day	kW	0.5	
quiescent power - night	kW	2	
performance degradation	%/yr	5	dust accumulation, thermal cycling, radiation
operating life	days	83	53 days needed
data rate required	Mbps		no data available
energy	kW-hr	1639	need calculated from power load as a function of time

Supportability			
availability factor	%	95	% time meeting load
deployability			integrated into vehicle design
EVA manhours required for set-up	hr	0.0	
IVA manhours required for set-up	hr	0.0	
Earth support manhours required for set-up	hr	0.0	
equipment necessary for set-up		none	
operability			
EVA manhours required for operations	hr/day	0.0	
IVA manhours required for operations	hr/day	0.0	
Earth support manhours required for operations	hr/day	0.1	incidental monitoring & control
health monitoring complexity		low	subjective value: fnc(# sensors, monitoring criticality)
maintainability			
EVA manhours required for maintenance	hr/day	0.0	
IVA manhours required for maintenance	hr/day	0.0	
Earth support manhours required for maintenance	hr/day	0.0	
equipment type necessary for maintenance		simple	
maintenance complexity		low	subjective value: fnc(#items, item type)
resource consumption			
spares/repair items	kg	0.0	
consumables	kg	0.0	
storage needs	cu. m	0.0	
resupply frequency	#/yr	0.0	
resupply integration complexity		none	subjective value: fnc(item type)

Cost

Development costs	$M	262	
Production costs	$M	69	
replacement unit cost	$M		

Piloted LTS

LTS+Crew Module+Ascent Stage+Descent Stage

Power System Data Sheet

Power System Type: Ag-Zn Batteries & Alkaline Fuel Cells

Schedule

development duration	yrs	3	time from ATP to available for use
schedule margin	yrs	2	time from available for use to need date (assumes '94 start)

Risk

Safety		
launch safety	low	high >> high risk relating to launch safety concerns
operational safety	low	
environment contamination	none	
Technical Risk		
complexity	low	
material availability	none	
TRL of least mature component	7	
average TRL	7.5	
decommissioning/disposal complexity	none	
Schedule risk	low	
Cost risk	low	

Evolution capability

currently not implemented

Note: launch escape system: nothing
crew module: Ag-Zn batteries
ascent stage: Fuel Cells
descent stage: EPDC

Note: PFC type: Shuttle derived PFC

Note: battery/PFC mass reference: Jim Masciarelli, 4/2 "Dry Run" presentation
[not consistent with costing mass estimate of 0+488+1425+117=2030 kg]

Note: battery/PFC volume reference: Bob Cataldo estimate 4/28/92 based on habitat system analysis

Note: battery/PFC transit power reference: Lee Mason/Bob Cataldo power integration estimates 2/26/92

Note: battery/PFC quiescent day power reference: Lee Mason/Bob Cataldo power integration estimates 2/26/92

Note: battery/PFC quiescent night power reference: Lee Mason/Bob Cataldo power integration estimates 2/26/92

Note: battery/PFC performance degradation reference: Lee Mason guesstimate

Note: battery/PFC operating life reference: JSC-24101 pgs 3-24,25 & 3-40

Note: battery/PFC energy reference: Lee Mason calculation of (transit power*transit time) + (quiescent day power*surface day time) + (quiescent night power*surface night time)

Note: battery/PFC availability factor reference: Lee Mason guesstimate

Note: battery/PFC EVA deploy reference: Lee Mason/EKS guesstimate based on integrated design

Note: battery/PFC IVA deploy reference: Lee Mason/EKS guesstimate based on integrated design

Note: battery/PFC ground support deploy reference: Lee Mason/EKS guesstimate based on integrated design

Note: battery/PFC deploy equipment type reference: Lee Mason/EKS guesstimate based on integrated design

Note: battery/PFC EVA operations reference: Lee Mason/EKS guesstimate (all operations performed from crew module or ground control)

Note: battery/PFC IVA operations reference: Lee Mason/EKS WAG (no operations necessary from crew module)

Note: battery/PFC Earth support operations reference: EKS WAG (same as PFC cargo vehicle - primarily monitor & control)

Note: battery/PFC health monitoring reference: EKS assessment based on JSC-24101 pg3-24,25 and 3-32

Note: battery/PFC EVA maintenance reference: EKS WAG necessary for return transit (1 hr/mission)

Note: battery/PFC IVA maintenance reference: EKS WAG necessary for return transit (2 hr/mission)

Note: battery/PFC Earth support maintenance reference: EKS WAG necessary for return transit (1/2 hr/mission)

Note: battery/PFC maintenance equipment type reference: EKS guesstimate based on component types

Note: battery/PFC maintenance complexity reference: EKS guesstimate based on component types

Note: battery/PFC consumables reference: EKS guesstimate based on mission time & maintainability

Note: battery/PFC resupply frequency reference: EKS guesstimate based on returning vehicle

Note: battery/PFC resupply complexity reference: EKS guesstimate based on returning vehicle

Note: battery/PFC development cost reference: Howard Ashley 3/25/92 includes batteries, PFC, EPDC

Note: batter/PFC production cost reference: Howard Ashley 3/25/92 includes batteries, PFC, EPDC

Note: battery/PFC development schedule reference: JSC-24101, Table 3-VI

Note: battery/PFC launch safety risk reference: EKS estimate based on JSC-24101 pg 4-3

Note: battery/PFC operational safety risk reference: EKS WAG based on component types

Note: battery/PFC contamination reference: EKS WAG based on component types

Note: battery/PFC technical complexity reference: JSC-24101 pg 4-67

Note: battery/PFC material availability reference: EKS guesstimate based on technology understanding

Note: battery/PFC lowest TRL reference: EKS estimate based on technology discussion in JSC-24101 pg 3-25, 3-29

Note: battery/PFC average TRL reference: EKS estimate based on technology discussion in JSC-24101 pg 3-25, 3-29

Note: battery/PFC disposal complexity reference: JSC-24101 pg 4-67

Note: battery/PFC schedule risk reference: EKS estimate based on component type [we've done these before]

Note: battery/PFC cost risk reference: EKS estimate based on component type [we've done these before]

Habitat/Airlock

Power System Data Sheet

Power System Type:		PV/RFC (Si, high press. reactants)	PV/RFC (Si, cryo stored reactants)	DIPS	
Performance					
System Utility					
mass	kg	5219	3000	1640	includes all necessary support structure burden
volume	cu. m	52	26	11	
day power	kW	10.5	10	10	4 units of 2.5kW DIPS provide 10kW continuous power
night power	kW	9.1	9.1	10	
quiescent power	kW	2	2	2.5	actual quiescent power requirement unknown
performance degradation	%/yr	2	2	5	dust accumulation, thermal cycling, radiation, isotope decay
operating life	yrs	3	3	10	
data rate required	Mbps				no data available
energy	kW-hr	3186	3186	continuous	need calculated from power load for night
Supportability					
availability factor	%	95	95	97	% time meeting load
deployability					deployed prior to crew arrival
EVA manhours required for set-up	hr	0.0	0.0	0.0	
IVA manhours required for set-up	hr	0.0	0.0	0.0	
Earth support manhours required for set-up	hr	12.0	12.0	2.0	
equipment necessary for set-up		none	none	off-loading	no data available
operability					
EVA manhours required for operations	hr/day	0.0	0.0	0.0	
IVA manhours required for operations	hr/day	0.2	0.2	0.3	
Earth support manhours required for operations	hr/day	0.2	0.2	0.3	quiescent period monitoring & control
health monitoring complexity		low/mod	mod/high	moderate	subjective value: fnc(# sensors, monitoring criticality)
maintainability					
EVA manhours required for maintenance	hr/day	0.2	0.2	0.1	
IVA manhours required for maintenance	hr/day	0.3	0.4	0.1	
Earth support manhours required for maintenance	hr/day	0.3	0.4	0.1	
equipment type necessary for maintenance		supporting	supporting	supporting	
maintenance complexity		high	high	low	subjective value: fnc(# items, item type)
resource consumption					
spares/repair items	kg	320.0	320.0	0.0	
consumables	kg	0.0	0.0	0.0	
storage needs	cu. m				no data available
resupply frequency	#/yr	0.3	0.3	0.1	
resupply integration complexity		moderate	low	moderate	subjective value: fnc(item type)
Cost					
Development costs	$M	86	101	120	
Production costs	$M	93	106	365	includes plutonium cost for first DIPS units
replacement unit cost	$M	65	78	20	uses original plutonium

Habitat/Airlock

Power System Data Sheet

Power System Type:		PV/RFC (St, high press. reactants)	PV/RFC (St, cryo stored reactants)	DIPS	
Schedule					
development duration	yrs	5.5	6	5	time from ATP to available for use
schedule margin	yrs	-0.5	-1	0	time from available for use to need date (assumes '94 start)
Risk					
Safety					
launch safety		low	low	mod/high	high >> high risk relating to launch safety concerns
operational safety		moderate	moderate	low/mod	
environment contamination		low	low	moderate	
Technical Risk					
complexity		moderate	mod/high	moderate	
material availability		none	none	high	
TRL of least mature component		4	3	4	
average system TRL		4.5	4	5	
decommissioning/disposal complexity		none	none	low/mod	
Schedule risk		low/mod	moderate	mod/high	
Cost risk		low/mod	moderate	high	
		limited to <80° latitude for reasonable masses	limited to <80° latitude for reasonable masses	global	

Evolution capability

not currently implemented

Note: PV/RFC type: SSF derived tracking PV array attached to the top of the habitat, 3000 psi reactant storage

Note: PV/cryo RFC type: SSF derived tracking PV array attached to the top of the habitat, cryogenic reactant storage

Note: DIPS type: plutonium-fueled GPHS with CBC (closed Brayton cycle) conversion

Note: PV/RFC mass reference: H. Chandler 3/23/92 as reflected in 4/2 "Dry Run" presentation

Note: PV/cryo RFC mass reference: Bob Cataldo estimate 4/28/92

Note: DIPS mass reference: Lee Mason estimate from LeRC 3/25/92 background data package

Note: PV/RFC volume reference: Bob Cataldo, LeRC FLO Sizing 3/17/92

Note: PV/cryo RFC volume reference: Bob Cataldo, LeRC FLO Sizing 3/17/92 [cryo system needs 1/2 volume of gaseous]

Note: DIPS volume reference: Bob Cataldo, LeRC FLO Sizing 3/17/92

Note: PV/RFC day power reference: Maus 3/24/92 as reflected in 4/2 "Dry Run" presentation
issue: 17.5 kW estimated by Bob Cataldo 4/28/92 as necessary if RFC recharging is to be done during crewed periods

Note: PV/cryo RFC day power reference: Lee Mason estimate from LeRC 3/25/92 background data package
issue: 23 kW estimated by Bob Cataldo 4/28/92 as necessary if RFC recharging is to be done during crewed periods

Note: DIPS day power reference: Lee Mason estimate from LeRC 3/25/92 background data package

Note: PV/RFC night power reference: Maus 3/24/92 as reflected in 4/2 "Dry Run" presentation

Note: PV/cryo RFC night power reference: Lee Mason estimate from LeRC 3/25/92 background data package

Note: DIPS night power reference: Lee Mason estimate from LeRC 3/25/92 background data package

Note: PV/RFC quiescent power reference: Maus 3/24/92 as reflected in 4/2 "Dry Run" presentation

Note: PV/cryo RFC quiescent power reference: Maus 3/24/92 as reflected in 4/2 "Dry Run" presentation

Note: DIPS quiescent power reference: Bob Cataldo estimate 4/28/92

Note: PV/RFC performance degradation reference: JSC-24101 pg 3-5

Note: PV/cryo RFC performance degradation reference: JSC-24101 pg 3-5

Note: DIPS performance degradation reference: Lee Mason guesstimate

Note: PV/RFC operating life reference: Don Williams 3/1/92 as reflected in 3/4 ExCB presentation; JSC-24101 pg 3-51

Note: PV/cryo RFC operating life reference: Don Williams 3/1/92 as reflected in 3/4 ExCB presentation; JSC-24101 pg 3-51

Note: DIPS operating life reference: JSC-24101 pg 4-3, 4-43

Note: PV/RFC energy reference: Lee Mason [probably calculation of night power*night time]

Note: PV/cryo RFC energy reference: Lee Mason [probably calculation of night power*night time]

Note: DIPS energy reference: continuous power generation requires no energy storage

Note: PV/RFC availability factor reference: Lee Mason geusstimate

Note: PV/cryo RFC availability factor reference: Lee Mason geusstimate

Note: DIPS availability factor reference: JSC-24101 pg 3-111

Note: PV/RFC EVA deploy reference: EKS/Lee Mason estimate based on deployment prior to crew arrival

Note: PV/cryo RFC EVA deploy reference: EKS/Lee Mason estimate based on deployment prior to crew arrival

Note: DIPS EVA deploy reference: EKS/Lee Mason estimate based on deployment prior to crew arrival

Note: PV/RFC IVA deploy reference: EKS/Lee Mason estimate based on deployment prior to crew arrival

Note: PV/cryo RFC IVA deploy reference: EKS/Lee Mason estimate based on deployment prior to crew arrival

Note: DIPS IVA deploy reference: EKS/Lee Mason estimate based on deployment prior to crew arrival

Note: PV/RFC Earth support deploy reference: Lee Mason guesstimate based on deploying the PV panels

Note: PV/cryo RFC Earth support deploy reference: Lee Mason guesstimate based on deploying the PV panels

Note: DIPS Earth support deploy reference: Lee Mason guesstimate based on offloading a "stand-alone" cart

Note: PV/RFC deploy equipment reference: EKS guesstimate based on assumption that deployment equipment integrated into design

Note: PV/cryo RFC deploy equipment reference: EKS guesstimate based on assumption that deployment equipment integrated into design

Note: DIPS deploy equipment reference: EKS guesstimate based on assumption that something will be needed to get the power supply off the lander

Note: PV/RFC EVA operations reference: EKS guesstimate based on judgement that EVA is not necessary for basic operations

Note: PV/cryo RFC EVA operations reference: EKS guesstimate based on judgement that EVA is not necessary for basic operations

Note: DIPS EVA operations reference: EKS guesstimate based on judgement that EVA is not encouraged around nuclear devices

Note: PV/RFC IVA operations reference: EKS guesstimate based on 10 minutes monitor & control per day for basic operations

Note: PV/cryo RFC IVA operations reference: EKS guesstimate based on 12 minutes monitor & control per day for basic operations (slightly higher for cryo storage than for high press)

Note: DIPS IVA operations reference: EKS guesstimate based on 20 minutes monitor & control per day for basic operations (slightly higher than cryo or high press PV/RFC systems)

Note: PV/RFC Earth support operations reference: EKS assumption of duplication of IVA monitor & control to aid in evolution to 100% IVA

Note: PV/cryo RFC Earth support operations reference: EKS assumption of duplication of IVA monitor & control to aid in evolution to 100% IVA

Note: DIPS Earth support operations reference: EKS assumption of duplication of IVA monitor & control to aid in evolution to 100% IVA

Note: PV/RFC health monitoring reference: EKS WAG based on relative complexity of tracking PV and high pressure storage devices

Note: PV/cryo RFC health monitoring reference: EKS WAG based on relative complexity of tracking PV and cryo storage devices (more complex than high press)

Note: DIPS health monitoring reference: EKS WAG based on relative complexity and criticality of nuclear devices

Note: PV/RFC EVA maintenance reference: EKS guesstimate based on Fisher-Price database*2 + 8 man-hrs for dusting in 45 day mission

Note: PV/cryo RFC EVA maintenance reference: EKS guesstimate based on Fisher-Price database*2 (rounded up due to reliquifaction) + 8 man-hrs for dusting in 45 day mission

Note: DIPS EVA maintenance reference: Eileen Stansbery estimate based on no failed hardware + 4 man-hrs for dusting in 45 day mission (easier config than PV on habitat)
issue: assumption of no hardware maintenance and repair is invalid and must be updated

Note: PV/RFC IVA maintenance reference: EKS WAG double database EVA time to estimate planning + monitor (real-time) all EVA maintenance

Note: PV/cryo RFC IVA maintenance reference: EKS WAG double database EVA time to estimate planning + monitor (real-time) all EVA maintenance

Note: DIPS IVA maintenance reference: EKS WAG double database EVA time to estimate planning + monitor (real-time) all EVA maintenance

Note: PV/RFC Earth support maintenance reference: EKS assumption of duplication of IVA monitor & control to aid in evolution to 100% IVA

Note: PV/cryo RFC Earth support maintenance reference: EKS assumption of duplication of IVA monitor & control to aid in evolution to 100% IVA

Note: DIPS Earth support maintenance reference: EKS assumption of duplication of IVA monitor & control to aid in evolution to 100% IVA

Note: PV/RFC maintenance equipment type reference: EKS WAG based on probability of electrostatic device (supporting tool) rather than wisk broom (simple tool)

Note: PV/cryo RFC maintenance equipment type reference: EKS WAG based on probability of electrostatic device (supporting tool) rather than wisk broom (simple tool)

Note: DIPS maintenance equipment type reference: EKS WAG based on probability of electrostatic device (supporting tool) rather than wisk broom (simple tool)

Note: PV/RFC maintenance complexity reference: Bob Cataldo based on inaccessibility of RFC package (EKS WAG of low/mod due to high pressure reactant system)

Note: Pv/cryo RFC maintenance complexity reference: Bob Cataldo estimate based on inaccessibility of RFC
[EKS WAG of moderate due to cryo reactant system more complex than high press; JSC-24101 pg 3-47]

Note: DIPS maintenance complexity reference: EKS WAG based on simple dusting

Note: PV/RFC spare reference: Lee Mason guesstimate of hardware (replace electrolizer and fuel cell stack)

Note: PV/cryo RFC spare reference: Lee Mason guesstimate of hardware (replace electrolizer and fuel cell stack)
issue: mass estimate needs to be verified - I used mass estimate from 3000 psi system

Note: DIPS spare reference: Lee Mason guesstimate based on operating life and maintenance

Note: PV/RFC consumable reference: Lee Mason/EKS WAG due to absence of available data; it is expected that some reactant resupply will be necessary

Note: PV/cryo RFC consumable reference: Lee Mason/EKS WAG due to absence of available data; it is expected that some reactant resupply will be necessary

Note: DIPS consumables reference: Lee Mason guesstimate based on design of GPHS

Note: PV/RFC resupply frequency reference: Lee Mason/EKS guesstimate anticipated replacement once every three years based on operating life

Note: PV/RFC resupply frequency reference: Lee Mason/EKS guesstimate anticipated replacement once every three years based on operating life

Note: DIPS resupply frequency reference: Lee Mason/EKS guesstimate anticipate replacement once every ten years

Note: PV/RFC resupply complexity reference: EKS guesstimate based on current accessibility to RFC elements

Note: PV/cryo RFC resupply complexity reference: EKS guesstimate based on replacing stack on surface

Note: DIPS resupply complexity reference: EKS guesstimate based on transferring plutonium to new hardware

Note: PV/RFC development cost reference: Kelley Cyr 3/16/92 cost estimate
fuel cells, tanks, hardware + electrolyser, H2O tank + solar arrays
issue: Bob Cataldo estimate 4/28/92 is $173M

Note: PV/cryo RFC development cost reference: Kelley Cyr 3/16/92 cost estimate of gaseous system with 20% overhead for cryo

Note: DIPS development cost reference: Lee Mason/Bob Cataldo cost estimate (info from DOE puts it between 120 and 150)

Note: PV/RFC production cost reference: Kelley Cyr 3/16/92 cost estimate
fuel cells, tanks, hardware + electrolyser, H2O tank + solar arrays
issue: Bob Cataldo estimate 4/28/92 is $87M

Note: PV/cryo RFC production cost reference: Kelley Cyr 3/16/92 cost estimate for gaseous system with 20% overhead for cryo

Note: DIPS production cost reference: Lee Mason/Bob Cataldo cost estimate algorithm for producing 4 2.5 kW modules [prod. cost = 34.85 * P(kW) + 4]

Note: PV/RFC replacement cost reference: Kelley Cyr 3/16/92 cost estimate

fuel cells, tanks, hardware + electrolyser, H2O tank

Note: PV/cryo RFC production cost reference: Kelley Cyr 3/16/92 cost estimate for gaseous system with 20% overhead for cryo

Note: DIPS replacement cost reference: Bob Cataldo cost estimate for producing one 2.5 kW module without plutonium

Note: PV/RFC development time reference: EKS estimate based on POP 90-2 T/AD data and Phil Dempsey guess based on Shuttle experience

Note: PV/cryo RFC development time reference: EKS estimate based on POP 90-2 T/AD data

Note: DIPS development schedule reference: Bob Cataldo estimate 4/28/92

Note: PV/RFC launch safety risk reference: EKS estimate based on JSC-24101 pg 4-21

Note: PV/cryo RFC launch safety risk reference: EKS estimate based on JSC-24101 pg 4-21

Note: DIPS launch safety reference: Bob Cataldo guesstimate based on plutonium load issue: JSC-24101 pg 3-101 tends to indicate low risk

Note: PV/RFC operational safety risk reference: EKS estimate based on JSC-24101 pg 4-21, modified for 3000 psi tank rupture risk

Note: PV/cryo RFC operational safety risk reference: Bob Cataldo estimate 4/28/92 (EKS estimate of low based on JSC-24101 pg 4-21)

Note: DIPS operational safety reference: Eileen Stansbery guesstimate based on JSC-24101 pg 3-101

Note: PV/RFC contamination risk reference: EKS estimate based on JSC-24101 pg 4-21

Note: PV/cryo RFC contamination risk reference: EKS estimate based on JSC-24101 pg 4-21

Note: DIPS contamination reference: Eileen Stansbery guesstimate (WAG)

Note: PV/RFC technical complexity reference: EKS estimate based on tracking mechanisms for PV and high pressure reliable RFC (JSC-24101 pg 3-5, 3-48)

Note: PV/cryo RFC technical complexity reference: EKS estimate based on tracking mechanisms for PV and cryogenic RFC in JSC-24101 pg 3-5, 3-47 (cryo RFC is more complex than high press. RFC)

Note: DIPS technical complexity reference: Lee Mason/Eileen Stansbery guesstimate (WAG)

Note: PV/RFC material availability reference: EKS estimate, should have no trouble obtaining parts

Note: PV/cryo RFC material availability reference: EKS estimate, should have no trouble obtaining parts

Note: DIPS material availability reference: Lee Mason/Eileen Stansbery guesstimate (WAG) based on Pu238 production

Note: PV/RFC lowest TRL reference: JSC-24101 pg 3-53 modified for work since published through discussions with Lee Mason

Note: PV/cryo RFC lowest TRL reference: JSC-24101 pg 3-53

Note: DIPS lowest TRL reference: JSC-24101 pg 4-7

Note: PV/RFC average TRL reference: Don Williams 3/1/92 as reflected in 3/4 ExCB presentation; PV is 6 (JSC-24101 pg 3-5) while RFC is ~4

Note: PV/cryo RFC average TRL reference: PV is 6 (JSC-24101 pg 3-5) while RFC is ~4 (JSC-24101 pg 3-53) but cryo systems are less mature than high pressure RFC

Note: DIPS system average TRL reference: JSC-24101 pg 4-7 modified by Bob Cataldo for work from 1989

Note: PV/RFC disposal complexity reference: JSC-24101 pg 4-21

Note: PV/cryo RFC disposal complexity reference: JSC-24101 pg 4-21

Note: DIPS disposal complexity reference: Lee Mason/Eileen Stansbery guesstimate due to isolation requirements

Note: PV/RFC schedule risk reference: EKS guesstimate based on lifetime testing of RFCs

Note: PV/cryo RFC schedule risk reference: EKS guesstimate based on lifetime testing of RFCs and cryo development

Note: DIPS schedule risk reference: Bob Cataldo estimate 4/28/92 based on "fast-track" program

Note: PV/RFC cost risk reference: EKS WAG

Note: PV/cryo RFC cost risk reference: EKS WAG

Note: DIPS cost risk reference: Eileen Stansbery guesstimate (WAG) due to technology immaturity and plutonium purchase uncertainties

Unpressurized Rover

Power System Data Sheet

Power System Type:		RFCs	NaS Battery	DHPS	
Performance					
System Utility					
mass	kg	203	170	410	
volume	cu. m	1	0.17	2.8	
power	kW	2	2	2.5	
performance degradation	%/yr	5	5	5	dust accumulation, thermal cycling, radiation, isotope decay
operating life	yrs	4	4	10	
data rate required	Mbps				no data available
energy	kW-hr	40	40	N/A	
Supportability					
availability factor	%	95	95	97	% time meeting load
deployability					integrated into vehicle, deployed radiator in baseline
EVA manhours required for set-up	hr	0.0	0.0	0.0	
IVA manhours required for set-up	hr	1.0	0.0	2.0	
Earth support manhours required for set-up	hr	1.2	0.0	2.4	
equipment necessary for set-up		none	none	off-loading	
operability					includes recharging requirements as necessary
EVA manhours required for operations	hr/day	0.2	0.2	0.0	
IVA manhours required for operations	hr/day	0.0	0.0	0.0	
Earth support manhours required for operations	hr/day	0.0	0.0	0.0	
health monitoring complexity		moderate	low	moderate	subjective value: fnc(# sensors, monitoring criticality)
maintainability					
EVA manhours required for maintenance	hr/day	0.1	0.0	0.08	
IVA manhours required for maintenance	hr/day	0.1	0.1	0.1	
Earth support manhours required for maintenance	hr/day	0.1	0.1	0.08	
equipment necessary for maintenance		supporting	simple	supporting	
maintenance complexity		low/mod	low	low	subjective value: fnc(item type)
resource consumption					
spares/repair items	kg	64.0	170.0	0.0	
consumables	kg	0.0	0.0	0.0	
storage needs	cu. m				no data available
resupply frequency	#/yr	0.3	0.3	0.1	
resupply integration complexity		low/mod	low/mod	moderate	subjective value: fnc(item type)
Cost					
Development costs	$M	40	12	0	
Production costs	$M	9	2	91	
replacement unit cost	$M	9	2	20	

Unpressurized Rover

Power System Data Sheet

Power System Type:		RFCs	NaS Battery	DIPS	
Schedule					
development duration	yrs	4	5	5	time from ATP to available for use
schedule margin	yrs	1	0	0	time from available for use to need date (assumes '94 start)
Risk					
Safety					
launch safety		low	low	mod/high	high >> high risk relating to launch safety concerns
operational safety		low	low	low/mod	
environment contamination		low	none	moderate	
Technical Risk					
complexity		low	low/mod	moderate	
material availability		none	none	high	
TRL of least mature component		4	3	4	
average TRL			3	5	
decommissioning/disposal complexity		none	none	low/mod	
Schedule risk		low/mod	low/mod	mod/high	
Cost risk		low	low/mod	high	

Evolution capability

not currently implemented

Note: NaS battery type: bipolar cell design providing 200 Wh/kg and >400 Wh/liter

Note: RFC type: alkaline fuel cell with gaseous storage

Note: DIPS type: plutonium-fueled GPHS with CBC (closed Brayton cycle) conversion on a trailer/cart (including a peaking battery)

Note: battery mass reference: LeRC's FLO Sizing description integration data from 3/17/92

Note: RFC mass reference: Bob Cataldo integration data from 3/10/92

Note: DIPS mass reference: Lee Mason estimate 3/24/92 based on m(kg) = 114 * P(kW) + 125 [DIPS vs. RTGs]

Note: battery volume reference: LeRC's FLO Sizing description integration data from 3/17/92

Note: RFC volume reference: Bob Cataldo integration data from 3/10/92

Note: DIPS volume reference: LeRC's FLO Sizing description 3/17/92

Note: battery power reference: LeRC's FLO Sizing description integration data from 3/17/92

Note: RFC power reference: PSS power profile and LeRC's FLO Sizing description 3/17/92

Note: DIPS power reference: Lee Mason estimate 3/25/92 based on cloning philosophy (design and develop 2.5 kW modules)

Note: battery performance degradation reference: Lee Mason guesstimate (discussion of battery degradation in JSC-24101 pg 3-26)

Note: RFC performance degradation reference: Lee Mason guesstimate

Note: DIPS performance degradation reference: Lee Mason guesstimate

Note: battery operating life reference: JSC-24101 pg 4-3 with sanity factor applied by Bob Cataldo 4/28/92

Note: RFC operating life reference: JSC-24101 pg 4-7

Note: DIPS operating life reference: JSC-24101 pg 4-7

Note: battery energy reference: LeRC's FLO Sizing description 3/17/92

Note: RFC energy reference: LeRC's FLO Sizing 3/17/92

Note: battery availability factor reference: Lee Mason guesstimate
issue: should be updated to accurately reflect recharging needs

Note: RFC availability factor reference: Lee Mason guesstimate
issue: should be updated to accurately reflect recharging needs

Note: DIPS availability factor reference: JSC-24101 pg 3-111

Note: [illegible]ery EVA deploy reference: EKS estimate based on integrated vehic[illegible] [illegible]esign

Note: RFC EVA deploy reference: EKS estimate based on automatically deploying radiator panels prior to crew use

Note: DIPS EVA deploy reference: EKS guesstimate based on teleoperating deployment of power cart

Note: battery IVA deploy reference: EKS estimate based on integrated vehicle design

Note: RFC IVA deploy reference: EKS WAG based on automatic/teleoperated radiator panel deployment from the habitat

Note: DIPS IVA operations reference: EKS WAG of off-loading necessary for additional power cart (same time estimate as habitat DIPS system offloading)

Note: battery Earth support operations reference: EKS assumption of duplication of IVA monitor & control (with additional 20% overhead for planning) to aid in evolution to 100% IVA

Note: RFC Earth support operations reference: EKS assumption of duplication of IVA monitor & control (with additional 20% overhead for planning) to aid in evolution to 100% IVA

Note: DIPS Earth support operations reference: EKS assumption of duplication of IVA monitor & control (with additional 20% overhead for planning) to aid in evolution to 100% IVA

Note: battery deploy equipment reference: EKS WAG based on assuming integrated vehicle design

Note: RFC deploy equipment reference: EKS WAG based on assuming integrated vehicle design

Note: DIPS deploy equipment reference: EKS WAG based on separate power cart design

Note: battery EVA operations reference: EKS guesstimate based on JSC-24101 pg 4-3 activation time estimate (TBD min. - using 10 minutes for fuel cell and RFC) to cover recharge responsibilities

Note: RFC EVA operations reference: EKS guesstimate based on JSC-24101 pg 4-7 activation time estimate (10 minutes) to cover recharge responsibilities

Note: DIPS EVA operations reference: EKS guesstimate based on judgement that EVA is not encouraged around nuclear devices

Note: battery IVA operations reference: EKS WAG assuming no monitor of EVA activation activity

Note: RFC IVA operations reference: EKS WAG assuming no monitor of EVA activation activity

Note: DIPS IVA operations reference: EKS WAG assuming no monitor of EVA activation activity

Note: battery Earth support operations reference: EKS SWAG assuming no incidental monitoring and control

Note: RFC Earth support operations reference: EKS SWAG assuming no incidental monitoring and control

Note: DIPS Earth support operations reference: EKS SWAG assuming no incidental monitoring and control

Note: battery health monitoring reference: EKS guesstimate based on JSC-24101 pg 3-23 and limited complexity of battery systems

Note: RFC health monitoring reference: EKS guesstimate based on JSC-24101 pg 3-35

Note: DIPS health monitoring reference: EKS WAG based on relative complexity and criticality of nuclear devices

Note: battery EVA maintenance reference: EKS WAG based on maintenance time from SSF database

Note: RFC EVA maintenance reference: EKS WAG based on radiator dusting (15 minutes every third rove) + maintenance time from SSF database (really flaky assumption)

Note: RFC EVA maintenance reference: EKS WAG based on radiator dusting (15 minutes every third rove) + maintenance time from SSF database (really flaky assumption)

Note: battery IVA maintenance reference: EKS WAG double database EVA time to estimate planning + monitor (real-time) all EVA maintenance

Note: RFC IVA maintenance reference: EKS WAG double database EVA time to estimate planning + monitor (real-time) all EVA maintenance

Note: DIPS IVA maintenance reference: EKS WAG double database EVA time to estimate planning + monitor (real-time) all EVA maintenance

Note: battery Earth support maintenance reference: EKS assumption of duplication of IVA monitor & control to aid in evolution to 100% IVA

Note: RFC Earth support maintenance reference: EKS assumption of duplication of IVA monitor & control to aid in evolution to 100% IVA

Note: DIPS Earth support maintenance reference: EKS assumption of duplication of IVA monitor & control to aid in evolution to 100% IVA

Note: battery maintenance equipment type reference: EKS WAG based on probability of only needing hand tools (simple tool) for recharging etc. [no dust-off specifically needed for batteries]

Note: RFC maintenance equipment type reference: EKS WAG based on probability of electrostatic device (supporting tool) rather than wisk broom (simple tool)

Note: DIPS maintenance equipment type reference: EKS WAG based on probability of electrostatic device (supporting tool) rather than wisk broom (simple tool)

Note: battery maintenance complexity reference: EKS WAG, design for maintenance

Note: RFC maintenance complexity reference: EKS WAG, size of RFC system relative to rover makes some complications inevitable

Note: DIPS maintenance complexity reference: EKS WAG, design for maintenance (separate cart allows for easy maintenance)

Note: battery spare reference: Lee Mason guesstimate
of replacing the batteries

Note: RFC spare reference: Lee Mason guesstimate
of electrolizer and stack

Note: DIPS spare reference: Lee Mason guesstimate
of no replacement necessary
issue: update for 10 year life

Note: ery consumable reference: Lee Mason/EKS guesstimate

Note: RFC consumable reference: Lee Mason/EKS guesstimate due to absence of data (assumes no leakage)

Note: DIPS consumables reference: Lee Mason guesstimate based on design of GPHS

Note: RFC resupply frequency reference: Lee Mason/EKS guesstimate anticipating replacement needed once every four years based on operating life

Note: RFC resupply frequency reference: Lee Mason/EKS guesstimate anticipating replacement needed once every four years based on operating life

Note: DIPS resupply frequency reference: Lee Mason/EKS guesstimate anticipating hardware replacement needed once every ten years

Note: battery resupply complexity reference: EKS WAG anticipating replacement of battery system

Note: RFC resupply complexity reference: EKS WAG anticipating accessible RFC design

Note: DIPS resupply complexity reference: EKS guesstimate based on transferring plutonium to new hardware

Note: battery development cost reference: JSC-24101 pg 3-29 [$8-12M over development period]

Note: RFC development cost reference: Jim Lee cost estimate 3/25 (assumed complexity = 1 >> new development)

Note: DIPS development cost reference: bookkept by habitat

Note: battery production cost reference: JSC-24101 pg 3-29 [$50k/kWh]

Note: RFC production cost reference: Jim Lee cost estimate 3/25

Note: DIPS production cost reference: Lee Mason/Bob Cataldo cost estimate algorithm for producing one 2.5 kW module [prod. cost = 34.85 * P(kW) + 4]

Note: battery production cost reference: EKS assumption of replacing entire unit; JSC-24101 pg 3-29 [$50k/kWh]

Note: RFC replacement cost reference: EKS assumption of replaceing entire unit; Jim Lee cost estimate 3/25

Note: DIPS replacement cost reference: Bob Cataldo cost estimate for producing one 2.5 kW module without plutonium

Note: battery development time reference: EKS WAG; JSC-24101 pg 3-29 schedule 8 yr written in 1989. [USAF Light Weight Aerospace Battery Program, contractor - Hughes]

Note: RFC development time reference: EKS WAG technology knowledge from POP90-2, data in JSC-24101, and based on Shuttle schedule discussion with Phil Dempsey

Note: DIPS development schedule reference: Bob Cataldo estimate 4/28/92

Note: battery launch safety risk reference: EKS estimate based on JSC-24101 pg 4-3 (launched frozen)

Note: RFC launch safety risk reference: EKS estimate based on JSC-24101 pg 4-21

Note: DIPS launch safety reference: Bob Cataldo guesstimate based on plutonium load issue: JSC-24101 pg 3-101 tends to indicate low risk

Note: battery operations safety risk reference: EKS estimate based on JSC-24101 pg 4-3

Note: RFC operations safety risk reference: EKS estimate based on JSC-24101 pg 4-21

Note: DIPS operational safety reference: Eileen Stansbery guesstimate based on JSC-24101 pg 3-101

Note: battery contamination risk reference: EKS estimate based on JSC-24101 pg 3-23 to 3-25

Note: RFC contamination risk reference: EKS estimate based on JSC-24101 pg 3-32

Note: DIPS contamination reference: Eileen Stansbery guesstimate (WAG)

Note: battery technical complexity reference: EKS guesstimate based on discussion in JSC-24101 pg 3-23 to 3-26

Note: RFC technical complexity reference: EKS guesstimate based on discussion in JSC-24101 pg 3-29 to 3-35

Note: DIPS technical complexity reference: Lee Mason/Eileen Stansbery guesstimate (WAG)

Note: battery material availability reference: EKS estimate, should have no trouble obtaining parts or chemistry

Note: RFC material availability reference: EKS estimate, should have no trouble obtaining parts

Note: DIPS material availability reference: Lee Mason/EKS guesstimate (WAG) based on plutonium

Note: battery lowest TRL reference: JSC-24101 pg 3-26

Note: RFC lowest TRL reference: Lee Mason estimate based on RFC readiness

Note: DIPS lowest TRL reference: JSC-24101 pg 4-7

Note: battery system average TRL reference: JSC-24101 pg 3-26

Note: DIPS system average TRL reference: JSC-24101 pg 4-7 modified by Bob Cataldo for work from 1989

Note: battery disposal complexity reference: JSC-24101 pg 4-3

Note: RFC disposal complexity reference: JSC-24101 pg 4-7

Note: DIPS disposal complexity reference: Lee Mason/Eileen Stansbery guesstimate due to isolation requirements

Note: battery schedule risk reference: JSC-24101 pg 3-26 discussion of USAF Light-Weight Aerospace Battery Program

Note: RFC schedule risk reference: EKS WAG based on understanding of lifetime and reliability testing from technology knowledge

Note: DIPS schedule risk reference: Bob Cataldo estimate 4/28/92 based on "fast-track" program

Note: ery cost risk reference: EKS WAG

Note: RFC cost risk reference: EKS WAG

Note: DIPS cost risk reference: Eileen Stansbery guesstimate (WAG) due to technology immaturity and plutonium purchase uncertainties

Geophysical Monitoring Package

Power System Data Sheet

Power System Type:		RTG	uses science utility power station	

Performance

System Utility				
mass	kg	22		
volume	cu. m	0.1		
power	W	96		
performance degradation	%/yr	3		dust accumulation, thermal cycling, radiation, isotope decay
operating life	yrs	12		
data rate required	Mbps			no data available
energy	kW-hr			

Supportability				
availability factor	%	95		% time meeting load
deployability				
EVA manhours required for set-up	hr	0		
IVA manhours required for set-up	hr	0		
Earth support manhours required for set-up	hr	0		
equipment necessary for set-up		none		
operability				
EVA manhours required for operations	hr/day	0		
IVA manhours required for operations	hr/day	0		
Earth support manhours required for operations	hr/day	0		
health monitoring complexity		moderate		subjective value: fnc(# sensors, monitoring criticality)
maintainability				
EVA manhours required for maintenance	hr/day	0		
IVA manhours required for maintenance	hr/day	0		
Earth support manhours required for maintenance	hr/day	0		
equipment type necessary for maintenance		none		
maintenance complexity		none		subjective value: fnc(item type)
resource consumption				
spares/repair items	kg	0		
consumables	kg	0		
storage needs	cu. m			no data available
resupply frequency	#/yr	0.08		
resupply integration complexity		high		subjective value: fnc(item type)

Cost

Development costs	$M	0		
Production costs	$M	18		includes plutonium cost for RTG
replacement unit cost	$M			

Geophysical Monitoring Package

Power System Data Sheet

Power System Type:		RTG	uses science utility power station	
Schedule				
development duration	yrs	2		time from ATP to available for use
schedule margin	yrs	3		time from available for use to need date (assumes '94 start)
Risk				
Safety				
launch safety		very high		high >> high risk relating to launch safety concerns
operational safety		moderate		
environment contamination		low		
Technical Risk				
complexity		low		
material availability		high		
TRL of least mature component		8		
average TRL		8		
decommissioning/disposal complexity		low/mod		
Schedule risk		low		
Cost risk		low		

Evolution capability

currently no implemented

Note: RTG type: plutonium fueled GPHS linked to Si-Ge thermoelectrics

Note: RTG mass reference: Lee Mason algorithm RTG(kg) = 224 * P(kW) + 0

Note: RTG volume reference: Bob Cataldo estimate 4/28/92 of 0.4m length and 0.5m diameter volume = pi*R^2*L

Note: RTG power reference: Dave McKay 2/27/92 presentation including 1/17/92 experiment package design

Note: RTG performance degradation reference: Bob Cataldo estimate 4/28/92

Note: RTG operating life reference: JSC-24101 pg 4-11

Note: RTG availability factor reference: Lee Mason guesstimate

Note: RTG EVA deploy reference: EKS WAG based on integrated design

Note: RTG IVA deploy reference: EKS WAG based on integrated design

Note: RTG Earth support deploy reference: EKS WAG based on integrated design

Note: RTG deploy equipment reference: EKS WAG based on integrated design

Note: RTG EVA operations reference: EKS guesstimate based on judgement that EVA is not encouraged around nuclear devices

Note: RTG IVA operations reference: EKS estimate based on no monitor & control necessary for normal operations

Note: RTG Earth support operations reference: EKS estimate based on no monitor & control necessary for normal operations

Note: RTG health monitoring reference: EKS WAG (simpler than DIPS, ~RFC)

Note: RTG EVA maintenance reference: EKS estimate based on JSC-24101 pg 4-11

Note: RTG IVA maintenance reference: EKS estimate based on JSC-24101 pg 4-11

Note: RTG Earth support maintenance reference: EKS estimate based on JSC-24101 pg 4-11

Note: RTG maintenance equipment type reference: EKS estimate based on JSC-24101 pg 4-11

Note: RTG maintenance complexity reference: EKS estimate based on JSC-24101 pg 4-11

Note: RTG spare reference: EKS guesstimate
of no replacement necessary
issue: update for 12 year life

Note: RTG consumables reference: EKS guesstimate based on design of GPHS (no consumables used)

Note: RTG resupply frequency reference: EKS guesstimate anticipating hardware replacement needed once every twelve years

Note: R.. resupply complexity reference: EKS guesstimate based on transfe...g plutonium to new hardware

Note: RTG development cost reference: EKS guesstimate RTGs are very common therefore no development necessary

Note: RTG production cost reference: Bob Cataldo algorithm $M = 187.6 * P(kW)
issue: Kelley Cyr cost estimate was $43M

Note: RTG development schedule reference: Bob Cataldo estimate based on scaling current available RTGs
(red tape and regulations takes ~5 yrs)

Note: RTG launch safety reference: Lee Mason guesstimate
issue: JSC-24101 pg 4-11 indicates very high safety for RTGs

Note: RTG operational safety reference: Lee Mason guesstimate
issue: JSC-24101 pg 4-11 indicates very high safety for RTGs

Note: RTG contamination reference: Lee Mason guesstimate

Note: RTG technical complexity reference: Lee Mason estimate, JSC-24101 pg 4-8

Note: RTG material availability reference: Lee Mason/EKS guesstimate (WAG) based on plutonium

Note: RTG lowest TRL reference: EKS WAG based on straight scaling of the GPHS that has already flown (flown = TRL9)

Note: RTG system average TRL reference: JSC-24101 pg 4-11

Note: RTG disposal complexity reference: EKS guesstimate due to isolation requirements

Note: RTG schedule risk reference: Lee Mason estimate [mature technology and designs]

Note: RTG cost risk reference: Lee Mason estimate [mature technology and designs]

Solar System Physics Experiment Package — Power System Data Sheet

Power System Type:		uses science utility power station	RTG	

Performance

System Utility				
mass	kg		32	
volume	cu. m		0.15	
power	W		145	
performance degradation	%/yr		3	dust accumulation, thermal cycling, radiation, isotope decay
operating life	yrs		12	
data rate required	Mbps			no data available
energy	kW-hr			

Supportability				
availability factor	%		95	% time meeting load
deployability				
EVA manhours required for set-up	hr		0	
IVA manhours required for set-up	hr		0	
Earth support manhours required for set-up	hr		0	
equipment necessary for set-up			none	
operability				
EVA manhours required for operations	hr/day		0	
IVA manhours required for operations	hr/day		0	
Earth support manhours required for operations	hr/day		0	
health monitoring complexity			moderate	subjective value: fnc(# sensors, monitoring criticality)
maintainability				
EVA manhours required for maintenance	hr/day		0	
IVA manhours required for maintenance	hr/day		0	
Earth support manhours required for maintenance	hr/day		0	
equipment necessary for maintenance			none	
maintenance complexity			none	subjective value: fnc(item type)
resource consumption				
spares/repair items	kg		0	
consumables	kg		0	
storage needs	cu. m			
resupply frequency	#/yr		0.08	
resupply integration complexity			high	subjective value: fnc(item type)

Cost

Development costs	$M		0	
Production costs	$M		27	includes plutonium cost for RTG
replacement unit cost	$M			

Solar System Physics Experiment Package Power System Data Sheet

Power System Type:		uses science utility power station	RTG	

Schedule

development duration	yrs		2	time from ATP to available for use
schedule margin	yrs		3	time from available for use to need date (assumes '94 start)

Risk

Safety			
launch safety		very high	
operational safety		moderate	
environment contamination		low	
Technical Risk			
complexity		low	
material availability		high	
TRL of least mature component		8	
average TRL		8	
decommissioning/disposal complexity		low/moderate	
Schedule risk		low	
Cost risk		low	

Evolution capability

currently not implemented

Note: RTG type: plutonium fueled GPHS linked to Si-Ge thermoelectrics

Note: RTG mass reference: Lee Mason algorithm RTG(kg) = 224 * P(kW) + 0

Note: RTG volume reference: Bob Cataldo estimate 4/28/92

Note: RTG power reference: Dave McKay 2/27/92 presentation including 1/17/92 experiment package design

Note: RTG performance degradation reference: Bob Cataldo estimate 4/28/92

Note: RTG operating life reference: JSC-24101 pg 4-11

Note: RTG availability factor reference: Lee Mason guesstimate

Note: RTG EVA deploy reference: EKS WAG based on integrated design

Note: RTG IVA deploy reference: EKS WAG based on integrated design

Note: RTG Earth support deploy reference: EKS WAG based on integrated design

Note: RTG deploy equipment reference: EKS WAG based on integrated design

Note: RTG EVA operations reference: EKS guesstimate based on judgement that EVA is not encouraged around nuclear devices

Note: RTG IVA operations reference: EKS estimate based on no monitor & control necessary for normal operations

Note: RTG Earth support operations reference: EKS estimate based on no monitor & control necessary for normal operations

Note: RTG health monitoring reference: EKS WAG (simpler than DIPS, ~RFC)

Note: RTG EVA maintenance reference: EKS estimate based on JSC-24101 pg 4-11

Note: RTG IVA maintenance reference: EKS estimate based on JSC-24101 pg 4-11

Note: RTG Earth support maintenance reference: EKS estimate based on JSC-24101 pg 4-11

Note: RTG maintenance equipment type reference: EKS estimate based on JSC-24101 pg 4-11

Note: RTG maintenance complexity reference: EKS estimate based on JSC-24101 pg 4-11

Note: RTG spare reference: EKS guesstimate
of no replacement necessary
issue: update for 12 year life

Note: RTG consumables reference: EKS guesstimate based on design of GPHS (no consumables used)

Note: RTG resupply frequency reference: EKS guesstimate anticipating hardware replacement needed once every twelve years

Note: [illegible]esupply complexity reference: EKS guesstimate based on transf[illegible] plutonium to new hardware

Note: RTG development cost reference: EKS guesstimate RTGs are very common therefore no development necessary

Note: RTG production cost reference: Bob Cataldo algorithm $M = 187.6 * P(kW)
issue: Kelley Cyr cost estimate was $43M

Note: RTG development schedule reference: Bob Cataldo estimate based on scaling current available RTGs
(red tape and regulations takes ~5 yrs)

Note: RTG launch safety reference: Lee Mason guesstimate
issue: JSC-24101 pg 4-11 indicates very high safety for RTGs

Note: RTG operational safety reference: Lee Mason guesstimate
issue: JSC-24101 pg 4-11 indicates very high safety for RTGs

Note: RTG contamination reference: Lee Mason guesstimate

Note: RTG technical complexity reference: Lee Mason estimate, JSC-24101 pg 4-8

Note: RTG material availability reference: Lee Mason/EKS guesstimate (WAG) based on plutonium

Note: RTG lowest TRL reference: EKS WAG based on straight scaling of the GPHS that has already flown (flown = TRL9)

Note: RTG system average TRL reference: JSC-24101 pg 4-11

Note: RTG disposal complexity reference: EKS guesstimate due to isolation requirements

Note: RTG schedule risk reference: Lee Mason estimate [mature technology and designs]

Note: RTG cost risk reference: Lee Mason estimate [mature technology and designs]

Lunar Transit Telescope

Power System Data Sheet

Power System Type:		uses science utility power station	RTG	

Performance

System Utility				
mass	kg		22	
volume	cu. m		0.1	
power	W		100	
performance degradation	%/yr		3	dust accumulation, thermal cycling, radiation, isotope decay
operating life	yrs		12	
data rate required	Mbps			no data available
energy	kW-hr			

Supportability				
availability factor	%		95	% time meeting load
deployability				
EVA manhours required for set-up	hr		0	
IVA manhours required for set-up	hr		0	
Earth support manhours required for set-up	hr		0	
equipment necessary for set-up			none	
operability				
EVA manhours required for operations	hr/day		0	
IVA manhours required for operations	hr/day		0	
Earth support manhours required for operations	hr/day		0	
health monitoring complexity			moderate	subjective value: fnc(# sensors, monitoring criticality)
maintainability				
EVA manhours required for maintenance	hr/day		0	
IVA manhours required for maintenance	hr/day		0	
Earth support manhours required for maintenance	hr/day		0	
equipment necessary for maintenance			none	
maintenance complexity			none	subjective value: fnc(item type)
resource consumption				
spares/repair items	kg		0	
consumables	kg		0	
storage needs	cu. m			
resupply frequency	#/yr		0.08	
resupply integration complexity			high	subjective value: fnc(item type)

Cost

Development costs	$M		0	
Production costs	$M		19	includes plutonium cost for RTG
replacement unit cost	$M			

Lunar Transit Telescope

Power S, m Data Sheet

Power System Type:		uses science utility power station	RTG	
Schedule				
development duration	yrs		2	time from ATP to available for use
schedule margin	yrs		3	time from available for use to need date (assumes '94 start)
Risk				
Safety				
launch safety			very high	
operational safety			moderate	
environment contamination			low	
Technical Risk				
complexity			low	
material availability			high	
TRL of least mature component			8	
average TRL			8	
decommissioning/disposal complexity			low/moderate	
Schedule risk			low	
Cost risk			low	

Evolution capability

currently not implemented

Note: RTG type: plutonium fueled GPHS linked to Si-Ge thermoelectrics

Note: RTG mass reference: Lee Mason algorithm RTG(kg) = 224 * P(kW) + 0

Note: RTG volume reference: Bob Cataldo estimate 4/28/92

Note: RTG power reference: Dave McKay 2/27/92 presentation including 1/17/92 experiment package design

Note: RTG performance degradation reference: Bob Cataldo estimate 4/28/92

Note: RTG operating life reference: JSC-24101 pg 4-11

Note: RTG availability factor reference: Lee Mason guesstimate

Note: RTG EVA deploy reference: EKS WAG based on integrated design

Note: RTG IVA deploy reference: EKS WAG based on integrated design

Note: RTG Earth support deploy reference: EKS WAG based on integrated design

Note: RTG deploy equipment reference: EKS WAG based on integrated design

Note: RTG EVA operations reference: EKS guesstimate based on judgement that EVA is not encouraged around nuclear devices

Note: RTG IVA operations reference: EKS estimate based on no monitor & control necessary for normal operations

Note: RTG Earth support operations reference: EKS estimate based on no monitor & control necessary for normal operations

Note: RTG health monitoring reference: EKS WAG (simpler than DIPS, ~RFC)

Note: RTG EVA maintenance reference: EKS estimate based on JSC-24101 pg 4-11

Note: RTG IVA maintenance reference: EKS estimate based on JSC-24101 pg 4-11

Note: RTG Earth support maintenance reference: EKS estimate based on JSC-24101 pg 4-11

Note: RTG maintenance equipment type reference: EKS estimate based on JSC-24101 pg 4-11

Note: RTG maintenance complexity reference: EKS estimate based on JSC-24101 pg 4-11

Note: RTG spare reference: EKS guesstimate
of no replacement necessary
issue: update for 12 year life

Note: RTG consumables reference: EKS guesstimate based on design of GPHS (no consumables used)

Note: RTG resupply frequency reference: EKS guesstimate anticipating hardware replacement needed once every twelve years

Note: R... resupply complexity reference: EKS guesstimate based on transfe... .g plutonium to new hardware

Note: RTG development cost reference: EKS guesstimate RTGs are very common therefore no development necessary

Note: RTG production cost reference: Bob Cataldo algorithm $M = 187.6 * P(kW)
issue: Kelley Cyr cost estimate was $43M

Note: RTG development schedule reference: Bob Cataldo estimate based on scaling current available RTGs
(red tape and regulations takes ~5 yrs)

Note: RTG launch safety reference: Lee Mason guesstimate
issue: JSC-24101 pg 4-11 indicates very high safety for RTGs

Note: RTG operational safety reference: Lee Mason guesstimate
issue: JSC-24101 pg 4-11 indicates very high safety for RTGs

Note: RTG contamination reference: Lee Mason guesstimate

Note: RTG technical complexity reference: Lee Mason estimate, JSC-24101 pg 4-8

Note: RTG material availability reference: Lee Mason/EKS guesstimate (WAG) based on plutonium

Note: RTG lowest TRL reference: EKS WAG based on straight scaling of the GPHS that has already flown (flown = TRL9)

Note: RTG system average TRL reference: JSC-24101 pg 4-11

Note: RTG disposal complexity reference: EKS guesstimate due to isolation requirements

Note: RTG schedule risk reference: Lee Mason estimate [mature technology and designs]

Note: RTG cost risk reference: Lee Mason estimate [mature technology and designs]

Small Research Telescope

Power System Data Sheet

Power System Type:		uses science utility power station	DIPS	

Performance

System Utility				
mass	kg		182	
volume	cu. m			
power	W		500	
performance degradation	%/yr			dust accumulation, thermal cycling, radiation, isotope decay
operating life	yrs		10	
data rate required	Mbps			no data available
energy	kW-hr			

Supportability				
availability factor	%		97	% time meeting load
deployability				
EVA manhours required for set-up	hr		0	
IVA manhours required for set-up	hr		0	
Earth support manhours required for set-up	hr		0	
equipment necessary for set-up			none	
operability				
EVA manhours required for operations	hr/day		0	
IVA manhours required for operations	hr/day		0	
Earth support manhours required for operations	hr/day		0	
health monitoring complexity			moderate	subjective value: fnc(# sensors, monitoring criticality)
maintainability				
EVA manhours required for maintenance	hr/day		0.1	
IVA manhours required for maintenance	hr/day		0.1	
Earth support manhours required for maintenance	hr/day		0.1	
equipment necessary for maintenance			supporting	
maintenance complexity			low	subjective value: fnc(item type)
resource consumption				
spares/repair items	kg		0	
consumables	kg		0	
storage needs	cu. m			
resupply frequency	#/yr		0.1	
resupply integration complexity			moderate	subjective value: fnc(item type)

Cost

Development costs	$M		120	
Production costs	$M		21	includes plutonium costs for DIPS and/or RTG
replacement unit cost	$M			

Small Research Telescope **Power System Data Sheet**

Power System Type:		uses science utility power station	DIPS	
Schedule				
development duration	yrs		5	time from ATP to available for use
schedule margin	yrs		0	time from available for use to need date (assumes '94 start)
Risk				
Safety				
launch safety			mod/high	
operational safety			low/mod	
environment contamination			moderate	
Technical Risk				
complexity			moderate	
material availability			high	
TRL of least mature component			4	
average TRL			5	
decommissioning/disposal complexity			low/mod	
Schedule risk			mod/high	
Cost risk			high	

Evolution capability

currently not implemented

Note: DIPS type: plutonium-fueled GPHS with CBC (closed Brayton cycle) conversion on a trailer/cart

Note: DIPS mass reference: Lee Mason algorithm DIPS(kg) = 114 * P(kW) + 125

Note: DIPS power reference: Dave McKay 2/27/92 presentation including 1/17/92 experiment package design

Note: DIPS operating life reference: Eileen Stansbery/Lee Mason estimate based on JSC-24101 pg 4-11

Note: DIPS availability factor reference: JSC-24101 pg 3-111

Note: DIPS EVA deploy reference: Eileen Stansbery guesstimate based on integrated design assumption

Note: DIPS IVA deploy reference: Eileen Stansbery guesstimate based on integrated design assumption

Note: DIPS Earth support deploy reference: Eileen Stansbery guesstimate based on integrated design assumption

Note: DIPS deploy equipment reference: Eileen Stansbery guesstimate based on integrated design assumption

Note: DIPS EVA operations reference: EKS guesstimate based on judgement that EVA is not encouraged around nuclear devices

Note: DIPS IVA operations reference: EKS WAG assuming no monitor of EVA activation activity

Note: DIPS Earth support operations reference: EKS SWAG assuming no incidental monitoring and control

Note: DIPS health monitoring reference: EKS WAG based on relative complexity and criticality of nuclear devices

Note: DIPS EVA maintenance reference: EKS estimate based on no failed hardware + 4 man-hrs for dusting in 45 day mission (similar to science utility station)
issue: assumption of no hardware maintenance and repair is invalid and must be updated

Note: DIPS IVA maintenance reference: EKS WAG double database EVA time to estimate planning + monitor (real-time) all EVA maintenance

Note: DIPS Earth support maintenance reference: EKS assumption of duplication of IVA monitor & control to aid in evolution to 100% IVA

Note: DIPS maintenance equipment type reference: EKS WAG based on probability of electrostatic device (supporting tool) rather than wisk broom (simple tool)

Note: DIPS maintenance complexity reference: EKS WAG based on simple dusting

Note: DIPS spare reference: Lee Mason guesstimate
of no replacement necessary
issue: update for 10 year life

Note: DIPS consumables reference: Lee Mason guesstimate based on design of GPHS

Note: DIPS resupply frequency reference: Lee Mason/EKS guesstimate anticipating hardware replacement needed once every ten years

Note: DIPS resupply complexity reference: EKS guesstimate based on transferring plutonium to new hardware

Note: DIPS development cost reference: Lee Mason/Bob Cataldo cost estimate (info from DOE puts it between 120 and 150)
issue: someone has to pay for development

Note: DIPS production cost reference: Lee Mason algorithm DIPS($M) = 34.85 * P(kW) + 4

Note: DIPS development schedule reference: Bob Cataldo estimate 4/28/92

Note: DIPS launch safety reference: Bob Cataldo guesstimate based on plutonium load
issue: JSC-24101 pg 3-101 tends to indicate low risk

Note: DIPS operational safety reference: Eileen Stansbery guesstimate based on JSC-24101 pg 3-101

Note: DIPS contamination reference: Eileen Stansbery guesstimate (WAG)

Note: DIPS technical complexity reference: Lee Mason/Eileen Stansbery guesstimate (WAG)

Note: DIPS material availability reference: Lee Mason/Eileen Stansbery guesstimate (WAG) based on Pu238 production

Note: DIPS lowest TRL reference: JSC-24101 pg 4-7

Note: DIPS system average TRL reference: JSC-24101 pg 4-7 modified by Bob Cataldo for work from 1989

Note: DIPS disposal complexity reference: Lee Mason/Eileen Stansbery guesstimate due to isolation requirements

Note: DIPS schedule risk reference: Bob Cataldo estimate 4/28/92 based on "fast-track" program

Note: DIPS cost risk reference: Eileen Stansbery guesstimate (WAG) due to technology immaturity and plutonium purchase uncertainties

Small Solar Telescope

Power System Data Sheet

Power System Type:		uses science utility power station	RTG	

Performance

System Utility				
mass	kg		11	
volume	cu. m		0.05	
power	W		50	
performance degradation	%/yr		3	dust accumulation, thermal cycling, radiation, isotope decay
operating life	yrs		12	
data rate required	Mbps			no data available
energy	kW-hr			

Supportability				
availability factor	%		95	% time meeting load
deployability				
EVA manhours required for set-up	hr		0	
IVA manhours required for set-up	hr		0	
Earth support manhours required for set-up	hr		0	
equipment necessary for set-up			none	
operability				
EVA manhours required for operations	hr/day		0	
IVA manhours required for operations	hr/day		0	
Earth support manhours required for operations	hr/day		0	
health monitoring complexity			moderate	subjective value: fnc(# sensors, monitoring criticality)
maintainability				
EVA manhours required for maintenance	hr/day		0	
IVA manhours required for maintenance	hr/day		0	
Earth support manhours required for maintenance	hr/day		0	
equipment necessary for maintenance			none	
maintenance complexity			none	subjective value: fnc(item type)
resource consumption				
spares/repair items	kg		0	
consumables	kg		0	
storage needs	cu. m			
resupply frequency	#/yr		0.08	
resupply integration complexity			high	subjective value: fnc(item type)

Cost

Development costs	$M		0	
Production costs	$M		9	includes plutonium cost for RTG
replacement unit cost	$M			

Small Solar Telescope

Power System Data Sheet

Power System Type:		usea science utility power station	RTG	
Schedule				
development duration	yrs		2	time from ATP to available for use
schedule margin	yrs		3	time from available for use to need date (assumes '94 start)
Risk				
Safety				
launch safety			very high	
operational safety			moderate	
environment contamination			low	
Technical Risk				
complexity			low	
material availability			high	
TRL of least mature component			8	
average TRL			8	
decommissioning/disposal complexity			low/moderate	
Schedule risk			low	
Cost risk			low	

Evolution capability

currently not implemented

Note: RTG type: plutonium fueled GPHS linked to Si-Ge thermoelectrics

Note: RTG mass reference: Lee Mason algorithm RTG(kg) = 224 * P(kW) + 0

Note: RTG volume reference: Bob Cataldo estimate 4/28/92

Note: RTG power reference: Dave McKay 2/27/92 presentation including 1/17/92 experiment package design

Note: RTG performance degradation reference: Bob Cataldo estimate 4/28/92

Note: RTG operating life reference: JSC-24101 pg 4-11

Note: RTG availability factor reference: Lee Mason guesstimate

Note: RTG EVA deploy reference: EKS WAG based on integrated design

Note: RTG IVA deploy reference: EKS WAG based on integrated design

Note: RTG Earth support deploy reference: EKS WAG based on integrated design

Note: RTG deploy equipment reference: EKS WAG based on integrated design

Note: RTG EVA operations reference: EKS guesstimate based on judgement that EVA is not encouraged around nuclear devices

Note: RTG IVA operations reference: EKS estimate based on no monitor & control necessary for normal operations

Note: RTG Earth support operations reference: EKS estimate based on no monitor & control necessary for normal operations

Note: RTG health monitoring reference: EKS WAG (simpler than DIPS, ~RFC)

Note: RTG EVA maintenance reference: EKS estimate based on JSC-24101 pg 4-11

Note: RTG IVA maintenance reference: EKS estimate based on JSC-24101 pg 4-11

Note: RTG Earth support maintenance reference: EKS estimate based on JSC-24101 pg 4-11

Note: RTG maintenance equipment type reference: EKS estimate based on JSC-24101 pg 4-11

Note: RTG maintenance complexity reference: EKS estimate based on JSC-24101 pg 4-11

Note: RTG spare reference: EKS guesstimate
of no replacement necessary
issue: update for 12 year life

Note: RTG consumables reference: EKS guesstimate based on design of GPHS (no consumables used)

Note: RTG resupply frequency reference: EKS guesstimate anticipating hardware replacement needed once every twelve years

Note: RTG resupply complexity reference: EKS guesstimate based on transferring plutonium to new hardware

Note: RTG development cost reference: EKS guesstimate RTGs are very common therefore no development necessary

Note: RTG production cost reference: Bob Cataldo algorithm $M = 187.6 * P(kW)
issue: Kelley Cyr cost estimate was $43M

Note: RTG development schedule reference: Bob Cataldo estimate based on scaling current available RTGs
(red tape and regulations takes ~5 yrs)

Note: RTG launch safety reference: Lee Mason guesstimate
issue: JSC-24101 pg 4-11 indicates very high safety for RTGs

Note: RTG operational safety reference: Lee Mason guesstimate
issue: JSC-24101 pg 4-11 indicates very high safety for RTGs

Note: RTG contamination reference: Lee Mason guesstimate

Note: RTG technical complexity reference: Lee Mason estimate, JSC-24101 pg 4-8

Note: RTG material availability reference: Lee Mason/EKS guesstimate (WAG) based on plutonium

Note: RTG lowest TRL reference: EKS WAG based on straight scaling of the GPHS that has already flown (flown = TRL9)

Note: RTG system average TRL reference: JSC-24101 pg 4-11

Note: RTG disposal complexity reference: EKS guesstimate due to isolation requirements

Note: RTG schedule risk reference: Lee Mason estimate [mature technology and designs]

Note: RTG cost risk reference: Lee Mason estimate [mature technology and designs]

Lunar Geologic Tool Set

Power System Data Sheet

Power System Type:		uses unpressurized rover power system	DIPS	

Performance

System Utility				
mass	kg		182	
volume	cu. m			
power	W	500	500	
performance degradation	%/yr			dust accumulation, thermal cycling, radiation, isotope decay
operating life	yrs		10	
data rate required	Mbps			no data available
energy	kW-hr			

Supportability				
availability factor	%		97	% time meeting load
deployability				integrated into vehicle
EVA manhours required for set-up	hr		0	
IVA manhours required for set-up	hr		0	
Earth support manhours required for set-up	hr		0	
equipment necessary for set-up			none	
operability				
EVA manhours required for operations	hr/day		0	
IVA manhours required for operations	hr/day		0	
Earth support manhours required for operations	hr/day		0	
health monitoring complexity			moderate	subjective value: fnc(# sensors, monitoring criticality)
maintainability				
EVA manhours required for maintenance	hr/day		0.08	
IVA manhours required for maintenance	hr/day		0.1	
Earth support manhours required for maintenance	hr/day		0.08	
equipment necessary for maintenance			supporting	
maintenance complexity			low	subjective value: fnc(item type)
resource consumption				
spares/repair items	kg		0	
consumables	kg		0	
storage needs	cu. m			
resupply frequency	#/yr		0.1	
resupply integration complexity			moderate	subjective value: fnc(item type)

Cost

Development costs	$M		0	development cost bookkept by science utility or habitat
Production costs	$M		21	includes plutonium cost
replacement unit cost	$M			

Lunar Geologic Tool Set

Power System Data Sheet

Power System Type:		uses unpressurized rover power system	DIPS	
Schedule				
development duration	yrs		5	time from ATP to available for use
schedule margin	yrs		0	time from available for use to need date (assumes '94 start)
Risk				
Safety				
launch safety			moderate/high	high >> high risk relating to launch safety concerns
operational safety			low/moderate	
environment contamination			moderate	
Technical Risk				
complexity			moderate	
material availability			high	
TRL of least mature component			4	
average TRL			5	
decommissioning/disposal complexity			low/moderate	
Schedule risk			moderate/high	
Cost risk			high	

Evolution capability

currently not implemented

Note: DIPS type: plutonium-fueled GPHS with CBC (closed Brayton cycle) conversion on a trailer/cart

Note: DIPS mass reference: Lee Mason algorithm DIPS(kg) = 114 * P(kW) + 125

Note: DIPS power reference: Lee Mason/Eileen Stansbery rounded sum of science power users

Note: DIPS operating life reference: Eileen Stansbery/Lee Mason estimate based on JSC-24101 pg 4-11

Note: DIPS availability factor reference: JSC-24101 pg 3-111

Note: DIPS EVA deploy reference: Eileen Stansbery guesstimate based on integrated design assumption

Note: DIPS IVA deploy reference: Eileen Stansbery guesstimate based on integrated design assumption

Note: DIPS Earth support deploy reference: Eileen Stansbery guesstimate based on integrated design assumption

Note: DIPS deploy equipment reference: Eileen Stansbery guesstimate based on integrated design assumption

Note: DIPS EVA operations reference: EKS guesstimate based on judgement that EVA is not encouraged around nuclear devices

Note: DIPS IVA operations reference: EKS WAG assuming no monitor of EVA activation activity

Note: DIPS Earth support operations reference: EKS SWAG assuming no incidental monitoring and control

Note: DIPS health monitoring reference: EKS WAG based on relative complexity and criticality of nuclear devices

Note: RFC EVA maintenance reference: EKS WAG based on radiator dusting (15 minutes every third rove) + maintenance time from SSF database (really flaky assumption)

Note: DIPS IVA maintenance reference: EKS WAG double database EVA time to estimate planning + monitor (real-time) all EVA maintenance

Note: DIPS Earth support maintenance reference: EKS assumption of duplication of IVA monitor & control to aid in evolution to 100% IVA

Note: DIPS maintenance equipment type reference: EKS WAG based on probability of electrostatic device (supporting tool) rather than wisk broom (simple tool)

Note: DIPS maintenance complexity reference: EKS WAG, design for maintenance (separate cart allows for easy maintenance)

Note: DIPS spare reference: Lee Mason guesstimate
of no replacement necessary
issue: update for 10 year life

Note: DIPS consumables reference: Lee Mason guesstimate based on design of GPHS

Note: DIPS resupply frequency reference: Lee Mason/EKS guesstimate anticipating hardware replacement needed once every ten years

Note: DIPS resupply complexity reference: EKS guesstimate based on transferring plutonium to new hardware

Note: L development cost reference: bookkept by science utility station

Note: DIPS production cost reference: Lee Mason/Bob Cataldo cost estimate algorithm for producing one 2.5 kW module [prod. cost = 34.85 * P(kW) + 4]

Note: DIPS development schedule reference: Bob Cataldo estimate 4/28/92

Note: DIPS launch safety reference: Bob Cataldo guesstimate based on plutonium load
issue: JSC-24101 pg 3-101 tends to indicate low risk

Note: DIPS operational safety reference: Eileen Stansbery guesstimate based on JSC-24101 pg 3-101

Note: DIPS contamination reference: Eileen Stansbery guesstimate (WAG)

Note: DIPS technical complexity reference: Lee Mason/Eileen Stansbery guesstimate (WAG)

Note: DIPS material availability reference: Lee Mason/Eileen Stansbery guesstimate (WAG) based on Pu238 production

Note: DIPS lowest TRL reference: JSC-24101 pg 4-7

Note: DIPS system average TRL reference: JSC-24101 pg 4-7 modified by Bob Cataldo for work from 1989

Note: DIPS disposal complexity reference: Lee Mason/Eileen Stansbery guesstimate due to isolation requirements

Note: DIPS schedule risk reference: Bob Cataldo estimate 4/28/92 based on "fast-track" program

Note: DIPS cost risk reference: Eileen Stansbery guesstimate (WAG) due to technology immaturity and plutonium purchase uncertainties

Robotic Science Rover

Power System Data Sheet

Power System Type:		DIPS	RTG	RFC	NaS Battery	
Performance						
System Utility						
mass	kg	239	224	330	1760	
volume	cu. m	1.5	0.6	3.4	1.1	
nominal power	W	1000	1000	600	600	
peak power	W	2000	1000	1000	1000	
performance degradation	%/yr					dust accumulation, thermal cycling, radiation, isotope decay
operating life	yrs	10	12	4	4	
data rate required	Mbps					no data available
energy	kW-hr			140	140	
Supportability						
availability factor	%	97	95	95	95	% time meeting load
deployability						integrated into vehicle
EVA manhours required for set-up	hr	0	0	0	0	
IVA manhours required for set-up	hr	0	0	1	0	
Earth support manhours required for set-up	hr	0	0	1.2	0	
equipment necessary for set-up		none	none	none	none	
operability						
EVA manhours required for operations	hr/day	0	0	0	0.17	
IVA manhours required for operations	hr/day	0	0	0.02	0	
Earth support manhours required for operatio	hr/day	0	0	0.02	0	
health monitoring complexity		moderate	moderate	moderate	low	subjective value: fnc(# sensors, monitoring criticality)
maintainability						
EVA manhours required for maintenance	hr/day	0.08	0	0.03	0.02	
IVA manhours required for maintenance	hr/day	0.1	0	0.07	0.1	
Earth support manhours required for mainten	hr/day	0.08	0	0.07	0.06	
equipment necessary for maintenance		supporting	none	supporting	simple	
maintenance complexity		low	none	low/mod	low	subjective value: fnc(item type)
resource consumption						
spares/repair items	kg	0	0	64	1760	
consumables	kg	0	0	0	0	
storage needs	cu. m					
resupply frequency	#/yr	0.1	0.08	0.25	0.25	
resupply integration complexity		moderate	high	low/mod	low/mod	subjective value: fnc(item type)
Cost						
Development costs	$M	0	0	40		development of DIPs is bookkept by Habitat for Option 1
Production costs	$M	39	187.6	9		DIPs and RTG values include plutonium
replacement unit cost	$M			9		

Ru .ic Science Rover Power System Data S. .

Power System Type:		DIPS	RTG	RFC	NaS Battery	
Schedule						
development duration	yrs	5	0	4	4	time from ATP to available for use
schedule margin	yrs	0	0	1	1	time from available for use to need date (assumes '94 start)
Risk						
Safety						
launch safety		mod/high	very high	low	low	high >> high risk relating to launch safety concerns
operational safety		low/mod	moderate	low	low	
environment contamination		moderate	low	low	none	
Technical Risk						
complexity		moderate	low	low	low/mod	
material availability		high	high	none	none	
TRL of least mature component		4	9	4	3	
average TRL		5	9		3	
decommissioning/disposal complexity		low/mod	low/mod	none	none	
Schedule risk		mod/high	low	low/mod	low/mod	
Cost risk		high	low	low	low/mod	

Evolution capability

currently not implemented

Note: RTG type: plutonium fueled GPHS linked to Si-Ge thermoelectrics

Note: RFC type: 220 hr ops time, 336 hr recharge time, 200 km range

Note: DIPS type: plutonium-fueled GPHS with CBC (closed Brayton cycle) conversion on a trailer/cart

Note: NaS battery type: bipolar cell design providing 200 Wh/kg and >400 Wh/liter

Note: RTG mass reference: Lee Mason algorithm RTG(kg) = 224 * P(kW) + 0

Note: RFC mass reference: LeRC's FLO Sizing description 3/17/92

Note: DIPS mass reference: Lee Mason algorithm DIPS(kg) = 114 * P(kW) + 125

Note: battery mass reference: LeRC's FLO Sizing description 3/17/92

Note: DIPS volume reference: Bob Cataldo estimate 4/28/92

Note: RFC volume reference: LeRC's FLO Sizing description 3/17/92

Note: DIPS volume reference: Bob Cataldo estimate 4/28/92

Note: battery volume reference: LeRC's FLO Sizing description 3/17/92

Note: RTG nominal power reference: Lee Mason estimate 3/25/92

Note: RFC nominal power reference: Lee Mason estimate 3/25/92

Note: DIPS nominal power reference: Lee Mason estimate 3/25/92

Note: battery nominal power reference: Lee Mason estimate 3/25/92

Note: DIPS peak power reference: Lee Mason estimate 3/25/92

Note: RFC peak power reference: Lee Mason estimate

Note: DIPS peak power reference: Lee Mason estimate (continuous)

Note: battery peak power reference: Lee Mason estimate

Note: RTG operating life reference: JSC-24101 pg 4-11

Note: RFC operating life reference: JSC-24101 pg 4-7

Note: DIPS operating life reference: Eileen Stansbery/Lee Mason estimate based on JSC-24101 pg 4-11

Note: battery operating life reference: JSC-24101 pg 4-3 with sanity factor applied by Bob Cataldo 4/28/92

Note : energy reference: LeRC's FLO Sizing 3/17/92

Note: battery energy reference: LeRC's FLO Sizing 3/17/92

Note: RTG availability factor reference: Lee Mason geusstimate

Note: RFC availability factor reference: Lee Mason guesstimate
issue: should be updated to accurately reflect recharging needs

Note: DIPS availability factor reference: JSC-24101 pg 3-111

Note: battery availability factor reference: Lee Mason guesstimate
issue: should be updated to accurately reflect recharging needs

Note: RTG EVA deploy reference: EKS WAG based on integrated design

Note: RFC EVA deploy reference: EKS estimate based on automatically deploying radiator panels

Note: DIPS EVA deploy reference: Eileen Stansbery guesstimate based on integrated design assumption

Note: battery EVA deploy reference: EKS estimate based on integrated vehicle design

Note: RTG IVA deploy reference: EKS WAG based on integrated design

Note: RFC IVA deploy reference: EKS WAG based on automatic/teleoperated radiator panel deployment from the habitat

Note: DIPS IVA deploy reference: Eileen Stansbery guesstimate based on integrated design assumption

Note: battery IVA deploy reference: EKS estimate based on integrated vehicle design

Note: RTG Earth support deploy reference: EKS WAG based on integrated design

Note: RFC Earth support operations reference: EKS assumption of duplication of IVA monitor & control (with additional 20% overhead for planning) to aid in evolution to 100% IVA

Note: DIPS Earth support deploy reference: Eileen Stansbery guesstimate based on integrated design assumption

Note: battery Earth support operations reference: EKS assumption of duplication of IVA monitor & control (with additional 20% overhead for planning) to aid in evolution to 100% IVA

Note: RTG deploy equipment reference: EKS WAG based on integrated design

Note: RFC deploy equipment reference: EKS WAG based on assuming integrated vehicle design

Note: DIPS deploy equipment reference: Eileen Stansbery guesstimate based on integrated design assumption

Note: battery deploy equipment reference: EKS WAG based on assuming integrated vehicle design

Note: RTG EVA operations reference: EKS estimate based on JSC-24101 pg 4-11

Note: RFC EVA operations reference: EKS guesstimate based on robotic philosophy

Note: DIPS EVA operations reference: EKS guesstimate based on judgement that EVA is not encouraged around nuclear devices

Note: battery EVA operations reference: EKS guesstimate based on JSC-24101 pg 4-3 activation time estimate (TBD min. - using 10 minutes for fuel cell and RFC) to cover recharge responsibilities

Note: RTG IVA operations reference: EKS estimate based on no monitor & control necessary for normal operations

Note: RFC Earth support operations reference: EKS guesstimate based on JSC-24101 pg 4-7 activation time estimate (30 minutes) to cover teleoperated recharge responsibilities every day/night cycle

Note: DIPS IVA operations reference: EKS WAG assuming no monitor of EVA activation activity

Note: battery IVA operations reference: EKS WAG assuming no monitor of EVA activation activity

Note: RTG Earth support operations reference: EKS estimate based on no monitor & control necessary for normal operations

Note: RFC Earth support operations reference: EKS assumption of duplication of IVA tasks to aid in evolution to 100% IVA

Note: DIPS Earth support operations reference: EKS SWAG assuming no incidental monitoring and control

Note: battery Earth support operations reference: EKS SWAG assuming no incidental monitoring and control

Note: RTG health monitoring reference: EKS WAG (simpler than DIPS, ~RFC)

Note: RFC health monitoring reference: EKS guesstimate based on JSC-24101 pg 3-35

Note: DIPS health monitoring reference: EKS WAG based on relative complexity and criticality of nuclear devices

Note: battery health monitoring reference: EKS guesstimate based on JSC-24101 pg 3-23 and limited complexity of battery systems

Note: RTG EVA maintenance reference: EKS estimate based on JSC-24101 pg 4-11

Note: RFC EVA maintenance reference: EKS WAG based on radiator dusting (15 minutes every recharge) + maintenance time from SSF database (really flaky assumption)

Note: RFC EVA maintenance reference: EKS WAG based on radiator dusting (15 minutes every third rove) + maintenance time from SSF database (really flaky assumption)

Note: battery EVA maintenance reference: EKS WAG based on maintenance time from SSF database

Note: RTG IVA maintenance reference: EKS estimate based on JSC-24101 pg 4-11

Note: RFC IVA maintenance reference: EKS WAG double database EVA time to estimate planning + monitor (real-time) all EVA maintenance

Note: DIPS Earth support maintenance reference: EKS assumption of duplication of IVA monitor & control to aid in evolution to 100% IVA

Note: RTG maintenance equipment type reference: EKS estimate based on JSC-24101 pg 4-11

Note: RFC maintenance equipment type reference: EKS WAG based on probability of electrostatic device (supporting tool) rather than wisk broom (simple tool)

Note: DIPS maintenance equipment type reference: EKS WAG based on probability of electrostatic device (supporting tool) rather than wisk broom (simple tool)

Note: battery maintenance equipment type reference: EKS WAG based on probability of only needing hand tools (simple tool)

Note: RTG maintenance complexity reference: EKS estimate based on JSC-24101 pg 4-11

Note: RFC maintenance complexity reference: EKS WAG, size of RFC system relative to rover makes some complications inevitable

Note: DIPS maintenance complexity reference: EKS WAG, design for maintenance (separate cart allows for easy maintenance)

Note: battery maintenance complexity reference: EKS WAG, design for maintenance

Note: RTG spare reference: EKS guesstimate
of no replacement necessary
issue: update for 12 year life

Note: RFC spare reference: Lee Mason guesstimate
of electrolizer and stack

Note: DIPS spare reference: Lee Mason guesstimate
of no replacement necessary
issue: update for 10 year life

Note: battery spare reference: Lee Mason guesstimate
of replacing the batteries

Note: RTG consumables reference: EKS guesstimate based on design of GPHS (no consumables used)

Note: RFC consumable reference: Lee Mason/EKS guesstimate due to absence of data (assumes no leakage)

Note: RTG resupply frequency reference: EKS guesstimate anticipating hardware replacement needed once every twelve years

Note: RFC resupply frequency reference: Lee Mason/EKS guesstimate anticipating replacement needed once every four years based on operating life

Note: DIPS resupply frequency reference: Lee Mason/EKS guesstimate anticipating hardware replacement needed once every ten years

Note: battery resupply frequency reference: Lee Mason/EKS guesstimate anticipating replacement needed once every four years based on operating life

Note: RTG resupply complexity reference: EKS guesstimate based on transferring plutonium to new hardware

Note: RFC resupply complexity reference: EKS WAG anticipating accessible RFC design

Note: DIPS resupply complexity reference: EKS guesstimate based on transferring plutonium to new hardware

Note: battery resupply complexity reference: EKS WAG anticipating replacement of battery system

Note: RTG development cost reference: EKS guesstimate RTGs are very common therefore no development necessary

Note: RFC development cost reference: Jim Lee cost estimate 3/25 (assumed complexity = 1 >> new development)
issue: possible double bookkeeping of development costs with manned rover

Note: DIPS development cost reference: bookkept by science utility station

Note: RTG production cost reference: Lee Mason algorithm RTG($M) = 123 * P(kW) + 30

Note: RFC production cost reference: Jim Lee cost estimate 3/25

Note: DIPS production cost reference: Bob Cataldo estimate algorithm; prod. cost = 34.85 * P(kW) + 4
issue: Kelley Cyr estimate $72M

Note: RFC replacement cost reference: EKS assumption of replaceing entire unit; Jim Lee cost estimate 3/25

Note: RTG development schedule reference: Bob Cataldo estimate based on using current available RTGs
(red tape and regulations takes ~5 yrs)

Note: RFC development time reference: EKS WAG technology knowledge from POP90-2, data in JSC-24101, and based on Shuttle schedule discussion with Phil Dempsey

Note: DIPS development schedule reference: Bob Cataldo estimate 4/28/92

Note: battery development time reference: EKS WAG batteries of this type are scheduled to be delivered to USAF within a year, but some modifications can be expected based on JSC-24101 pg 3-26

Note: RTG launch safety reference: Lee Mason guesstimate

issue: JSC-24101 pg 4-11 indicates very high safety for RTGs

Note: RFC launch safety risk reference: EKS estimate based on JSC-24101 pg 4-21

Note: DIPS launch safety reference: Bob Cataldo guesstimate based on plutonium load
issue: JSC-24101 pg 3-101 tends to indicate low risk

Note: battery launch safety risk reference: EKS estimate based on JSC-24101 pg 4-3 (launched frozen)

Note: RTG operational safety reference: Lee Mason guesstimate
issue: JSC-24101 pg 4-11 indicates very high safety for RTGs

Note: RFC operations safety risk reference: EKS estimate based on JSC-24101 pg 4-21

Note: DIPS operational safety reference: Eileen Stansbery guesstimate based on JSC-24101 pg 3-101

Note: battery operations safety risk reference: EKS estimate based on JSC-24101 pg 4-3

Note: RTG contamination reference: Lee Mason guesstimate

Note: RFC contamination risk reference: EKS estimate based on JSC-24101 pg 3-32

Note: DIPS contamination reference: Eileen Stansbery guesstimate (WAG)

Note: battery contamination risk reference: EKS estimate based on JSC-24101 pg 3-23 to 3-25

Note: RTG technical complexity reference: Lee Mason estimate, JSC-24101 pg 4-8

Note: RFC technical complexity reference: EKS guesstimate based on discussion in JSC-24101 pg 3-29 to 3-35

Note: DIPS technical complexity reference: Lee Mason/Eileen Stansbery guesstimate (WAG)

Note: battery technical complexity reference: EKS guesstimate based on discussion in JSC-24101 pg 3-23 to 3-26

Note: RTG material availability reference: Lee Mason/EKS guesstimate (WAG) based on plutonium

Note: RFC material availability reference: EKS estimate, should have no trouble obtaining parts

Note: DIPS material availability reference: Lee Mason/Eileen Stansbery guesstimate (WAG) based on Pu238 production

Note: battery material availability reference: EKS estimate, should have no trouble obtaining parts or chemistry

Note: RTG lowest TRL reference: these have flown (flown = TRL9), there are 3 GPHS RTGs available

Note: RFC lowest TRL reference: Lee Mason estimate based on RFC readiness

Note: DIPS lowest TRL reference: JSC-24101 pg 4-7

Note: battery lowest TRL reference: JSC-24101 pg 3-26

Note: RTG system average TRL reference: JSC-24101 pg 4-11

Note: DIPS system average TRL reference: JSC-24101 pg 4-7 modified by Bob Cataldo for work from 1989

Note: battery system average TRL reference: JSC-24101 pg 3-26

Note: RTG disposal complexity reference: EKS guesstimate due to isolation requirements

Note: RFC disposal complexity reference: JSC-24101 pg 4-7

Note: DIPS disposal complexity reference: Lee Mason/Eileen Stansbery guesstimate due to isolation requirements

Note: battery disposal complexity reference: JSC-24101 pg 4-3

Note: RTG schedule risk reference: Lee Mason estimate [mature technology and designs]

Note: RFC schedule risk reference: EKS WAG based on understanding of lifetime and reliability testing from technology knowledge

Note: DIPS schedule risk reference: Bob Cataldo estimate 4/28/92 based on "fast-track" program

Note: battery schedule risk reference: JSC-24101 pg 3-26 discussion of USAF Light-Weight Aerospace Battery Program

Note: RTG cost risk reference: Lee Mason estimate [mature technology and designs]

Note: RFC cost risk reference: EKS WAG

Note: DIPS cost risk reference: Eileen Stansbery guesstimate (WAG) due to technology immaturity and plutonium purchase uncertainties

Note: battery cost risk reference: EKS WAG

performance degradation	%/yr		2			dust accumulation, thermal cycling, radiation, isotope decay
oper life	yrs	10	3	10	12	
data required	Mbps					no data available
energy	kW-hr					

Supportability						
availability factor	%	97	95	97	95	% time meeting load
deployability						
EVA manhours required for set-up	hr	0	0.5	0	0	
IVA manhours required for set-up	hr	0	0.5	0	0	
Earth support manhours required for set-up	hr	0	0.5	0	0	
equipment necessary for set-up		none	none	none	none	
operability						
EVA manhours required for operations	hr/day	0	0	0	0	
IVA manhours required for operations	hr/day	0	0.03	0	0	
Earth support manhours required for operations	hr/day	0	0.03	0	0	
health monitoring complexity		moderate	moderate	moderate	moderate	subjective value: fnc(# sensors, monitoring criticality)
maintainability						
EVA manhours required for maintenance	hr/day	0.1	0.2	0.1	0	
IVA manhours required for maintenance	hr/day	0.1	0.4	0.1	0	
Earth support manhours required for maintenance	hr/day	0.1	0.4	0.1	0	
equipment necessary for maintenance		supporting	supporting	supporting	none	
maintenance complexity		low	moderate	low	none	subjective value: fnc(item type)
resource consumption						
spares/repair items	kg	0	64	0	0	
consumables	kg	0	0	0	0	
storage needs	cu. m					
resupply frequency	#/yr	0.1	0.3	0.1	0.08	
resupply integration complexity		moderate	low	moderate	high	subjective value: fnc(item type)

Cost

Development costs	$M	120	0	0	0	development of DIPs must be accounted for in the baseline
Production costs	$M	39	10.8	39	188	DIPs and RTG values include plutonium
replacement unit cost	$M		10.8			

Risk

Safety					
launch safety	mod/high	low	high	very high	high >> high risk relating to launch safety concerns
operational safety	low/mod	moderate	low/mod	moderate	
environment contamination	moderate	low	moderate	low	
Technical Risk					
complexity	moderate	mod/high	moderate	low	
material availability	high	none	high	high	
TRL of least mature component	4	3	4	9	
average TRL	5	4	5	9	
decommissioning/disposal complexity	low/mod	none	low/mod	low/mod	
Schedule risk	mod/high	moderate	mod/high	low	
Cost risk	high	moderate	high	low	

Evolution capability

not currently implemented

Note: r . / cryo RFC type: SSF derived tracking PV array attached to the top o. the habitat, cryogenic reactant storage (same type as habitat)

Note: DIPS type: plutonium-fueled GPHS with CBC (closed Brayton cycle) conversion on a trailer/cart

Note: RTG type: plutonium fueled GPHS linked to Si-Ge thermoelectrics

Note: PV/cryo RFC mass reference: LeRC's FLO Sizing description 3/17/92

Note: DIPS mass reference: Lee Mason algorithm DIPS(kg) = 114 * P(kW) + 125

Note: RTG mass reference: Lee Mason algorithm RTG(kg) = 224 * P(kW) + 0

Note: PV/cryo RFC volume reference: LeRC's FLO Sizing description 3/17/92 (for gaseous) divided by 2 (for approximate cryo equivalence)

Note: DIPS volume reference: data from Bob Cataldo 4/28/92

Note: DIPS volume reference: data from Bob Cataldo 4/28/92

Note: PV/cryo RFC power reference: Lee Mason/Eileen Stansbery rounded sum of science power users

Note: DIPS power reference: Lee Mason/Eileen Stansbery rounded sum of science power users

Note: RTG power reference: Lee Mason/Eileen Stansbery rounded sum of science power users

Note: PV/cryo RFC performance degradation reference: JSC-24101 pg 3-5

Note: PV/cryo RFC operating life reference: JSC-24101 pg 3-51

Note: DIPS operating life reference: Eileen Stansbery/Lee Mason estimate based on JSC-24101 pg 4-11

Note: RTG operating life reference: JSC-24101 pg 4-11

Note: PV/cryo RFC availability factor reference: Lee Mason geusstimate

Note: DIPS availability factor reference: JSC-24101 pg 3-111

Note: RTG availability factor reference: Lee Mason geusstimate

Note: PV/cryo RFC EVA deploy reference: EKS estimate based on deploying radiator panels

Note: DIPS EVA deploy reference: Eileen Stansbery guesstimate based on integrated design assumption

Note: RTG EVA deploy reference: EKS WAG based on integrated design

Note: PV/cryo RFC IVA deploy reference: EKS WAG monitor (real-time) all EVA maintenance

Note: DIPS IVA deploy reference: Eileen Stansbery guesstimate based on integrated design assumption

Note: RTG IVA deploy reference: EKS WAG based on integrated design

Note: PV/cryo RFC Earth support deploy reference: EKS assumption of duplication of IVA monitor & control to aid in evolution to 100% IVA

Note: DIPS Earth support deploy reference: Eileen Stansbery guesstimate based on integrated design assumption

Note: RTG Earth support deploy reference: EKS WAG based on integrated design

Note: PV/cryo RFC deploy equipment reference: EKS WAG based on assuming integrated design

Note: DIPS deploy equipment reference: Eileen Stansbery guesstimate based on integrated design assumption

Note: RTG deploy equipment reference: EKS WAG based on integrated design

Note: PV/cryo RFC EVA operations reference: EKS guesstimate based on judgement that EVA is not necessary for basic operations

Note: DIPS EVA operations reference: EKS guesstimate based on judgement that EVA is not encouraged around nuclear devices

Note: RTG EVA operations reference: EKS estimate based on JSC-24101 pg 4-11

Note: PV/cryo RFC IVA operations reference: EKS guesstimate based on 2 minutes monitor & control per day for basic operations (slightly higher for cryo storage than for high press)

Note: DIPS IVA operations reference: EKS WAG assuming no monitor of EVA activation activity

Note: RTG IVA operations reference: EKS estimate based on no monitor & control necessary for normal operations

Note: PV/cryo RFC Earth support operations reference: EKS assumption of duplication of IVA monitor & control to aid in evolution to 100% IVA

Note: DIPS Earth support operations reference: EKS SWAG assuming no incidental monitoring and control

Note: RTG Earth support operations reference: EKS estimate based on no monitor & control necessary for normal operations

Note: PV/cryo RFC health monitoring reference: EKS WAG based on relative complexity of tracking PV and cryo storage devices (more complex than high press)

Note: DIPS health monitoring reference: EKS WAG based on relative complexity and criticality of nuclear devices

Note: RTG health monitoring reference: EKS WAG (simpler than DIPS, ~RFC)

Note: PV/cryo RFC EVA maintenance reference: EKS guesstimate based on Fisher-Price database*2 (rounded up due to reliquifaction) + 8 man-hrs for dusting in 45 day mission

Note: DIPS EVA maintenance reference: EKS estimate based on no failed hardware + 4 man-hrs for dusting in 45 day mission (easier config than PV on habitat)
issue: assumption of no hardware maintenance and repair is invalid and must be updated

Note: RTG EVA maintenance reference: EKS estimate based on JSC-24101 pg 4-11

Note: PV/cryo RFC IVA maintenance reference: EKS WAG double database EVA time to estimate planning + monitor (real-time) all EVA maintenance

Note: DIPS IVA maintenance reference: EKS WAG double database EVA time to estimate planning + monitor (real-time) all EVA maintenance

Note: RTG IVA maintenance reference: EKS estimate based on JSC-24101 pg 4-11

Note: PV/cryo RFC Earth support maintenance reference: EKS assumption of duplication of IVA monitor & control to aid in evolution to 100% IVA

Note: DIPS Earth support maintenance reference: EKS assumption of duplication of IVA monitor & control to aid in evolution to 100% IVA

Note: RTG Earth support maintenance reference: EKS estimate based on JSC-24101 pg 4-11

Note: PV/cryo RFC maintenance equipment type reference: EKS WAG based on probability of electrostatic device (supporting tool) rather than wisk broom (simple tool)

Note: DIPS maintenance equipment type reference: EKS WAG based on probability of electrostatic device (supporting tool) rather than wisk broom (simple tool)

Note: RTG maintenance equipment type reference: EKS estimate based on JSC-24101 pg 4-11

Note: PV/cryo RFC maintenance complexity reference: EKS WAG based on cryo reactant system more complex than high press; JSC-24101 pg 3-47

Note: DIPS maintenance complexity reference: EKS WAG based on simple dusting

Note: RTG maintenance complexity reference: EKS estimate based on JSC-24101 pg 4-11

Note: RFC spare reference: Lee Mason guesstimate
of electrolizer and stack

Note: DIPS spare reference: Lee Mason guesstimate
of no replacement necessary
issue: update for 10 year life

Note: RTG spare reference: EKS guesstimate
of no replacement necessary
issue: update for 12 year life

Note: RFC consumable reference: Lee Mason/EKS guesstimate due to absence of data (assumes no leakage)

Note: DIPS consumables reference: Lee Mason guesstimate based on design of GPHS

Note: RTG consumables reference: EKS guesstimate based on design of GPHS (no consumables used)

Note: RFC resupply frequency reference: Lee Mason/EKS guesstimate anticipating replacement needed once every three years based on operating life

Note: DIPS resupply frequency reference: Lee Mason/EKS guesstimate anticipating hardware replacement needed once every ten years

Note: RTG resupply frequency reference: EKS guesstimate anticipating hardware replacement needed once every twelve years

Note: RFC resupply complexity reference: EKS WAG anticipating accessible RFC design

Note: DIPS resupply complexity reference: EKS guesstimate based on transferring plutonium to new hardware

Note: RTG resupply complexity reference: EKS guesstimate based on transferring plutonium to new hardware

Note: PV/cryo RFC development cost reference: bookkept by habitat in this option

Note: baseline development cost reference: Lee Mason/Bob Cataldo cost estimate (info from DOE puts it between 120 and 150) - bookkept by habitat for this option

Note: RTG development cost reference: EKS guesstimate RTGs are very common therefore no development necessary

Note: PV/cryo RFC production cost reference: RFC - Jim Lee cost estimate 3/25

Note: DIPS production cost reference: algorithm from Bob Cataldo $M = 34.8 * P(kWe) + 4
issue: Kelley Cyr cost estimate ($32M)

Note: RTG production cost reference: Bob Cataldo algorithm $M = 187.6 * P(kW)
issue: Kelley Cyr cost estimate was $43M

Note: PV/cryo RFC replacement cost reference: EKS assumption of replacing entire RFC unit; RFC - Jim Lee cost estimate 3/25

Note: PV/cryo RFC development time reference: EKS estimate based on POP 90-2 T/AD data

Note: DIPS development schedule reference: Bob Cataldo estimate 4/28/92

Note: RTG development schedule reference: Bob Cataldo estimate based on using current available RTGs
(red tape and regulations takes ~5 yrs)

Note: PV/cryo RFC launch safety risk reference: EKS estimate based on JSC-24101 pg 4-21

Note: DIPS launch safety reference: Lee Mason guesstimate
issue: JSC-24101 pg 3-101 tends to indicate low risk

Note: RTG launch safety reference: Lee Mason guesstimate
issue: JSC-24101 pg 4-11 indicates very high safety for RTGs

Note: PV/cryo RFC operational safety risk reference: Bob Cataldo estimate 4/28/92 (EKS estimate of low based on JSC-24101 pg 4-21)

Note: DIPS operational safety reference: Eileen Stansbery guesstimate based on JSC-24101 pg 3-101

Note: RTG operational safety reference: Lee Mason guesstimate
issue: JSC-24101 pg 4-11 indicates very high safety for RTGs

Note: PV/cryo RFC contamination risk reference: EKS estimate based on JSC-24101 pg 4-21

Note: DIPS contamination reference: Eileen Stansbery guesstimate (WAG)

Note: RTG contamination reference: Lee Mason guesstimate

Note: PV/cryo RFC technical complexity reference: EKS estimate based on tracking mechanisms for PV and cryogenic RFC in JSC-24101 pg 3-5, 3-47 (cryo RFC is more complex than high press. RFC)

Note: DIPS technical complexity reference: Lee Mason/Eileen Stansbery guesstimate (WAG)

Note: RTG technical complexity reference: Lee Mason estimate, JSC-24101 pg 4-8

Note: PV/cryo RFC material availability reference: EKS estimate, should have no trouble obtaining parts

Note: DIPS material availability reference: Lee Mason/Eileen Stansbery guesstimate (WAG) based on plutonium

Note: RTG material availability reference: Lee Mason/EKS guesstimate (WAG) based on plutonium

Note: PV/cryo RFC lowest TRL reference: JSC-24101 pg 3-53

Note: DIPS lowest TRL reference: JSC-24101 pg 4-7

Note: RTG lowest TRL reference: these have flown (flown = TRL9), there are 3 GPHS RTGs available

Note: PV/cryo RFC average TRL reference: PV is 6 (JSC-24101 pg 3-5) while RFC is ~4 (JSC-24101 pg 3-53) but cryo systems are less mature than high pressure RFC

Note: DIPS system average TRL reference: JSC-24101 pg 4-7 modified by Bob Cataldo for work from 1989

Note: RTG system average TRL reference: JSC-24101 pg 4-11

Note: PV/cryo RFC disposal complexity reference: JSC-24101 pg 4-21

Note: DIPS disposal complexity reference: Lee Mason/Eileen Stansbery guesstimate due to isolation requirements

Note: RTG disposal complexity reference: EKS guesstimate due to isolation requirements

Note: PV/cryo RFC schedule risk reference: EKS guesstimate based on lifetime testing of RFCs and cryo development

Note: DIPS schedule risk reference: Bob Cataldo estimate 4/28/92 based on "fast-track" program

Note: RTG schedule risk reference: Lee Mason estimate [mature technology and designs]

Note: PV/cryo RFC cost risk reference: EKS WAG

Note: DIPS cost risk reference: Eileen Stansbery guesstimate (WAG) due to technology immaturity and plutonium purchase uncertainties

Note: RTG cost risk reference: Lee Mason estimate [mature technology and designs]

2. Mission Design & Analysis

Abstract

First Lunar Outpost (FLO) Conceptual Flight Profile

This section of the report contains two conceptual flight profiles (CFPs) for the First Lunar Outpost (FLO) study. These CFPs were developed by the Systems Engineering Division in support of the FLO study and the Exploration Programs Office (EXPO) at the Johnson Space Center (JSC).

Included are flight profiles for the initial cargo flight that delivers the lunar habitat for FLO and the subsequent piloted flight that delivers the first crew to the outpost and returns them to Earth. These flight profiles contain trajectory and flight-specific information portraying typical events, maneuvers, and associated trajectory parameters for FLO missions. Also, the flight profile design and rationale are captured, with supporting data and information, in Appendix C. Currently, flight profiles are described for an outpost at Mare Smythii, which is an equatorial eastern limb lunar site. A future version will contain flight profiles for an outpost at Aristarchus, which is a near-side mid-latitude lunar site. Discussions of abort strategies will also be provided in a future version of this document.

Questions concerning this section should be directed to XE/Kent Joosten (713/283-5467) or XI/Al DuPont (713/283-5358).

JSC-25880

First Lunar Outpost (FLO) Conceptual Flight Profile

Engineering Directorate
Systems Engineering Division

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NASA
National Aeronautics and
Space Administration

Lyndon B. Johnson Space Center

JSC-25880

SPACE SHUTTLE PROGRAM

First Lunar Outpost
Conceptual Flight Profile

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Acknowledgement

This Conceptual Flight Profile document is intended to describe and define the various flight phases for a typical mission of the First Lunar Outpost Study. These flight profiles were developed by a team in the Systems Engineering Division, which included support contractors from McDonnell Douglas Space System Company–Houston Division. Special thanks are extended to the contributors for their time and expertise in the development of this document. Thanks are also extended to the members in the Exploration Program Office for their guidance.

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ACRONYMS

ACRV	assured crew return vehicle
CFP	conceptual flight profile
EI	entry interface
ETO	Earth to orbit
EVA	extravehicular activity
ExPO	Exploration Program Office (JSC)
FLO	First Lunar Outpost
FLORG	First Lunar Outpost Requirements and Guidelines
FY	fiscal year
GMT	Greenwich mean time
Isp	specific impulse
KSC	Kennedy Space Center
LAN	longitude of ascending node
LEO	low Earth orbit
LM	Apollo lunar module
LOI	lunar orbit insertion
MET	mission elapsed time
n. mi.	nautical mile
SEI	Space Exploration Initiative
STS	Space Transportation System
TBD	to be determined
TEI	trans-Earth injection
TLI	translunar injection

SECTION 1
SUMMARY

This document contains two conceptual flight profiles (CFPs) for the First Lunar Outpost (FLO) Study. These CFPs were developed by the Systems Engineering Division in support of the FLO Study and the Exploration Program Office (ExPO) at the Johnson Space Center.

The objective of a CFP has been to document typical flight parameters and evaluate design requirements by putting together a complete end-to-end mission. This CFP develops concepts that the ExPO has identified into end-to-end flight scenarios. These scenarios pertain to the space transportation (trajectory) design portion of the FLO Study and portray typical or nominal flights.

Included are flight profiles for the initial cargo flight that delivers the lunar habitat for FLO and the subsequent piloted flight that delivers the first crew to the outpost and returns them to Earth. These flight profiles contain trajectory and flight-specific information portraying typical events, maneuvers, and associated trajectory parameters for FLO missions. Also, the flight profile design and rationale are captured, with supporting data and information, in appendix C. Currently, flight profiles are described for an outpost at Mare Smythii, which is an equatorial eastern limb lunar site. A future version will contain flight profiles for an outpost at Aristarchus, which is a near side mid-latitude lunar site. Discussions of abort strategies will also be provided in a future version of this document.

Guidelines used for the enclosed profiles are based on the First Lunar Outpost Requirements and Guidelines (FLORG) document, currently in review draft version 1.0. These assumptions and guidelines were made for the purpose of expediting this study. However, the ExPO recognizes that further trade studies are needed before these assumptions and guidelines can be developed into requirements.

Analyses of these CFPs will provide insight into deriving and developing mission-driven requirements for the FLO Study.

SECTION 2
INTRODUCTION

2.1 Background

The Space Exploration Initiative (SEI) encompasses a series of exploration missions that seeks to achieve the goal of expanding human presence to the Moon and Mars. FLO is a strategy that provides the functional capabilities required for the human exploration missions that will lead to the achievement of expanding human presence on the Moon. The FLO mission establishes a human-tended, exploration capability on the lunar surface. Developed by the ExPO from a "lunar campsite" strategy, FLO utilizes the results of studies dating from the 90-Day Study and the Synthesis Report.

Section 2.4, Overview of First Lunar Outpost Mission, includes the goals, objectives, and architecture of the FLO Study. The FLO strategy uses a pre-integrated, man-tended outpost landed on an unmanned cargo lander requiring minimum crew setup or assembly. The outpost provides facilities for up to a 45-day stay by a crew of four and will allow for complementary human and robotic surface activities.

The FLO Study was initiated in December 1991 as a definition project to develop a mature concept that could be submitted into the FY93 budget cycle. ExPO established six study teams for the FLO Study, and this document was initiated in support of the Mission Design and Analysis Team. The ultimate goal of the FLO Study is to develop a reference mission including element concepts and operations plans, schedules, costs, and a set of requirements.

2.2 Purpose and Scope

CFPs provide trajectory and flight-specific information portraying typical events, maneuvers, associated trajectory parameters, and design rationale. This information can then be used to develop and evaluate mission-driven requirements or guidelines, to identify issues impacting vehicle design, and to verify that the mission can be put together in an integrated end-to-end fashion.

These descriptions involve transfer vehicles traveling from Earth orbit to the lunar surface and then returning to Earth. Scenarios are developed that assume certain vehicle concepts; however, the vehicles are not described herein. These vehicle descriptions can be found in the FLO Vehicle Design report being generated by the Space Transportation and Lunar Lander Team.

Flights described in this document detail typical (or nominal) flight profiles for the FLO Study. Therefore, the trajectory parameters are not necessarily the worst-case values and should not be used as design drivers. While these flight profiles are for nominal missions, abort considerations and discussions will be included in a future version of this document. Lunar surface activities are not covered in this CFP. Also, only a brief discussion is given for the Earth to orbit (ETO) launch segment.

The flight profiles describe the cargo and piloted flights for the outpost mission. The candidate lunar site used for this CFP is Mare Smythii, which is on the equatorial east limb located 1.7° north and 85.8° east. Aristarchus, a near side mid-latitude site (23° north and 48° west), is intended for a future version of this document. These sites were chosen to demonstrate both the similarities and differences experienced when going to different lunar locations.

2.3 Document Layout

This CFP discusses (sec. 3) the FLO mission in general, the groundrules used for the study, and the scenarios for the cargo and piloted flights. Also discussed in section 3 are the mission phases, which are common for the cargo and piloted flights. For each mission phase, the flight design logic and rationale are presented. The flight profile is described in detail in section 4: the overall flight description, the flight profile sequence of major events, and the mission phase data. These sections contain pertinent data and charts.

Assumptions and guidelines used to develop the flight design profiles of these missions are discussed in appendix C.

2.4 Overview of First Lunar Outpost Mission

The reference mission, identified by the ExPO in the FLORG document, describes an automated delivery of the lunar outpost, activation and deployment of the outpost, arrival of the first crew, and science and technology surface activities. However, before the reference mission is outlined, it is necessary to discuss the background and the development of the objectives that led to this particular choice of reference mission.

In any program, it is important to establish the reasons for undertaking the project. For FLO, four main goals have been detailed: (1) to establish a human-tended exploration capability on the lunar surface; (2) to conduct important scientific experiments and prepare for future science experiments; (3) to demonstrate extraction and use of local materials on the Moon; and (4) to expand and enhance the capabilities of humans on the Moon.

The FLO concept that the ExPO developed includes these primary objectives: to enable the crew to return to Earth at any time during the mission; to have global access to the lunar surface; to maximize the mass of useful payload to the lunar surface; and to minimize the number of elements in the mission. The need to provide abort capability at all times, or the ability to leave the lunar surface and return to Earth quickly, is driven by crew safety. The objective to access any latitude and longitude on the lunar surface (i.e., global access of the lunar surface) is viewed as a necessity, since it is not known which experiments will be the most valuable or where to find useful materials on the Moon (to accomplish the goals 2 and 3 above).

The philosophy of ExPO is to keep the cost of FLO down, yet strive to do it better, at less expense, and more quickly. A program that will take 5 years from startup to operations is considered not only feasible and attractive, but essential in gaining budget acceptance. It will also accomplish a return to the Moon by the end of the decade. The desire to keep the cost of the program down led to the objective of minimizing the number of vehicle elements in the program. Crew modules or pressure vessels are considered large drivers in program cost. Therefore, it is desirable to have a single pressure vessel on the piloted flight.

The number of ETO launch vehicles needed will be a driver to the mission definition, since it is desirable to minimize the amount of on-orbit assembly of the FLO elements. The "lunar campsite" strategy, from which FLO was developed, uses a split-mission architecture. A cargo flight will deliver the habitat, and a subsequent piloted flight will deliver the crew. Preliminary studies at ExPO suggested that a heavy-lift launch vehicle in the 200 mt class will deliver the target mass to the lunar surface for the FLO Study (about 27 mt), allowing a single ETO launch for both the cargo and piloted flights. A single ETO launch will avoid the complexities of on-orbit rendezvous and assembly and reduce launch rate concerns at the launch site.

Combining the primary objectives of anytime return and global lunar access, along with the desire for a single crew module and minimum orbital operations (rendezvous and docking), led to the choice of a direct mission mode to the lunar surface. Direct mode means that the vehicle will descend to the lunar surface with its return stage and propellant. While the penalty of a direct mode will show up in the amount of useful payload to the lunar surface, it was chosen over other modes involving rendezvous because of its ability to meet the other objectives and the perception of increased probability of mission success.

Another attractive feature of the "lunar campsite" strategy is that the habitat would be delivered as a pre-integrated structure requiring no construction and minimizing the amount of lunar surface activities needed to prepare the outpost for occupancy. Previous studies have highlighted the problem of orbital alignment with lunar transfer and return vehicles when using the Space Station Freedom (SSF). Also, the need to reduce the amount of on-orbit assembly and to minimize orbital operations would negate any advantage of using the SSF.

The reference mission described in the FLORG document is accomplished via a split mission profile. The lunar habitat will be delivered to the landing site by a separate cargo flight and will be checked out remotely before the crew leaves Earth on the piloted flight. Each flight will be accomplished by a heavy-lift launch vehicle capable of injecting a fully integrated cargo or manned spacecraft into a translunar trajectory with a single launch.

The piloted round-trip flight will not involve rendezvous in lunar orbit. The complete manned spacecraft will descend to the lunar surface, thus avoiding the need for two separate pressure vessels and facilitating anytime return to the Earth. The surface stay time will be up to 45 days. This is a lunar day-night-day cycle. For this stay time, required consumables to support a crew of four were considered achievable within the target mass to the lunar surface.

Return to Earth will be accomplished with an Apollo-type atmospheric entry module, which will be used for a direct entry and landing. An Apollo-derived atmospheric entry module will reduce development costs. Currently, land-landing at Earth return is being considered as a guideline for the FLO mission. While issues have emerged with this guideline, it is the option used for this study.

SECTION 3
FIRST LUNAR OUTPOST MISSION

From a mission design perspective, the basic requirements for the FLO Mission are: (1) The transportation system must be designed to provide access to any arbitrary point on the lunar surface. (2) In the interest of economy, the transportation system will include only a single manned pressure vessel. (3) The flight crew will stay on the lunar surface a maximum of 45 days. (4) At any arbitrary lunar lift-off time, the flight crew must have the option of returning to a selected Earth landing site on land within 5 days. These "groundrules and guidelines" (as they are called by the program office) are discussed in the next subsection, in addition to the assumptions used in the mission design and development of the flight profiles included in this document.

3.1 Groundrules and Guidelines

Following is a summary of the groundrules and guidelines found in the FLORG document that are important to the trajectory design. Appendix B contains a reprint of the applicable requirements from the FLORG document with the associated index number.

a. Assume a single launch vehicle will be utilized for each flight to the Moon. The use of a single launch vehicle for each flight reduces on-orbit operations by eliminating rendezvous and assembly. Using a single launch vehicle also increases the probability of performing translunar injection (TLI) during the first window of opportunity.

b. Whenever practical, commonality is desired for the cargo and piloted flights to reduce costs by reducing the number of elements. Individual flights will be launched by the same type of heavy-lift Earth launch vehicle with the capability to inject a fully integrated cargo or manned spacecraft into a translunar trajectory with a single Earth launch. Also, the lunar descent stage of the transportation system will be common, capable of landing the habitat or the flight crew on the surface of the Moon.

c. A single crew module (pressure vessel) will be used to transport the crew into lunar orbit, to and from the lunar surface, and back to Earth. Crew module time in transit and on orbit will be minimized. Consumables and propellant needed to support the crew while in orbit result in sizeable penalties in required mass to low Earth orbit.

d. The manned round-trip flight will not involve rendezvous in lunar orbit. The complete manned spacecraft will descend to the lunar surface, thus avoiding the need for two separate pressure vessels. Also, this direct method facilitates anytime return to the Earth, since no waiting is required for alignment of the rendezvous orbit.

e. At any arbitrary lunar lift-off time, the flight crew must have the option of returning to a selected landing site on land within 5 days. This anytime lunar lift-off provides the flight crew with an abort capability at all times following lunar landing. After abort declaration, 1 day is allowed on the surface for ascent stage preparation, and up to 4 days are allowed for the return trip. Land-landings at Earth return are presumed to cost less than water landings in terms of recovery and refurbishment of the crew module.

f. The desire for the FLO Study is to return to the Moon before the year 2000. The first launch in 1999 was chosen to drive schedules and is considered technically achievable.

g. The transportation system will be designed to access any lunar latitude and longitude. Science research and space resource utilization objectives drive this guideline. Global access is essential to provide a capability to reach areas of highest interest and greatest potential return.

h. Sixty days will elapse between cargo launches and piloted flights. This 60-day timespan reduces the dormant period of the habitat on the lunar surface and includes the turnaround time needed between launches by the Earth launch site. Also, the time allows for remote checkout of all habitat systems prior to commitment to the piloted flight.

i. Nominal outbound flight time for the piloted flight is 4 days (96 hours). This is a compromise between minimum energy requirements and minimum mass requirements based on crew cabin occupancy time.

j. Piloted lunar landings will occur soon after lunar sunrise (within 4 days). Landing at the lunar outpost near lunar dawn maximizes the time available for science and exploration on the lunar surface. Also, since the crew will walk from the spacecraft to the outpost, extravehicular activity (EVA) constraints have to be considered. Temperature extremes experienced in lunar darkness and around lunar noon (depending on the latitude of the lunar site) lead to a desire to perform EVA sometime after lunar dawn and before lunar noon.

k. The maximum lunar surface stay time in the habitat is 45 days, 3 days of which are reserved for contingencies that might delay lunar departure (e.g., solar flares). This time allows for a lunar day-night-day cycle.

l. Lunar lift-off for the return to Earth occurs prior to lunar sunset. The crew returns to the spacecraft and the associated launch operations should occur in daylight.

m. Inbound flight time for the piloted flight is 4 days ± 12 hours (84 to 108 hours). This variation in flight time allows full longitude control of the Earth landing point. Note that the maximum energy requirements, budgeted for the return trip, occur at 84 hours.

n. Earth return will be accomplished with an Apollo-type atmospheric entry module, which will be used for a direct entry and landing. A direct entry mode has a tendency to reduce the propellant required by the spacecraft. Circularization of the crew module at Earth results in a mass penalty due to the additional propellant that must be lowered into and lifted out of the lunar gravity well. Use of an Apollo-type entry module should minimize development cost.

3.2 Flight Descriptions

The FLO Mission is accomplished in two flights: (1) the automated delivery of the habitat (i.e., the cargo flight) and (2) the delivery of the crew (i.e., the piloted flight). The flight profiles for both are similar and are described in sections 3.2.1 and 3.2.2. The cargo outbound flight time is 5 days and was chosen to minimize the propulsive energy requirements. The piloted outbound flight time is 4 days, a compromise between reducing crew cabin occupancy time and minimizing propulsive energy requirements. The piloted flight has an inbound flight time of 4 days ± 12 hours (the cargo flight does not have an inbound leg).

3.2.1 Cargo flight description

The first flight in the split-mission profile is the automated delivery of the lunar habitat. All necessary hardware for the lunar outpost, including the habitat, is delivered on a single heavy-lift launch vehicle to low Earth orbit. The desired circular Earth parking orbit has an altitude of 185 km (100 n. mi.). At this point in the mission, the spacecraft consists of a TLI stage and the lunar descent stage with the cargo (lunar habitat). At the appropriate time in the parking orbit, the TLI stage executes a propulsive maneuver to place the spacecraft on a transfer trajectory to the Moon. After this maneuver, the TLI stage is separated from the rest of the spacecraft and maneuvered to be placed on a different trajectory.

The translunar coast phase takes 5 days, during which time midcourse maneuvers are executed as appropriate. Upon arrival at the Moon, the lunar descent stage executes the lunar orbit insertion (LOI) maneuver and inserts into a temporary 100 km circular lunar parking orbit. Soon after the LOI maneuver, the lunar descent stage executes a deorbit maneuver, coasts through a 181° transfer angle, and performs a powered flight descent to the landing site.

The scenario for the cargo flight is shown in figure 3-1.

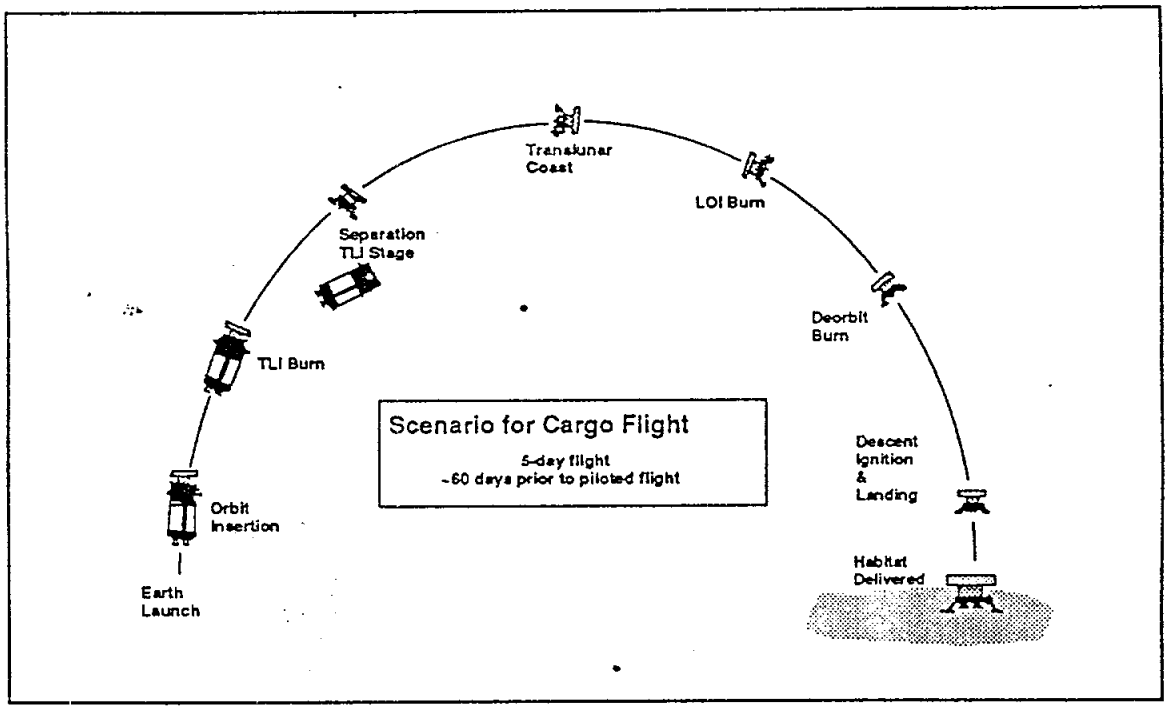


Figure 3-1. FLO: delivery of the habitat.

3.2.2 Piloted Flight Description

The piloted flight follows the cargo flight by 60 days. This flight delivers a crew of 4 to the lunar habitat and, after a 42-day surface stay, returns the crew to Earth. All the necessary hardware for this flight, along with the flight crew, is delivered on a single heavy-lift launch vehicle to low Earth orbit (185 km altitude). The spacecraft for the piloted flight consists of a TLI stage, a lunar descent stage, a crew module, and an ascent stage. The crew module is both the pressure vessel for the crew and the Apollo-style Earth return portion. At the appropriate time in the Earth parking orbit, the TLI stage executes a propulsive maneuver to place the spacecraft on a transfer trajectory to the Moon. After this maneuver, the TLI stage is separated from the rest of the spacecraft and maneuvered to be placed on a different trajectory.

The translunar coast phase lasts 4 days, during which time midcourse maneuvers are executed as appropriate. Upon arrival at the Moon, the lunar descent stage executes the LOI maneuver and inserts into a temporary 100 km circular parking orbit. Soon after the LOI maneuver, the descent stage executes a deorbit maneuver, coasts through a 181° transfer angle, and performs a powered flight descent to the landing site.

The maximum stay on the lunar surface is 45 days. When the lunar surface mission is completed, the crew leaves the habitat and enters the spacecraft. The ascent stage performs the powered flight ascent, leaving the lunar descent stage on the surface. After termination of the powered flight phase, the spacecraft coasts to an altitude of 100 km and executes the circularization maneuver. About 2 hours later, the ascent stage performs the trans-Earth injection (TEI) maneuver. The trans-Earth coast phase lasts 4 days ± 12 hours, depending on the selection of the Earth landing site. During trans-Earth coast, midcourse maneuvers are executed as appropriate. Prior to atmospheric entry, the ascent stage is separated from the crew module, which reenters the Earth atmosphere in a direct entry mode. The Earth return entry is targeted for a land-landing.

The scenario for the piloted flight is shown in figure 3-2.

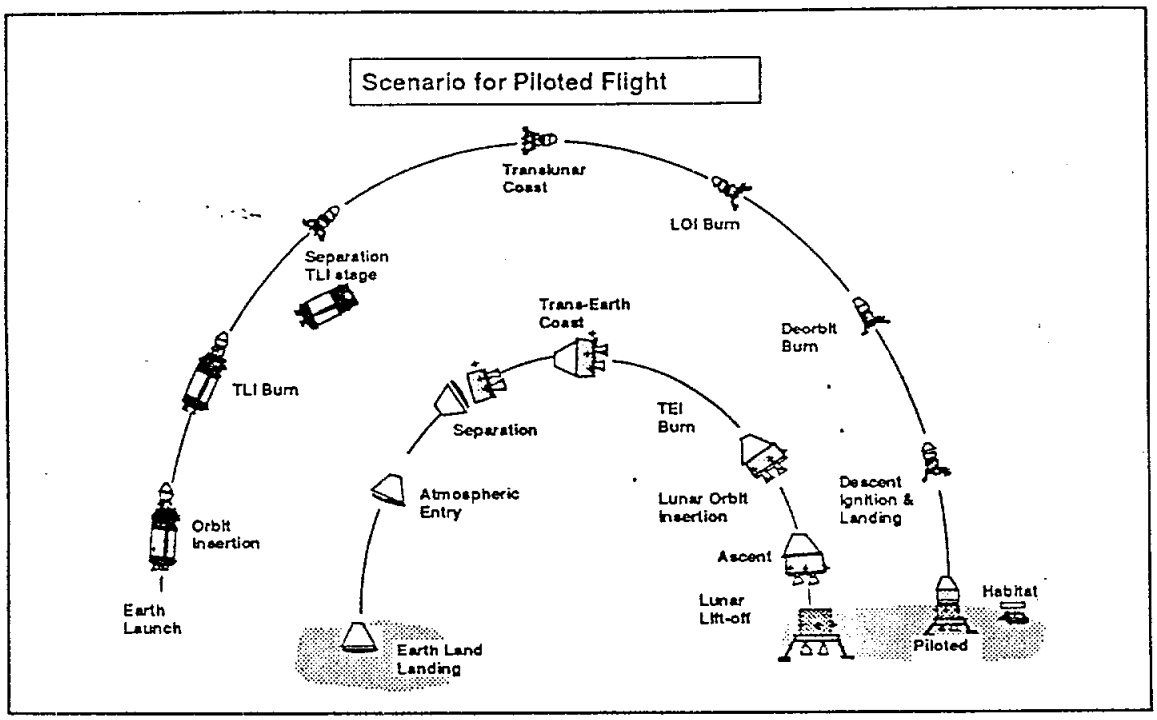


Figure 3-2. FLO: delivery of the crew.

SECTION 4
GENERIC MISSION PHASE DISCUSSION

This section contains a description of each mission phase of a flight (cargo and piloted). Also, the flight design rationale used in developing the CFPs is explained. More detailed discussion and data supporting the flight design rationale are found in appendix C.

4.1 Earth Launch Phase

The FLO elements are assumed to be launched from the Kennedy Space Center (KSC). All the elements for each flight are delivered to the Earth parking orbit by a single launch vehicle (ref: initial guidelines and groundrules from the ExPO). The use of a single launch reduces on-orbit operations by eliminating rendezvous and assembly. This increases the probability of performing TLI during the first window of opportunity. Trade studies are being performed to compare the use of a single and dual heavy-lift launch vehicle for each flight to the Moon.

On the average, during any 25-hour period and for a given range of launch azimuths from KSC, two distinct opportunities occur for a coplanar departure to the Moon. (See section 3.4 of appendix C for further discussion). During the Apollo Program, the two opportunities were characterized as "Pacific injection" and "Atlantic injection". One results in a translunar injection maneuver out of Earth orbit in the vicinity of the Pacific Ocean, and the other opportunity results in a maneuver out of Earth orbit in the vicinity of the Atlantic Ocean (fig. 31, appendix C).

To minimize propulsive energy requirements for lifting the vehicle from the Earth surface and injecting it onto the translunar trajectory, it is generally best to launch nearly due east (90° launch azimuth, direction measured from north) and insert the payload into a temporary circular parking orbit just above the atmosphere. However, a wide contiguous launch window is desirable for accommodating local weather conditions and coping with last-minute spacecraft or launch system problems. For this study, the range of launch azimuths from KSC is 72° to 108°. Although there were variations, this was the usual range of launch azimuths for Apollo missions. This range of launch azimuths yields a maximum Earth orbit inclination of about 33° with respect to the equator and a contiguous launch window width of approximately 4.5 hours for each opportunity. The 72° launch azimuth is always the first opportunity. As the launch site rotates to the east, the launch direction moves from northeast to east and to southeast until the 108° launch azimuth is encountered. If launch occurs at the opening of the launch window (72° azimuth), the inclination of the parking orbit will be 33.26°.

Usually, only one of these opportunities results in a daylight launch from KSC. If daylight launch is a constraint, using a single launch vehicle provides one daily opportunity for Earth launch. If launch is delayed past the window (e.g., due to weather), the following daily opportunities occur progressively later in the day due to the motion of the Moon antipode; also, lighting conditions (Sun angles) at lunar landing will be different.

Current FLORG guidelines specify launch of the piloted flight in 1999, with launch of the cargo flight occurring 60 days earlier. For this study, December (the last opportunity in 1999) was chosen for the piloted flight. Thus, October (60 days earlier) is the launch month for the cargo flight. The actual day of launch was selected by lighting conditions at the lunar landing site. The time of launch is determined based on desired LOI time and translunar flight time. Given a particular launch azimuth, the launch time must be selected so that the position of the Moon, advanced to LOI time, will lie in the parking orbit plane.

4.2 Earth Parking Orbit Phase

With the parking orbit determined by launch time and launch azimuth, there is a particular point in the orbit where a horizontal, in-plane, TLI will give the spacecraft the required energy and the necessary direction of motion for transfer to the Moon. The point in the orbit where this optimum impulse occurs varies with launch azimuth and with the declination of the Moon; therefore, a variable coast time in the parking orbit is essential. The longest coast time needed is equal to the period of a low-altitude Earth orbit, which is about 90 minutes. Longer coast times may be desirable to accommodate navigation updates and systems checkout prior to the TLI maneuver.

An altitude of 185 km was used for this study, primarily since this altitude is usually the lowest altitude shown in launch vehicle performance data. Thus it is considered a reasonable Earth parking orbit altitude for this study (ref: sec. 3.4, appendix C for more discussion).

The TLI maneuver is scheduled to occur at the first opportunity during the initial Earth parking orbit revolution. If TLI ignition is delayed more that a few seconds past the planned time, an expensive performance penalty makes it necessary to coast through a full revolution in the parking orbit, about 90 minutes, before trying to initiate the TLI maneuver. During this time, the Earth orbit will regress a fraction of a degree to introduce a plane change penalty. This penalty becomes prohibitive unless the transit time is shortened enough to keep the Moon position (advanced to arrival time) contained within the regressing plane of the parking orbit. (A more detailed presentation of the TLI window is given in section 3.4.1 of appendix C.)

A maximum of 4.5 hours is allowed in Earth orbit, which results in three TLI opportunities occurring during the first three Earth orbit revolutions. The 4.5 hours is based on TLI stage design limitations.

4.3 Translunar Phase

After the lunar injection maneuver, the TLI stage is separated from the rest of the lunar spacecraft. At this point the lunar spacecraft consists of either the lunar descent stage and the habitat for the cargo flight or the lunar descent stage, crew module (pressure vessel and Earth entry capsule), and ascent stage for the piloted flight. The best disposal method of the TLI stage has not been addressed. A potential recontact of the TLI stage with the rest of the lunar spacecraft or an impact of the TLI stage at an undesirable lunar surface location can occur. For a small energy cost, the TLI stage could be targeted to a gravity swingby of the Moon out of the Earth-Moon system or targeted to impact the Moon at a desired selenographic location.

The nominal translunar coast (from TLI to LOI) takes 5 days for the cargo flight and 4 days for the piloted flight. The 5-day flight time minimizes energy requirements, but the piloted flight has to include crew module mass, which increases with time due to consumables. During this time, the vehicle position and velocity are monitored, midcourse correction maneuvers are applied as necessary (a total budget of 30 m/s), and vehicle systems are checked out in preparation for the LOI maneuver.

4.4 Lunar Orbit Phase

The vehicle reaches a speed of about 2400 m/s at the point of closest approach to the Moon (perilune). A low-altitude temporary circular orbit is preferred over a direct hyperbolic descent, because the direct descent (1) generally requires a higher delta-V, (2) does not provide global access to the lunar surface, and (3) would require perilune of the approach trajectory to be located below the lunar surface for most landing sites. Item (3) is a flight safety consideration, since an

ignition failure would result in a lunar surface impact. The arrival at the Moon is discussed in section 3.3 of appendix C.

The LOI maneuver is designed to brake the vehicle into a 100 km circular lunar orbit. Low lunar orbit altitudes have lower propulsion requirements than do higher altitudes. For overall minimization of spacecraft mass, the parking orbit should be as low as possible. But considering possible navigation errors, maneuver execution errors, and orbit perturbations, a reasonable altitude margin above the lunar surface is desired. The nominal parking orbit altitude for Apollo was 108 km; currently, no altitude requirements are identified that would increase this value. For these reasons, 100 km was chosen for missions under consideration for this study.

The LOI maneuver typically consists of a single burn that simultaneously corrects the plane of the approach hyperbola and circularizes the orbit. In this study, however, the translunar trajectory was designed to be coplanar to minimize energy and mass requirements.

The deorbit maneuver is performed at the first opportunity, provided there is adequate time (at least 15 minutes) between LOI and the deorbit maneuver. If that time is insufficient, 1 revolution is added and deorbit is performed at the second opportunity.

The lunar parking orbit is temporary, usually about 0.5 revolution. For the piloted flight, however, the crew module is required to have the capability to stay in orbit for a period of 12 hours. Navigation requirements and vehicle systems checkout will occur during this time, but the primary reason for the 12-hour stay capability is to accommodate a crew sleep cycle. The possibility exists that at LOI the crew day cycle will be coming to an end. Time in lunar orbit can be spent such that, prior to the deorbit maneuver, the crew will have had an opportunity to sleep.

4.5 Lunar Descent and Landing Phase

After a sufficient wait time in the lunar parking orbit, the lunar descent stage performs a deorbit maneuver. This maneuver puts the spacecraft onto a 100 x 18.5 km transfer orbit. The spacecraft coasts through approximately a 181° transfer angle to a point just past the perilune of the transfer orbit, where it begins the powered descent to the lunar surface.

The lunar powered descent trajectory was designed to provide fuel efficiency subject to all terrain clearance constraints. The terrain clearance capability of the powered descent should be equal to or greater than that of Apollo lunar module (LM) descent. FLO descent performance was fairly insensitive to the choice of periapse altitude of the transfer orbit. A perilune altitude of 18.5 km was chosen because it provided better terrain clearance than the Apollo descent for a small performance impact. The nominal powered descent trajectory should allow for navigation state updates as well as for sufficient abort capability. The current nominal descent design has no navigation considerations incorporated. Also, while abort studies have been done, no abort requirements have been identified that would drive the powered descent trajectory design.

Lunar landing is timed to be soon after lunar sunrise (within 4 days, according to the guidelines) to maximize the crew time spent on surface science and exploration activities.

4.6 Lunar Surface Stay

Nominal lunar surface stay is 42 days for the piloted flight. The crew will land within walking distance of the habitat. The habitat will have been deployed and activated autonomously and checkout of the habitat systems will have been completed before committing to the piloted flight launch.

Return to Earth is planned to occur near the end of the second lunar day (42 days). For outpost sites on the eastern limb (i.e., Mare Smythii), leaving the Moon during this phase will result in a daylight landing on Earth. However, for other lunar sites, leaving the Moon at the end of a lunar day may result in a nighttime landing on Earth.

4.7 Lunar Ascent Phase

Lunar ascent from the outpost site begins with an Apollo-like vertical rise and follows a minimum fuel pitch profile inserting into a 100 x 18.5 km transfer orbit. The ascent trajectory provides fuel optimality subject to terrain clearance constraints. These constraints are a vertical rise to 50 fps and then the first pitch segment (10 sec) limited to a 20° pitchover from the vertical. After the powered flight phase, the spacecraft coasts to apolune where it performs an insertion maneuver to achieve a 100 km altitude circular lunar orbit. (The initial ascent from the lunar surface is detailed in section 3.2.2 of appendix C.)

4.8 Lunar Orbit Phase

After orbit insertion, wait time in lunar orbit is kept to a minimum prior to the TEI maneuver. One additional lunar orbit revolution may be required if the TEI maneuver occurs too close to the orbit insertion (due to the geometry between the TEI location and the lunar outpost site). For this study, if TEI occurred within 15 minutes of the insertion into lunar orbit, an additional revolution was added. Other considerations, such as navigation requirements or crew sleep cycles, may increase time spent in lunar orbit before the TEI maneuver.

The ascent stage reignites to execute the TEI maneuver. The trans-Earth trajectory was designed to be coplanar to minimize energy and mass requirements.

4.9 Trans-Earth Phase

The nominal trans-Earth coast (from TEI to atmospheric entry) takes 4 days ± 12 hours. This 24-hour variation in transit time allows the Earth to rotate in the inertial frame to change the longitude of the geographic touchdown point. Such a change in transit time has little effect on the solar illumination of the landing site. The local time of day at touchdown depends on the phase of the Moon at the time of lunar departure (section 3.1, figure 1, of Appendix C). Also, an anytime return requirement (i.e., the crew is forced to leave the lunar surface at any time after lunar landing) implies the necessity of night landings at Earth return for some portions of the mission. For the nominal mission (42-day surface stay), with the lunar site on the eastern limb, Earth landings will occur sometime between dawn and noon, Earth local time.

During this phase, the vehicle position and velocity are monitored, midcourse correction maneuvers are applied as necessary, and vehicle systems are checked out in preparation for atmospheric entry at Earth. Prior to atmospheric entry, the ascent stage separates from the crew module (pressure vessel and Earth return capsule), which reenters the Earth atmosphere in a direct entry mode. The Earth return entry is targeted for a land-landing.

4.10 Earth Entry and Landing Phase

The Earth return perigee point is generally located within a few degrees of the geocentric position occupied by the lunar antipode. Changing the return transit time has little effect on the geocentric transfer angle, but does allow the Earth to rotate in the inertial frame and thus change the longitude of the perigee point.

Such a change has little effect on the solar illumination of the landing site, however, because the local time of day at perigee remains the same. If an anytime abort is required from the Moon, which currently is a groundrule, this implies the necessity of anytime lighting conditions at Earth landing.

As the vehicle approaches the Earth, the ascent/return stage is separated from the crew module. The vehicle positions itself into the correct entry attitude for the atmospheric entry phase.

An Apollo-style vehicle is assumed for the atmospheric reentry and landing portion of the flight. The atmospheric flight time from entry interface to landing was derived from Apollo data. The nominal target, 2380 km (1285 n. mi.) downrange from the entry interface point, is 15 minutes away in flight time. The downrange distance is used for a Great Circle arc from the projection of the entry point on the Earth surface to the landing site.

For this study, the vacuum perigee latitude of the return trajectory determined the landing site chosen. The landing site lies within a landing footprint that ranges from 1778 km (960 n. mi.) to 3167 km (1710 n. mi.) downrange from the entry interface point. The landing sites were chosen from an assured crew return vehicle (ACRV) landing study that resulted in a set of landing sites spread throughout the globe. The sites chosen were just examples used for this CFP.

SECTION 5
FLIGHT PROFILE DATA FOR CANDIDATE LUNAR SITES
MARE SMYTHII

5.1 Cargo Flight to Mare Smythii

5.1.1 Flight description

The cargo flight, which delivers the habitat for the FLO, consists of the TLI stage, the lunar descent stage, and the cargo (habitat). Launch occurs on October 5, 1999, at 9:30 Greenwich mean time (GMT). This is before sunrise at KSC; if a daylight launch is required, then launch would occur a few days later. A single heavy-lift launch vehicle delivers the spacecraft to an Earth parking orbit at an altitude of 185 km (100 n. mi.) and at an inclination of 33.26°. The sequence of major events for the lunar outpost delivery flight appears in table 5-I.

At 10:51 GMT (1 hour and 21 minutes after launch), the TLI stage executes the TLI maneuver. The impulsive delta-V for this burn is 3140 m/s, which places the spacecraft on a minimum energy trajectory to the Moon. Soon after the lunar transfer maneuver, the TLI stage is separated from the spacecraft, leaving the lunar descent stage and the cargo (outpost habitat) on the translunar trajectory. The translunar coast phase takes 5 days, during which midcourse maneuvers are executed as appropriate. Approximately 30 m/s delta-V is budgeted for these maneuvers.

Upon arrival at the Moon, the descent stage executes the LOI maneuver on October 10, at 10:51 GMT (5 days, 1 hour, 21 minutes mission elapsed time (MET)). The LOI maneuver inserts the spacecraft into a temporary 100 km circular lunar parking orbit at an inclination of 41.1° and a longitude of ascending node of 92.3° west (selenographic coordinates).

Approximately 1 hour after the LOI maneuver, the descent stage executes the deorbit maneuver (19 m/s delta-V at 11:49 GMT), coasts through a 181° transfer angle, and performs a powered flight descent (1853 m/s) to the landing site (Mare Smythii). Landing of the outpost habitat occurs at 12:53 GMT on October 10, 1999 (5 days, 3 hours, 23 minutes MET).

Table 5-I. Sequence of Major Events for the Lunar Outpost Delivery Flight

Event	GMT hr:min	MET day\hr:min	Data Summary	
Earth Launch*	1999 Oct 5 9:30	0\00:00	Launch azimuth, deg	72 †
Earth Orbit Insertion	--	--	Altitude, km Inclination, deg	185 33.26
Translunar Injection (TLI)	10:51	0\01:21	ΔV, mps	3140
Translunar Coast Midcourse Corrections	--	--	Duration, days ΔV, mps	5 ~30
Lunar Orbit Insertion (LOI)	1999 Oct 10 10:51	5\01:21	ΔV, mps	809
Lunar Parking Orbit	--	--	Altitude, km Inclination, deg LAN, deg Wait time, min	100 41.1 -92.3 58
Deorbit	11:49	5\02:19	ΔV, mps	19
Landing Ignition	12:46	5\03:16		
Lunar Landing	1999 Oct 10 12:53	5\03:23	ΔV, mps	1853

*Determination of lift-off times uses a model of powered flight ascent arc. Accuracy of lift-off time is within ±5 minutes

† Impulsive inertial azimuth

5.1.2 Earth launch phase

The nominal Earth launch azimuth (direction) from KSC is 72° east of north. This is the opening of the launch window (approximately 4 hours and 30 minutes long) and occurs on October 5, 1999, at 9:30 GMT. The date was selected by backing off 60 days from the piloted flight. (This rationale was discussed in the previous section.) Coordinates for KSC are geodetic latitude 28.6° and longitude 279.4°.

5.1.3 Earth orbit phase

Earth parking orbit insertion occurs at an altitude of 185 km with an inclination of 33.26°. The inclination is a result of the 72° launch azimuth from KSC.

The TLI maneuver (impulsive delta-V of 3140 m/s) occurs at 10:51 GMT or 1 hour and 21 minutes MET. This is the first opportunity of the TLI window. There are two more opportunities, each a revolution apart.

Maneuver information for the cargo flight to Mare Smythii is shown in the following table. These data are for an impulsive TLI without gravity loss.

Table 5-II. Mare Smythii Cargo Flight TLI Maneuver Parameters

TLI Departure Date/Time	Oct 5, 1999 / 10:51 GMT
Delta-V/Plane Change	3140 m/s / 0 °
TLI Latitude/Longitude	6.4 ° / -145.6 °

TLI maneuver geometry is shown in figure 5-1. The altitude, velocity, latitude, and longitude parameters for the impulsive maneuver are listed in the TLI box. Also shown is the location of the Moon at TLI with the associated Sun direction. In figure 5-2, the altitude of the spacecraft is plotted for the first 12 hours after TLI. Also shown are the geodetic latitude and longitude of the spacecraft.

5.1.4 Translunar phase

After the lunar transfer maneuver, the TLI stage separates from the spacecraft (i.e., lunar descent stage and the cargo). The spacecraft continues on the nominal translunar coast phase (from TLI to LOI), which takes 5 days. During this time, the vehicle position and velocity are monitored, midcourse correction maneuvers are applied as necessary (total budgeted 30 m/s), and vehicle systems are checked out in preparation for LOI.

In figure 5-3, the vehicle position during the translunar coast is plotted in geodetic latitude versus longitude coordinates. The groundtrack of the translunar trajectory overlayed on the continents is depicted, with the TLI and LOI positions. In figure 5-4, the distance from the Earth to the spacecraft is plotted out to 120 hours. Also shown are the geodetic latitude and longitude.

The next two figures (figs. 5-5 and 5-6) pertain to elevation angles above the Earth limb and the Moon limb. The first chart depicts the solar elevation angle above the Earth limb. This angle was obtained using position vectors for the vehicle and the Sun. The Earth was assumed to be spherical when computing the angular location of the respective limb. The diagram associated with the solid line in figure 5-5 illustrates this geometry. The elevation angles for the Sun angle above

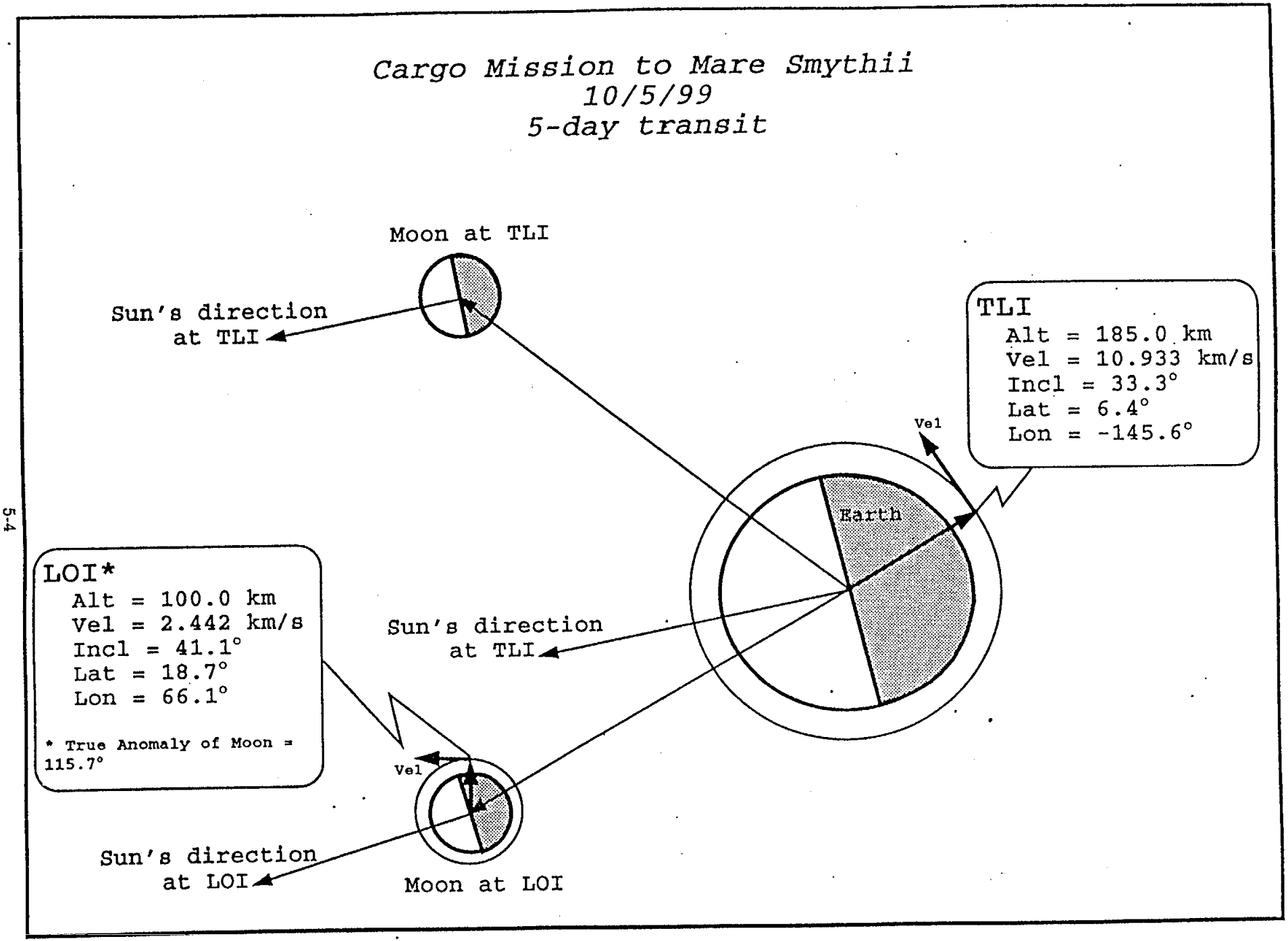


Figure 5-1. TLI and LOI maneuver geometry.

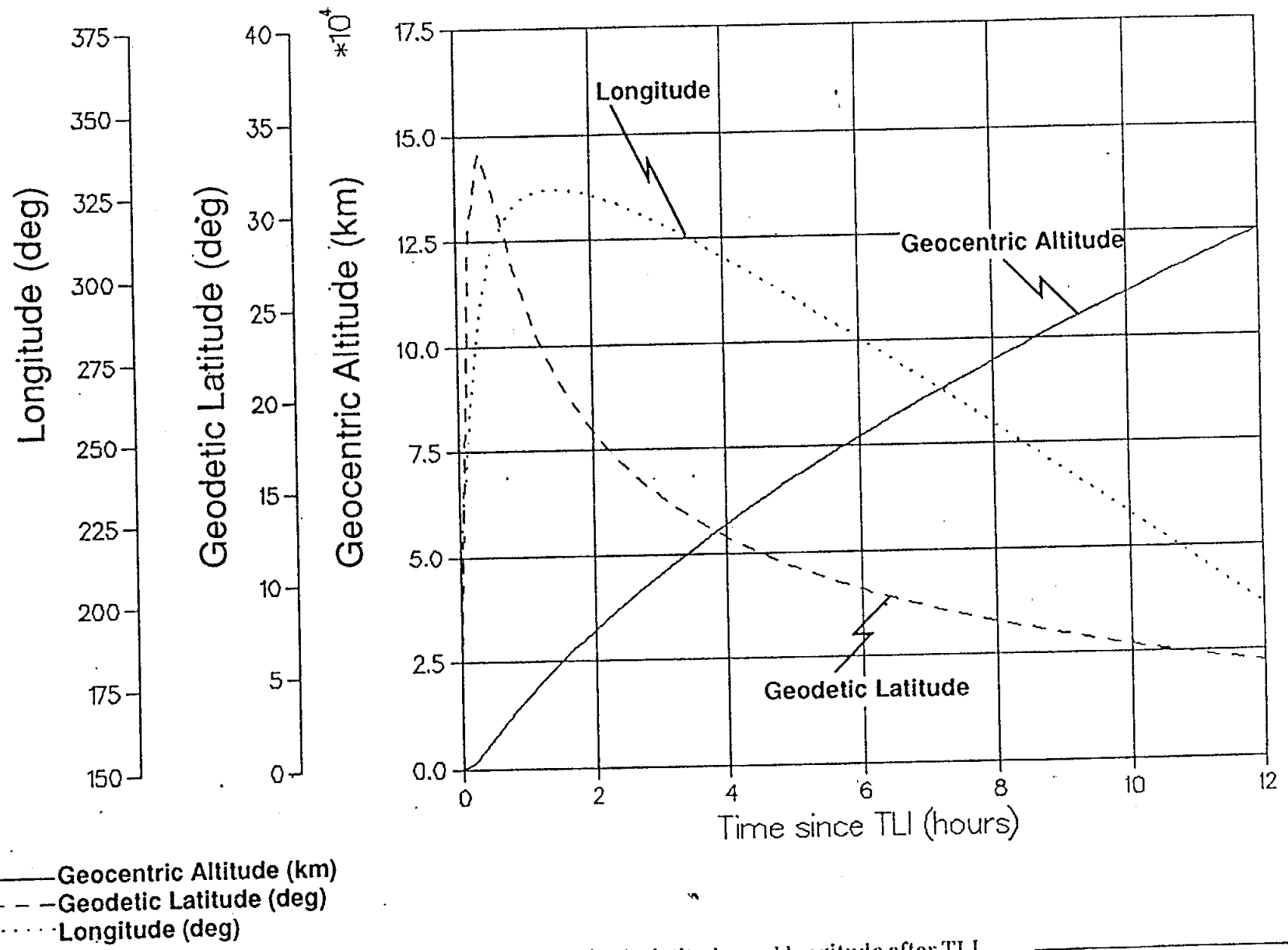


Figure 5-2. Spacecraft altitude, latitude, and longitude after TLI.

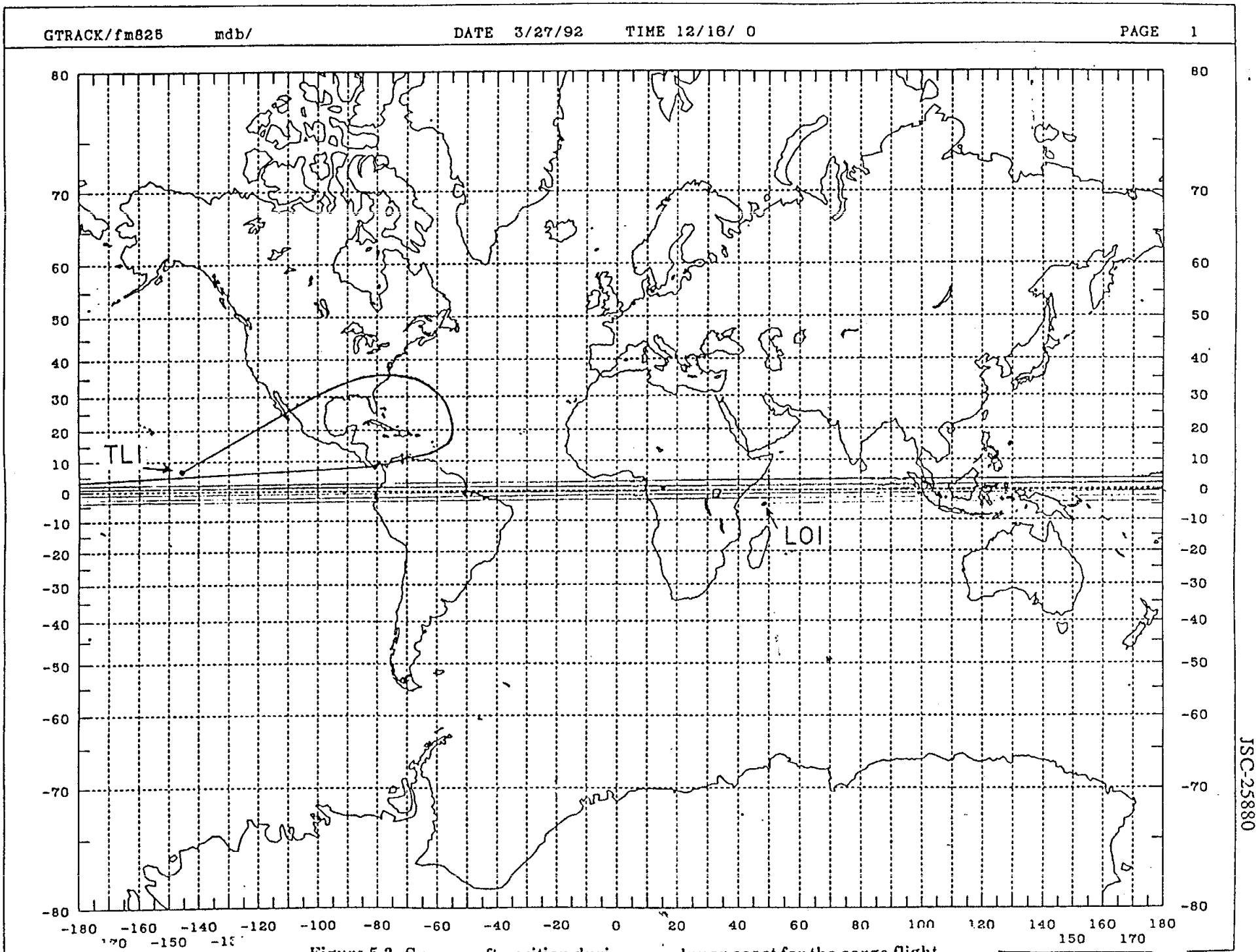


Figure 5-3. Spacecraft position durin nslunar const for the cargo flight.

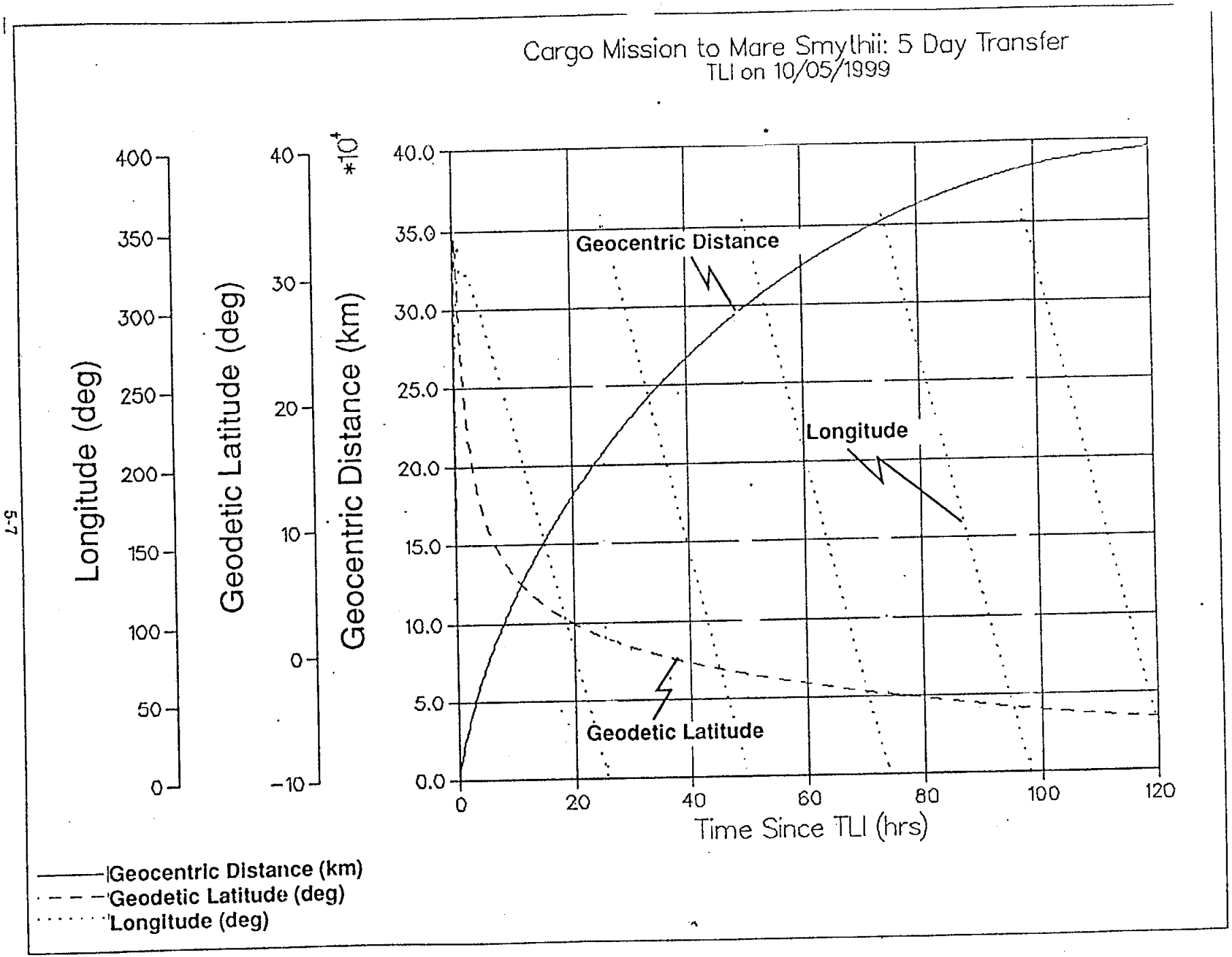


Figure 5-4. Spacecraft distance, latitude, and longitude during translunar coast.

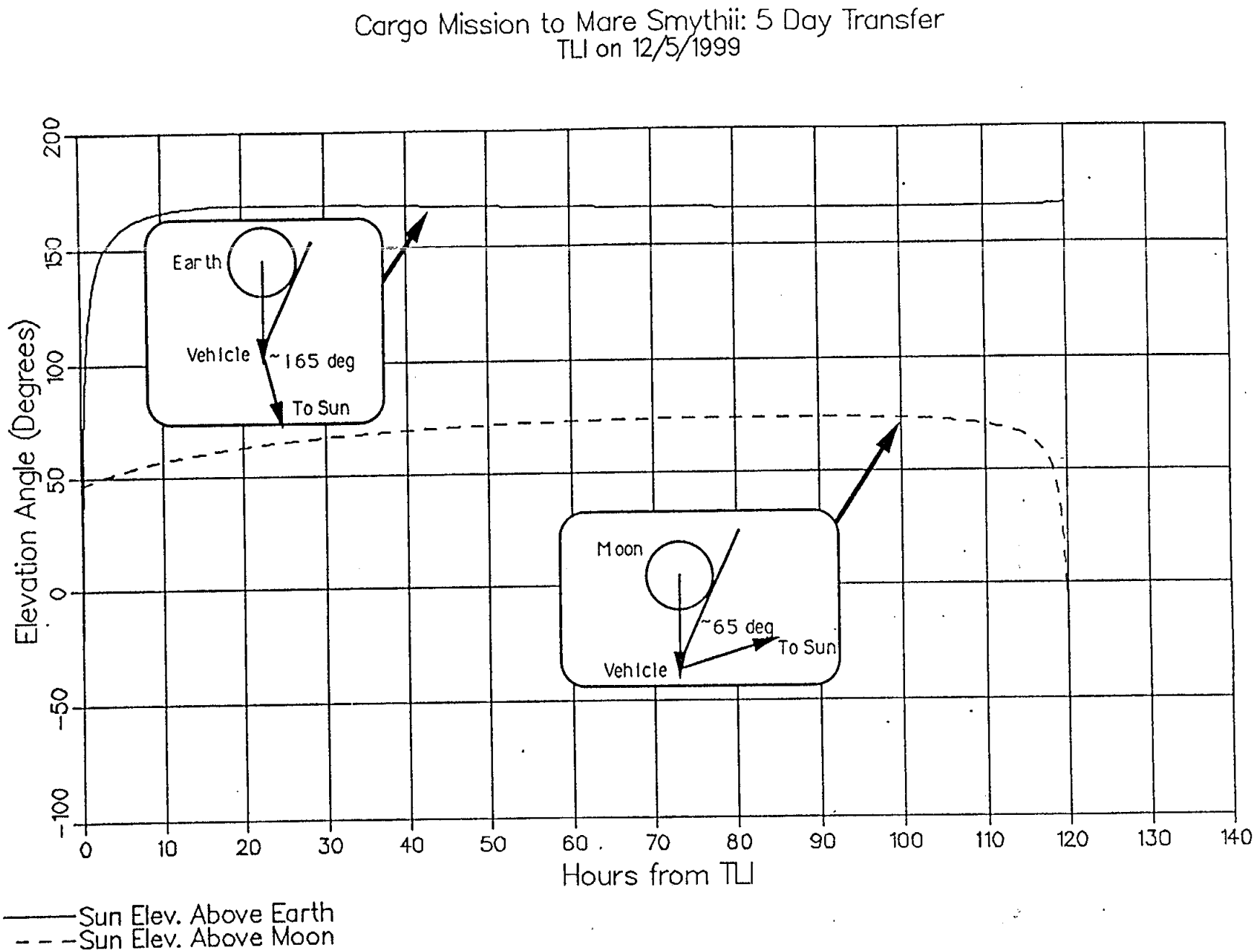


Figure 5-5. Spacecraft Sun elevation angles during translunar coast.

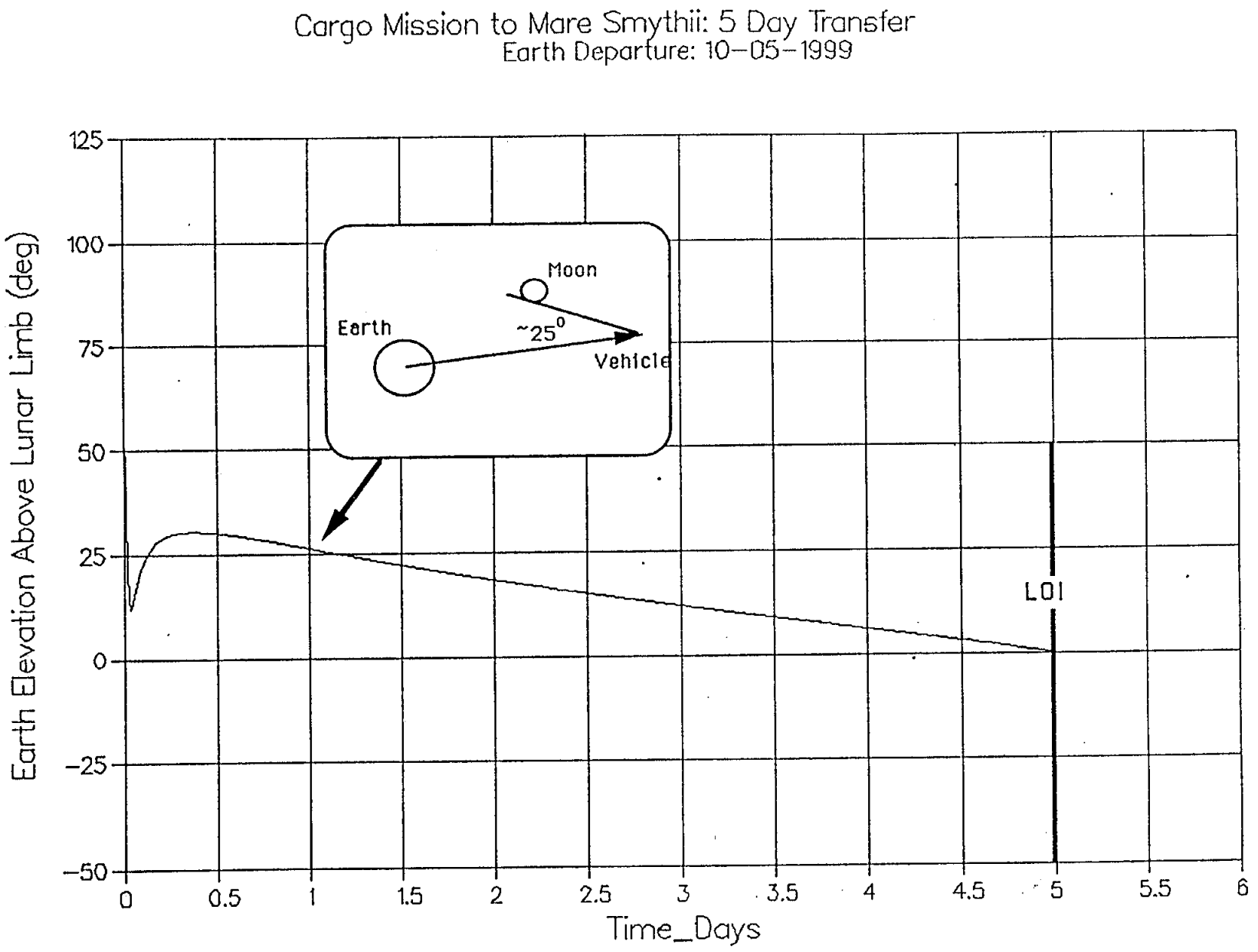


Figure 5-6. Spacecraft Earth elevation angle above Moon during translunar coast.

the lunar limb and the Earth elevation above the lunar limb are similar. Figures 5-5 and 5-6 contain diagrams illustrating these relationships.

In figure 5-7, the vehicle position, latitude, and longitude are plotted in selenographic coordinates as it approaches the Moon.

5.1.5 Lunar orbit phase

The lunar descent stage executes the LOI maneuver (impulsive delta-V of 809 m/s) at 10:51 GMT on October 10, 1999. This is an MET of 5 days, 1 hour, and 21 minutes. The maneuver geometry is shown in figure 5-1.

The temporary lunar parking orbit the spacecraft enters is at 100 km altitude, 41.1° inclination, and an ascending node 92.3° west selenographic longitude. The spacecraft is in this orbit for approximately 1 hour. Maneuver information for the cargo flight to Mare Smythii are shown in table 5-III.

Table 5-III. Mare Smythii Cargo Flight LOI Maneuver Parameters

LOI Date/Time	Oct 10, 1999 / 10:51 GMT
Delta-V/Plane Change	809 m/s / 0 °
LOI Latitude/Longitude	18.7 ° / 66.1 °

5.1.6 Lunar descent and landing phase

The lunar descent stage executes the deorbit maneuver (19 m/s) 58 minutes after LOI. This maneuver puts the spacecraft into a 100 x 18.5 km transfer orbit. After 57 minutes and a transfer angle of about 181°, the descent stage begins a 7-minute powered descent to the surface. The first 335 seconds of the powered descent are devoted primarily to removing downrange velocity. Then the spacecraft performs a 60-second pitchover/throttledown maneuver. At the conclusion of this maneuver, the spacecraft has a thrust acceleration of 1.2 lunar g (0.2 Earth g) and is oriented along the radial for a final 100 m (24-second) vertical descent to touchdown.

The descent profile and times for the cargo flight, at an inclination of 41.1°, will be similar to the piloted flight, at an inclination of 1.7°. Section 5.2.6 provides tabular and plot data for the descent profile. The higher inclination orbit will have a slightly higher delta-V cost due to some loss of lunar angular velocity advantage.

Lunar landing at Mare Smythii occurs on October 10, 12:53 GMT, an MET of 5 days 3 hours 23 minutes. Total maneuver cost for landing is 1853 m/s.

5.1.7 Lunar surface phase

The habitat is deployed and activated autonomously. It will be checked out remotely prior to launch of the crew. The design of the outpost will be such that it will not be necessary to unload the outpost from the lander. The crew will be able to move in and start science and exploration activities as soon as possible after arrival.

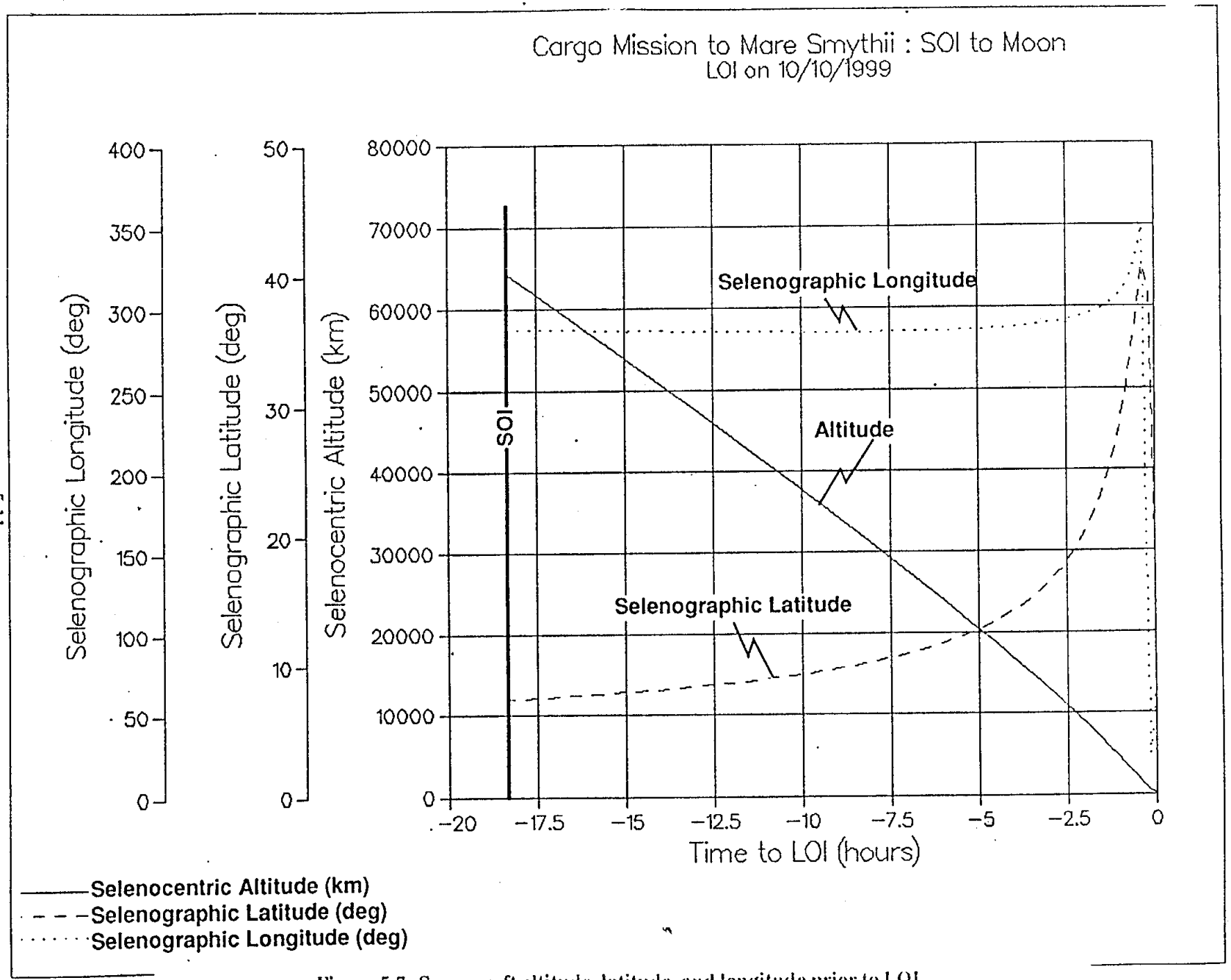


Figure 5-7. Spacecraft altitude, latitude, and longitude prior to LOI.

5.2 PILOTED FLIGHT TO MARE SMYTHII

5.2.1 Flight description

The piloted flight consists of the TLI stage, the lunar descent stage, the ascent stage, and the crew module (pressure vessel and Earth return capsule). The sequence of major events for the piloted flight to Mare Smythii appears in table 5-IV. This flight delivers a crew of four to the lunar outpost at Mare Smythii, which was delivered on the cargo flight 60 days earlier. Launch occurs on December 5, 1999, at 8:06 GMT. A single, heavy-lift launch vehicle delivers the spacecraft (crew included) to an Earth parking orbit at an altitude of 185 km (100 n. mi.) and an inclination of 33.26°.

At 9:34 GMT (1 hour and 28 minutes MET), the TLI stage executes the lunar injection maneuver. The impulsive delta-V for this burn is 3144 m/s, which places the spacecraft on a 4-day transfer trajectory to the Moon. Soon after the lunar transfer maneuver, the TLI stage is dropped, and the remaining spacecraft consists of the lunar descent stage, ascent stage, and crew module.

The translunar coast phase takes 4 days, during which time midcourse maneuvers are executed as appropriate. Approximately 30 m/s delta-V is budgeted for these maneuvers.

Upon arrival at the Moon, the descent stage executes the LOI maneuver on December 9, at 9:34 GMT (4 days, 1 hour, and 28 minutes MET). The LOI maneuver inserts the spacecraft into a temporary 100 km circular lunar parking orbit at an inclination of 1.7° and a longitude of ascending node of 6.7° east (selenographic coordinates).

After approximately 1 hour in orbit, the lunar descent stage executes the deorbit maneuver (19 m/s delta-V at 10:33 GMT), putting the vehicle on a 100 x 18.5 km transfer orbit. After coasting through a 181° transfer angle, the descent stage performs a powered flight descent (1853 m/s) to the landing site (Mare Smythii). Landing occurs on December 9 at 11:37 GMT (4 days, 3 hours, and 31 minutes MET). Lunar sunrise at Mare Smythii occurs approximately 27 hours earlier. The crew will walk to the habitat, which will have already been activated and checked out remotely.

The duration of the lunar surface stay is 42 days. After this stay time, the crew returns to the spacecraft for departure. The ascent stage performs the lunar ascent, leaving the lunar descent stage on the surface. Lift-off from the lunar surface occurs January 20, 2000, at 11:33 GMT.

The powered flight ascent takes about 6 minutes (1831 m/s). The vehicle coasts to an altitude of 100 km and performs a circularization burn (19 m/s, 12:36 GMT) to place the vehicle into a 100 km circular lunar orbit. After 2 hours and 12 minutes, the ascent stage performs the TEI maneuver (837 m/s, 14:47 GMT).

The trans-Earth coast phase lasts 4 days, 1 hour, and 32 minutes, during which time midcourse maneuvers are executed as appropriate (30 m/s budget). Before atmospheric entry, the crew module separates from the ascent stage and prepares for entry interface. The Earth return landing is targeted for a land-landing in Chile, at a latitude of 21.5° south and a longitude of 69.4° west.

Table 5-IV. Sequence of Major Events: Piloted Flight to Mare Smythii

Event	GMT hr:min	MET day\hr:min	Data Summary	
Earth Launch*	1999 Dec 5 8:06	0\00:00	Launch Azimuth, deg	72 †
Earth Orbit Insertion	--	--	Altitude, km Inclination, deg	185 33.26
Translunar Injection (TLI)	9:34	0\01:28	ΔV, mps	3144
Translunar Coast Midcourse Corrections	--	--	Duration, days ΔV, mps	4 ~30
Lunar Orbit Insertion (LOI)	1999 Dec 9 9:34	4\01:28	ΔV, mps	825
Lunar Parking Orbit	--	--	Altitude, km Inclination, deg LAN, deg Wait, min	100 1.7 6.7 59
Deorbit	10:33	4\02:27	ΔV, mps	19
Landing Ignition	11:30	4\03:24		
Landing	1999 Dec 9 11:37	4\03:31	ΔV, mps	1853
Lunar Surface Stay	--	--	Duration, days	42

*Determination of lift-off times uses a model of powered flight ascent arc. Accuracy of lift-off time is within ±5 minutes.
† Impulsive inertial azimuth

Table 5-IV. Concluded

Event	GMT hr:min	MET day\hr:min	Data Summary	
Lift-off Lunar Surface	2000 Jan 20 11:33	46\03:27		
Ascent Cutoff	11:39	46\03:33	ΔV, mps (100x18.5 km transfer orbit)	1831
Lunar Orbit Circularization	12:36	46\04:30	Altitude, km ΔV, mps	100 19
Lunar Parking Orbit	--	--	Inclination, deg LAN, deg Wait time, min	98.2 -94.4 132
Trans-Earth Injection (TEI)	2000 Jan 20 14:47	46\06:41	ΔV, mps	837
Trans-Earth Coast Midcourse Corrections	--	--	Duration, days ΔV, mps	4.06 ~30
Atmospheric Entry	2000 Jan 24 16:17	50\08:11	Relative entry speed, km/s Azimuth, deg	10.5 96.0
Earth Touchdown	2000 Jan 24 16:31	50\08:25	Latitude, deg Longitude, deg	21.5 S 69.4 W

5.2.2 Earth launch phase

The nominal Earth launch azimuth (direction) from KSC is 72° east of north. This is the opening of the launch window (approximately 4 hours and 30 minutes long). Launch of the piloted flight to Mare Smythii occurs on December 5, 1999, at 8:06 am GMT. This date was selected by choosing an opportunity in December that would assure that landing at Mare Smythii would occur just after lunar sunrise. This rationale was discussed in the previous section. Coordinates for KSC are geodetic latitude 28.6° and longitude 279.4°.

5.2.3 Earth orbit phase

Earth parking orbit insertion occurs at an altitude of 185 km with an inclination of 33.26°. The inclination is a result of the 72° launch azimuth from KSC.

The TLI maneuver (impulsive delta-V of 3144 m/s) occurs at 9:34 GMT or 1 hour and 28 minutes MET. This is the first opportunity of the TLI window. There are two more opportunities, each a revolution apart.

Maneuver information for the cargo flight to Mare Smythii are shown in table 5-V. These data are for an impulsive TLI, without gravity loss.

Table 5-V. Mare Smythii Piloted Flight
TLI Maneuver Parameters

TLI Departure Date/Time	Dec 5, 1999 / 9:34 GMT
Delta-V/Plane Change	3144 m/s / 0 °
TLI Latitude/Longitude	22.1 ° / -119.4 °

The TLI maneuver geometry for piloted flight to Mare Smythii is shown in figure 5-8. The altitude, velocity, latitude, and longitude parameters for the impulsive maneuver are listed in the TLI box. Also shown is the Moon location at TLI with the associated Sun direction. In figure 5-9, the altitude of the spacecraft is plotted for the first 12 hours after TLI. Also shown are the geodetic latitude and longitude of the spacecraft.

5.2.4 Translunar phase

After the lunar transfer maneuver, the TLI stage separates from the rest of the spacecraft (i.e., the lunar descent stage, ascent stage, and crew module). The spacecraft continues on the nominal translunar coast phase (from TLI to LOI), which takes 4 days. During this time, the vehicle position and velocity are monitored, midcourse correction maneuvers are applied as necessary, and vehicle systems are checked out in preparation for LOI.

In figure 5-10, the vehicle position during translunar coast is plotted in geodetic latitude versus longitude coordinates. The groundtrack of the translunar trajectory overlayed on the continents is depicted with the TLI and LOI positions. In figure 5-11, the distance from the Earth to the spacecraft is plotted out to 96 hours. Also shown are the geodetic latitude and longitude.

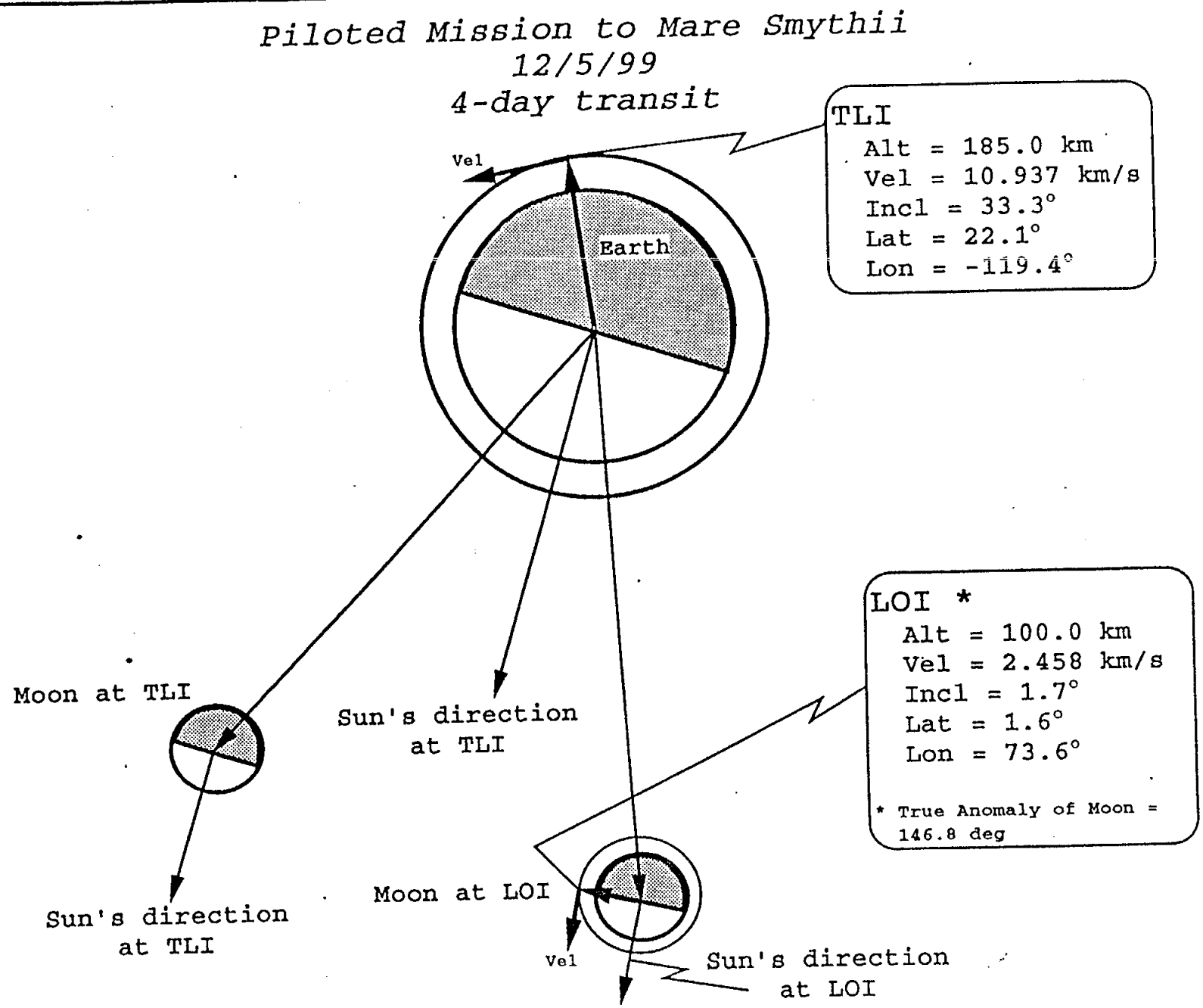


Figure 5-8. TLI and LOI maneuver geometry.

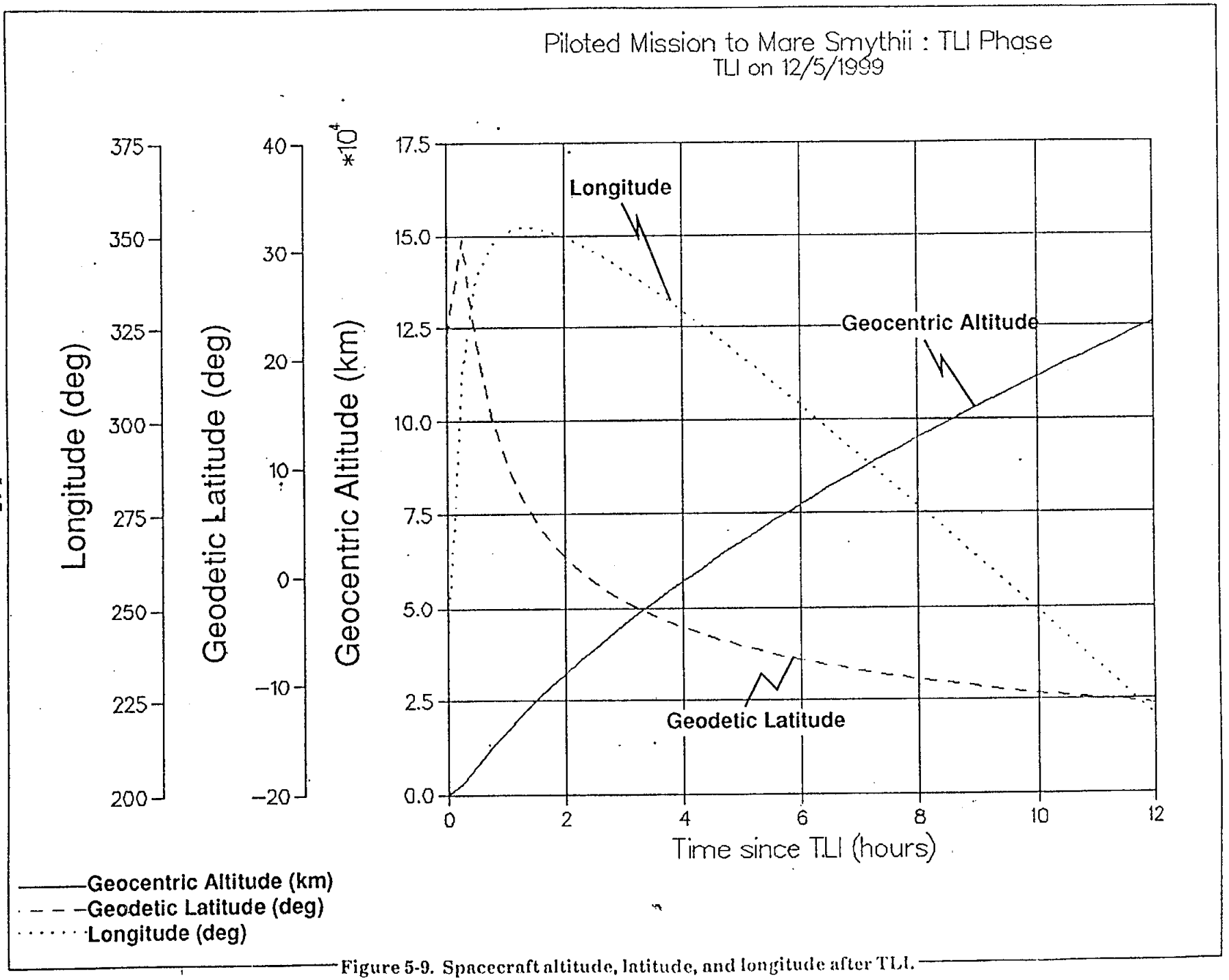


Figure 5-9. Spacecraft altitude, latitude, and longitude after TLI.

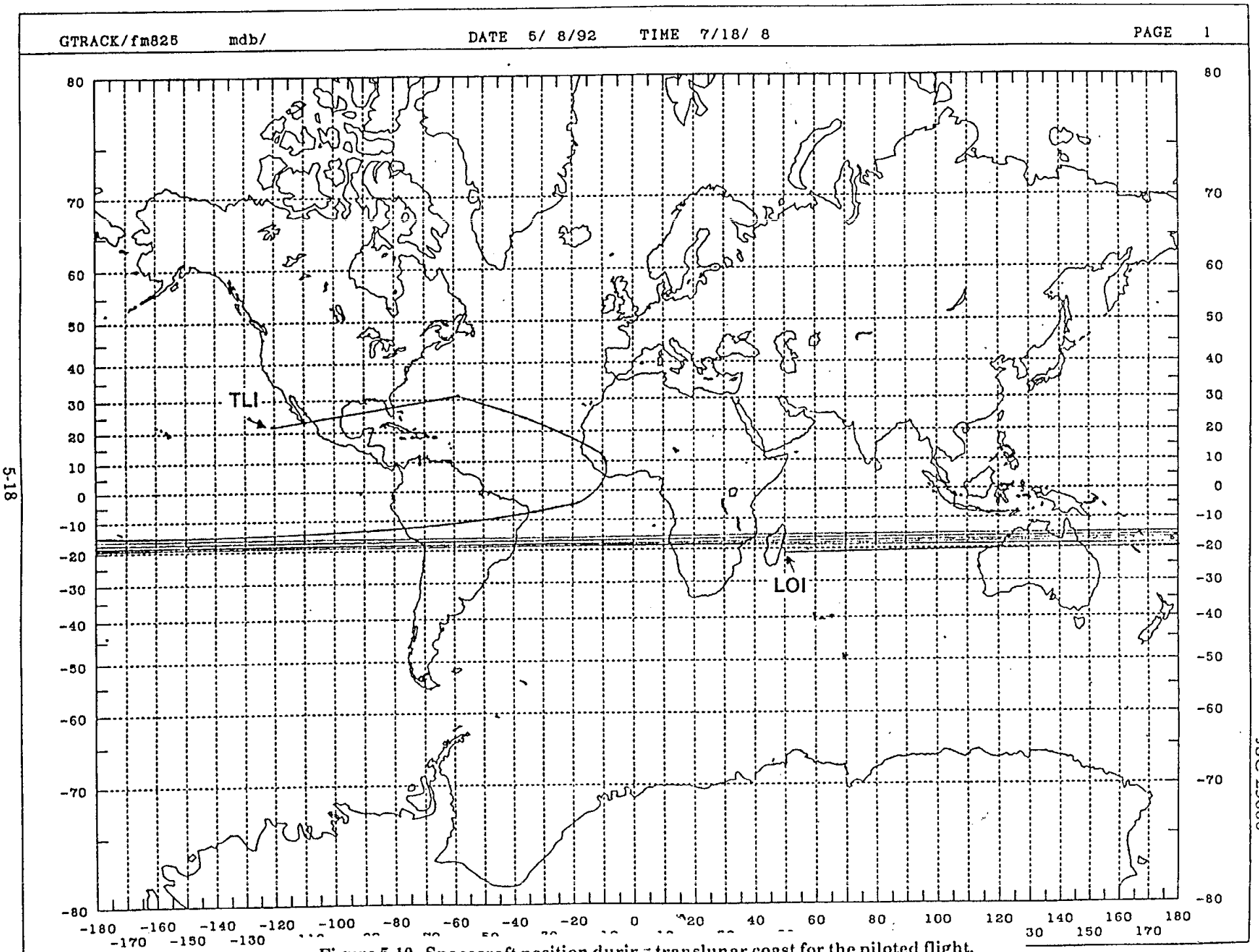


Figure 5-10. Spacecraft position durir translunar coast for the piloted flight.

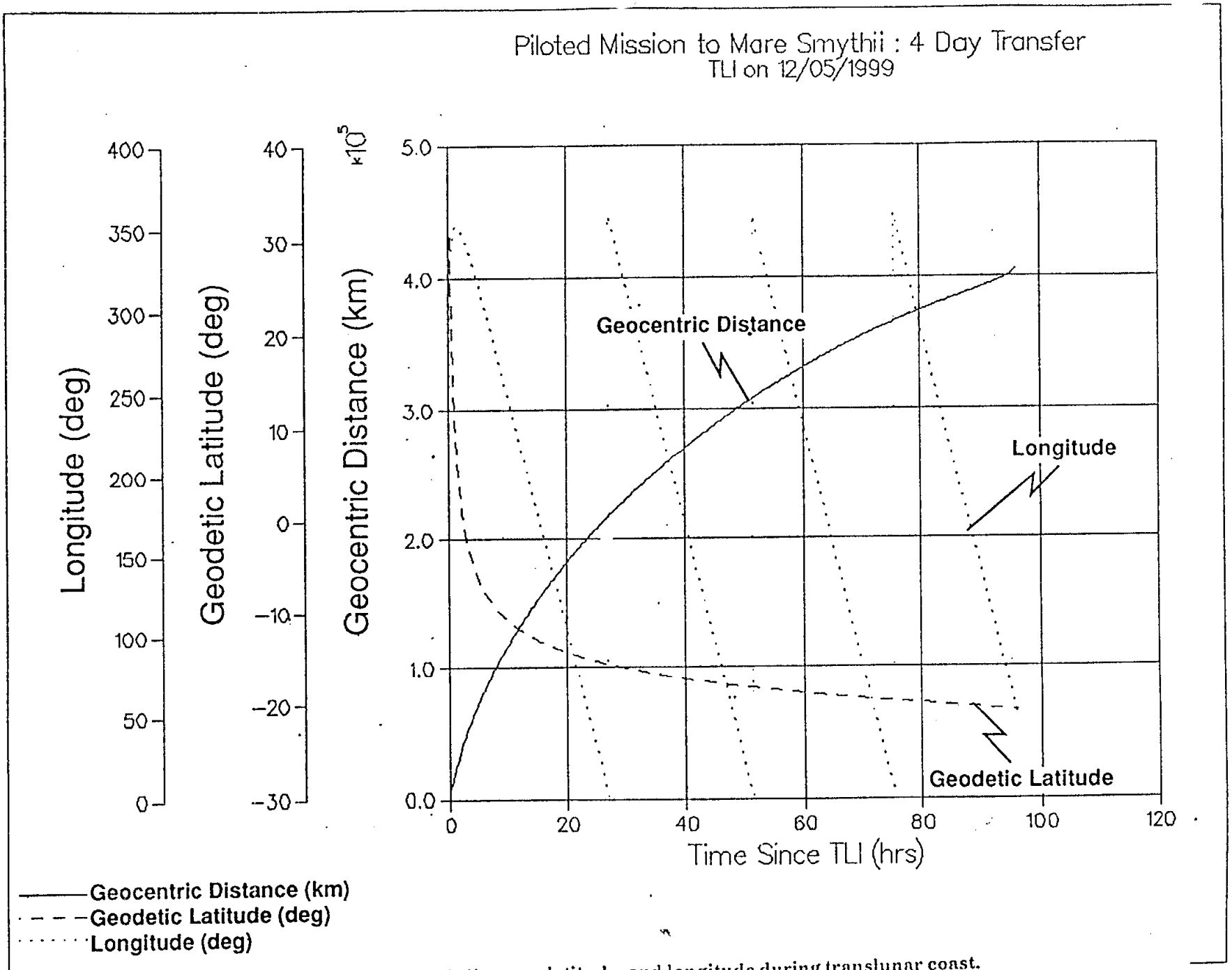


Figure 5-11. Spacecraft distance, latitude, and longitude during translunar coast.

The next two figures (figs. 5-12 and 5-13) pertain to elevation angles relative to the vehicle. The first chart depicts the solar elevation angle above the Earth limb. This angle was obtained using position vectors for the vehicle and the Sun. The Earth was assumed to be spherical when computing the angular location of the respective limb. The diagram associated with the solid line in figure 5-12 illustrates this geometry. The elevation angles for the Sun angle above the lunar limb and the Earth elevation above the lunar limb are similar. Figures 5-12 and 5-13 illustrate these relationships.

The vehicle position as it approaches the Moon is illustrated in figure 5-14, along with the selenographic latitude and longitude.

5.2.5 Lunar orbit phase

The lunar descent stage executes the LOI maneuver (impulsive delta-V of 825 m/s) at 9:34 GMT on December 9, 1999. This is an MET of 4 days, 1 hour, and 28 minutes. The maneuver geometry is shown in figure 5-8.

The temporary lunar parking orbit the spacecraft enters is at an altitude of a 100 km and at an inclination of 1.7°, with an ascending node of 6.7° east selenographic longitude. The spacecraft waits in this orbit for 59 minutes. Maneuver information for the piloted flight to Mare Smythii is shown in table 5-VI.

Table 5-VI. Mare Smythii Piloted Flight LOI Maneuver Parameters

LOI Date/Time	Dec 9, 1999 / 9:34 GMT
Delta-V/Plane Change	825 m/s / 0 °
LOI Latitude/Longitude	1.6 ° / 73.6 °

5.2.6 Lunar descent and landing phase

The descent stage executes the deorbit maneuver (19 m/s) 59 minutes after LOI. This maneuver puts the spacecraft into a 100 x 18.5 km transfer orbit. After 57 minutes and a transfer angle of about 181°, the descent stage begins a 7-minute powered descent to the surface. The first 335 seconds of powered descent are devoted primarily to removing downrange velocity. Then the spacecraft performs a 60-second pitchover/throttledown maneuver. At the conclusion of this maneuver, the spacecraft has a thrust acceleration of 1.2 lunar g (0.2 Earth g) and is oriented along the radial for a final 100 m (24 second) vertical descent to touchdown. Table 5-VII contains the lunar descent parameter data by phase elapsed time (PET) from deorbit to touchdown. This profile will be similar for descents to other lunar sites.

Piloted lunar landing at Mare Smythii occurs on December 9, 11:37 GMT, an MET of 4 days, 3 hours, and 31 minutes. This is 1 day and 3 hours after lunar sunrise. The total landing maneuver cost is 1852 m/s.

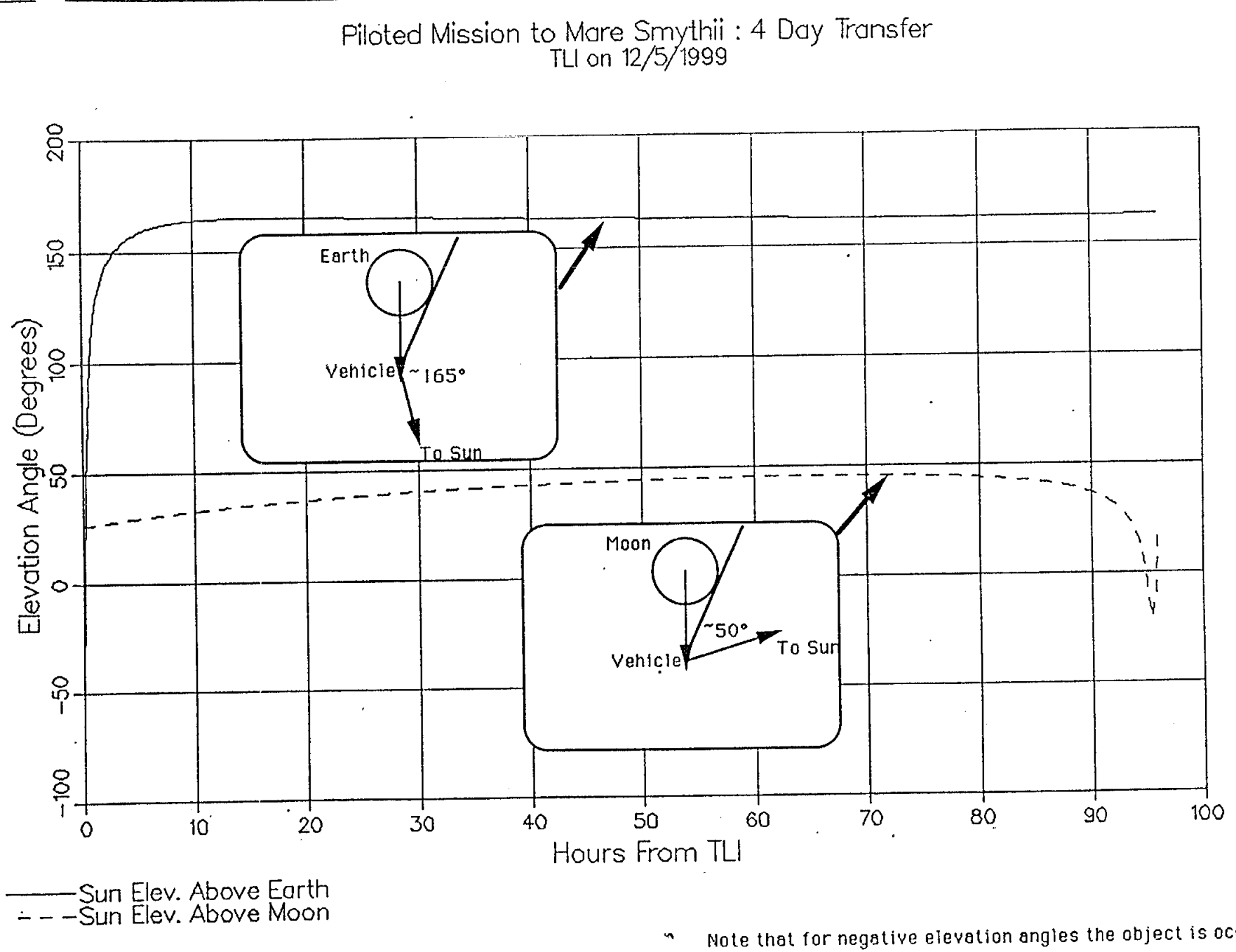
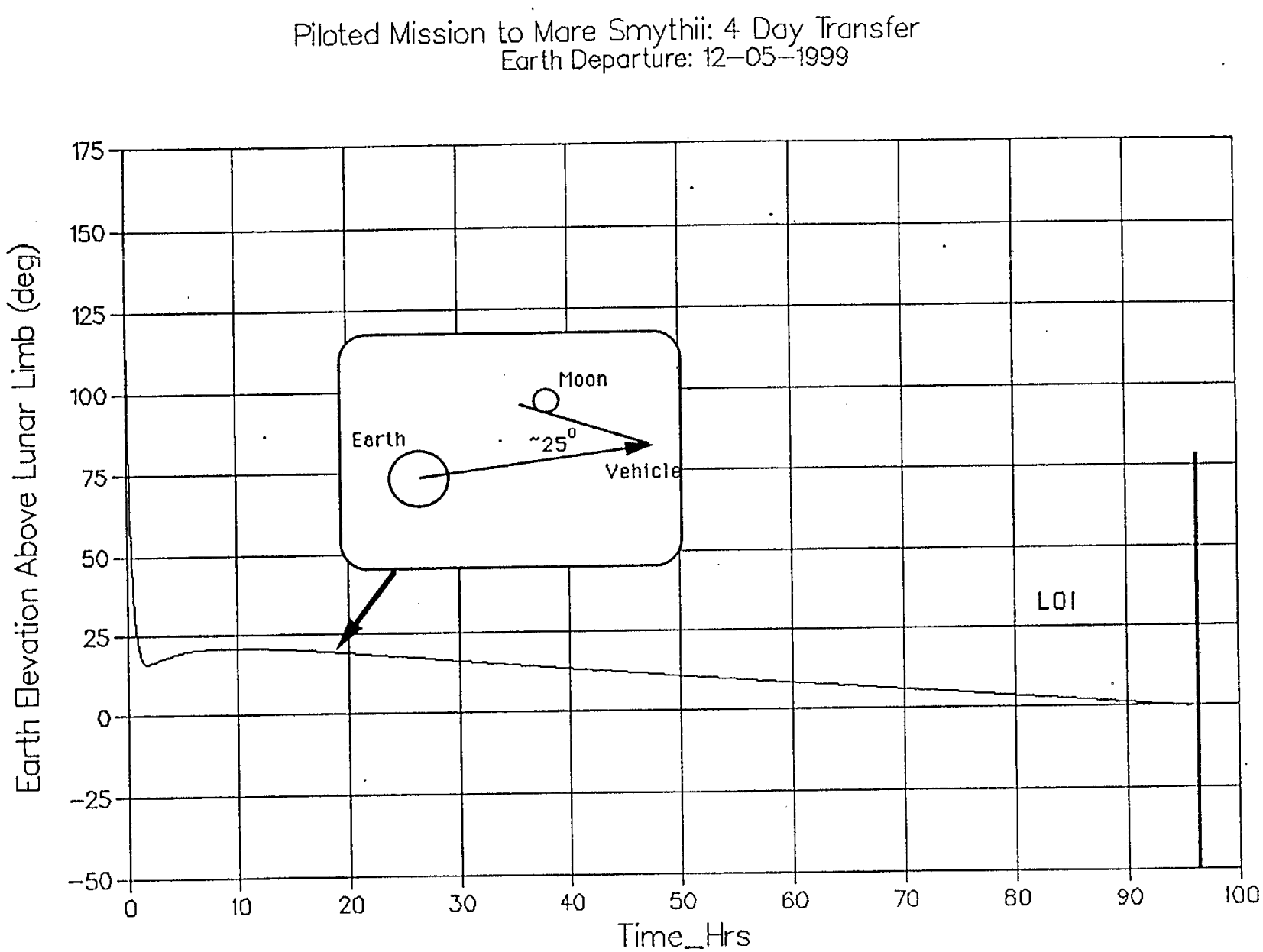


Figure 5-12. Spacecraft Sun elevation angles during the translunar coast.



Note that for negative elevation angles the object is occulted

Figure 5-13. Spacecraft Earth elevation angle above the Moon during translunar coast.

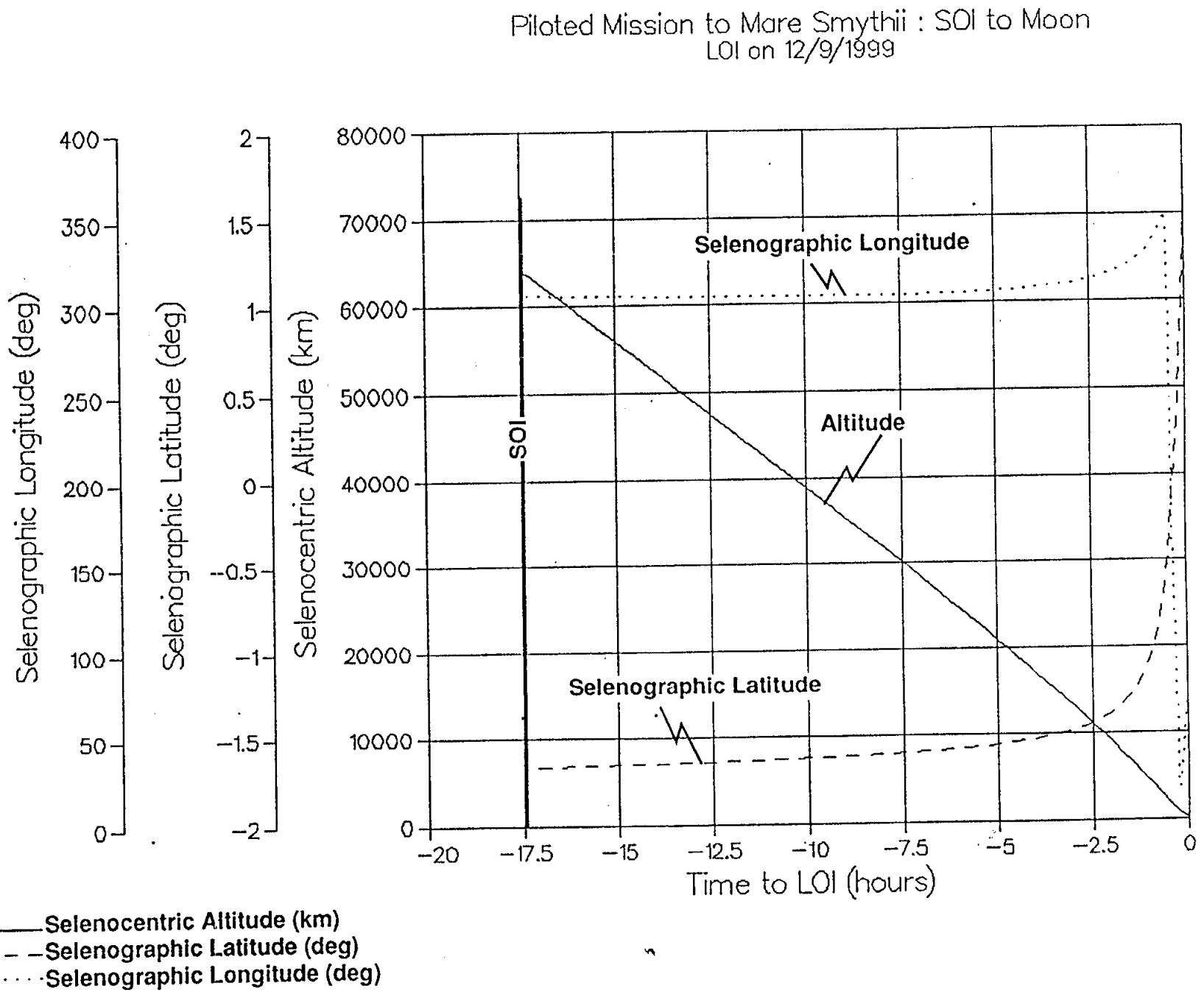


Figure 5-14. Spacecraft altitude, latitude, and longitude prior to LOI.

Table 5-VII. Lunar Descent Data

PET (sec)	Ha (km)	Hp (km)	ΔVideal (m/s)	Mass (kg)	Comment
0	100	100	0	74776.0	Deorbit
4.7	100	18.5	18.6	74457.3	Begin Coast
3440.9	101.1*	17.5*	18.6	74457.3	Beg Powered Descent
3596.0	14.57	-1321	677.5	64007.4	Pitch #1
3656.0	11.68	-1537	961.7	59965.3	Pitch #2
3716.0	7.63	-1667	1265.8	55923.1	Pitch #3
3776.0	3.79	-1731	1592.7	51881.0	Pitch #4
3836.0	0.1	-1738	1824.2	49171.3	Pitchup/Throttle-down
3860.0	0.0	-1738	1871.3	48627.9	Touchdown

Isp 444 max (100% throttle)
*Program uses osculating elements for descent simulation; therefore, there will be some variation in orbital elements with time.

Figure 5-15 depicts the lunar descent scenario from deorbit to landing. Figure 5-16 depicts the powered flight descent phase.

5.2.7 Lunar surface phase

After lunar landing, the crew will walk to the lunar outpost and begin checkout of the lunar habitat in preparation for lunar surface stay. The nominal lunar surface stay is 42 days (the maximum stay is 45 days with 3 days reserved for contingencies).

5.2.8 Lunar ascent phase

The 328-second lunar ascent from Mare Smythii begins with an Apollo-like vertical rise that terminates 6 seconds later when the vehicle reaches an altitude rate of 15.24 m/s (50 fps). The vehicle flies a minimum fuel pitch profile inserting into a 100 x 18.5 km transfer orbit. The vehicle then coasts to apolune, where it performs a maneuver to achieve a 100 km altitude circular orbit. The vehicle is modeled with three engines flown at full throttle (131438 N). This gives the ascent vehicle an initial thrust-to-weight of 0.43 Earth g (2.6 lunar g).

Table 5-VIII contains the lunar ascent parameter data in PET from lift-off to ascent cutoff.

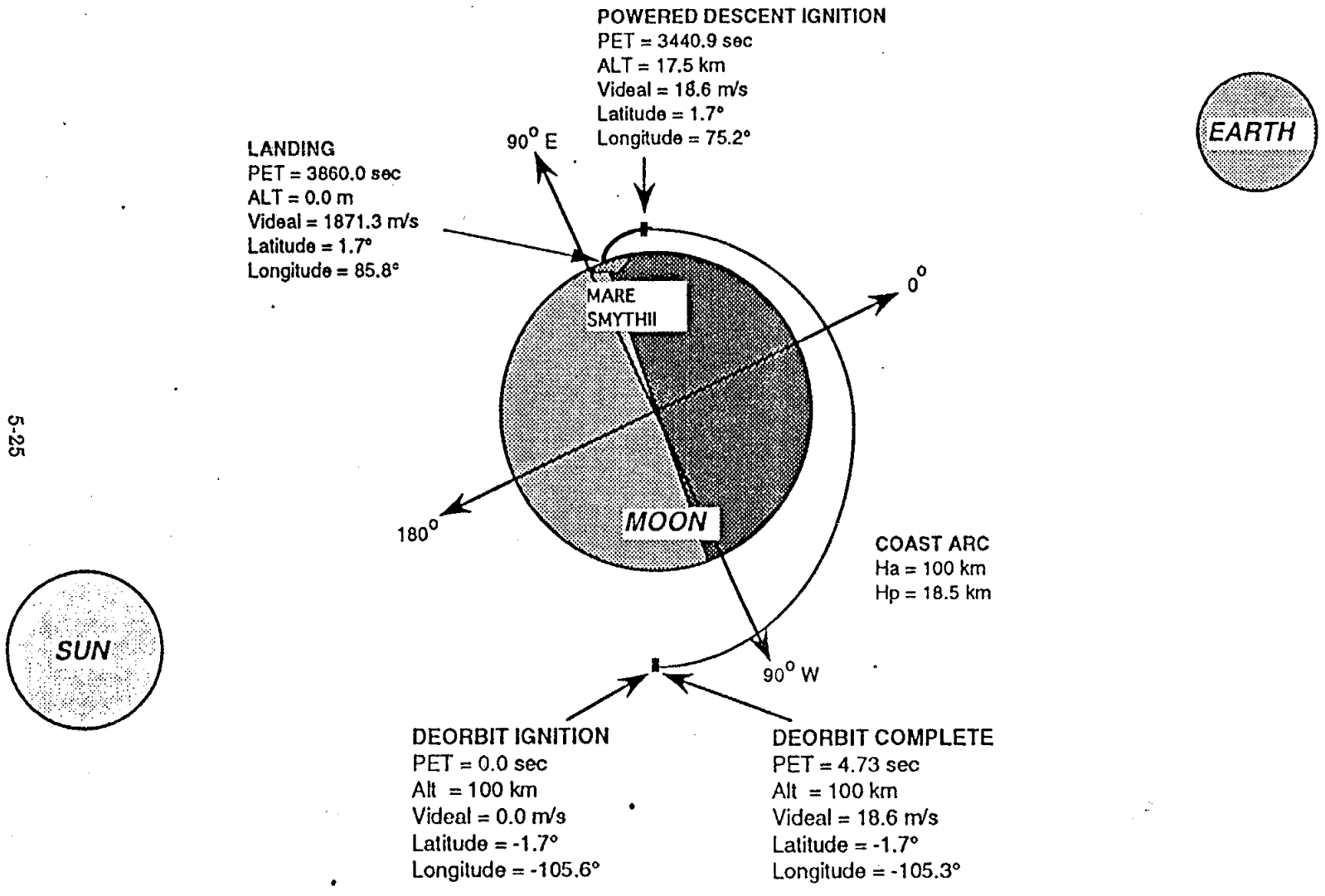


Figure 5-15. Lunar descent scenario.

NOMINAL LUNAR DESCENT - POWERED FLIGHT PHASE

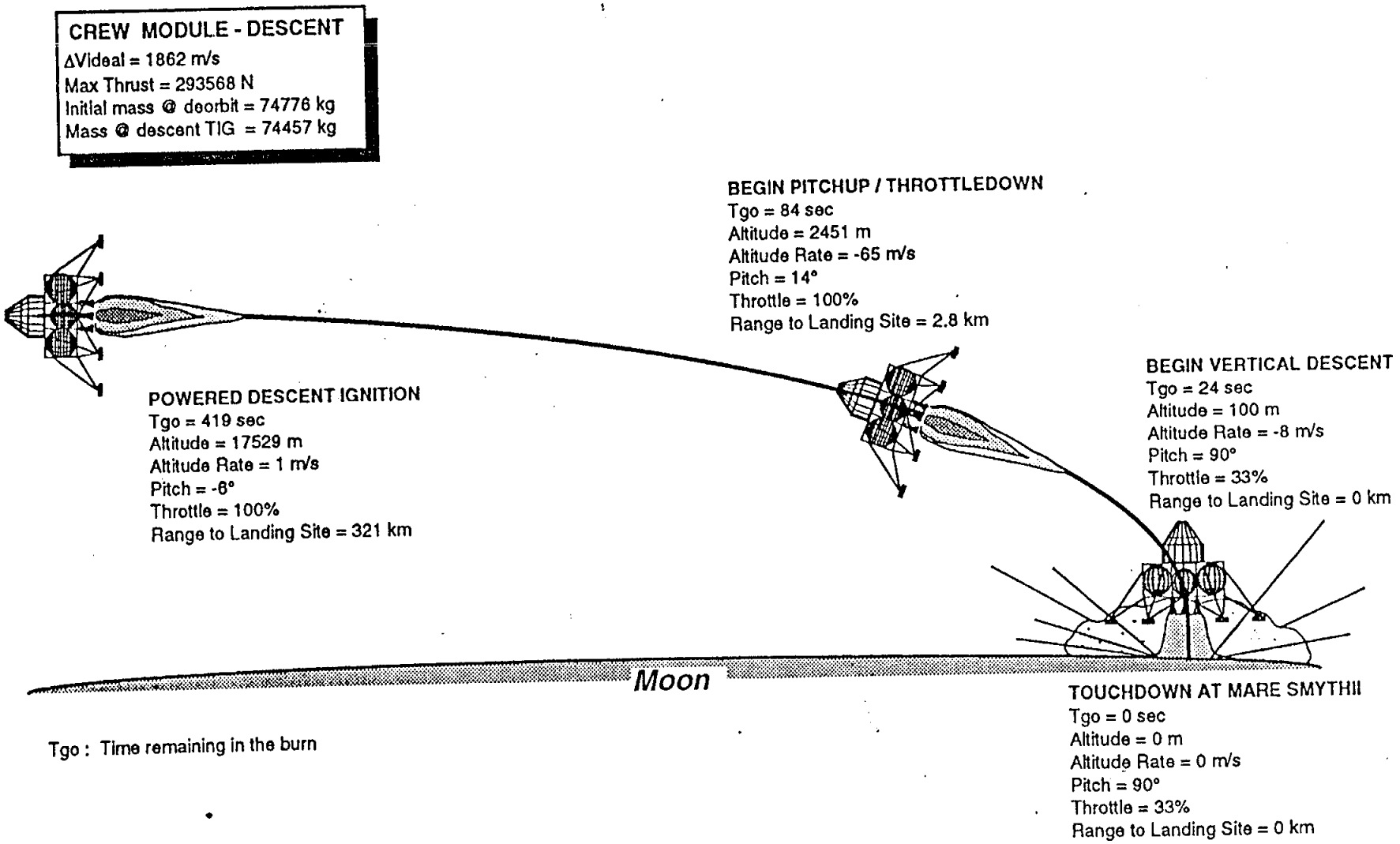


Figure 5-16. Powered flight descent phase.

Table 5-VIII. Lunar Ascent Data

MET (sec)	Ha (km)	Hp (km)	ΔVideal (m/s)	Mass (kg)	Comment
0	0	-1738.2	0	31063.0	Lift-off Ignition
5.8	0.1	-1738.2	24.7	30819.8	End vertical rise
15.8	0.9	-1738.2	67.6	30401.0	End Pitch #1
35.8	3.1	-1737.1	155.3	29563.4	End Pitch #2
75.8	6.1	-1721.7	338.3	27888.2	End Pitch #3
175.8	14.9	-1563.2	849.0	23700.3	End Pitch #4
327.9	100	18.4	1831.0	17331.8	End Pitch Rate

Isp is 320 sec (nonmodeled).

5.2.9 Lunar orbit phase

The temporary lunar parking orbit that the spacecraft enters is at an altitude of 100 km with an inclination of 98.2° retrograde, and an ascending node of 94.4° west selenographic longitude. Stay time in lunar orbit is 2 hours and 12 minutes prior to the TEI maneuver (one additional revolution of lunar orbit wait time was required).

The ascent stage reignites to execute the TEI maneuver (impulsive delta-V of 837 m/s) at 14:47 GMT on January 20, 2000. This is an MET of 46 days, 6 hours, and 41 minutes. The TEI maneuver geometry is shown in figure 5-17. As with the outbound trajectory, the trans-Earth trajectory was designed to be coplanar to minimize energy and mass requirements.

Figure 5-18 illustrates the altitude, latitude, and longitude as the spacecraft leaves the Moon for the piloted flight to Mare Smythii. Table 5-IX contains the Mare Smythii piloted flight TEI maneuver parameters.

Table 5-IX. Mare Smythii Piloted Flight TEI Maneuver Parameters

TEI Date/Time	Jan 20, 2000 / 14:47 GMT
Delta-V/Plane Change	837 m/s / 0 °
TEI Latitude/Longitude	45.1 ° / -104.5 °

The TEI maneuver geometry for the piloted flight return to Earth is shown in figure 5-17. The altitude, velocity, latitude, and longitude for the impulsive maneuver are listed in the TEI box. Figure 5-18 illustrates the altitude, selenographic latitude and longitude as the vehicle leaves the Moon.

5.2.10 Trans-Earth phase

The nominal trans-Earth coast (from TEI to atmospheric entry) for the Mare Smythii piloted flight takes 4 days and 92 minutes to vacuum perigee (55 minutes to entry interface (EI)). During this

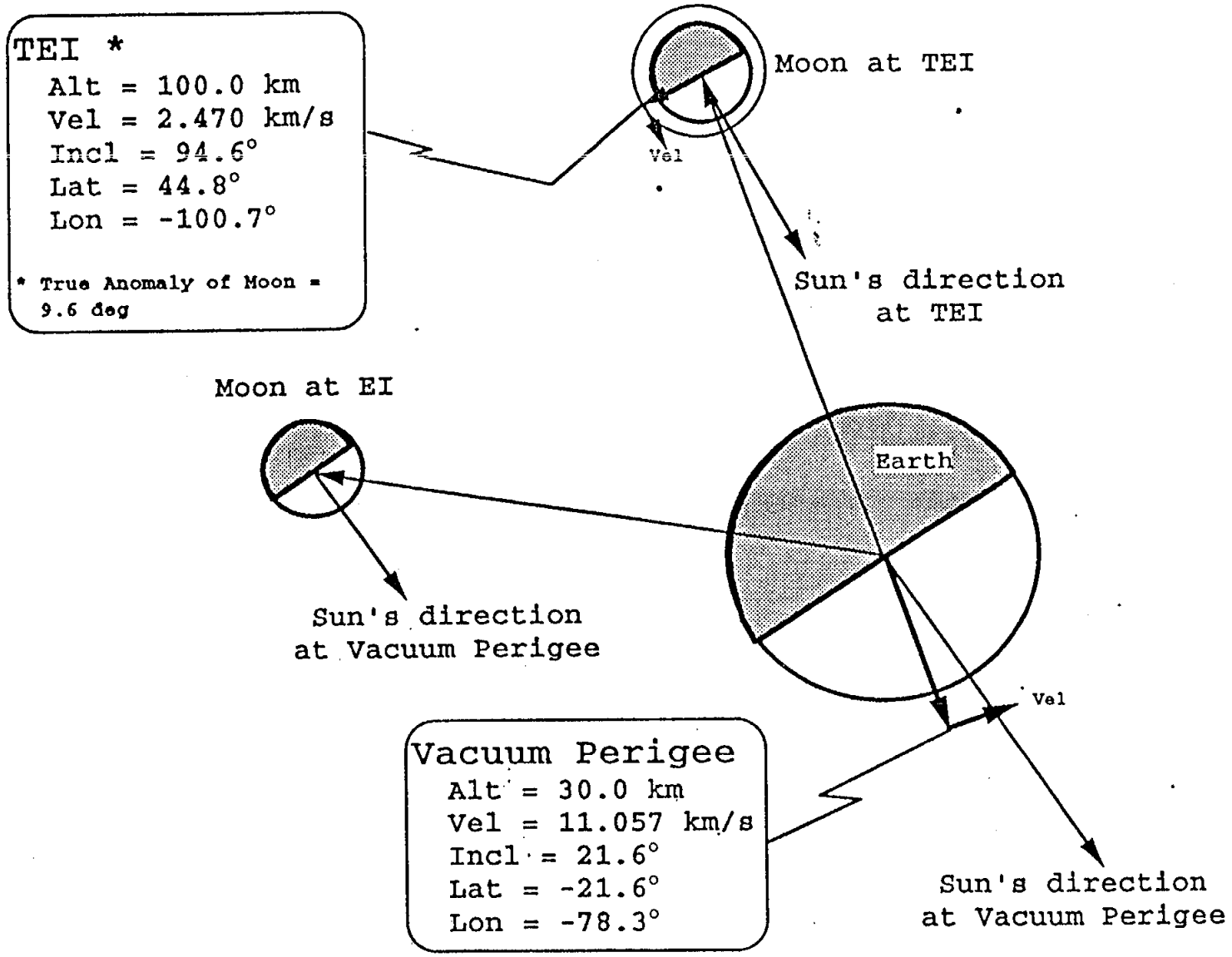


Figure 5-17. TEI and Earth return geometry.

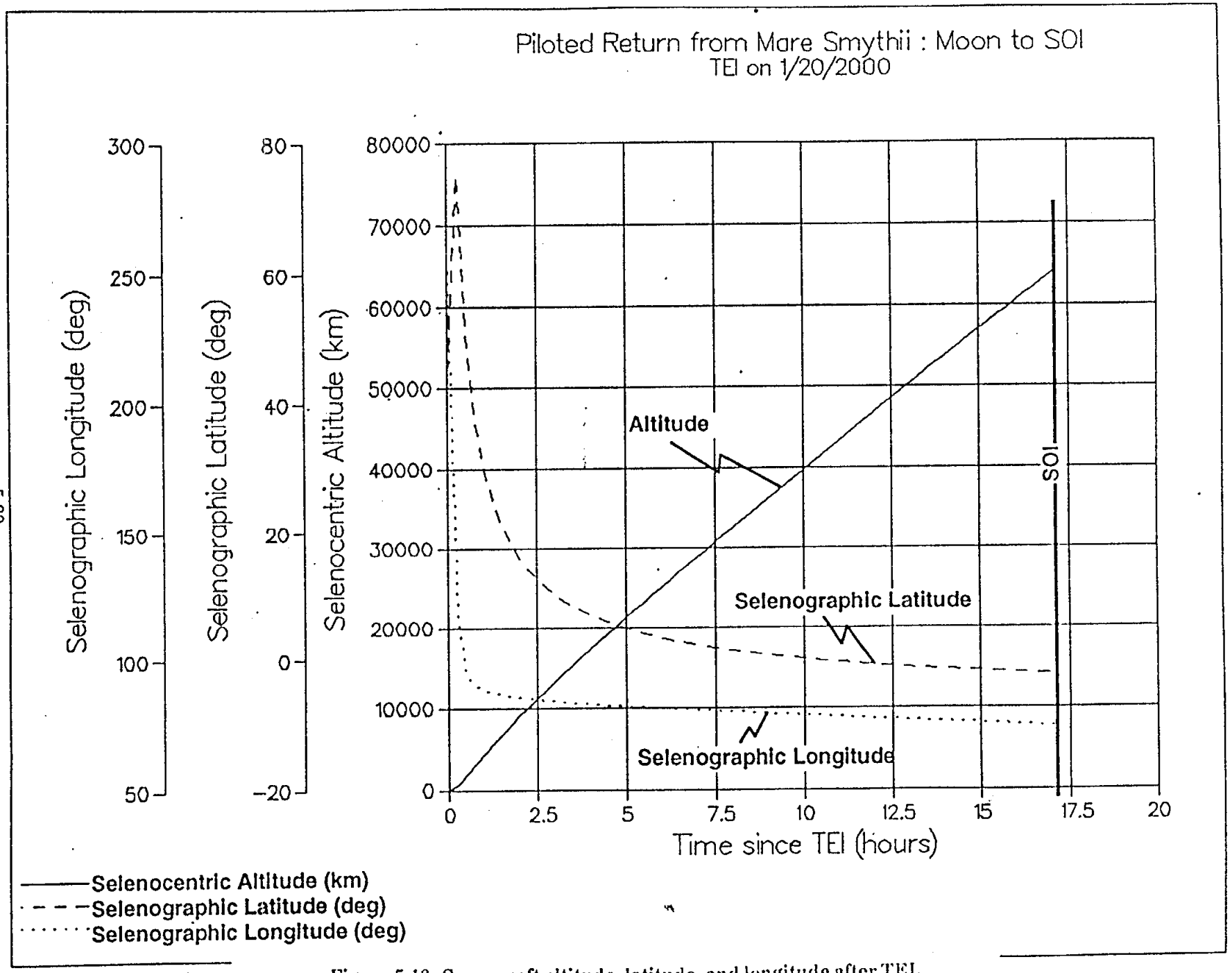


Figure 5-18. Spacecraft altitude, latitude, and longitude after TEI.

time, the vehicle position and velocity are monitored, midcourse correction maneuvers are applied as necessary, and vehicle systems are checked out in preparation for atmospheric entry. The ascent stage separates from the crew module just prior to atmospheric entry.

In figure 5-19, the vehicle position during trans-Earth coast is plotted in geodetic latitude versus longitude coordinates. The TEI and EI positions are shown. Figure 5-20 depicts the distance from the Earth to the spacecraft for the return from the Moon. Also shown are the geodetic latitude and longitude.

The next two figures (figs. 5-21 and 5-22) pertain to elevation angles relative to the vehicle. The first chart depicts the solar elevation angle above the Moon limb. The Moon was assumed to be spherical when computing the angular location of the respective limb. The diagram associated with the dotted line in figure 5-21 illustrates this geometry. The solid line in this figure depicts the Earth limb above the Sun. The elevation angles for the Earth angle above the lunar limb is illustrated in figure 5-22.

5.2.11 Earth entry phase

An Apollo-style vehicle is assumed for the atmospheric reentry and landing portion of the flight. As the vehicle approaches the Earth, the ascent stage is separated from the crew module. The vehicle positions itself into the correct entry attitude for the atmospheric phase.

The nominal target, 2380 km (1285 n. mi.) downrange from the EI point, is 15 minutes away in flight time. The downrange distance is used for a Great Circle arc from the projection of the entry point on the Earth surface to the landing site. For the nominal downrange target, the vehicle lands at coordinates 21.5° south and 69.4° west. The target lies within a landing footprint that ranges from 1778 km (960 n. mi.) to 3167 km (1710 n. mi.) downrange from the EI point. Figure 2.11-1 illustrates the Apollo downrange footprint.

Atmospheric entry occurs at 16:17 GMT on January 24, 2000. This is an MET of 50 days, 8 hours, and 11 minutes. The relative entry speed is 10.54 km/s at an azimuth of 96.0°. Based on the vacuum perigee latitude of the return trajectory, an example landing site was chosen in Chile. The perigee coordinates are 21.5° south and 69.4° west.

In figure 5-24, the altitude from the Earth to the spacecraft is plotted against time to vacuum perigee. Also shown are the geodetic latitude and longitude.

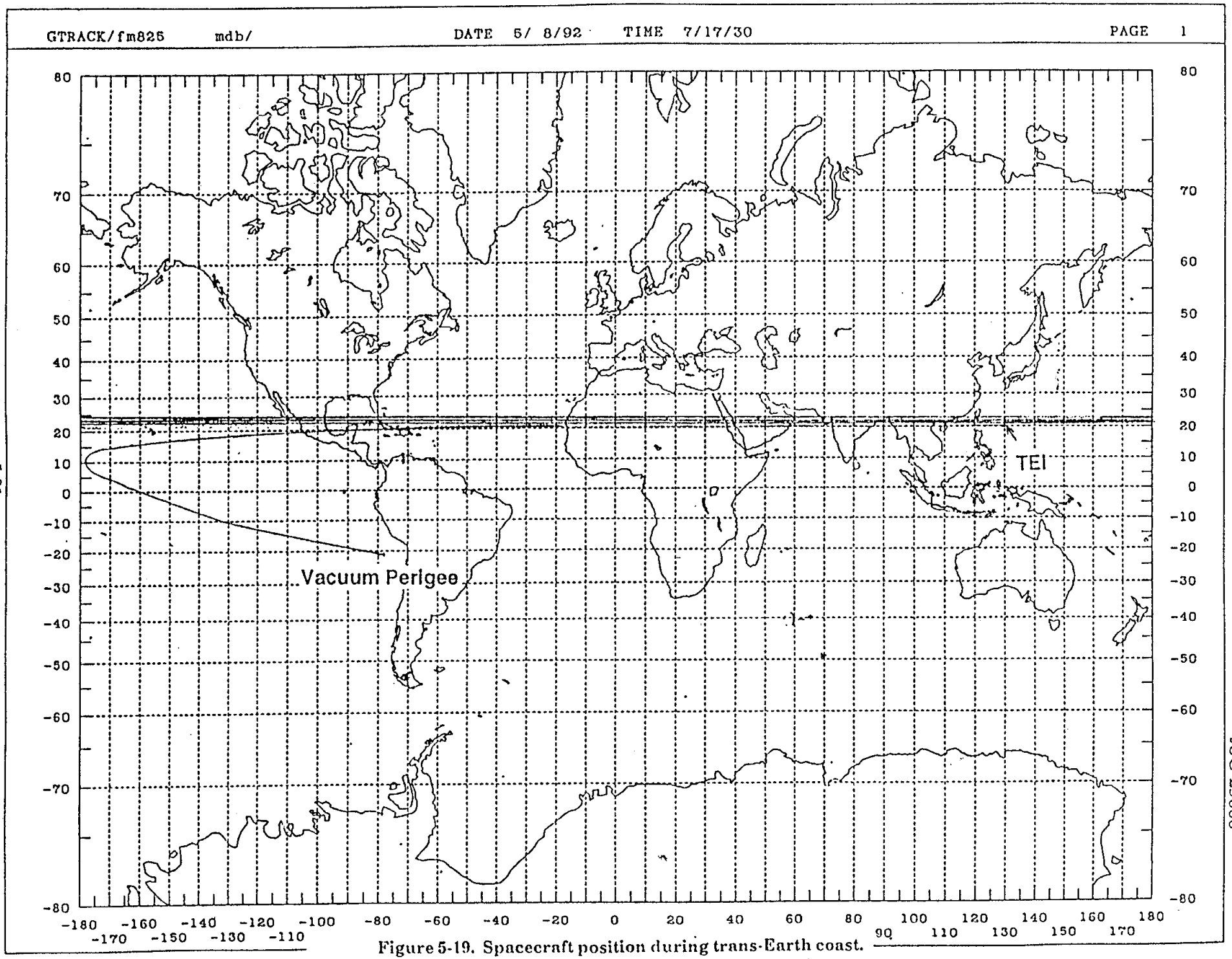


Figure 5-19. Spacecraft position during trans-Earth coast.

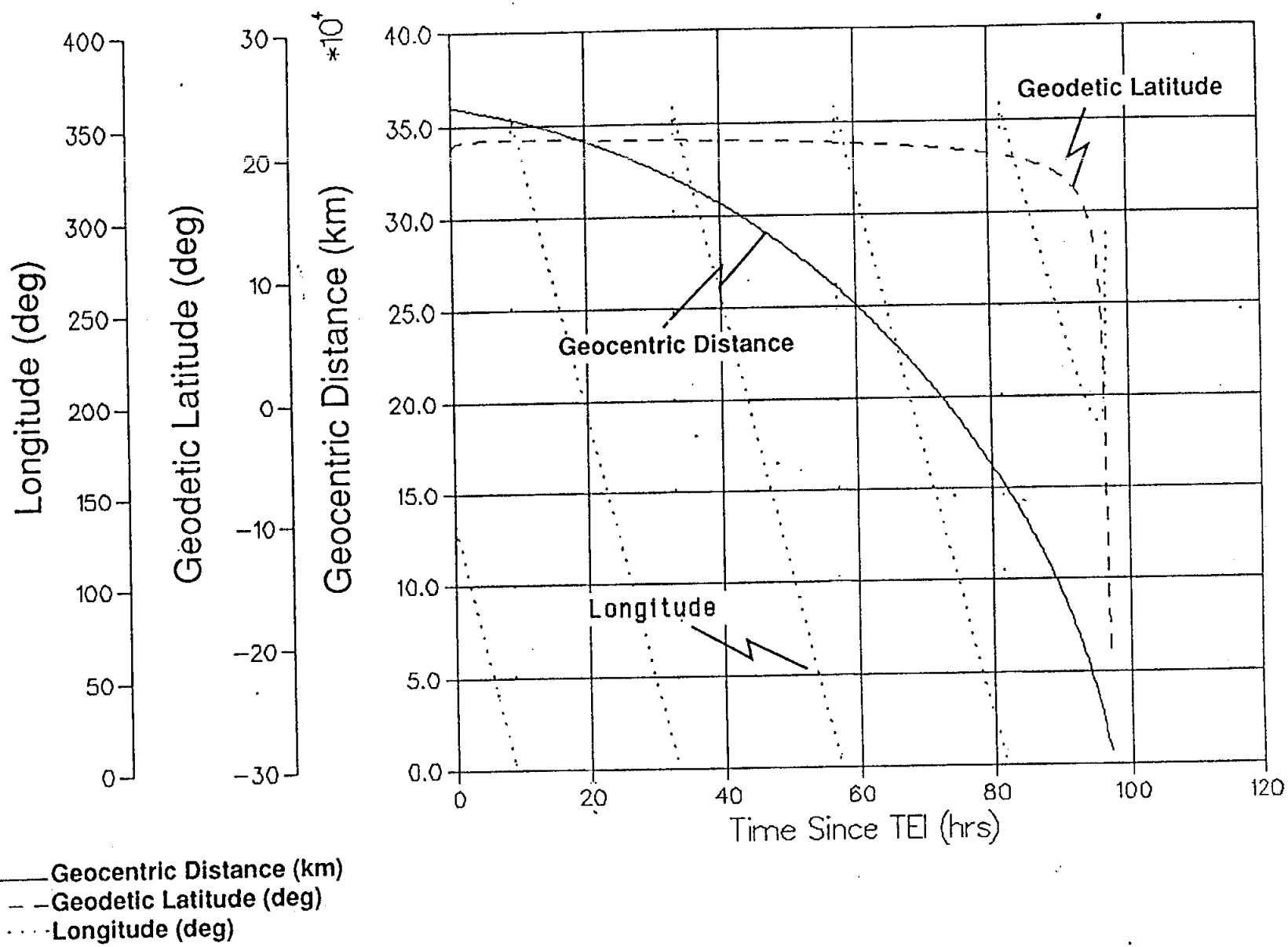


Figure 5-20. Spacecraft distance, latitude, and longitude during trans-Earth coast.

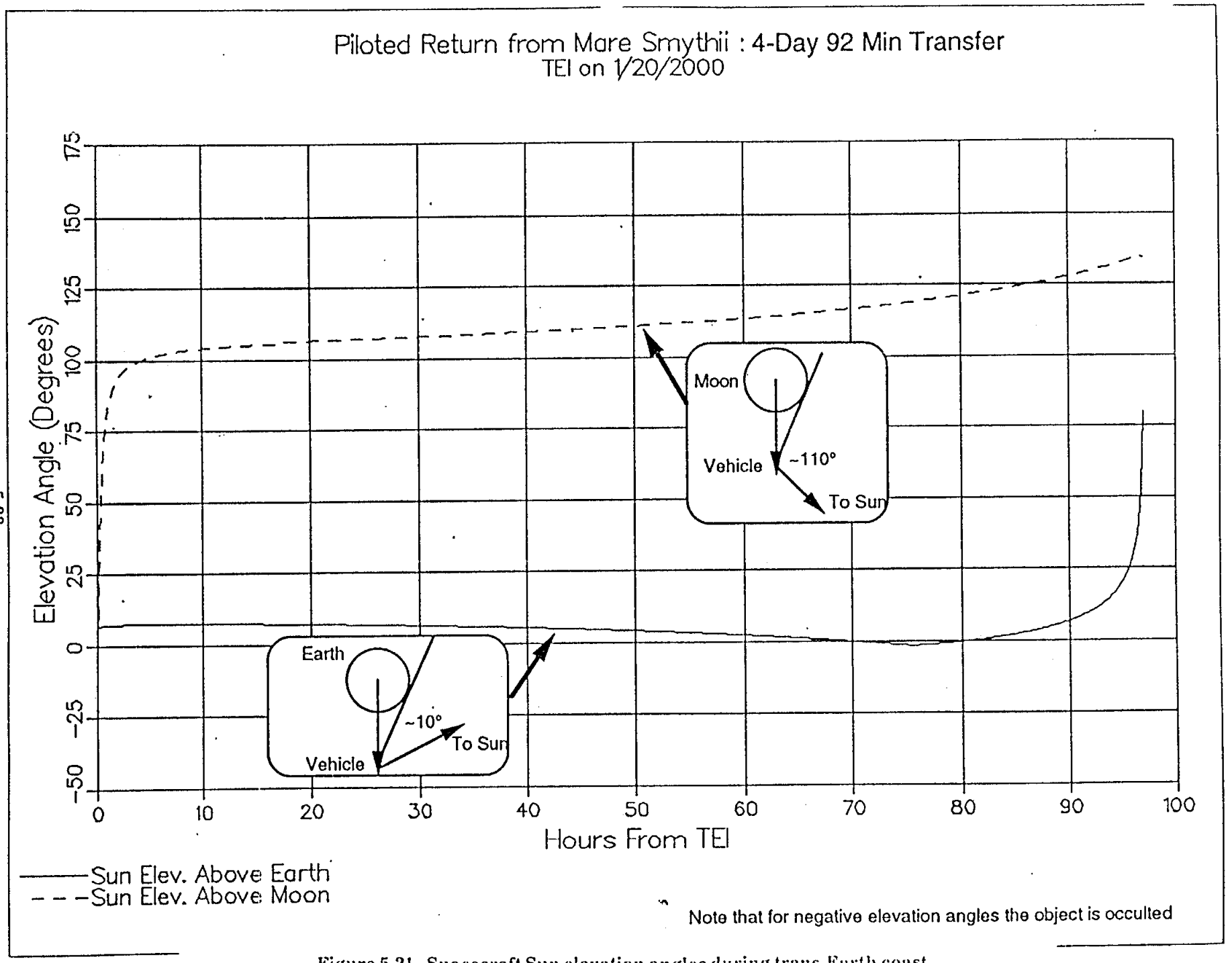
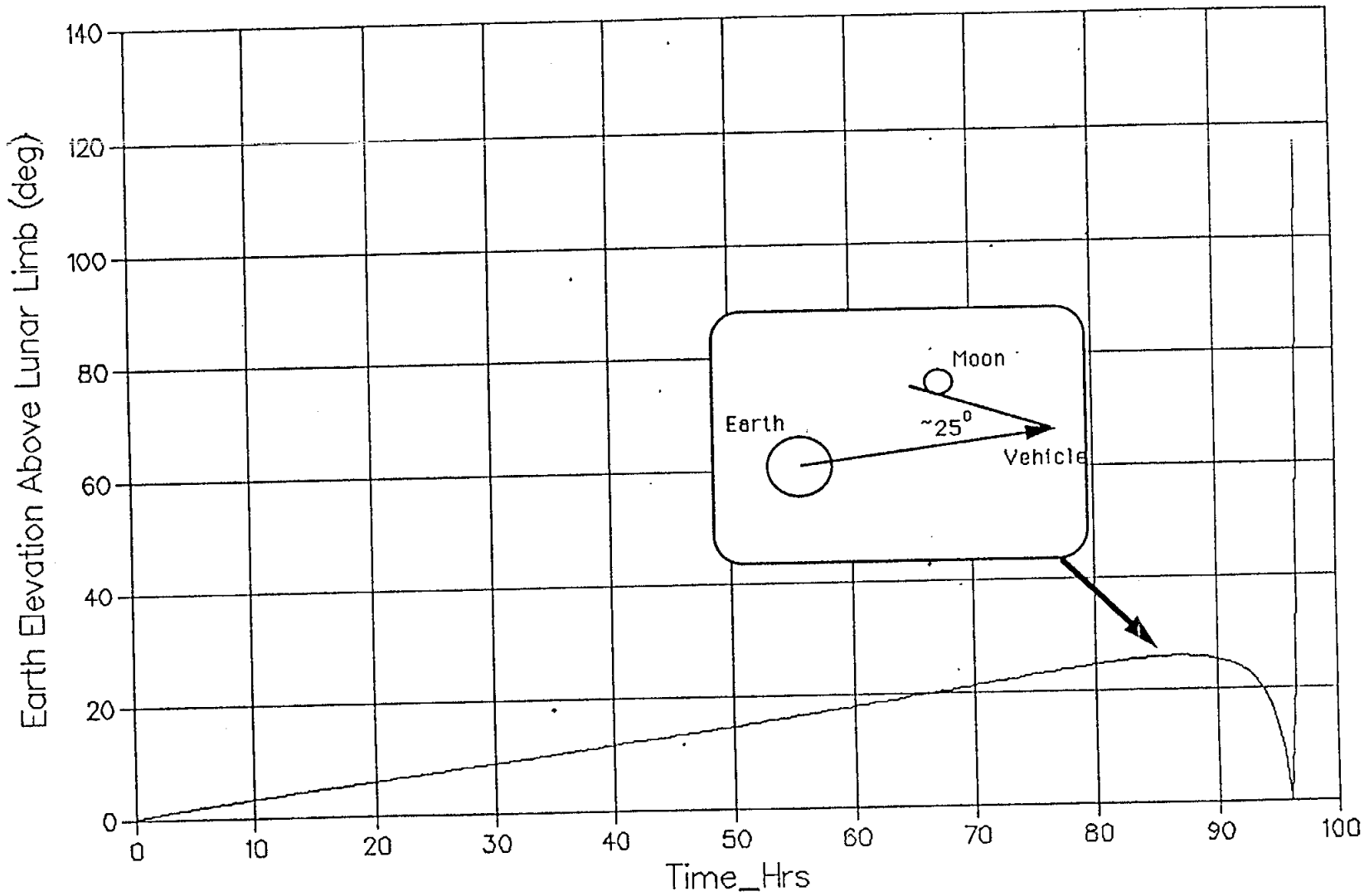


Figure 5-21. Spacecraft Sun elevation angles during trans-Earth coast.



Note that for negative elevation angles the object is occulted

Figure 5-22. Spacecraft Earth elevation angle during trans-Earth coast.

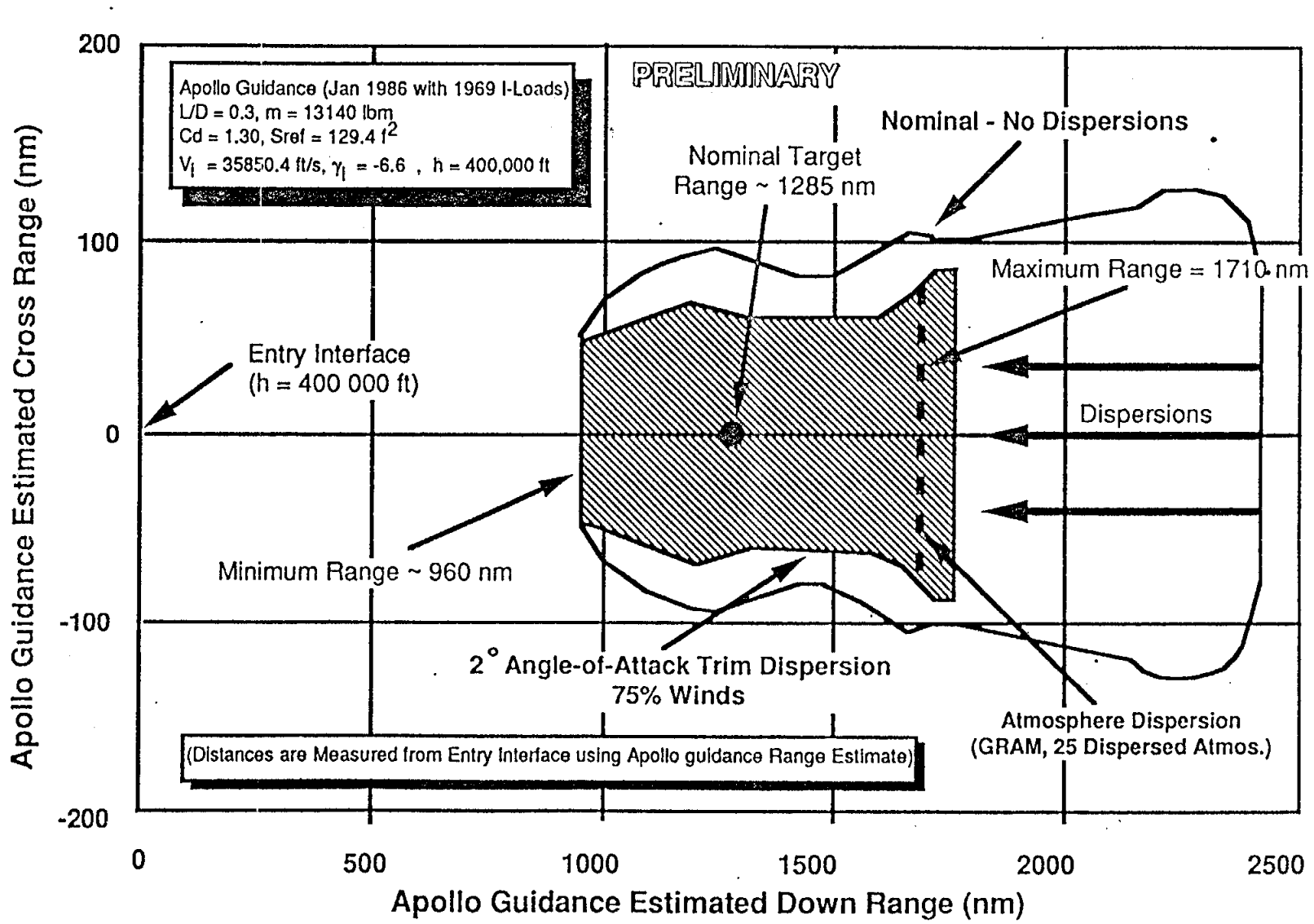


Figure 5-23. Apollo downrange versus crossrange.

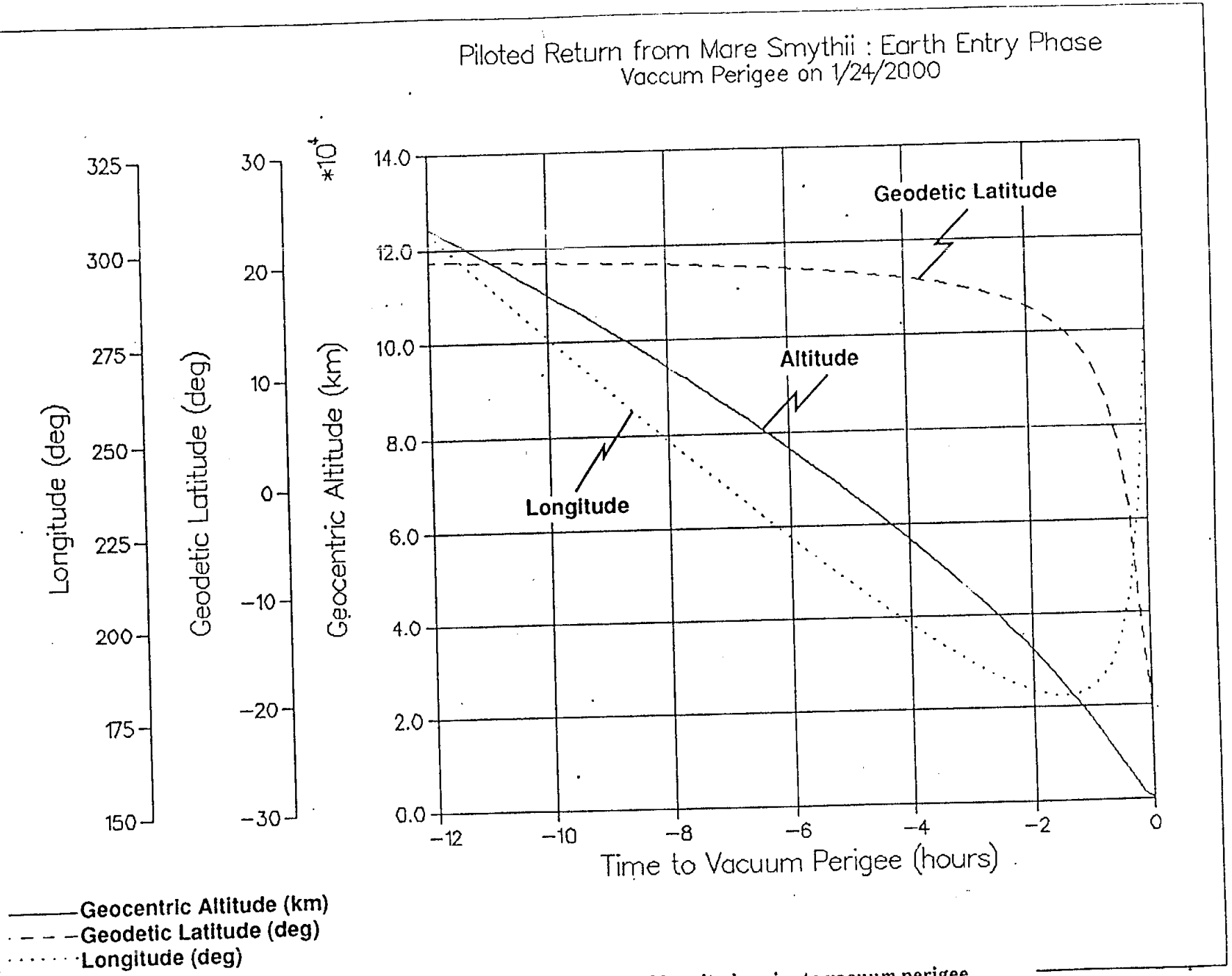


Figure 5-24. Spacecraft altitude, latitude, and longitude prior to vacuum perigee.

APPENDIX A
DELTA-V BUDGET

This delta-V budget for the space transportation system of the First Lunar Outpost is preliminary. The rationale and justification for these numbers can be found in appendix C.

Maneuver	ΔV (m/s)
Translunar Injection (TLI)	3200
Outbound midcourse	30
Lunar Orbit Insertion (LOI)	892
Deorbit	20
Descent	1853
Ascent	1832
Circularization maneuver	20
Trans-Earth Injection (TEI)	848
Inbound midcourse	30

The TLI, LOI, and TEI numbers are based on the worst-case Earth-Moon geometry over the 18.6-year cycle. Launch window penalties are included in the budget. The values in the above table, with the exception of the descent and ascent values, are impulsive delta-Vs.

APPENDIX B
GUIDELINES AND CONSTRAINTS FOR THE FIRST LUNAR OUTPOST TRAJECTORY

The guidelines used for this conceptual flight profile (CFP) reside in the review draft of the First Lunar Outpost Requirements and Guidelines (FLORG) document. The number in parentheses is the FLORG index number, and the words in italics are the exploration office rationale. Only those requirements and guidelines pertaining to the trajectory design for the First Lunar Outpost (FLO) study are shown below:

Minimize crew module time in orbit. (376)
Sizeable penalties in required mass to low Earth orbit (LEO) result from consumables and propellant needed to support the crew while they are in orbit.

Minimize orbital operations such as rendezvous, docking, and assembly for piloted and cargo vehicles. (377)
Orbital operations such as rendezvous, docking, and assembly can lead to sizeable penalties in required mission complexity, time in orbit, and mass to LEO.

Assume a single Earth launch vehicle will be utilized for each flight to the Moon. (246A)
The use of a single launch vehicle for each flight minimizes on-orbit operations by eliminating rendezvous and assembly, thus increasing the probability of performing translunar insertion (TLI) during the first window of opportunity.

Assume 60 days between cargo and piloted flights. (379)
This design goal minimizes the dormant period of the habitation system on the lunar surface between delivery of this system and first occupancy by a crew. It also sets a minimum time-spacing guideline for launch support.

Assume the necessary precursor missions will have been performed to sufficiently characterize the lunar landing site prior to outpost delivery. (244A)
A lunar video mapping mission is being planned and managed separately from the FLO mission and is expected to meet this assumption.

Assume piloted landings on the lunar surface will occur within 4 days following lunar sunrise. (495)
During lunar noon and lunar midnight, lunar surface temperatures reach extremes. In addition, the Sun angle at lunar noon washes out lunar surface features and causes extravehicular activity (EVA) to become complex. The assumption that piloted landings will occur within 4 days of lunar sunrise is based upon the desire to maximize the time available for science and exploration activities, many of which require crew EVA.

Assume that flight elements will land at least 1 km away from critical elements on the lunar surface to minimize blast ejecta damage. (380)
Preliminary studies indicate that 1 km limits launch/landing blast ejecta to acceptable levels.

For nominal missions and revisit missions, assume piloted landers will provide the capability to land within 100 m of the designated lunar landing site. (381)
Designated landing sites for the piloted flights will be chosen based upon the maximum safe walking distance between the piloted lander and the habitat, and the minimum walking distance between the piloted lander and the habitat.

The nominal launch date for the first mission will be in 1999. (309)
This date was chosen to drive schedules and is considered technically achievable.

Provide the capability for launch on any day of the lunar cycle. (411)
Launch windows are not constrained by unfavorable Earth/Moon geometries.

Nominal lunar surface stays will consist of a lunar day-night-day cycle (45 Earth days inclusive of contingencies). (147)
Provides for significant exploration capability per mission and falls within the life support and design capabilities that are realistically achievable by the year 2000.

Provide the capability to access any lunar latitude/longitude. (149A)
Derived from the science research and space resource utilization objectives. Global access is essential to provide a capability to reach areas of highest interest and potential return.

Provide the capability for abort during all mission phases. (383)
Derived from crew safety requirements. It does not imply continuous (at any time) abort capability, but rather a capability to declare and execute aborts during each mission phase.

Provide the capability for direct Earth reentry of the crew module. (386)
Based upon the desire to minimize propellant required by flight segment elements. Circularization of the crew module at Earth incurs mass to LEO penalties due to the additional propellant, which must be delivered to and extracted from the lunar gravity well.

Provide the capability to perform Earth landings on land at TBD locations. (498)
Returning crew and payloads to Earth is a primary capability to meet the mission objectives of returning the crew safely and enabling research. This requirement does not imply landing on all terrain conditions; therefore, specific landing locations will be established. Landing on land is the nominal option.

Provide the capability to perform Earth water landings. (373A)
Range safety considerations dictate launch trajectories that necessitate water landings in the case of an Earth launch abort.

Nominal Earth return landings will take place on land. (257A)
The current assumptions are (1) that refurbishment of the crew module is less costly for land-landings than for water landings, and (2) that crew and crew module recovery costs are less costly for land-landings than for water landings. Trades must be performed before this guideline can become firm.

The crew will be launched with the space transfer vehicle crew module. (248B)
Based upon the desire to avoid Space Transportation System (STS) launch and rendezvous for crew delivery. This guideline implies that the Earth launch vehicle will be man-rated.

Provide the capability to perform Earth landings under any lighting conditions. (256A)
Direct Earth reentry dictates that landing site lighting will be governed by the geometry of the Earth, Moon, and Sun system at the time of Earth entry interface. Due to the requirement to provide abort capability during all mission phases, landings can occur during darkness as well as daylight.

Provide the capability to support the crew for a total of 13 Earth days during a nominal mission. (388)
Based on the total number of days that the flight segment is required to support the crew on the Earth surface prior to launch, during transit between Earth and the Moon, on the lunar surface, and on the Earth surface following landing.

Provide the capability for abort at all times during piloted lunar descent. (414)
Derived from requirement to provide abort during all mission phases.

Provide the capability for landing site aimpoint redesignation. (415)
Required when the spacecraft controllers can see that the primary landing site is obstructed or contains a landing hazard.

Provide the capability for abort at all times following piloted lunar landing within 24 hours of abort declaration. (148A)
Based on the desire to provide an abort capability at all times following lunar landing without requiring that the piloted flight system be capable of lift-off at any time following lunar landing.

Nominal transit time for lunar transfer will be 96 hours and for Earth return transfer will be 108 hours. (394 and 399)
Based on a compromise to minimize crew cabin occupancy time without a large ΔV penalty. The nominal Earth return trip, 4 days $\pm$ 12 hours, is needed for longitude control.

Upon arrival at the Moon, the vehicle will inject into a temporary lunar parking orbit followed by deorbit.
Using a lunar parking orbit is safer, has operational advantages, has lower ΔV cost, and allows total landing site accessibility when compared to a direct (hyperbolic) descent.

APPENDIX C
TRAJECTORY-RELATED MISSION ANALYSIS FOR THE FIRST LUNAR OUTPOST

Trajectory-Related Mission Analysis for the First Lunar Outpost

1. REQUIREMENTS

The transportation system must be designed to provide access to any arbitrary point on the lunar surface.

In the interest of economy, the transportation system shall include only a single manned pressure vessel.

Nominally, the flight crew will stay on the lunar surface for 45 days.

The flight crew must have the option of returning to a selected site on terra firma within 5 days of any arbitrary lunar liftoff time.

2. ASSUMPTIONS

A lunar habitat will be delivered to the landing site by a separate cargo flight, and checked out remotely before the flight crew leaves earth on the manned flight. The nominal time between landing of the habitat and the arrival of the first crew is 60 days.

Individual flights will be launched by a heavy-lift vehicle big enough to inject a fully integrated cargo or manned spacecraft into a translunar trajectory with a single launch.

The manned round-trip flight will not involve rendezvous in lunar orbit. The complete manned spacecraft will descend to the lunar surface, thus avoiding the need for two separate pressure vessels, and facilitating any-time return to the earth.

Return to earth will be accomplished with an Apollo-type atmospheric entry module, which will be used for a direct entry and landing.

3. NOMINAL OPERATIONS

3.1. Atmospheric Reentry and Landing

The lift-to-drag ratio (L/D) for the Apollo entry module is about 0.3, which provides enough aerodynamic control to vary the down-track position of the landing point by several hundred kilometers. However, it does not provide a very significant cross-track maneuvering capability.

In an inertial frame of reference like that illustrated in Figure 1, the nominal Apollo touchdown point is generally located within a few degrees of the geocentric position occupied by the lunar antipode at the time of departure from the moon. Changing the return transit time has very little effect on the geocentric transfer angle, but it does allow the earth to rotate in the inertial frame

and thus change the longitude of the touchdown point. Such a change has little effect on solar illumination of the landing site, because the local time of day at touchdown remains very nearly constant. As shown in Figure 1, the illumination of the earth landing site can be visualized pretty accurately by observing (from the earth, at the time of lunar departure) the illumination at the center of the moon's visible hemisphere. It is obvious that anytime-return from the moon implies the necessity of night landings on the earth.

Figure 2 shows how the inertial speed at the atmospheric entry altitude (121.92 km, by long-standing convention) is affected by the duration of the return transit from the moon. For reasonable transit times, this variation is insignificant compared to that which is imposed on the entry airspeed by variations in the entry azimuth.

For a fixed lunar departure date, maximum control of the earth landing latitude is attained by varying the atmospheric entry coazimuth (i.e., the azimuth measured from east instead of north) over a range of ±90 degrees, as illustrated in Figure 3. The two extremes of this range are obtained on two geocentric polar trajectories, one passing over the south pole during the return from the moon, and the other passing over the north pole.

3.2. Departure from the Moon

When the inclination of the lunar orbit is at its maximum value of 28.5 degrees, Figure 4 shows that the ability to attain both of the aforesaid extremes is dependent on the ability to change the azimuth of the geocentric return trajectory, at the point of lunar departure, by as much as 118.5 degrees relative to the azimuth of the lunar orbit. This relative azimuth corresponds to the coazimuth of lunar departure in a quasi-inertial geocentric frame sometimes identified by the abbreviation EMS (for earth-moon system), which is referenced to the instantaneous position of the moon and the plane of its orbit.

Figure 5 shows, for the moon at each extreme of its geocentric distance, how the selenocentric departure energy varies with transit time when the EMS departure coazimuth is targeted for its most economical value of zero degrees. Figure 6 depicts similar data for the situation where the required magnitude of the EMS departure coazimuth is 118.5 degrees.

For hyperbolic trajectories (like those under consideration here), the orbital energy parameter C_3 is equal to the square of V_∞, which is the magnitude of the limiting velocity of the spacecraft as its distance from the center of attraction approaches infinity. The direction of the limiting velocity is parallel to the departure asymptote of the hyperbola. As indicated in Figure 7, the selenographic longitude of the asymptote direction is strongly dependent on the inbound transit time. The selenographic latitude of the asymptote direction depends strongly on the EMS departure coazimuth, and has a limiting magnitude of about 18 degrees.

The propulsive velocity increment required to attain a particular departure energy depends on the thrust-to-weight ratio of the propulsion stage, and on the strategy used to get into the lunar departure hyperbola. For some strategies, it also depends on the relative orientation of the departure V_∞ vector and the position vector of the lunar site. The propulsive velocity increment, ΔV_{IDEAL}, is that which the propulsion stage would be able to deliver in the absence of a gravity

Figure 1.
Effect of Lunar Phase on Lighting at Earth Landing Site

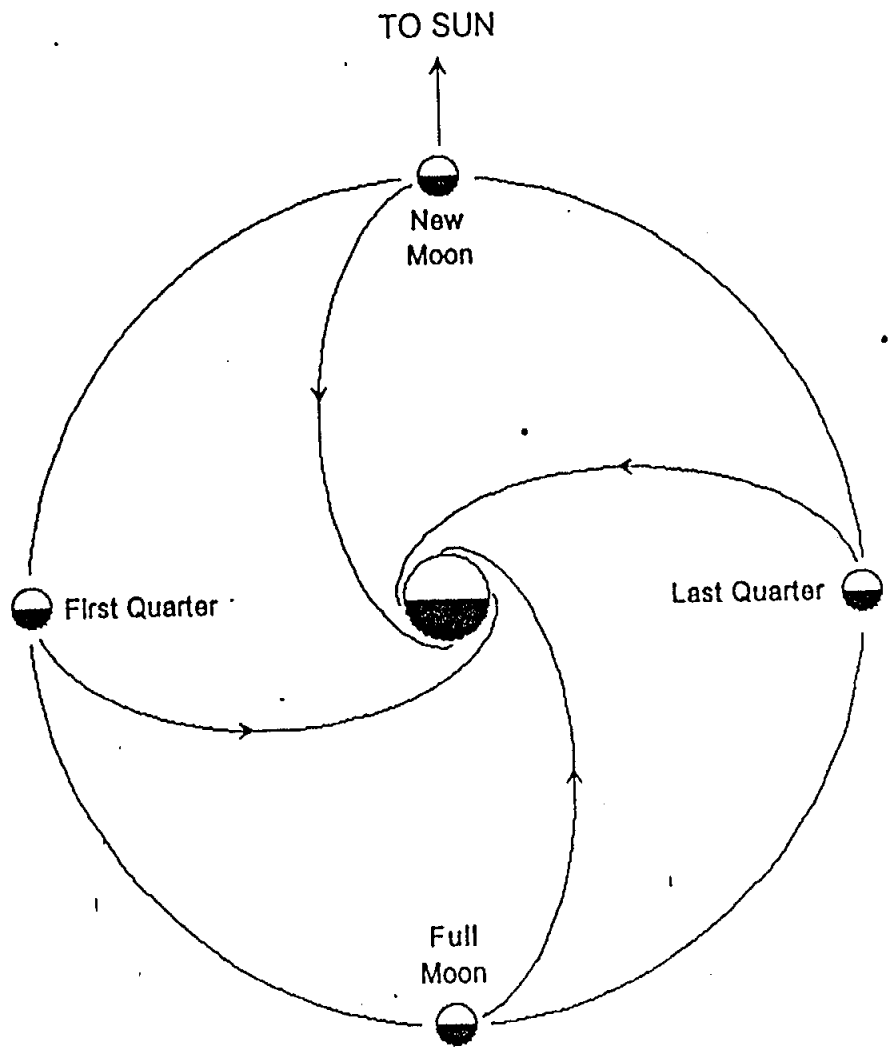


Figure 2.
Atmospheric Entry Speed
Transfer Orbit Incl. = Zero wrt Earth-Moon Plane

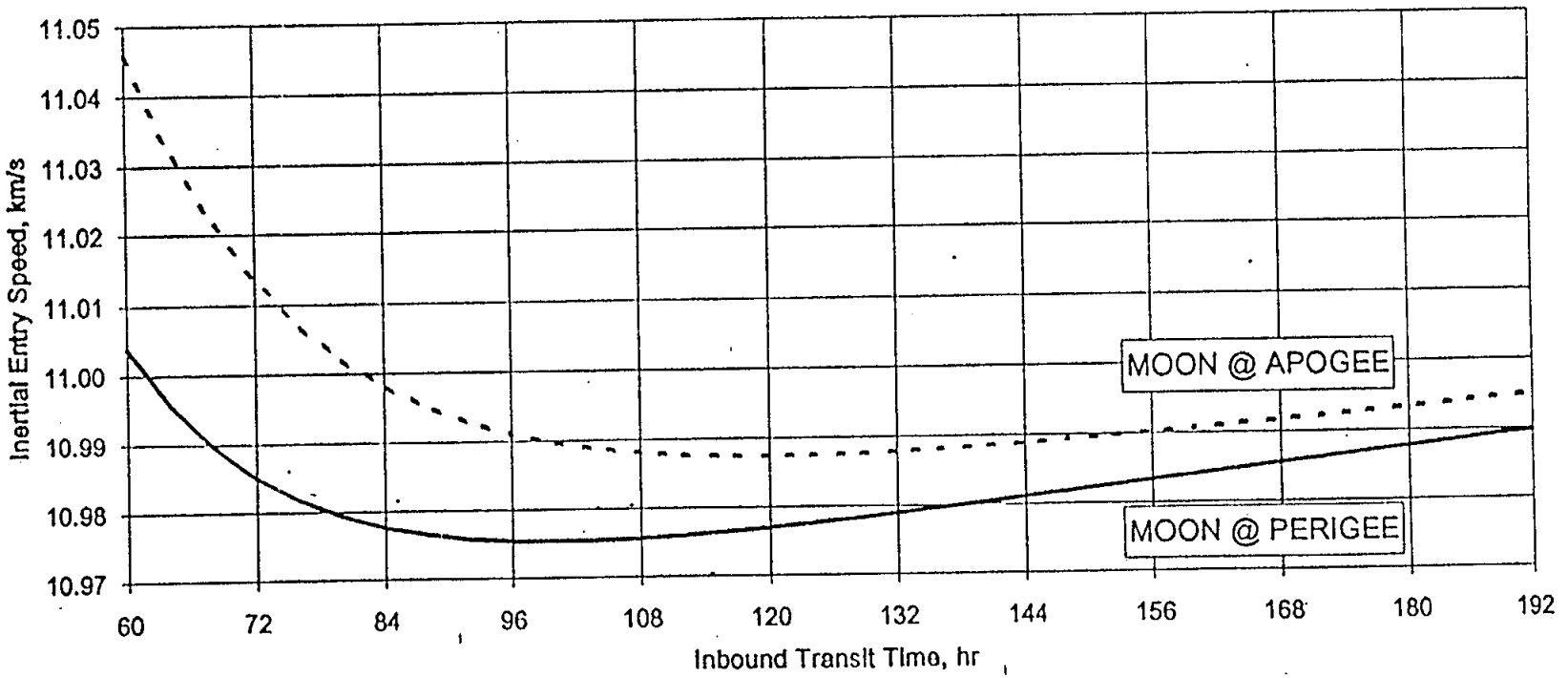


Figure 3.
Entry and Landing Ground Track

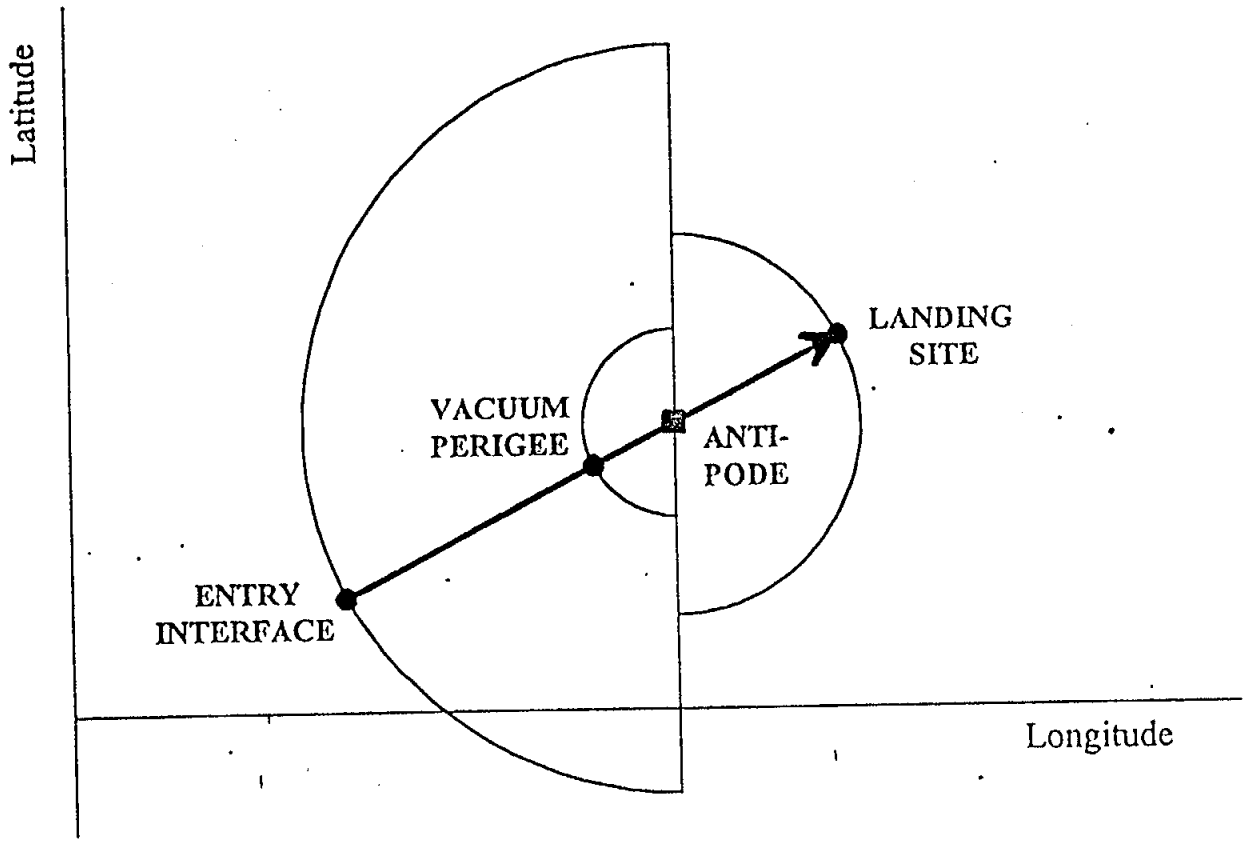


Figure 4.
Geocentric Plane-Change Extremes at TEI

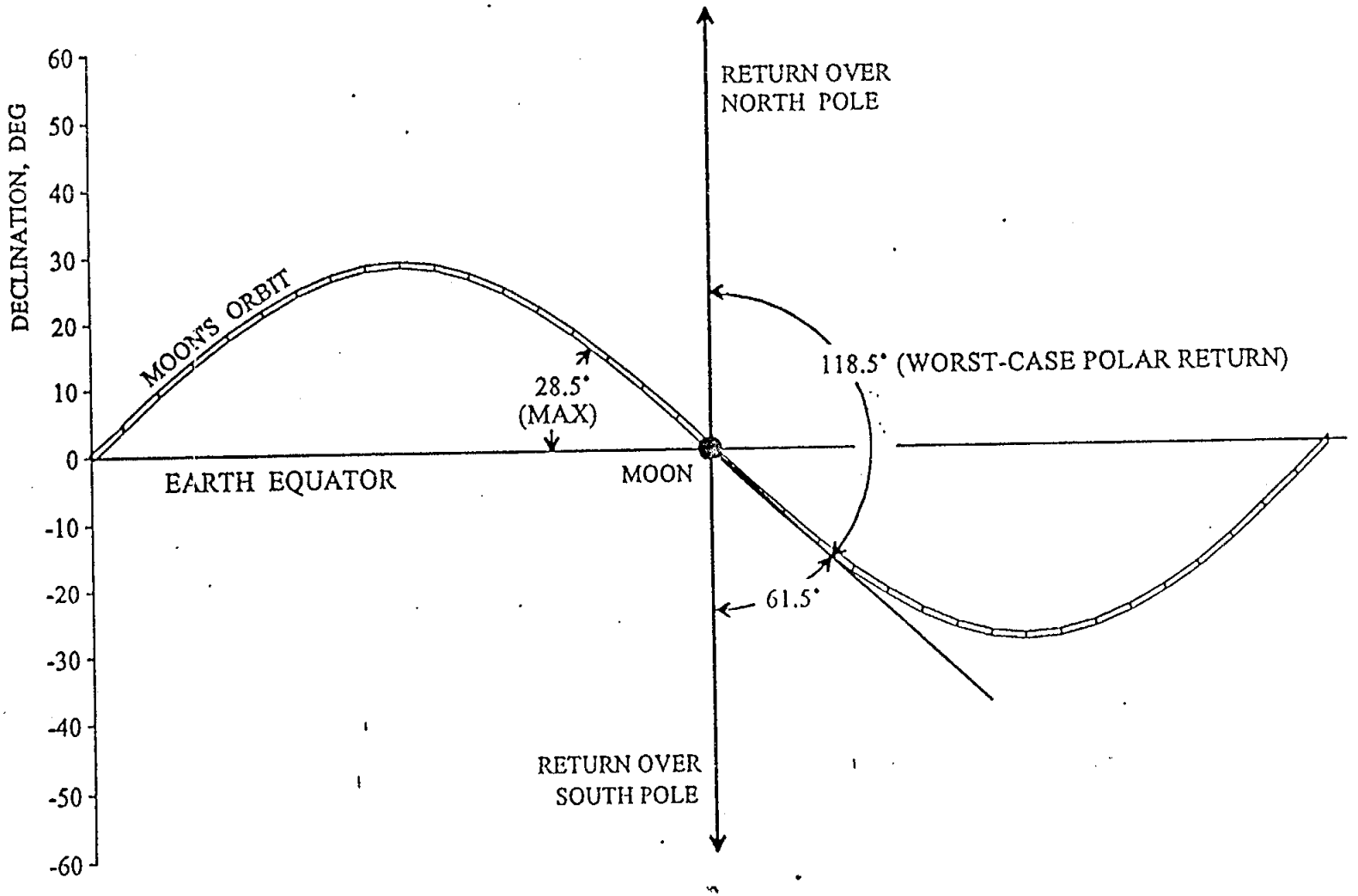


Figure 5.
Selenocentric Departure Energy
Transfer Orbit Incl. = Zero wrt Earth-Moon Plane

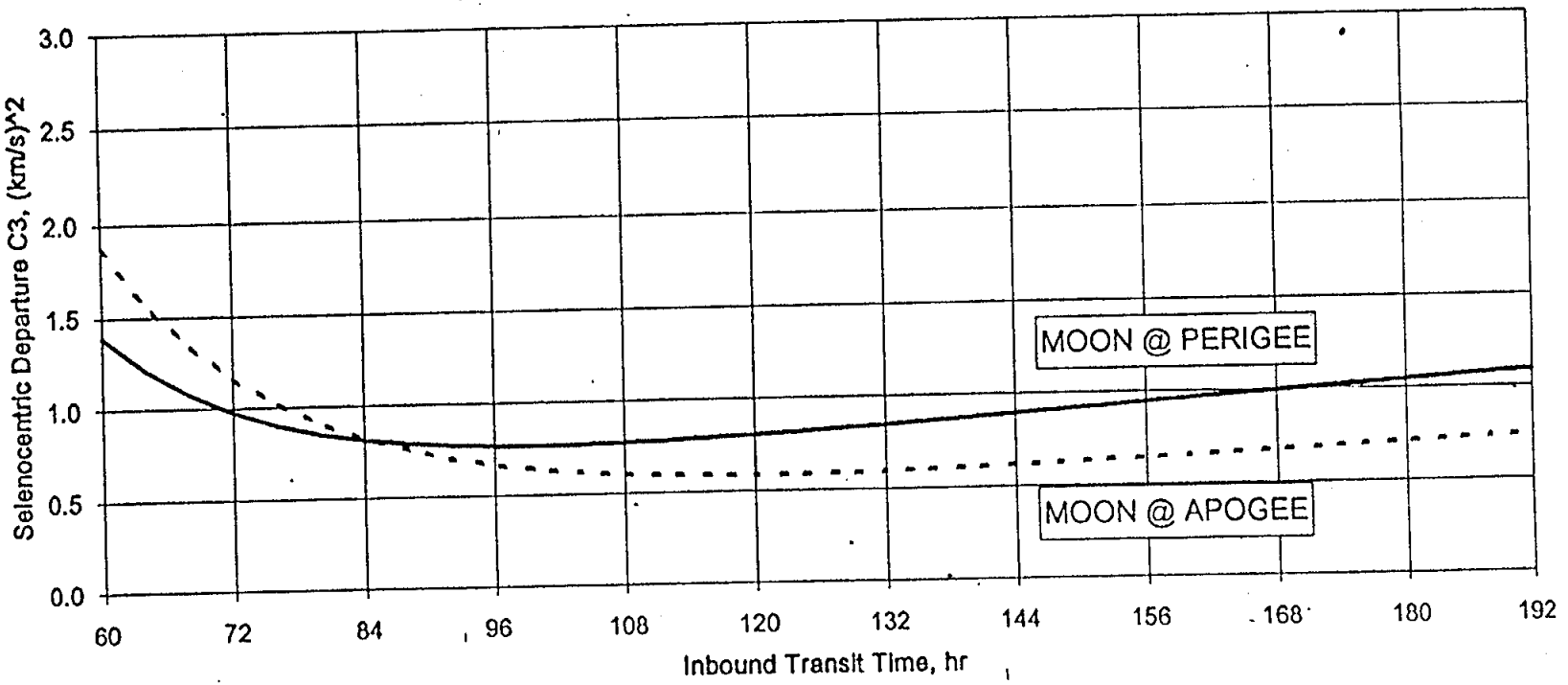


Figure 6.
Selenocentric Departure Energy
Transfer Orbit Incl. = 118.5° wrt Earth-Moon Plane

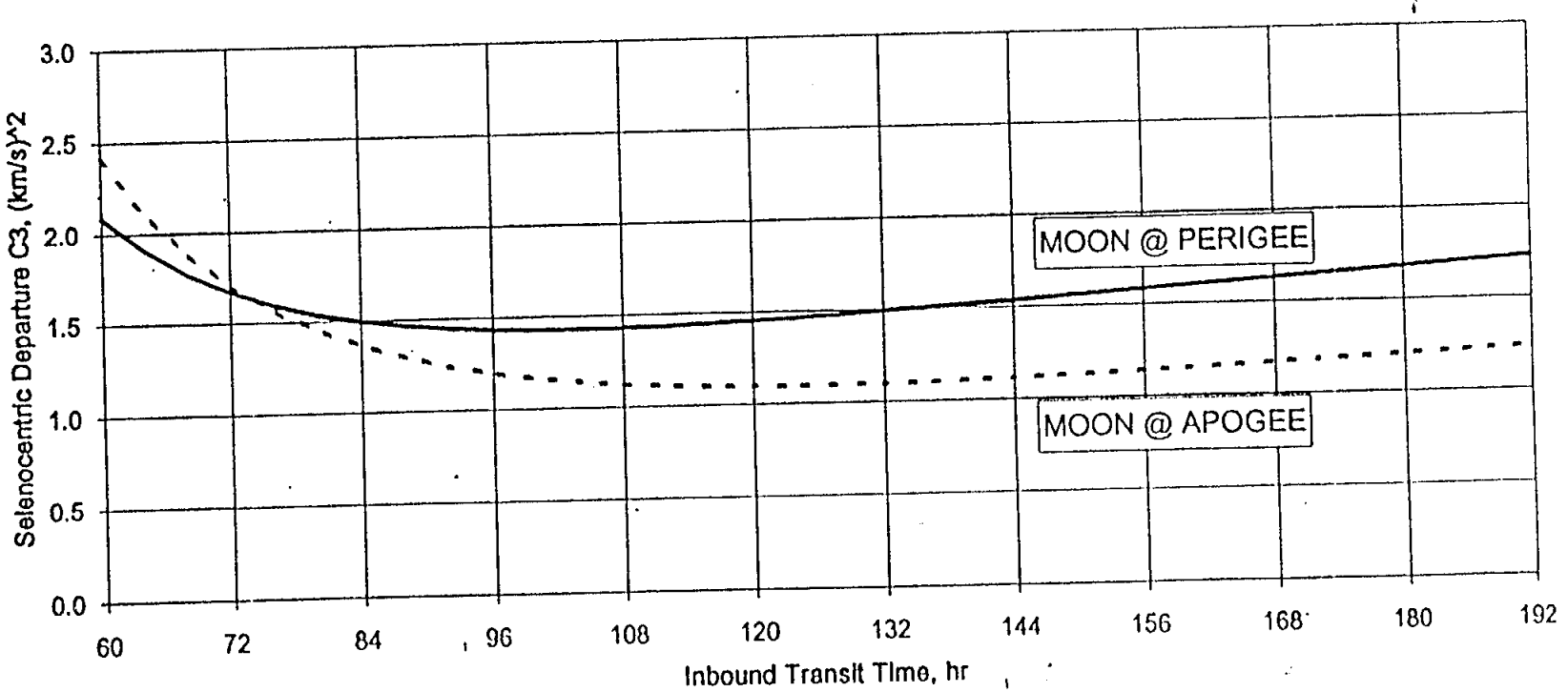
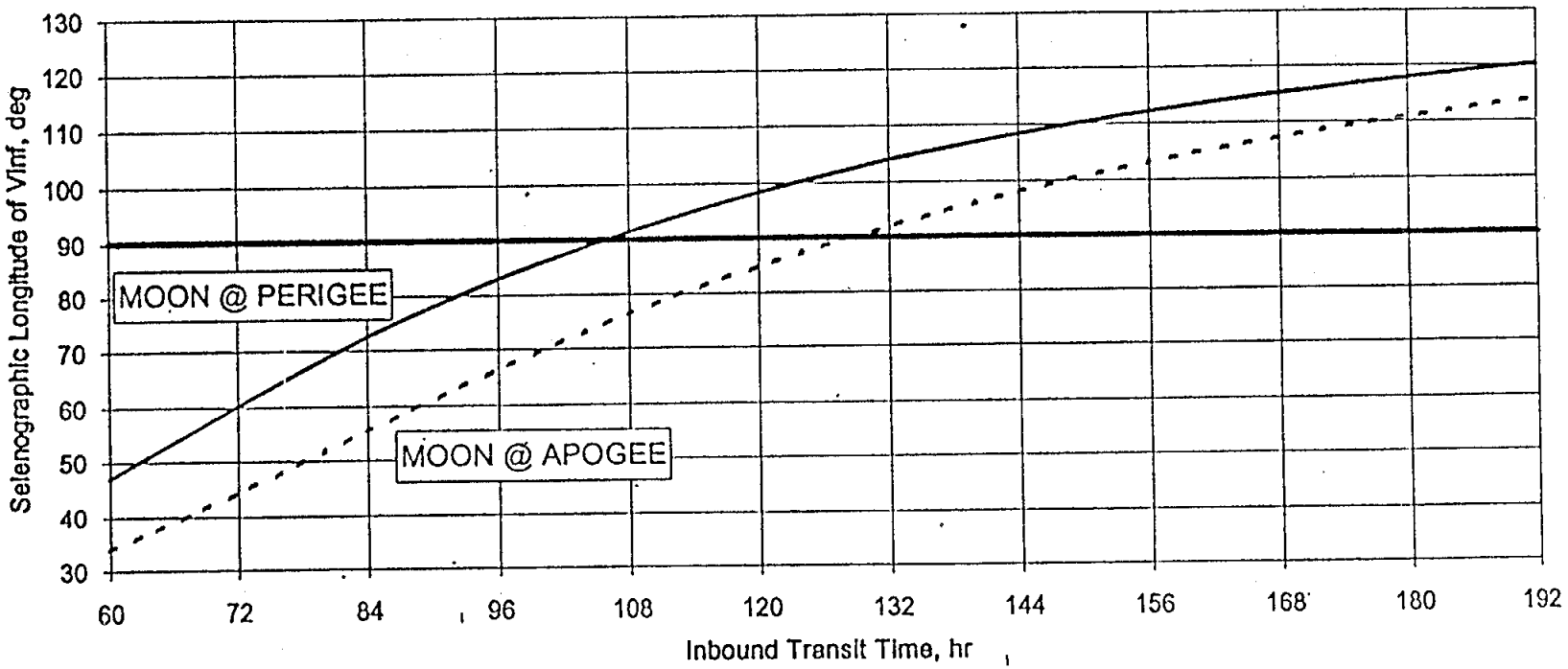


Figure 7.
Direction of Departure Asymptote
Transfer Orbit Incl. = Zero wrt Earth-Moon Plane



field. Its magnitude, along with burnout weight and specific impulse, determines how much propellant is required for a particular powered-flight phase.

Conceptually, the simplest solution is to compute the hyperbolic velocity needed at the liftoff position (as if the V_∞ vector were to be attained in coasting flight from that point) and then make the engine thrust large enough to attain the required velocity in a negligible amount of time. This approach is impractical, of course, because of the prohibitive engine weight that is necessary to produce the required thrust. It also fails to meet the requirement for global access to the moon because, for a given return-to-earth transit time, there is a fairly large region from which it would be necessary to fly beneath the lunar surface to reach the V_∞ vector.

When realistic engines are considered, it is found that ΔV_{IDEAL} increases as thrust becomes smaller, because of the necessity for overcoming lunar gravity during an increasing duration of powered flight. The finite-thrust analogue of the impulsive solution described in the preceding paragraph therefore involves a tradeoff between engine weight and the weight of the extra propellant required to counteract the "gravity loss" (i.e., the increase in ΔV_{IDEAL}) that occurs with smaller engines.

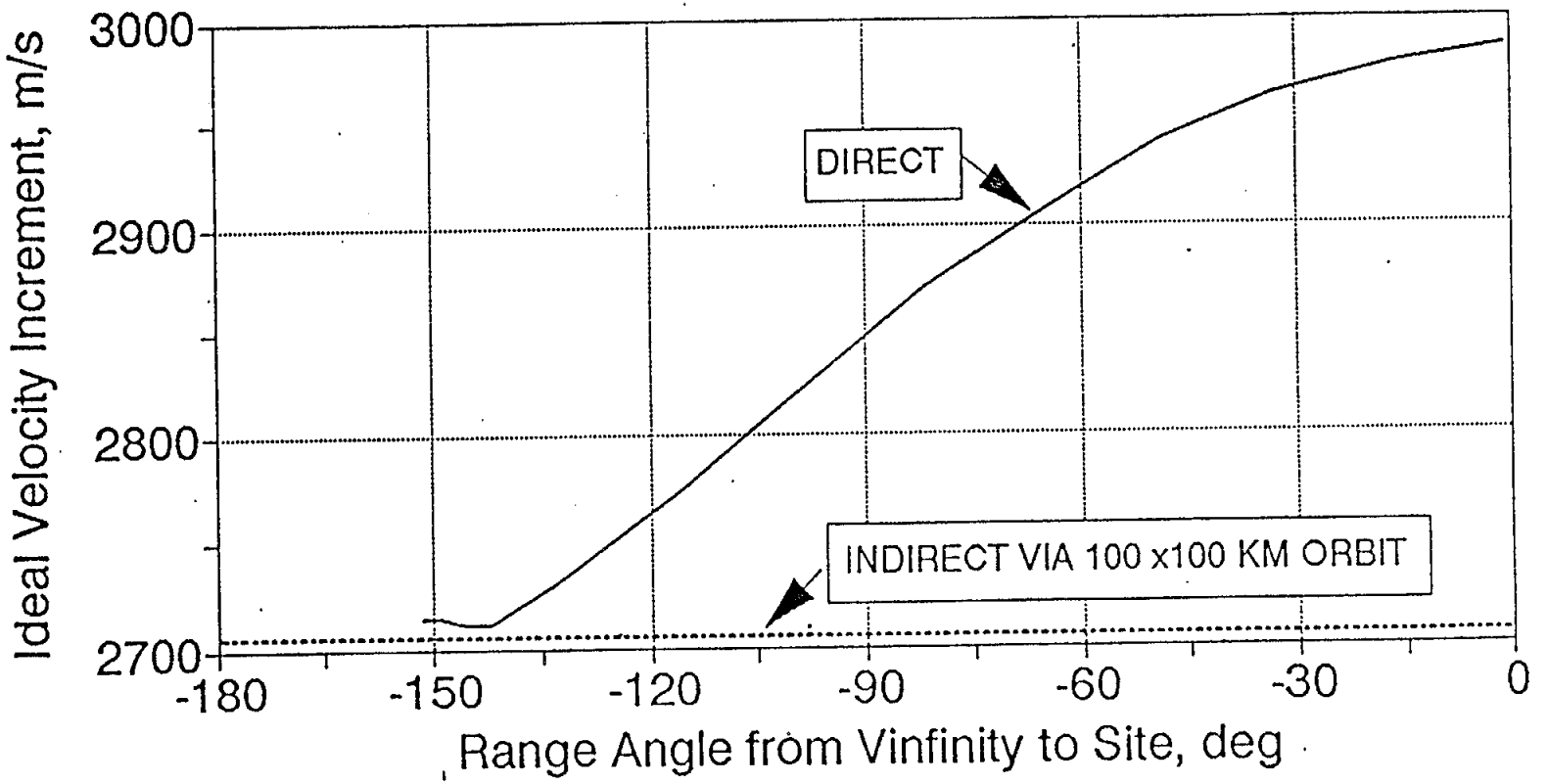
It is not only the magnitude of the engine thrust that needs to be optimized, but also its direction as a function of time. When the position vector of the lunar site is aligned with the V_∞ vector, the optimum thrust direction is straight up from the surface for the total duration of the burn, and the gravity loss is rather large. As the angular separation between the two vectors is increased, the gravity loss becomes smaller until the optimum separation is reached at an angle (somewhere between 90 and 180 degrees) whose value depends on the magnitude of the V_∞ vector. Figure 8 shows, for a representative V_∞ magnitude and a near-optimum thrust-to-weight ratio, how the required propulsive velocity increment varies with the angular separation under consideration. It should be noted that the minimum propulsion requirement for direct injection into the hyperbola is very nearly equal to that which is required for indirect injection, i.e., by way of a low-altitude intermediate circular orbit.

The important thing to understand about Figure 8 is that the minimum value of ΔV_{IDEAL} enables direct injection only *from a narrow annulus of sites*, whereas the same value enables indirect injection *from every point on the lunar surface*. The only possible disadvantages of indirect injection that can be brought to mind are that (1) it requires two burns instead of one, and (2) for a given injection energy, it can require as much as a two hours longer to reach earth, because of the required coasting time in the intermediate orbit. Neither of these is thought to be serious enough to override the advantage of the smaller propulsive velocity increment.

3.2.1. Transearth Injection (TEI)

Figures 9 and 10 show the TEI velocity increment magnitudes that correspond to the energy parameter values shown in Figures 5 and 6, assuming impulsive coplanar departure from a circular parking orbit having an altitude of 100 km above the lunar surface. Coplanarity of the parking orbit and the departure hyperbola requires only that the ascent azimuth from the lunar site be chosen properly. The propulsive velocity requirement for ascending from the lunar surface to the parking orbit is addressed in Section 3.2.2.

Figure 8.
Delta-V for Lunar Departure
Vinf = 1000 m/s, (F/W)o = 0.45 Earth g



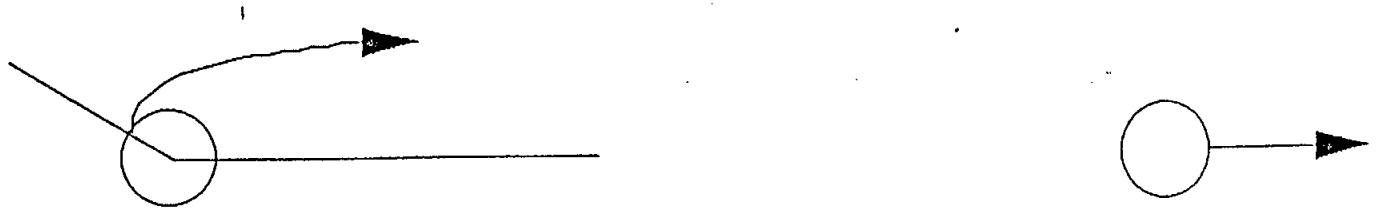


Figure 9.
TEI Delta-V from 100x100 km Lunar Orbit
Transfer Orbit Incl. = Zero wrt Earth-Moon Plane

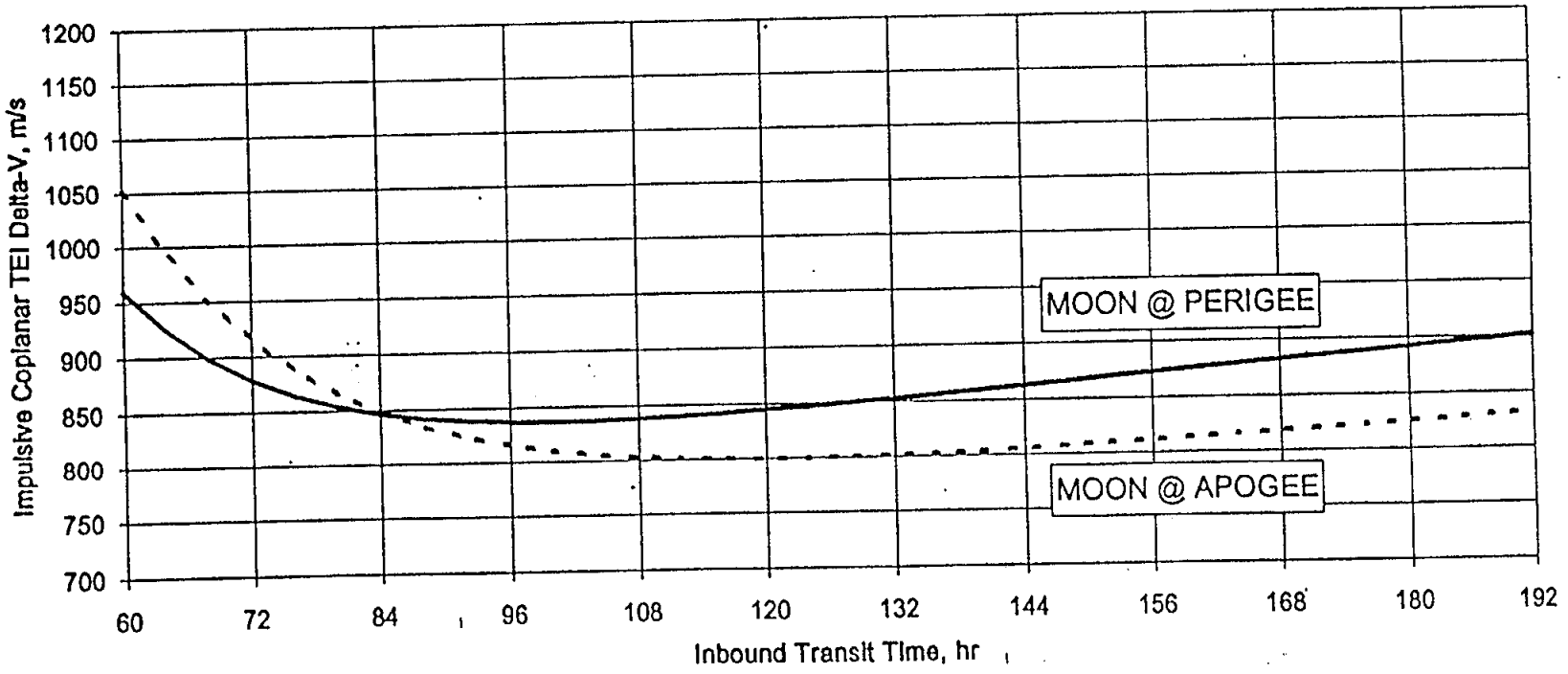
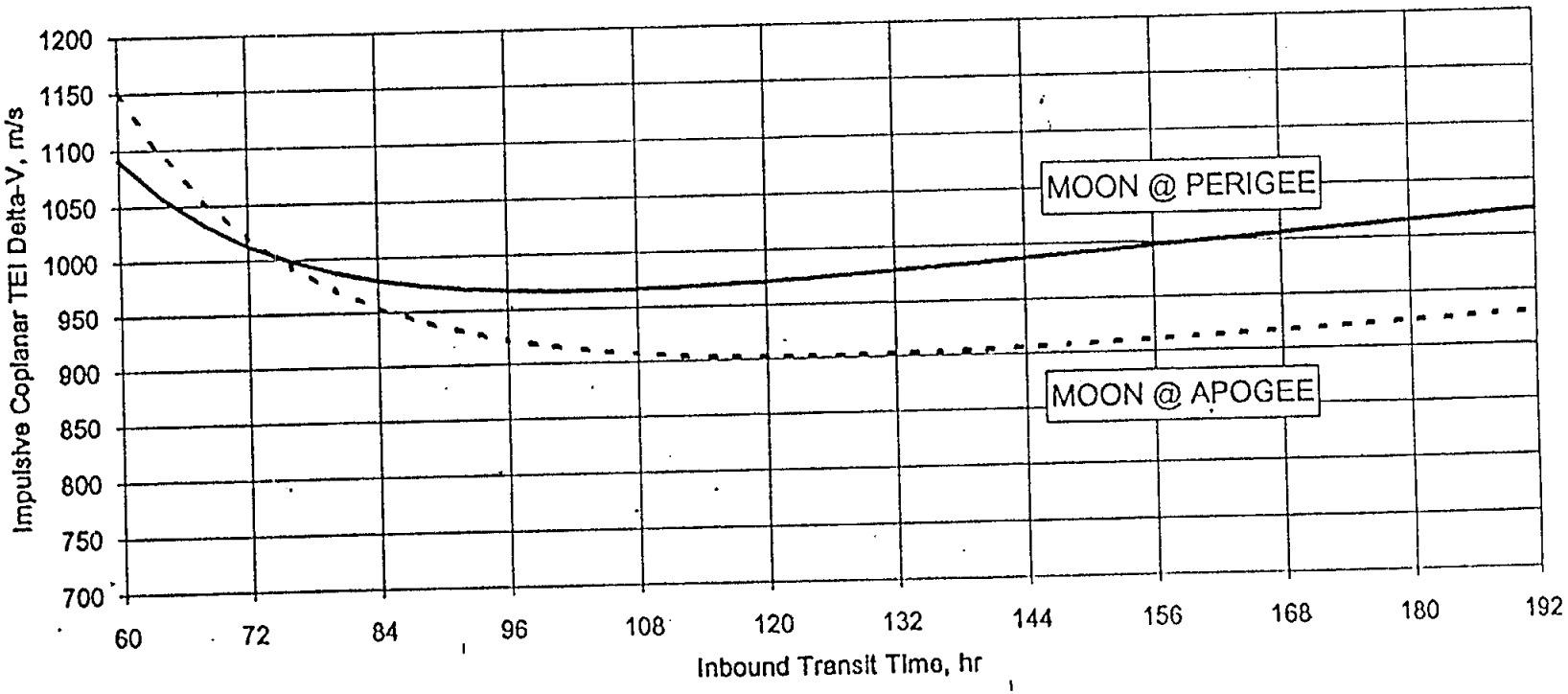


Figure 10.
TEI Delta-V from 100x100 km Lunar Orbit
Transfer Orbit Incl. = 118.5° wrt Earth-Moon Plane



Assuming TEI is executed by restarting the ascent stage, the thrust-to-weight ratio during that burn should be great enough to limit the error of the impulsive approximation to something less than 2 m/sec. Nevertheless, the data in Figures 9 and 10 (and also in Figure 2) should not be assumed to have a greater accuracy than about 10 m/sec. That much variation can occur because of differences between retrograde and posigrade departures from the moon (either of which is possible from any particular site), monthly and seasonal variations in solar perturbation of the transfer trajectory, and changes in the effects of lunar and earth oblateness that occur with different trajectory orientations.

3.2.2. Initial Ascent from the Lunar Surface

Figure 11 shows, as a function of liftoff thrust-to-weight ratio, the ΔV_{IDEAL} required for ascent from the lunar surface to circular orbits having altitudes of 50, 100, and 150 km. The assumed value for specific impulse (Isp) is 300 seconds, which is representative of storable hypergolic propellants. As can be seen in Figure 12, Isp variations on the order of ±10 seconds have an insignificant effect on the propulsive velocity increment (but not on the propellant requirement, which is more important). The data shown in Figures 11 and 12 are for ascent trajectories that were lofted, for the sake of terrain clearance, above those that would have required the absolute minimum propulsive velocity increments. Figure 13 shows a typical altitude vs range profile.

Determination of a thrust-to-weight ratio for the ascent stage that will minimize the spacecraft weight is a complicated process involving Isp, engine weight variation with thrust, and various structural factors. When off-the-shelf engines are to be used, such a determination is complicated by the necessity of treating thrust as a discrete variable rather than a continuous one. Therefore it is not possible, on the basis of trajectory considerations alone, to say what value of ΔV_{IDEAL} ought to be used to design the ascent stage.

It is obvious in Figure 11 that low orbit altitudes impose lower propulsion requirements for this phase of the departure than do the higher ones. This advantage is partially offset by greater TEI ΔV magnitudes at the lower altitudes, but not entirely so. For overall minimization of spacecraft weight, then, the parking orbit should be as low as possible. But considering possible errors in navigation and maneuver execution, and of orbital perturbations, there is some risk of striking a lunar mountain peak if the nominal altitude is set too low. The swath of lunar terrain visible from orbit also becomes narrower with decreasing altitude, which may be significant in relation to navigation procedures involving landmark or surface beacon tracking. The nominal parking orbit altitude for Apollo missions was 108 km, and it seems unlikely that the missions under consideration here would require any higher value.

3.3. Arrival at the Moon

Selenocentric arrival trajectories are essentially mirror images of their departure counterparts, the plane of symmetry being that of the lunar prime meridian. In line with the detailed discussion of departure considerations in Section 3.2, descent by way of a low-altitude intermediate circular orbit is preferred over direct hyperbolic descent, because the direct descent

Figure 11.
Effect of Orbit Alt on Ascent Delta-V
Isp = 300 sec

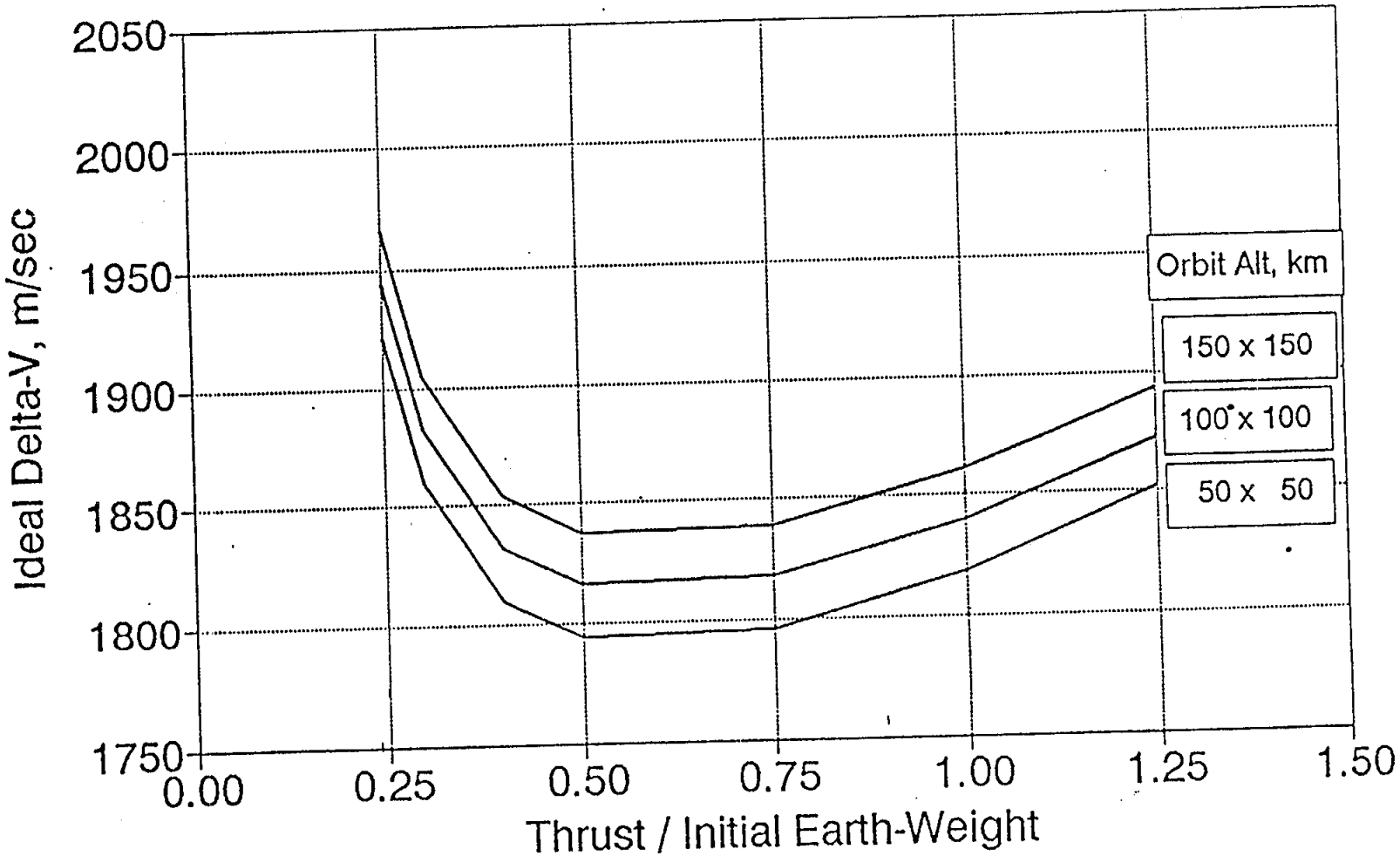


Figure 12.
Effect of Isp on Ascent Delta-V
100x100 km Orbit

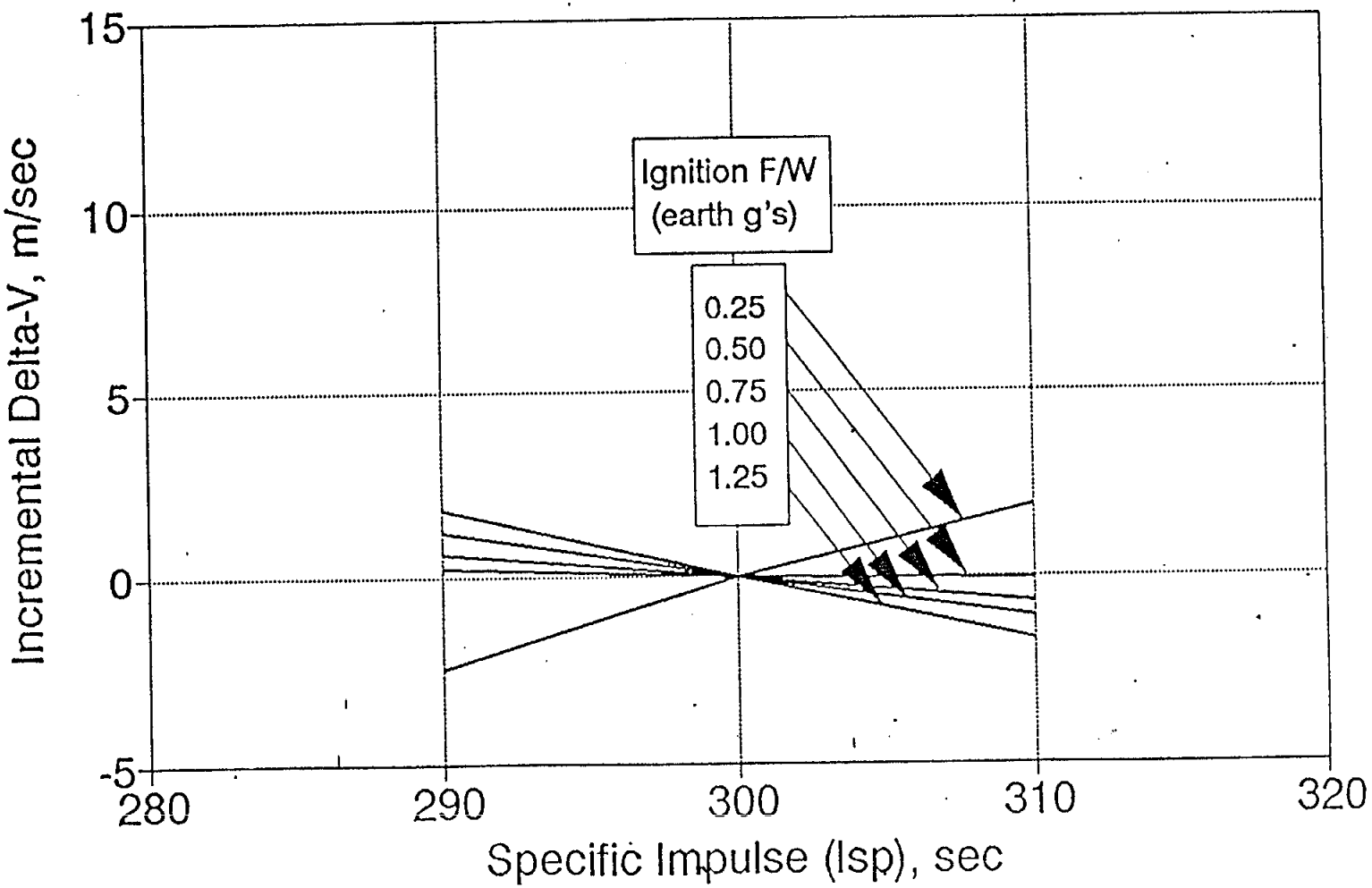
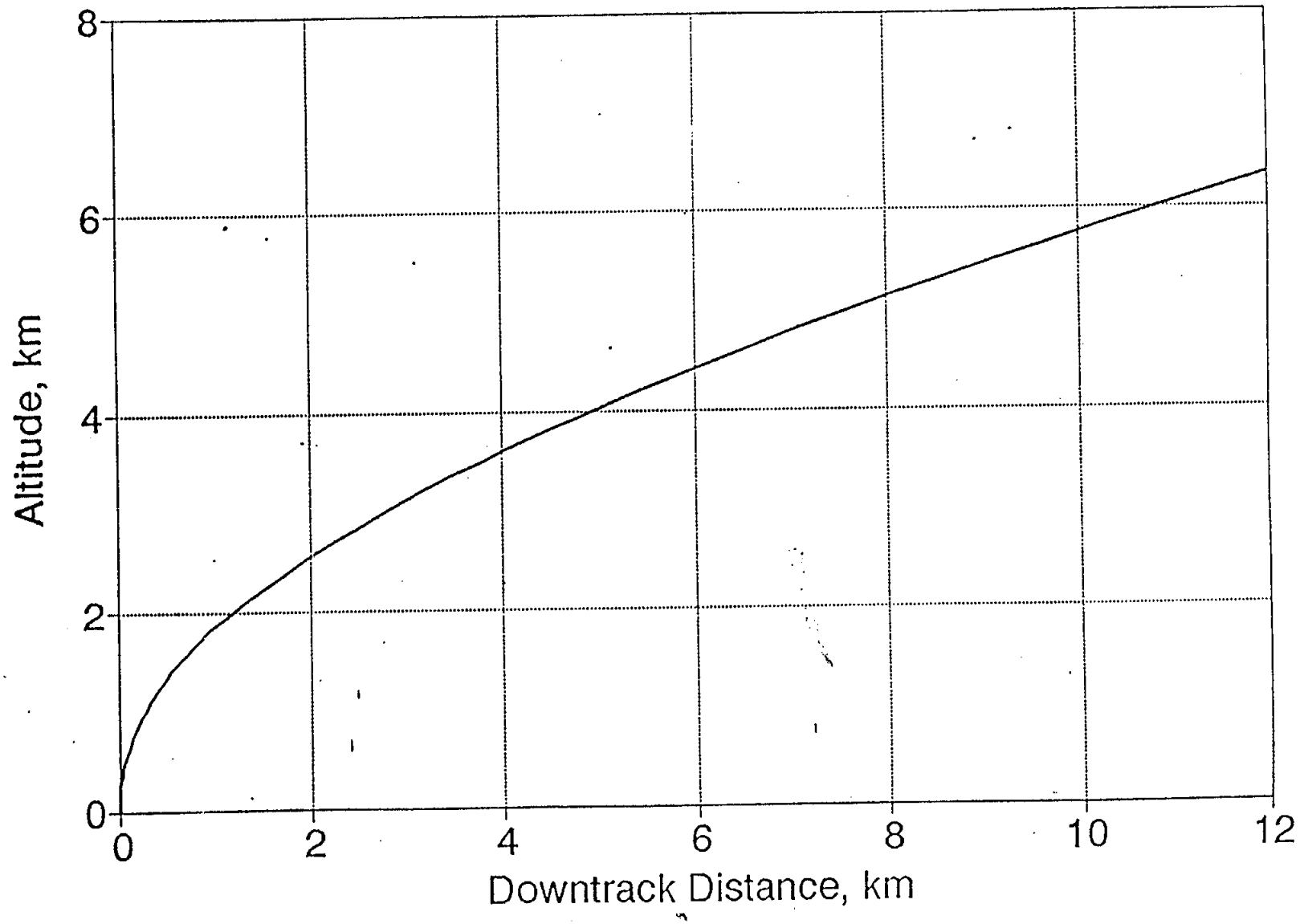


Figure 13
Ascent Altitude/Range Profile



(1) generally requires a higher propulsive velocity increment and (2) does not provide global access to the lunar surface.
In the case of arrival, there is also a flight safety consideration. For most landing sites, direct hyperbolic descent would require perisel of the approach trajectory to be located below the lunar surface --- sometimes very far below it. In such cases, impact is imminent at the time of descent stage ignition, and an ignition failure would almost certainly be catastrophic.

With an indirect approach, perisel altitude of the approach hyperbola coincides with the altitude of the parking orbit, and a similar ignition failure (at the beginning to the lunar orbit insert burn) is relatively benign in its effect. In such an event, flyby abort studies show that the propulsive ΔV of the (separate) ascent stage is more than enough to achieve a safe return to earth if the abort maneuver is executed within 12 hours or so of perisel passage.

3.3.1. Final Descent to the Lunar Surface

Figure 14 shows a typical altitude vs range profile for the lunar descent trajectories. They have been lofted to assure terrain clearance in the vicinity of the landing site, and also to provide a clear view of the site during terminal approach.

Figure 15 shows how ΔV_{IDEAL} varies with thrust-to-weight ratio of the descent stage, which is assumed to use cryogenic propellants. It also shows the effect of differences in the specific impulse, which is very small insofar as the propulsive velocity increment is concerned. Figure 16 shows how ΔV_{IDEAL} is affected by different altitudes of the initial capture orbit. It should be noted in both of these figures that thrust-to-weight ratio is based on *nominal* (full-throttle) thrust and *final* earth-weight (i.e., what the vehicle would weigh at the instant of lunar touchdown if it could be transported magically to the earth surface and weighed there). Using a throttling profile similar to that of the Apollo descent stage, the actual thrust was varied during the descent so that the acceleration due to the engine(s) almost exactly counterbalanced that due to lunar gravity at the end of the descent.

3.3.2. Lunar Orbit Insertion

Figure 17 shows, for the moon at the extremes of its geocentric distance, how the longitude of the lunar approach asymptote varies with transit time from the earth. For a hypothetical observer at the center of the moon, the asymptote points in the direction of the spacecraft at a very distant point on the approach hyperbola. This direction is opposite to that of the arrival V_∞ vector (i.e., the moon-relative velocity of the approaching spacecraft).

The symmetry of the approach and departure asymptotes in the selenographic reference system can be seen by comparing Figures 7 and 17. It is important to understand that the longitude coordinates in both of these figures are referenced to the selenographic frame *at the instant of perisel passage*. Since both of the asymptotes are fixed in inertial space, and the selenographic frame is rotating, the longitude of either asymptote is continually decreasing at a rate of 0.549 degrees per hour. This variation should be taken into account if the cited figures are used, for example, to locate beacons on the lunar surface to support navigation during the lunar approach phase.

Figure 14.
Descent Altitude/Range Profile

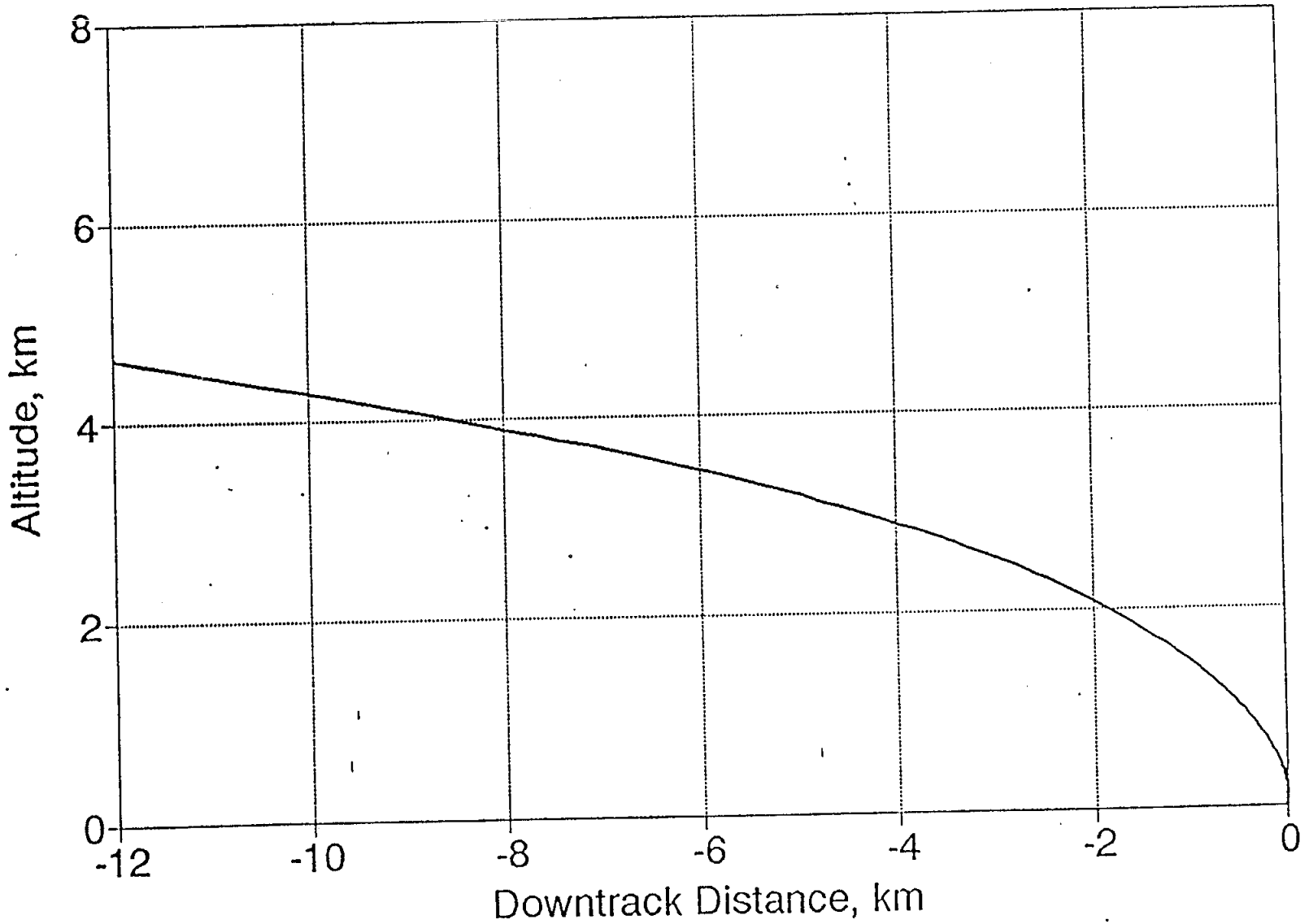


Figure 15.
Effect of Isp on Descent Delta-V
100x100 km Orbit

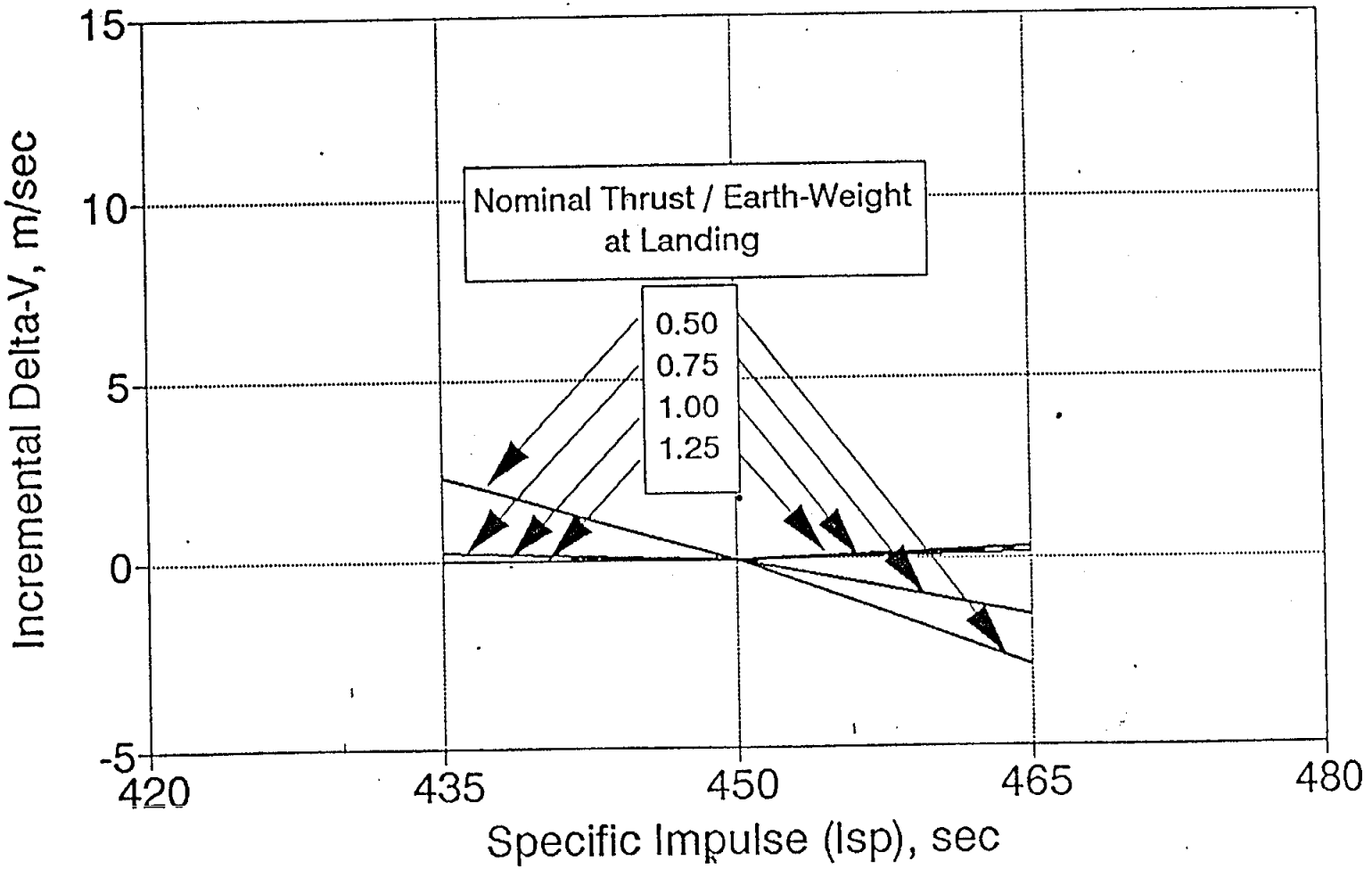


Figure 16.
Effect of Orbit Alt on Descent Delta-V
Isp = 450 sec

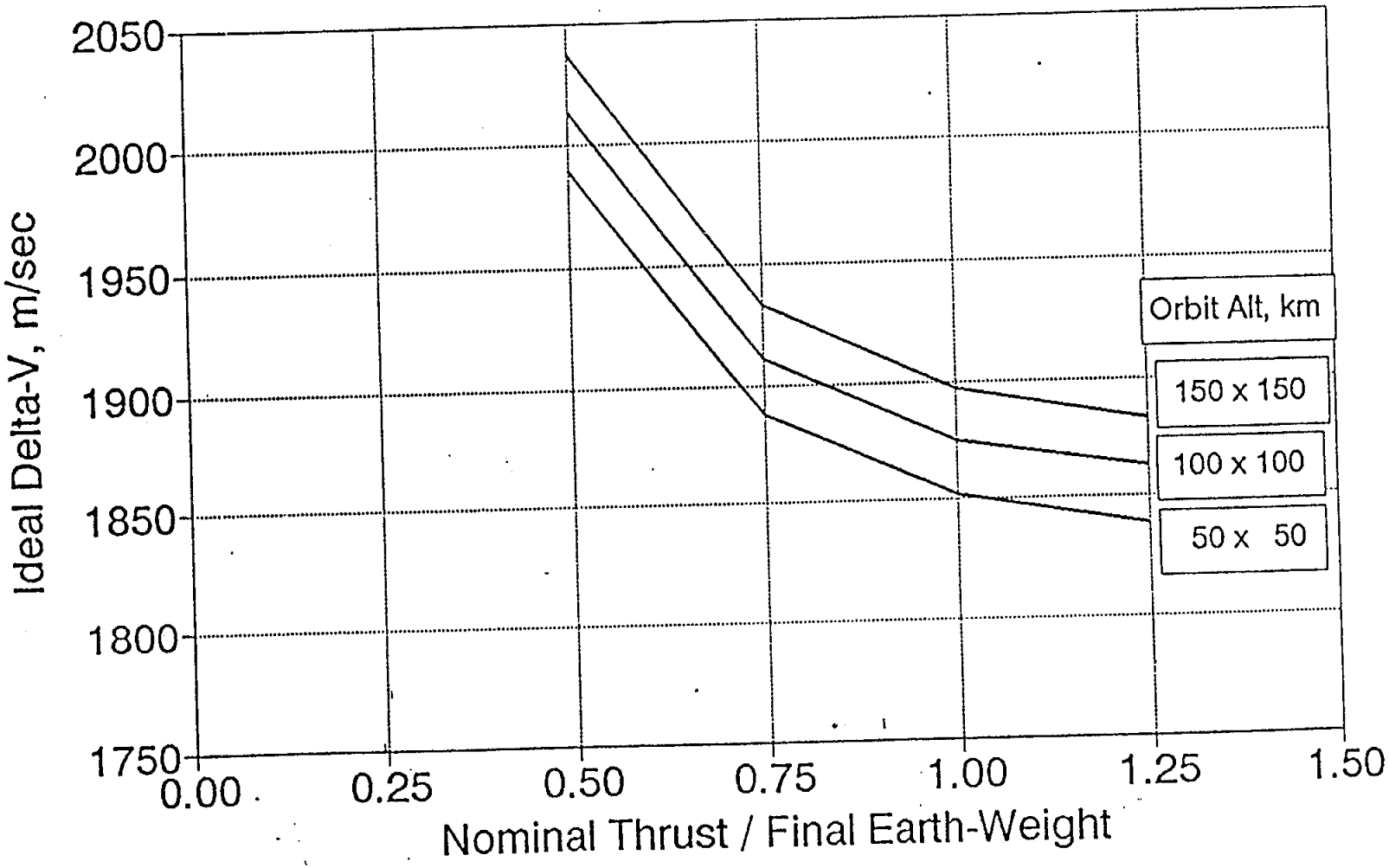
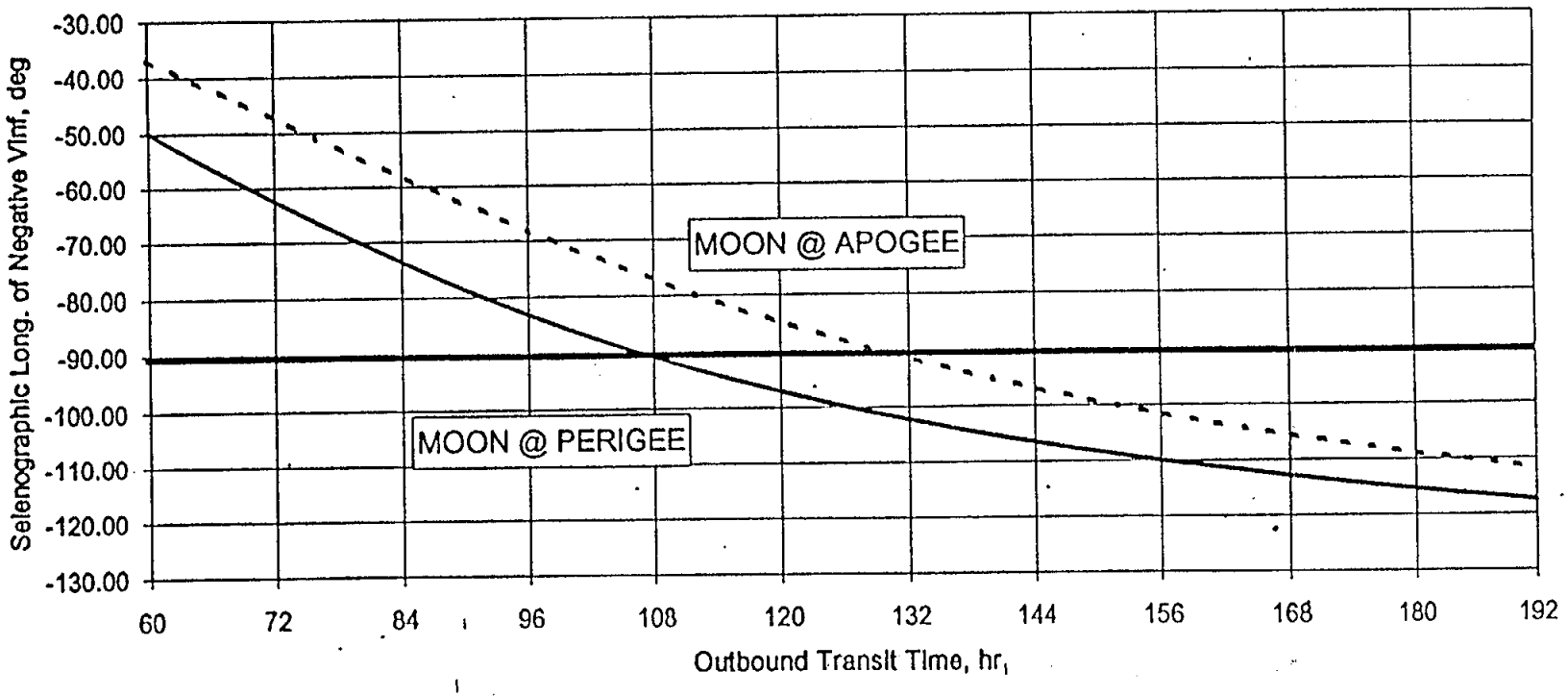


Figure 17.
Direction of Approach Asymptote
Transfer Orbit Incl. = 61.5° wrt Earth-Moon Plane



The effect of transit time on the selenocentric arrival energy parameter, C_3, is shown in Figure 18. The arrival energy depends not only upon the transit time and the geocentric distance of the moon, but also on the inclination of the geocentric transfer orbit relative to the earth-moon plane. The energy is minimized when this inclination is zero.

The data in Figure 18 are conservative in that they are based on a relative inclination of 61.5 degrees, which is the sum of two orbit plane inclinations measured with respect to the earth equator: 28.5 degrees for the maximum possible inclination of the moon's geocentric orbit, and 33 degrees for an assumed operational limit on the spacecraft's geocentric parking orbit inclination. To avoid excessive earth-departure plane change penalties, it is necessary for the plane of the transfer orbit to coincide very closely with that of the geocentric parking orbit. To attain a daily launch window width on the order of 4 hours from Cape Canaveral, the parking orbit inclination must be allowed to go as high as 33 degrees.

Figure 19 shows the ΔV_{IDEAL} requirement for lunar orbit insertion (LOI), assuming an impulsive coplanar maneuver. If LOI is performed by the lunar descent stage, gravity loss during the LOI burn should not exceed 2 m/sec. With proper planning and trajectory control, a coplanar insertion is always possible.

3.4. Departure from the Earth

The energy requirement for geocentric departure is shown in Figure 20. As in the case of the three preceding figures, these data were computed for a transfer orbit inclination of 61.5 degrees relative to the earth-moon plane. However, unlike the *selenocentric arrival* energy, the *geocentric departure* energy is quite insensitive to orientation of the transfer orbit plane, so the values would be very nearly the same for any inclination.

To minimize the overall propulsive velocity increment that is required to lift the lunar payload from the earth surface and inject it into the translunar trajectory, generally it is best to launch nearly due east, inserting the payload initially into a temporary circular parking orbit just above the atmosphere. Given a particular launch azimuth, the launch time must be selected so that the position of the moon, advanced to LOI time, will lie in the parking orbit plane. There will then be a particular point in the orbit where a horizontal, in-plane, translunar injection (TLI) velocity increment will give the lunar spacecraft the required geocentric energy and the necessary direction of motion for the transfer to the moon. The geocentric angle between the launch site and this optimum TLI impulse point varies with launch azimuth and with the declination of the moon. Therefore, a variable coast time in the parking orbit is an essential element for economical transfer of the spacecraft from the earth to the moon. The longest coast time that is needed to optimize the geometrical aspects of the transfer is equal to the period of a low-altitude geocentric orbit, which is about 90 minutes. Longer coast times may be desirable to accommodate navigation and systems checkout before TLI.

The propulsive velocity increment requirement for coplanar TLI (alone) depends on earth-moon transit time, the level of thrust acceleration, and the altitude of the predeparture orbit. Figure 21 shows the influence of the latter two parameters in the case of a 4-day transit. Down to the point

Figure 18.
Selenocentric Arrival Energy
Transfer Orbit Incl. = 61.5° wrt Earth-Moon Plane

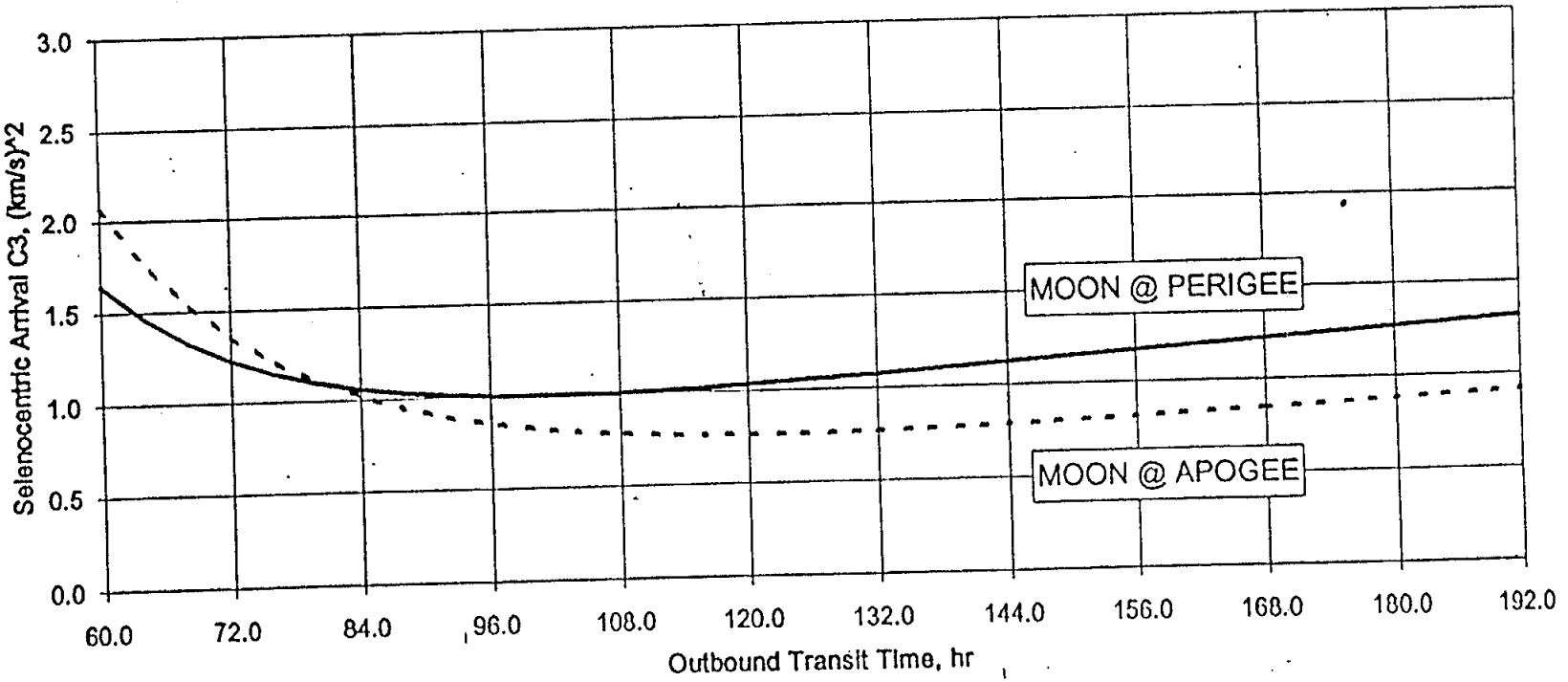


Figure 19.
LOI Delta-V into 100x100 km Lunar Orbit
Transfer Orbit Incl. = 61.5° wrt Earth-Moon Plane

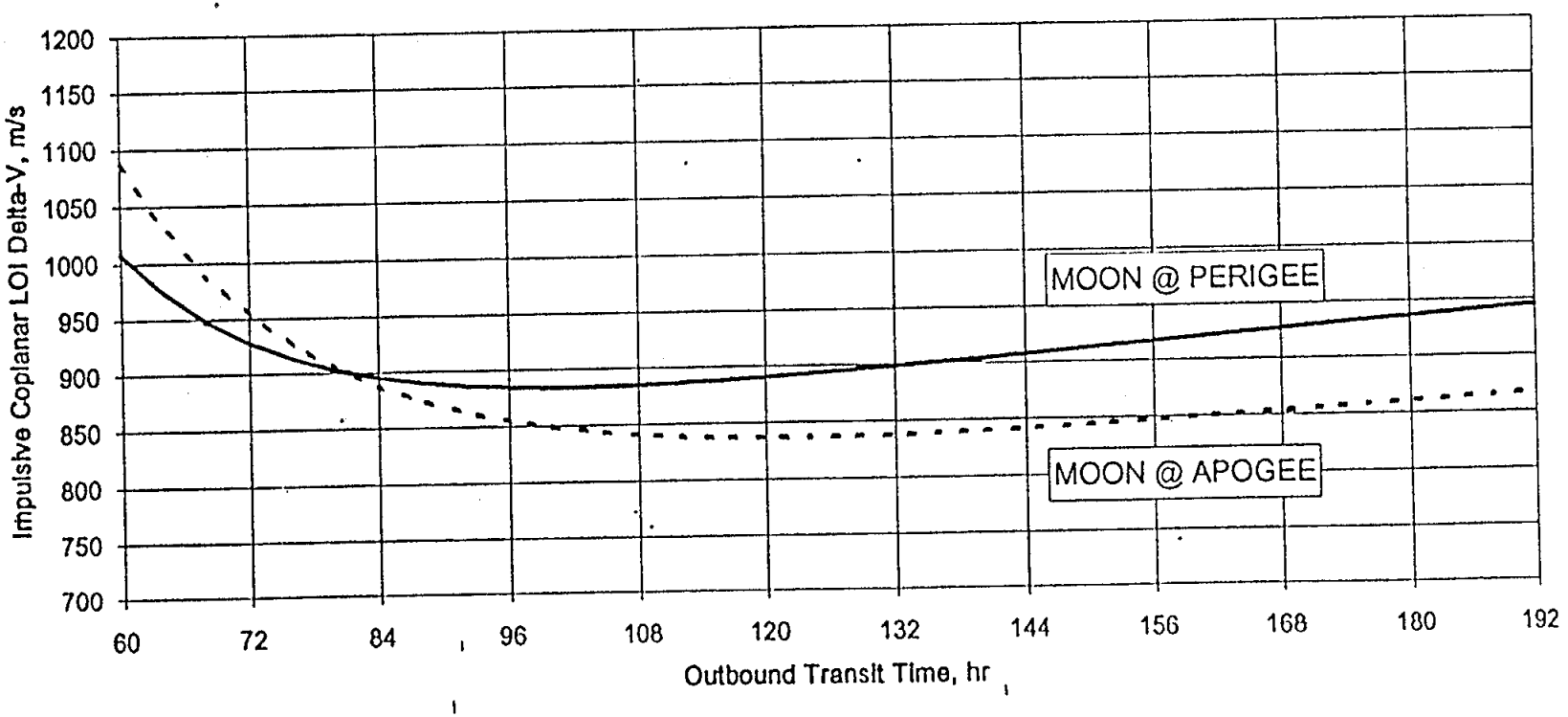


Figure 20.
Geocentric Departure Energy
Transfer Orbit Incl. = 61.5° wrt Earth-Moon Plane

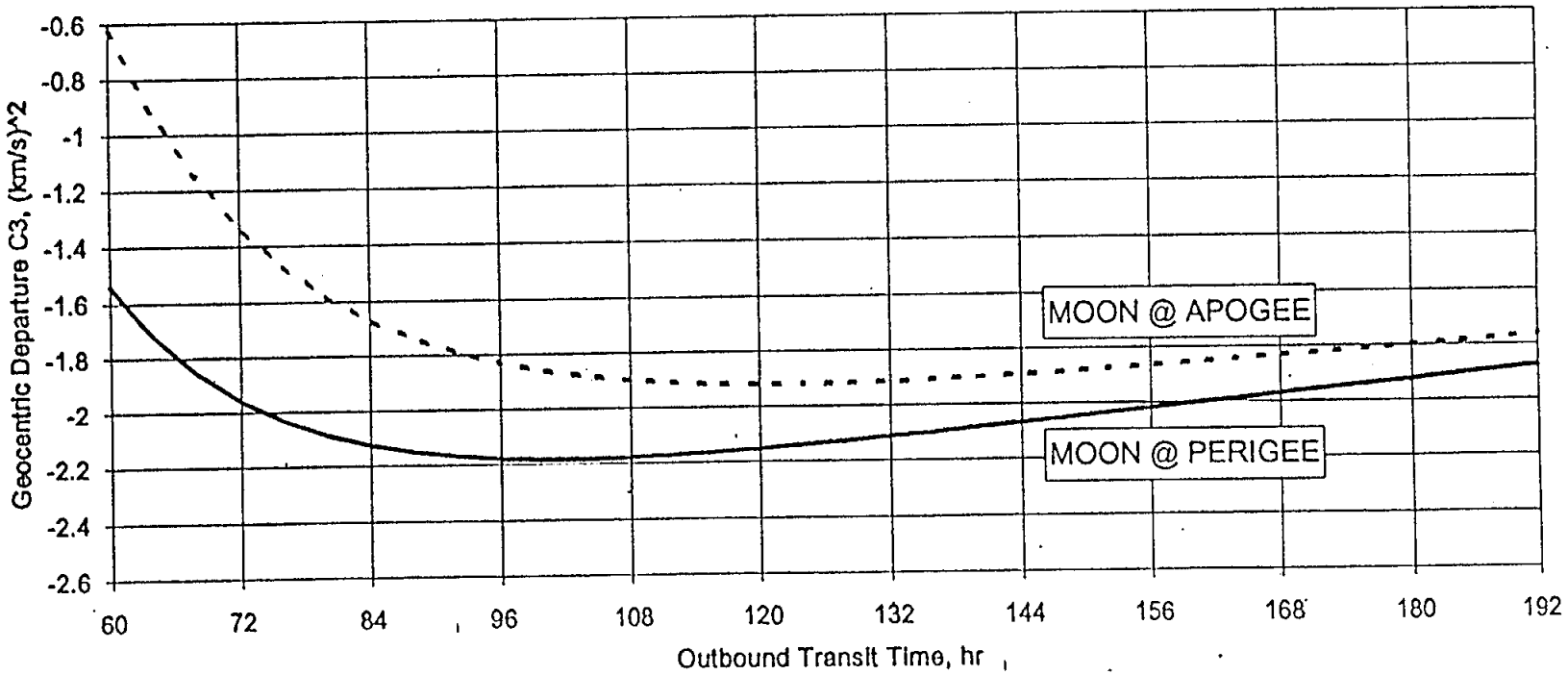
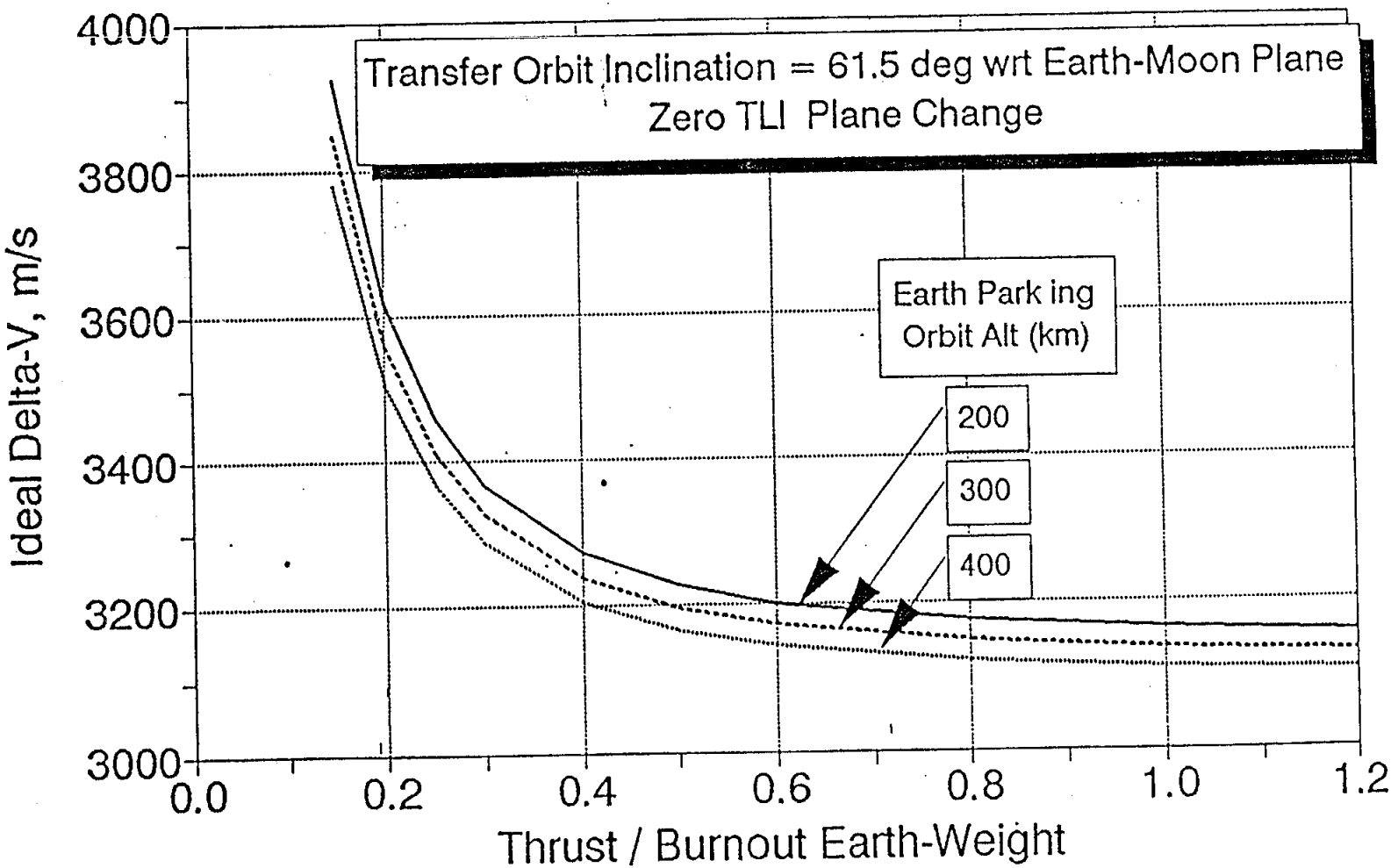


Figure 21.
Effect of TLI Thrust-to-Weight Ratio
96-Hr Flt to Moon @ Apogee, Isp = 450



where aerodynamic drag becomes dominant, the reduction of velocity losses during ascent to lower parking orbit altitudes is greater than the associated increase in TLI ΔV that is shown in Figure 21. The optimum altitude — where opposing effects on the lunar payload are exactly balanced — is a function of parking orbit stay time and aerodynamics and launch vehicle thrust magnitude, but usually it comes out near 200 km when the stay time is no more than a few hours. Figure 22 shows the variation of impulsive coplanar TLI ΔV magnitude with transit time, assuming a parking orbit altitude of 200 km.

3.4.1. Translunar Injection Window

After the parking orbit has been established, if TLI ignition is delayed more than a few seconds past the planned time, an expensive alteration of the injection flight path angle is required to rendezvous with the moon. To avoid an excessive performance penalty, it becomes necessary to coast through a full revolution in the parking orbit, waiting about 90 minutes before trying again to initiate TLI. However, during that interval, earth oblateness will make the parking orbit node regress through a significant fraction of a degree. This introduces a plane-change penalty that soon becomes prohibitive unless the transit time is shortened enough to keep the moon's position (advanced to arrival time) contained within the regressing plane of the parking orbit.

The longer TLI is delayed, the earlier the spacecraft needs to arrive at the moon. For transit times shorter than about 5 days, this means that the required C_3 increases with TLI delay. However, the penalty resulting from the higher energy requirement is far smaller than the plane-change penalty that would otherwise be incurred. This fact is illustrated in Figure 23, which shows how other strategies would affect the variation of impulsive ΔV with TLI time. In this figure, and in others to follow, the time of TLI is treated as if it were a continuous variable even though real injection opportunities are essentially discrete events separated by intervals of about 90 minutes.

Figure 24 shows how LOI ΔV varies with TLI time for the different strategies. Here there is no clear-cut advantage for the coplanar TLI scheme, but neither is there a disadvantage. (When comparing this figure with the preceding one, the great difference in the scale of ΔV should be noted.) Figures 25 and 26 show the variations of transit time and the time of lunar arrival with TLI time.

Table I was constructed to evaluate the effect of TLI window width on basic design requirements for the manned lunar spacecraft. Minimum transit time (column A) was taken as a fundamental variable, for which an optimum value ultimately should be selected on the basis of minimizing the spacecraft mass. Both the outbound and the return leg of the round-trip mission were subjected to the same numerical value for this trajectory constraint. The values in column B were obtained by adding 24 hours to those in column A, and represent the upper bounds on return transit time needed to obtain a 360-degree variation in the longitude of the atmospheric entry point. The values in column C were taken from Figure 2, and represent the maximum inertial entry speed for the transit time limits defined in the two preceding columns. Column D contains the associated maximum values of impulsive coplanar TEI ΔV, which came from Figure 10.

Figure 22.
TLI Delta-V from 200x200 km Earth Orbit
Transfer Orbit Incl. = 61.5° wrt Earth-Moon Plane

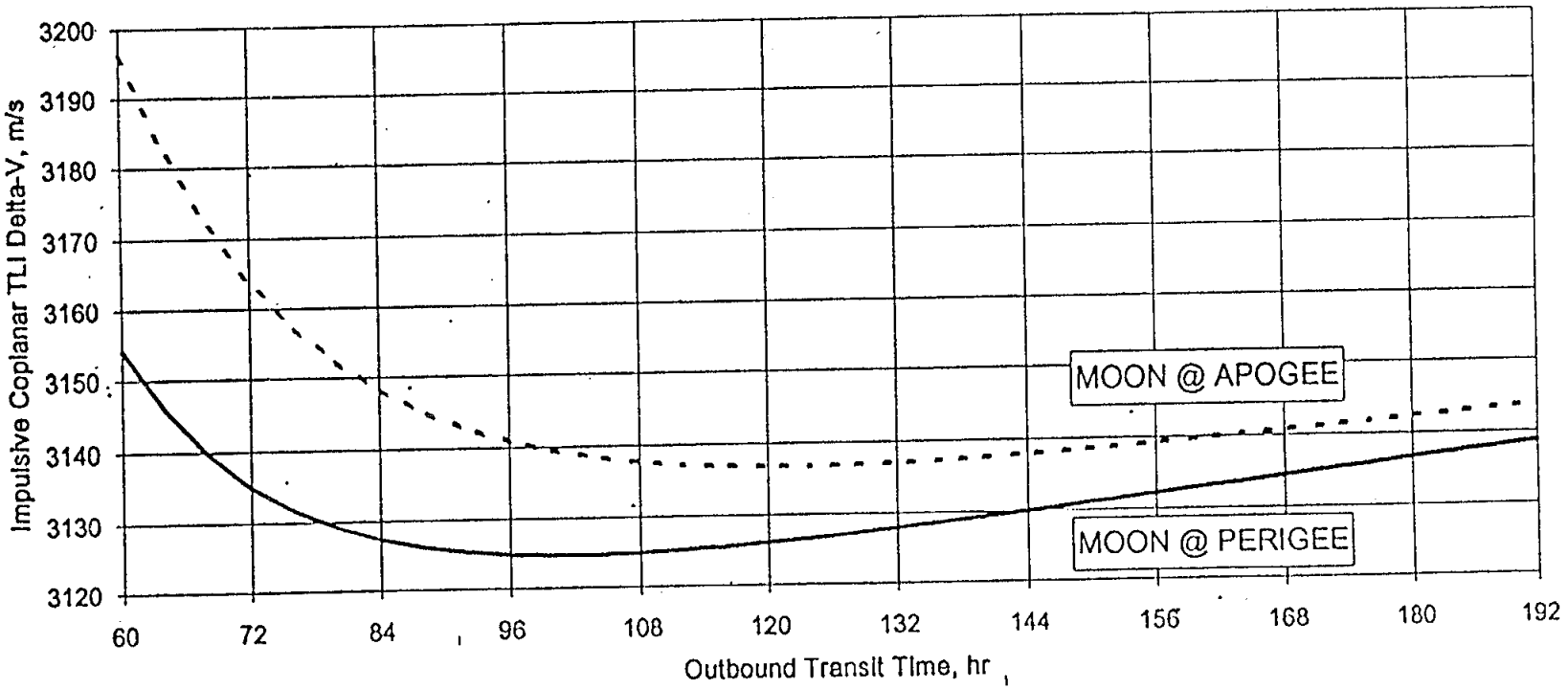


Figure 23.
Translunar Injection Window
TLI Delta-V

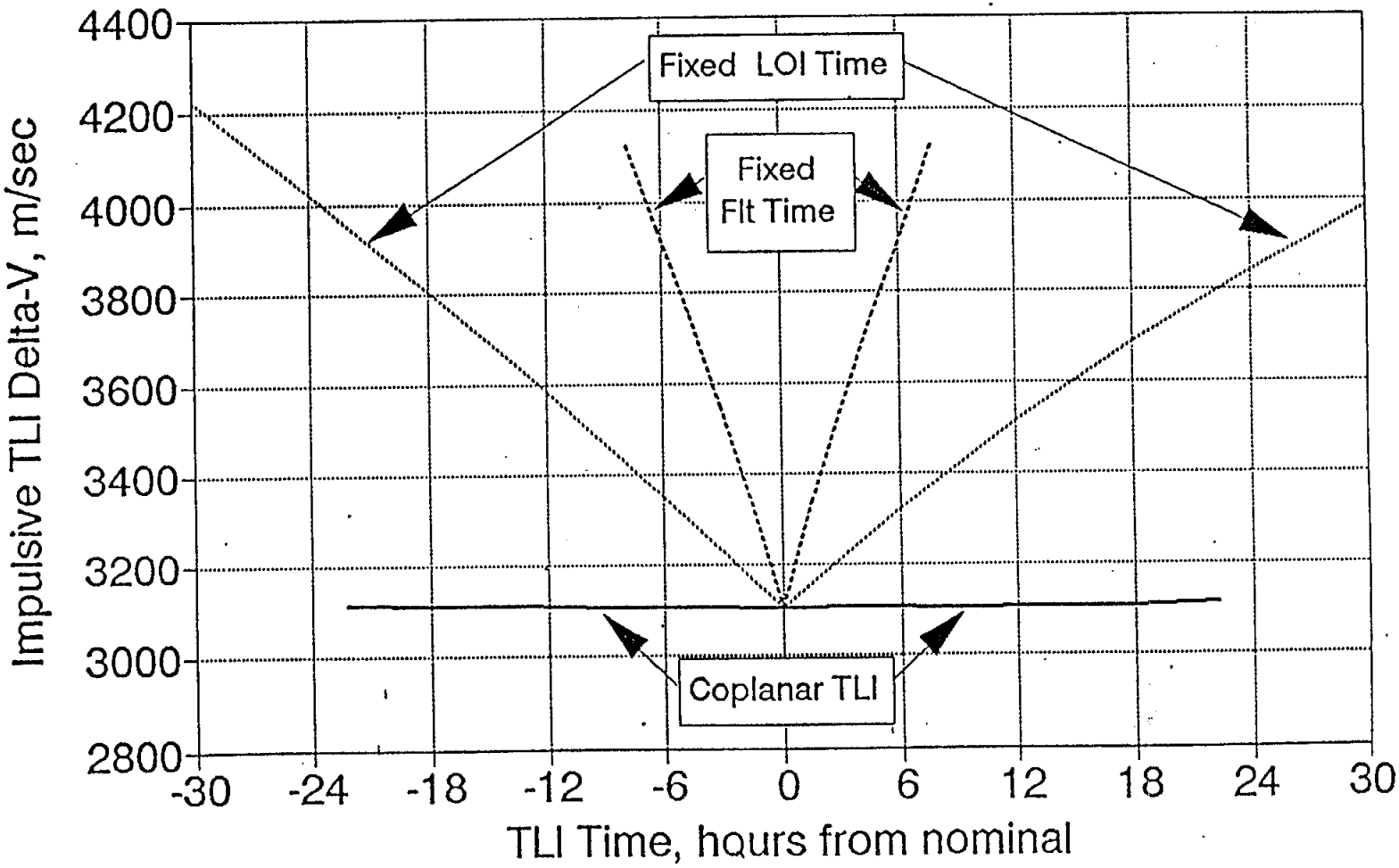


Figure 24.
Translunar Injection Window
LOI Delta-V

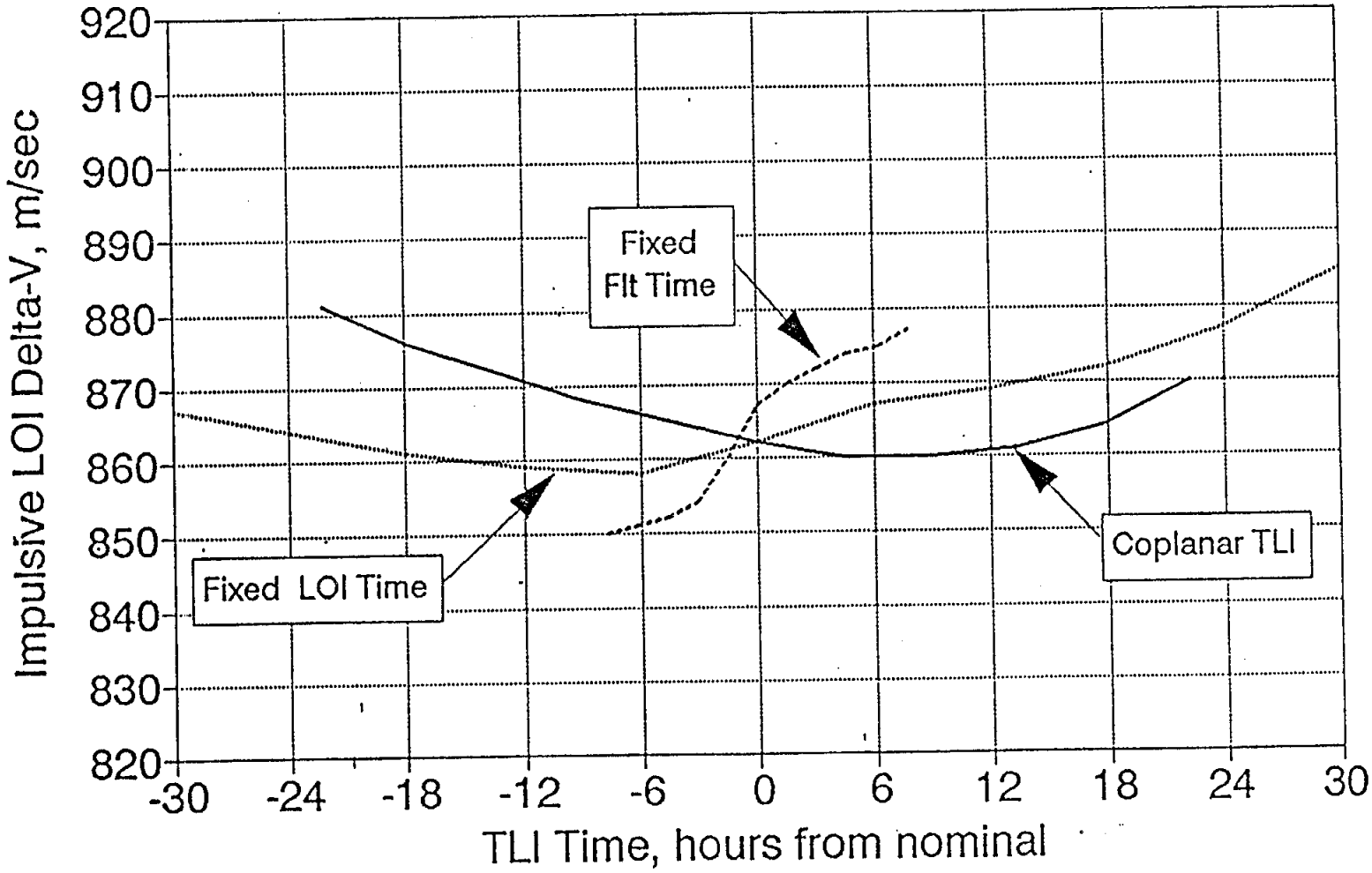


Figure 25.
Translunar Injection Window
Earth-to-Moon Flight Time

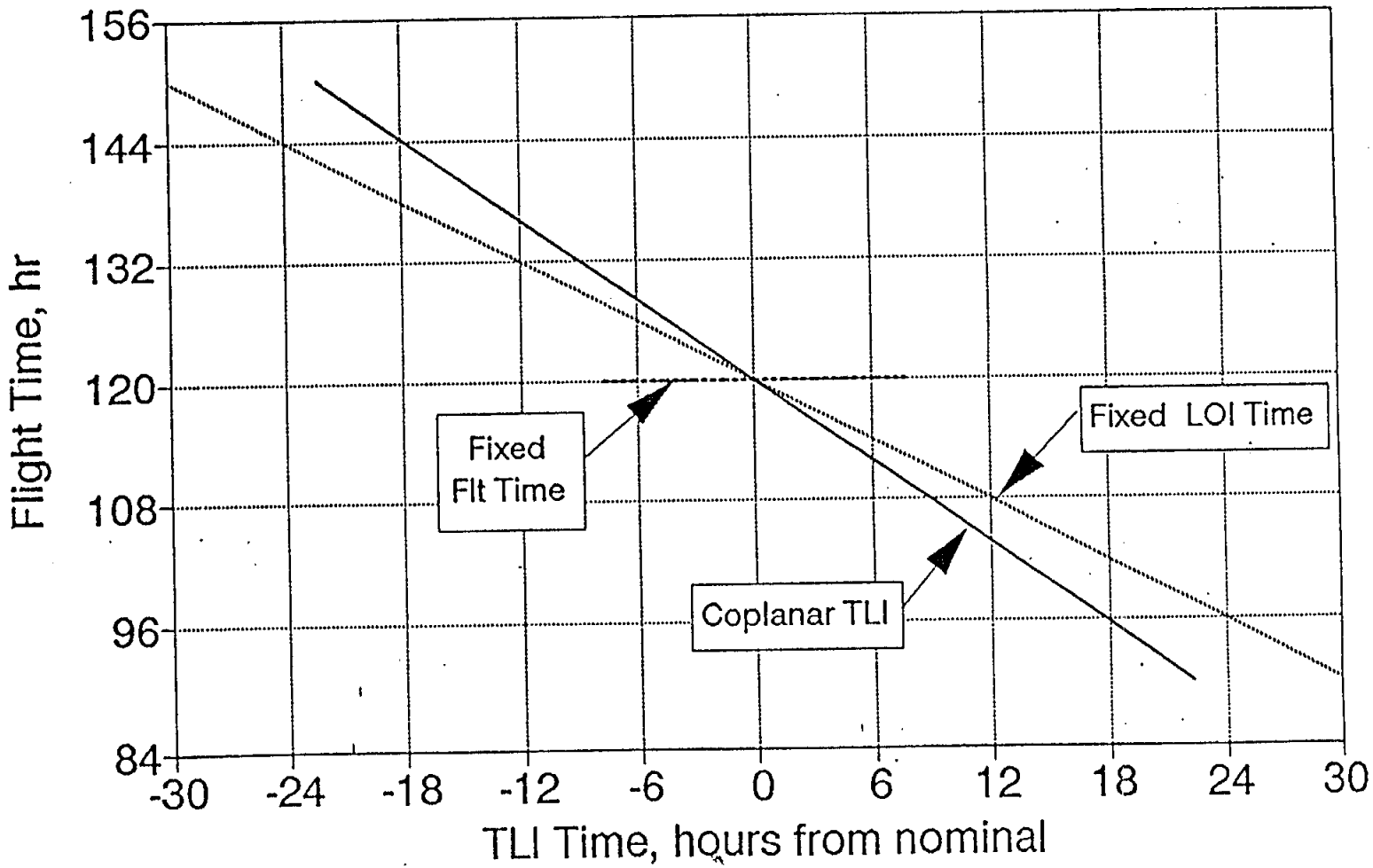


Figure 26.
Translunar Injection Window
LOI Time

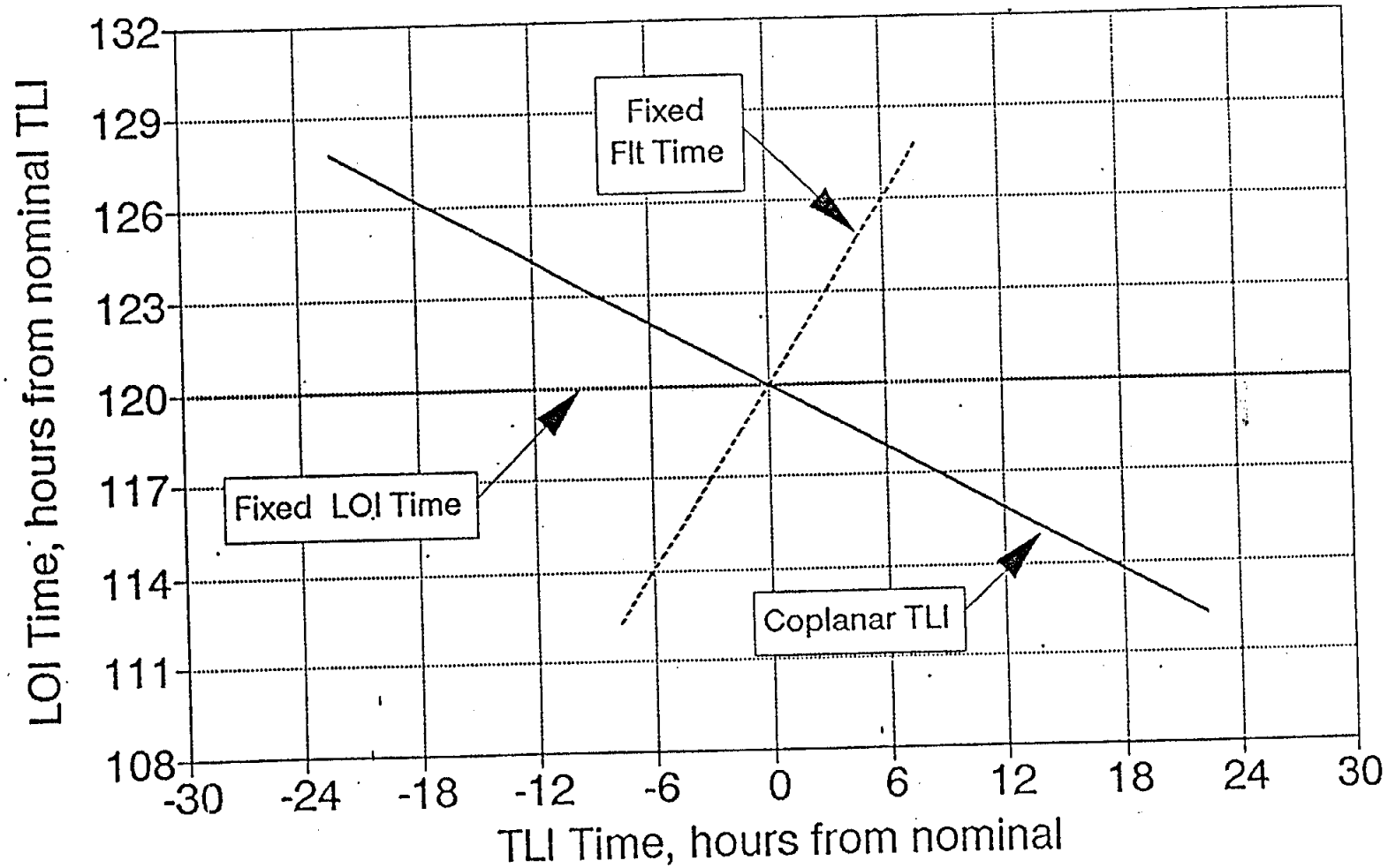


Table 1.
Time and Velocity Data for Earth-Moon Round Trip

A	B	C	D	E	F	G	H	I
Min Transit Time (hr)	Max Inbound Transit (hr)	Max Entry Speed (m/sec)	Max TEI Delta-V (m/sec)	TLI Window Width (hr)	Max Outbound Transit (hr)	Max Round Trip (hr)	Max TLI Delta-V (m/sec)	Max LOI Delta-V (m/sec)
60	84	11,046	1,149	7.5	72	162	3,196	1,086
72	96	11,014	1,020	7.5	84	186	3,164	954
84	108	10,998	979	7.5	96	210	3,148	895
96	120	10,991	971	7.5	108	234	3,141	884
108	132	10,988	979	7.5	120	258	3,138	890
60	84	11,046	1,149	32	111.2	201.2	3,196	1,086
72	96	11,014	1,020	32	123.2	225.2	3,164	954
84	108	10,998	979	32	135.2	249.2	3,148	900
96	120	10,991	971	32	147.2	273.2	3,141	910
108	132	10,988	979	32	159.2	297.2	3,140	919
	(A+24)	(Fig. 2)	(Fig. 10)		(A+1.6E)	(B+F+6)	(Fig. 22)	(Fig. 19)

Column E in Table 1 contains the TLI window width, which was treated as a parameter. Multiplying this value by 1.6 and adding it to the value in column A yields the approximate maximum outbound transit time (column F) required to avoid a TLI plane change. The proper multiplier of window width varies with time of month and the inclination of the moon's orbit around the earth. However, it usually does not exceed the value used in Table I, and never has to do so unless (1) solar illumination constraints are imposed on the ascent trajectory, or (2) constraints are imposed on the geographic position of the TLI maneuver.

Column G contains the sum of the values in columns B and F, with 6 hours added to account for time spent in orbit around the moon after arrival and before departure. It represents the maximum duration of in-transit occupancy of the crew module, from the opening of the TLI window to the approximate time of landing on the earth. It does not account for occupancy during ascent and nominal coast in the geocentric parking orbit, nor on the lunar surface, nor additional revs in lunar parking orbit such as might be required for navigation or other purposes. The values in columns H and I are taken from Figures 22 and 19, respectively.

The values in column G are represented on the horizontal axes of Figures 27-30, which show how velocity increment magnitude and entry speed requirements (columns H, I, D, C) vary with occupancy time in the crew cab. These are the basic quantities that determine trajectory-dependent variations in the lunar spacecraft mass.

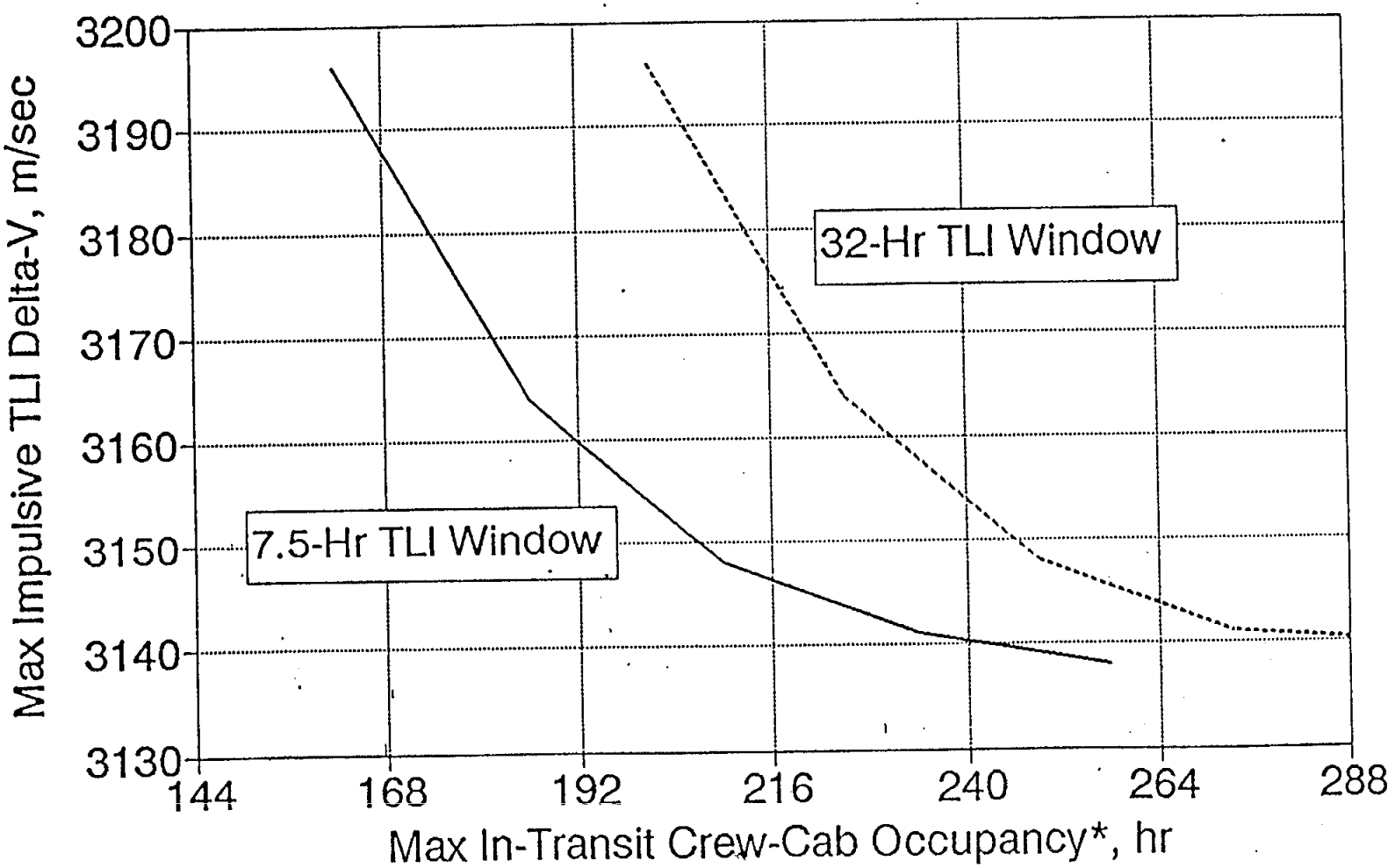
3.4.2. Launch Window

Specifying a time of arrival (LOI) at the moon, along with a time of launch from Cape Canaveral, makes it possible to determine an optimum launch azimuth that will nullify all plane change penalties during ascent, geocentric parking orbit insertion, and translunar injection. Leaving out less important subtleties, the basic problem is solved by finding a plane in inertial space that contains the position of the launch site at launch time as well as the position of the moon at LOI time, both positions being measured from the center of the earth.

There are two distinct possibilities for the direction of motion in any such plane: posigrade and retrograde. These two choices involve a smaller and a greater subtended angle between the launch site and the moon. Suppose the launch azimuth is always chosen to yield, say, the smaller subtended angle. Then the azimuth will vary monotonically through a range of 0 to 360 degrees in a time interval that varies slightly with the time of month, but which averages just under 25 hours. However, if the direction of motion is chosen more realistically to be always posigrade, then the azimuth will traverse the range of 0 to 180 degrees twice during the same period of time. Therefore, on the average, during any 25-hour period there will be two distinct opportunities for coplanar departure to the moon within a given range of launch azimuths from Cape Canaveral.

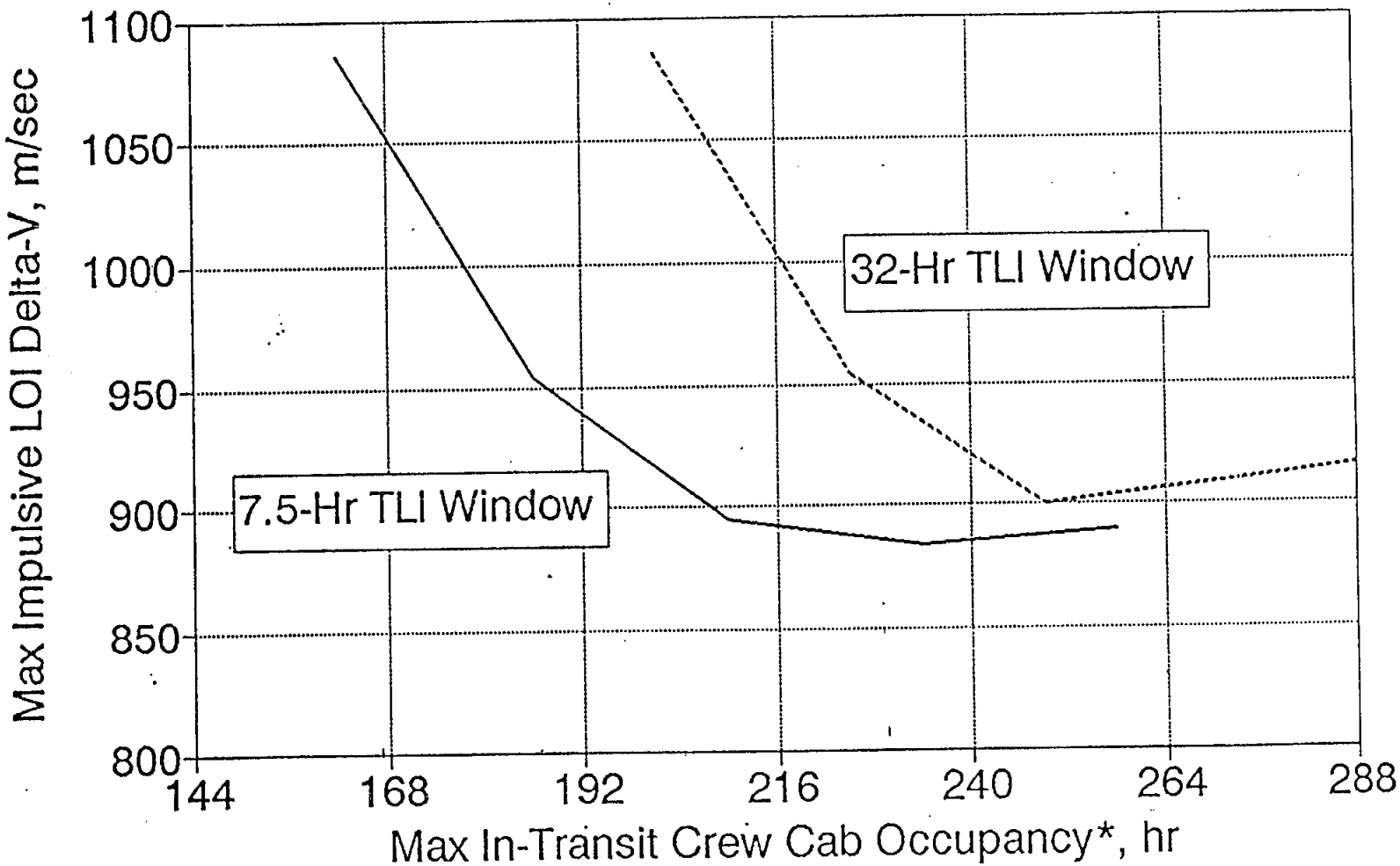
During the Apollo program, the two opportunities were characterized as "Atlantic injection" and "Pacific injection" opportunities. As can be seen in Figure 31, these designations suggest themselves by the general location of the TLI maneuver. Although there were variations, 72 to 108 degrees was the usual range of launch azimuths for Apollo missions. This yields a

Figure 27.
Impulsive TLI Delta-V
200x200 km Earth Assembly Orbit



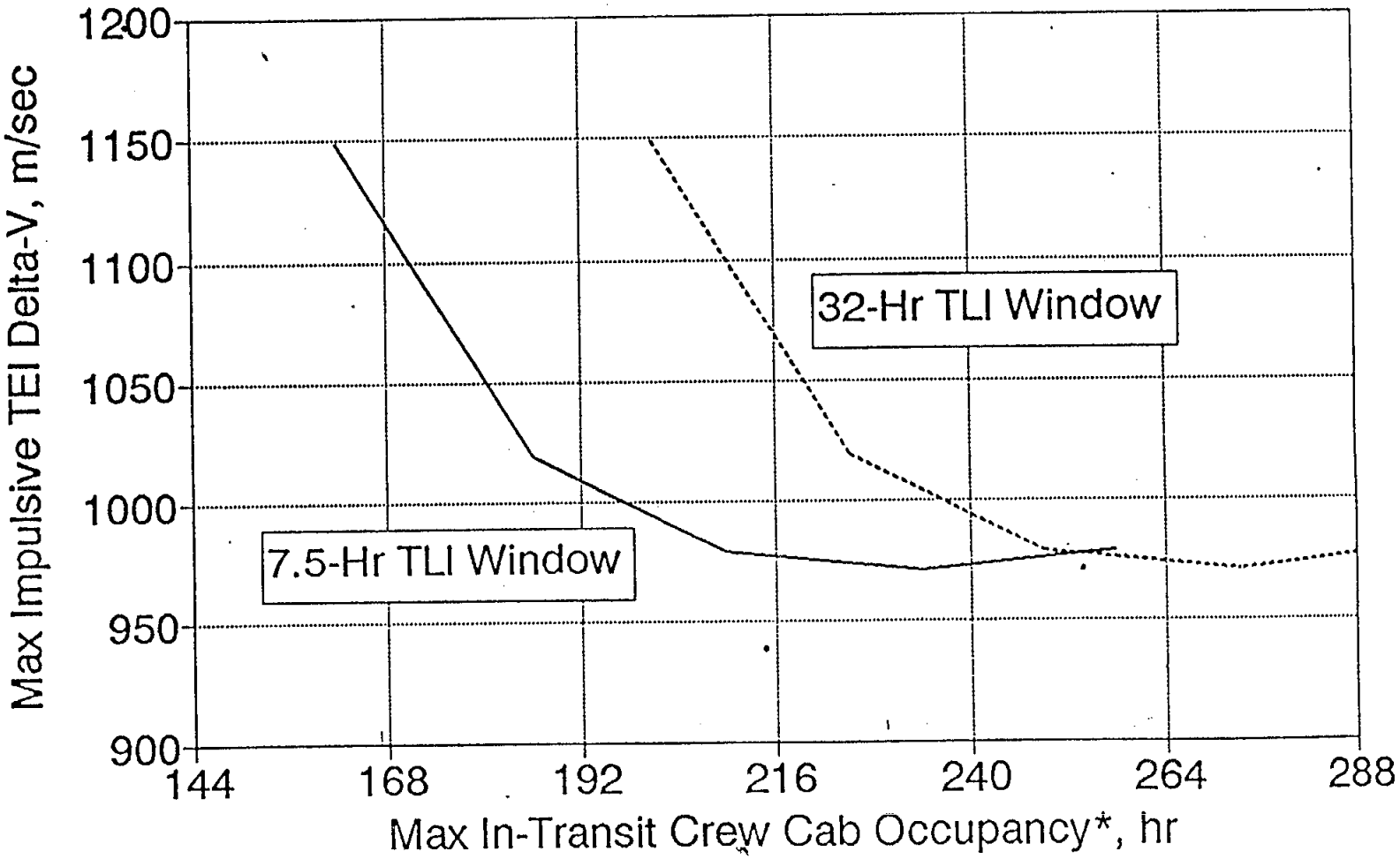
*(Earliest TLI -> Lu Landing) + (Liftoff -> Reentry)

Figure 28.
Impulsive LOI Delta-V
100 x 100 km Lunar Capture Orbit



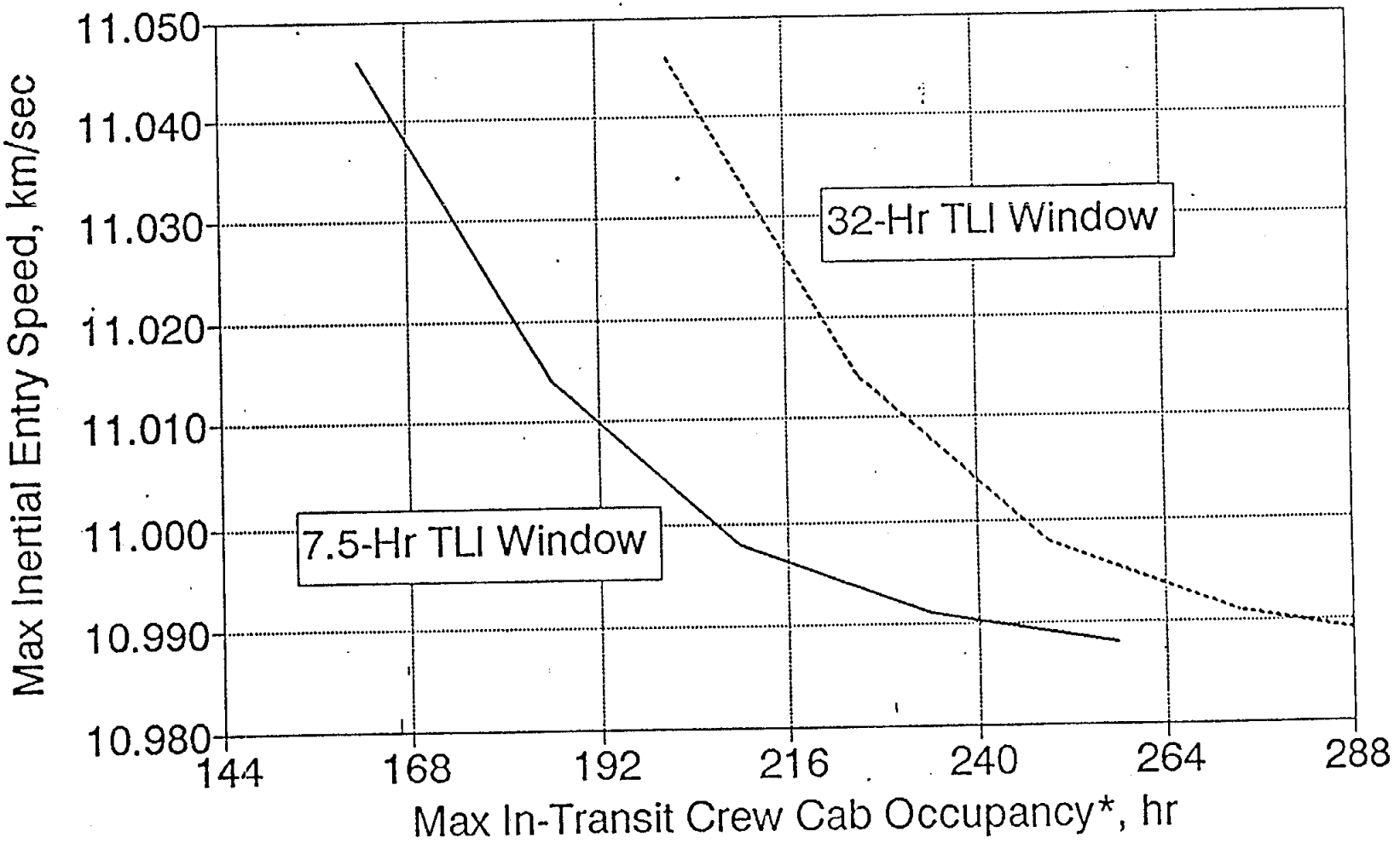
*(Earliest TLI -> Lunar Landing) + (Liftoff -> Reentry)

Figure 29.
Impulsive TEI Delta-V
100 x 100 km Lunar Departure Orbit



*(Earliest TLI -> Lur Landing) + (Liftoff -> Reentry)

Figure 30.
Inertial Entry Speed
121.92-km Atmospheric Entry Altitude



*(Earliest TLI -> Lunar Landing) + (Liftoff -> Reentry)

Figure 31.

NASA

Lunar Launch Window
(Single Launch)

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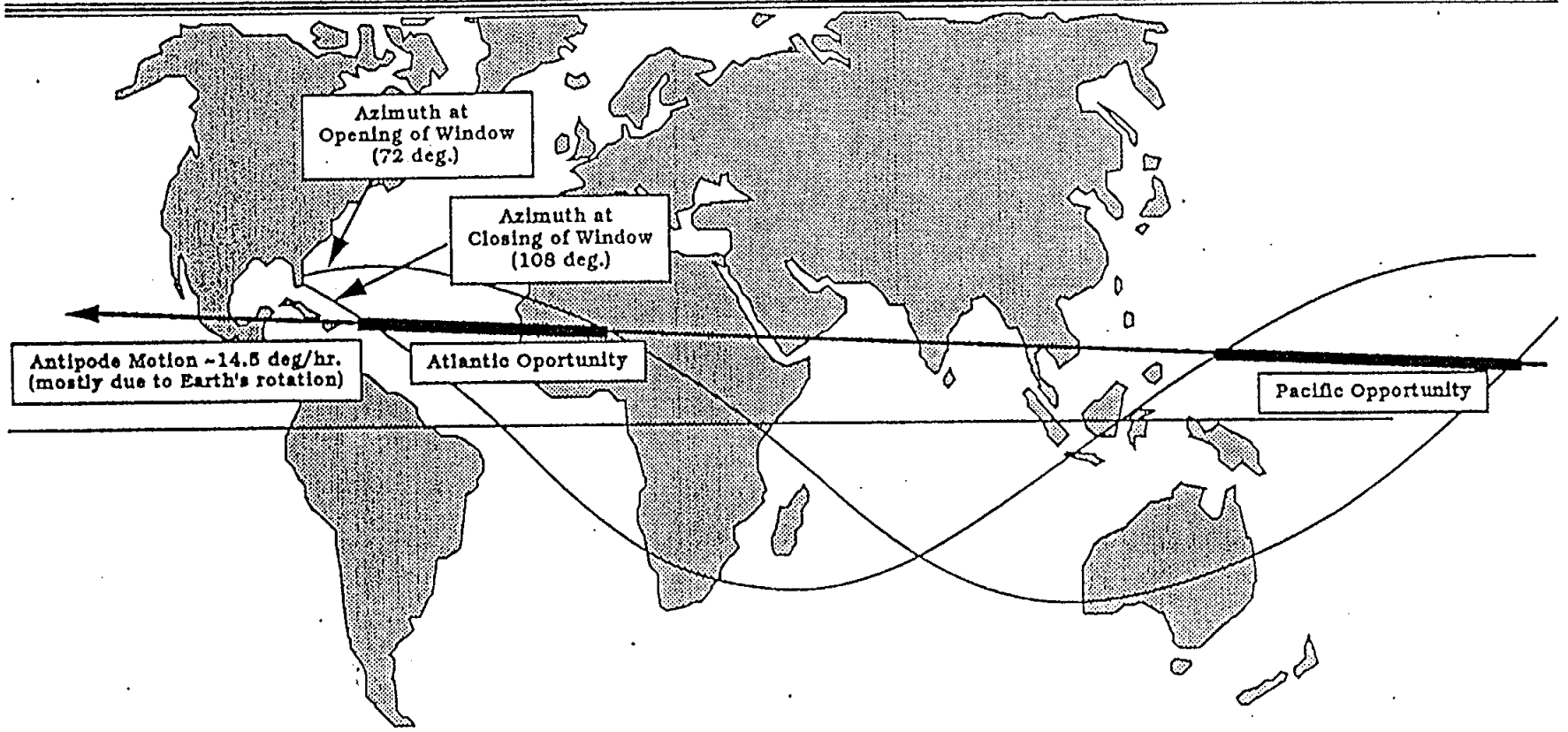

Two daily launch windows occur

- Relatively long windows are achievable by varying launch azimuth.
- Antipode moves westward at ~14.5°/hr; window opens with northerly azimuth, closes with southerly
- Apollo achieved 4.5 hr. windows with 72 - 108° deg. azimuth (33° maximum inclination)

naximum parking orbit inclination of about 33 degrees with respect to the equator, and a contiguous launch window width of approximately 4.5 hours for each opportunity. Wide windows are desirable for coping with last-minute glitches in the spacecraft and launch systems, but even more so for accommodating local weather conditions. Aside from range safety constraints, the main problem is that ascent vehicle performance is degraded as the launch azimuth deviates more and more from due east, and anything very far outside the Apollo range is likely to be unacceptable.

Abstract

First Lunar Outpost (FLO) Navigation Data Book

This section of the report is intended to document the navigation efforts as of April 2, 1992. Included in this section are the Design Guidelines and Assumptions as of 3/20/92, a menu of navigation options completed 3/31/92, and the results of ongoing navigation analyses supporting the navigation architecture concept definition.

The navigation Design Guidelines and Assumptions (DG&A) document identifies the design drivers for conceptual development of a navigation architecture for the FLO. It uses a top-down approach for defining guidelines for the FLO architecture concepts, precursor missions, and the flight phases (including aborts).

The menu of navigation options has been tailored for the FLO to serve as a mission planning tool for identifying navigation options for each flight phase.

The navigation analyses provide system performance accuracy and requirements for the navigation concept options. The studies address such issues as element measurement accuracy, nav system accuracy, system complexity, and infrastructure emplacement and build-up.

Questions concerning this section should be directed to XE (CSDL)/Gregg Barton (713/283-5470).

The Charles Stark Draper Laboratory, Inc.

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CSDL/ExPO/XE

Telephone (713) 283-5470
Fax (713) 283-5818

ESB-92-305
LMI-92-039

To: ExPO
From: Gregg Barton / Navigation
Date: 6/4/92
Subj: First Lunar Outpost Navigation Data Book - April 2, 1992

Please find enclosed the First Lunar Outpost (FLO) navigation data book. This book is intended to document the navigation efforts as of April 2, 1992. When completed, the navigation efforts will have defined the requirements for an integrated end-to-end navigation architecture for the FLO. A top-down systems process which will achieve this end product is outlined below. The first two activities have been completed and are included in the data book. The analyses supporting the navigation architecture concept definition (including a one page summary for select analyses) complete the remainder of the data to date. The rest of the activities / products (4-7) are currently being worked.

1) Design Guidelines and Assumptions - Completed 3/20/92

The navigation DG&A identifies the design drivers for conceptual development of a navigation architecture for the FLO. The DG&A uses a top-down approach for defining guidelines for the FLO architecture concepts, precursor missions, and the flight phases (including aborts).

2) Menu of Navigation Options - Completed 3/31/92

A menu of navigation options has been tailored for the FLO to serve as a mission planning tool for identifying navigation options for each flight phase. The menu enables the mission planner the ability to select navigation techniques from a comprehensive list of viable options for different methodologies of navigation.

3) Results of the navigation analyses - Ongoing activity

The navigation analyses provide system performance accuracy and requirements for the nav concept options. The studies address such issues as element measurement accuracy, nav system accuracy, system complexity and infrastructure emplacement and buildup.

4) Navigation Methodology Options

The navigation methodology provides a high level system description of each of the navigation element options. This product uses the navigation menu as the element list and the results of the navigation analyses to develop the system requirements and conceptual description. The Methodology options serve as the building blocks for the navigation strawman.

5) Navigation Strawman Concepts

The navigation strawman defines the integrated end-to-end navigation architecture utilizing elements from different methodologies of navigation. The strawman will address such topics as precursor navigation knowledge and emplacement requirements, nav concepts and systems during the initial FLO missions, and evolution of the nav architecture.

6) Navigation Requirements

The navigation requirements for the FLO will flow-down from the navigation strawman. Navigation requirements will be written for the precursor missions (knowledge and infrastructure buildup), transportation and surface systems, support elements, and infrastructure evolution.

7) Navigation Operational Description

The navigation operational description identifies the top-level navigation operational support to the FLO reference mission. The detailed reference mission and operational concepts will include an operational description of the primary and backup navigation support systems and elements from the strawman for each mission phase.

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ESB-92-192
LMI-92-021

To: Navigation Team
From: Gregg Barton
Date: 4/9/92
Subj: First Lunar Outpost Navigation Architecture Concepts

The enclosed presentation entitled "First Lunar Outpost -- Abstract of the Navigation Architecture Concepts" was presented to Mike Griffin and Jay Greene (Associate and Deputy Administrator for Exploration) on 2 April, 1992. The objective of the presentation was to provide an overview of the navigation architecture concepts and to status the work to date. Due to the time allocation, the details of the navigation guidelines and assumptions, menu of navigation options, navigation strawmen and summary of analyses will be reviewed at a later date.

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First Lunar Outpost

Abstract of the

Navigation Architecture Concepts

presented to

Mike Griffin / Jay Greene

Gregg Barton / CSDL

April 2, 1992

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FLO Navigation Architecture Concepts
FLO Navigation Team

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Tim Brand	CSDL	Nav Strawman Concepts
Judy Miller	"	Menu of Nav Options
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FLO Navigation Architecture Concepts
"The Big Picture"

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- Design Guidelines and Assumptions (Complete)
- Menu of navigation options (Complete)
- Navigation Strawman Concepts (Documentation)
- Selection of methodology options from strawmen (Documentation)
- Results of navigation analyses (In progress)
- Navigation architecture concepts and operational description (To be worked)
- Issues / Future Work (In progress)

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FLO Navigation Architecture Concepts
Design Guidelines & Assumptions

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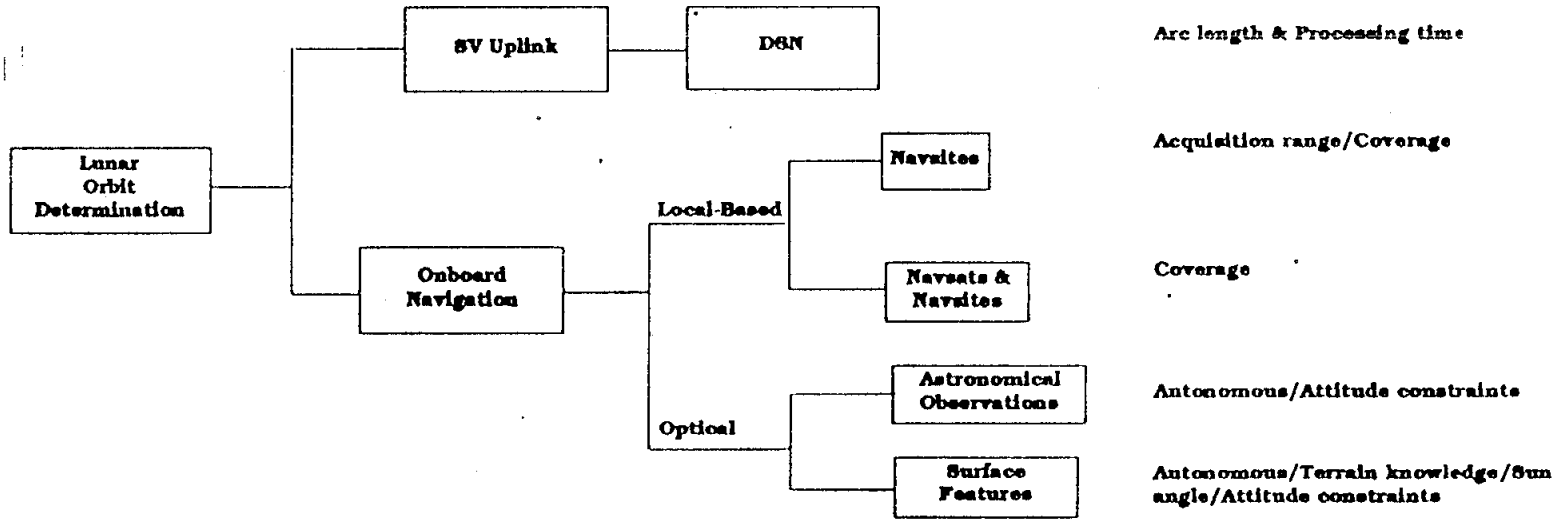
- Navigation design guidelines and assumptions are documented in FLO Detailed Guidelines & Assumptions Document*
- Main objective of Nav DG&A is to identify the design drivers for defining the navigation architecture concepts
  - Specific implementation will be defined by navigation strawman concepts
- Nav DG&A is organized in a top-down approach defining guidelines for architecture concepts, precursor missions, and the flight phases (including aborts)
  - Identifies the critical areas for navigation
  - Identifies key interfaces required with communication, transportation, surface, and science

* "First Lunar Outpost Detailed Asumptions", Working Drafts, 20 March 1992

FLO Navigation Architecture Concepts
Menu of Navigation Options

- Menu of navigation options has been tailored for FLO as outlined in the Navigation Design Guidelines & Assumptions
  - Only options that are viable and advocated by the navigation team for FLO are included
- Menu objective is to enable mission planner to select for each flight phase different methodologies of navigation
  - Earth-based infrastructure (DSN, TDRSS radiometrics)
  - Onboard autonomy (GPS radiometrics, astronomical observations, optics, radar, etc)
  - Lunar-based infrastructure (navsats, navsites radiometrics)
- Menu of options consists of 2 distinct levels for each phase
  - 1st level: Top level option of methodologies
  - 2nd level: Infrastructure element options and details

FLO Navigation Architecture Concepts
Menu options / Lunar Orbit (Cargo*/Piloted**)



* Cargo from Lunar Approach
** Piloted from Lunar Approach and Earth Return

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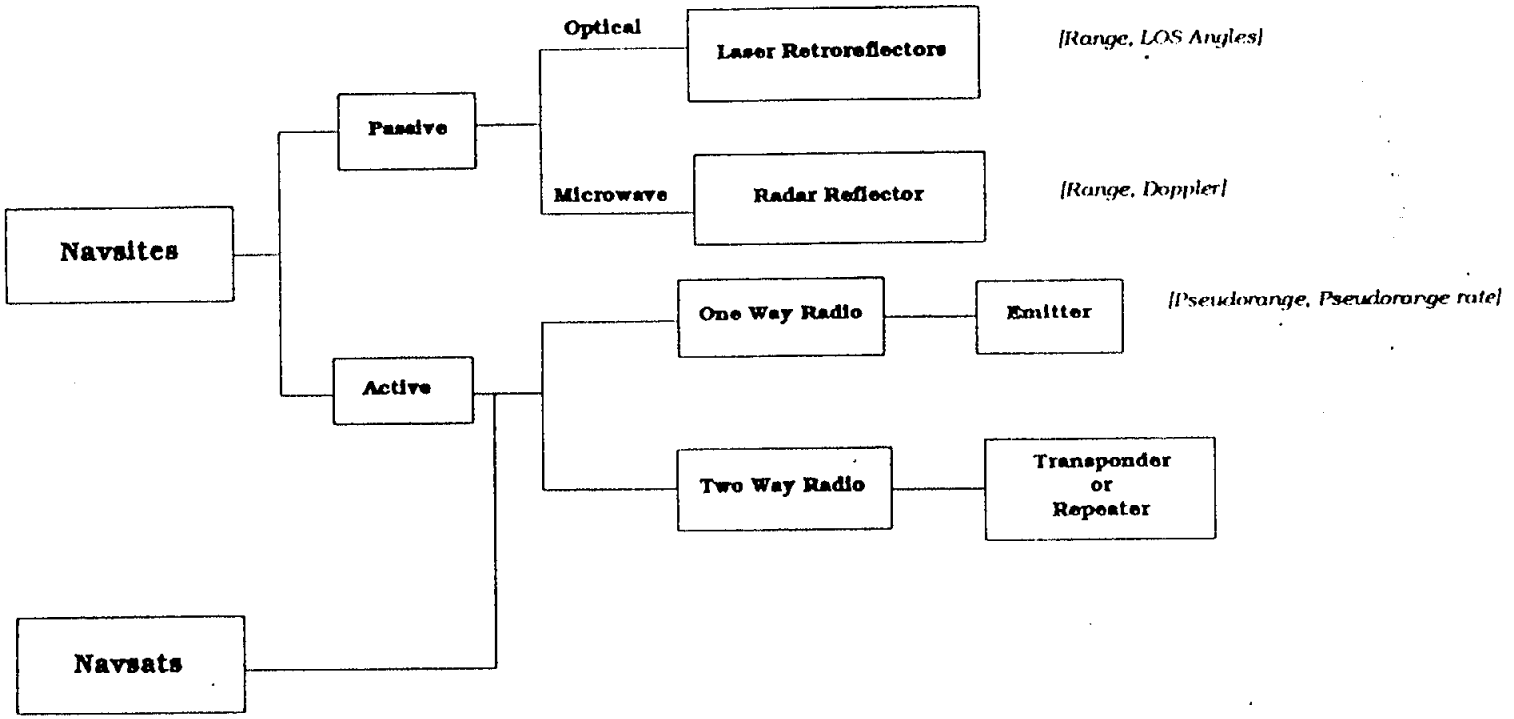
FLO Navigation Architecture Concepts
Menu Options / Local-based Navaids



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FLO Navigation Architecture Concepts
Navsite Options

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TYPE	SITE HARDWARE	USER HARDWARE	COVERAGE	EMPLACEMENT	SURVEY	ADVANTAGES	DISADVANTAGES
Laser Retro reflectors	Size Number of surfaces	Laser Transmitter & Receiver	Number Location	Hard landers Artemis Lander Rovers	Earth-Based (front side only) Survey Satellite LTS	- Passive devices - Low cost - May be able to survey from Earth - Fixed sites	- User laser pointing & tracking/stability/power - Incompatible with navsats - Emplacement/survey
Radar Reflector	Size Number of surfaces	Antenna Radar	Number Location	Hard landers Artemis Lander Rovers	Survey Satellite LTS	- Passive devices - Low cost - Fixed sites	- User radar power - Incompatible with navsats - Emplacement/survey
Emitter	Power Frequency Ref Clock Antenna size Antenna type Omni Directional	Antenna Receiver Frequency Ref Clock	Number Location	Hard landers Artemis Lander Cargo/Piloted Landers Rovers	DSN (front side only) Survey Satellite LTS	- Unlimited number of users - Fixed sites	- Requires power source (solar array & battery) - Frequency/time bias monitor/data transfer - Emplacement/survey - Life expectancy
Transponder or Repeater	Power Frequency Ref Antenna size Antenna type Omni Directional	Antenna Transmitter Receiver Frequency Ref Clock	Number Location	Hard landers Artemis Lander Cargo/Piloted Landers Rovers	DSN (front side only) Survey Satellite LTS	- No data transfer necessary - Fixed sites	- Requires power source (solar array & battery) - Noise - Emplacement/survey - Life expectancy

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FLO Navigation Architecture Concepts
Navigation Strawman

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- Top-level strategy
  - "Grow as you go" buildup / Evolve to global coverage
- DSN
  - Proven system and associated technology
  - Performance of ground-based radio nav has improved substantially since Apollo era (≈ 10x)
  - Support will be limited to lunar front-side without far-side comm relay
  - Scheduling
- Navsite
  - Low power two-way range/doppler omni transponder on lunar surface
  - Can be used to maintain accurate orbital ephemeris
  - Provides high accuracy landing aid (≈ 10 m)
- Navsat
  - Moderate altitude orbiting two-way range/doppler transponder
  - Provides primary update for lunar approach, orbital ops, descent, surface nav and departure
  - Function as communication link
- Optical
  - Not currently a primary nav architecture option

FLO Navigation Architecture Concepts
Navigation Strawman (cont)

Navigation strawmen concepts

- Lunar-based infrastructure - autonomy emphasis

  GPS - Support for LEO, post-TLI and Earth return

  Navsites - Single navsite

  - Cone of coverage at 100 km altitude is small (8-10° effective half-cone angle) but lunar rotation is slow (≈1.1°/rev)
  - Landing area beacon can support once/rev state update for at least 6 revs prior to descent and following ascent

  Navsats - small constellation

  - Three low inclination navsats could provide continuous comm/nav coverage at latitudes below 45°
  - Moderate inclination navsat (≈45°) could cover poles

- Earth-based infrastructure - ground based emphasis
  - Utilize existing 26 and 34 meter subnet at Canberra, Madrid and Goldstone stations
  - Available data types are 2-way range and doppler (S-band) and antenna angle
  - New hardware for sub-meter range measurement capability
  - Infrastructure capable of supporting comm/nav from LEO to lunar surface

FLO Navigation Architecture Concepts
Navigation Analyses

- Limited number of navigation analyses have been completed to support the strawman concepts and the infrastructure element options
- Analyses to date have concentrated on radiometrics
  - Near Earth and Earth departure with GPS
  - Lunar approach and on-orbit with navsats and navsites
  - Cargo and piloted landing performance capabilities
- Analyses are also addressing gravity and terrain knowledge precursor requirements
  - Uncertainty in the gravity uncertainty is the major problem hindering performance analyses
  - Map tie error and hazard probabilities are major drivers for terrain knowledge

FLO Navigation Architecture Concepts
Navigation Analyses (cont)

1. Brand, Tim; McClain, Wayne; "Preliminary Analysis of High Altitude Lunar Orbiter Lifetimes", CSDL ESB-92-150, LMI-92-012, March 16,1992.
2. Brand, Tim; Shepperd, Stan; "Lunar Navigation State Uncertainty Growth Rates Resulting from Higher Order Gravity Effects", CSDL, Jan 15, 1992.
3. Brand, Tim; Shepperd, Stan; "Habitat Lander Deorbit to Landing Navigation Study", CSDL ESB-92-149, LMI-92-013, March 17, 1992.
4. Brand, Tim; Shepperd, Stan; "Manned Lander Deorbit to Landing Navigation Study - Preliminary", CSDL LMI-92-015, March 17, 1992.
5. Brand, Tim; Shepperd, Stan; "Habitat and Manned Landers Deorbit to Landing Navigation Study Updates", CSDL, March 19,1992.
6. Cuthbert, Peter; "Lunar Approach with Lunar Navigation Satellite (5000 Km circ)", JSC/SED, March 12, 1992.
7. Hoffman, David; "The Control of Periodic Trajectories about the Unstable Equilibrium Points of the Earth-Moon System - A Status Report), JSC/SED, Feb 21, 1992.
8. Kreimendahl, Frank; "Lunar Mission: Using GPS During Transfer", CSDL ESB-92-146, LMI-92-014, March 13,1992.
9. McClain, Wayne; Brand, Tim; "Impact of lunar Gravity Knowledge on Lunar Missions", CSDL ESB-92-050 LMI-92-004, January 31,1992.
10. Munk, Michelle; "Lunar Surface Beacon Navigation Study - Summary of Results", JSC/SED, Feb 20, 1992.

FLO Navigation Architecture Concepts Issues

- Degree of autonomy at start-up
- Surface and EVA navigation requirements
- Gravity uncertainty
  - Uncertainty in current knowledge
  - Uncertainty in knowledge following precursor mission
- Safe landing for habitat
  - Surface knowledge to support elimination of target redesignation
  - Autonomous target site redesignation capability (hazard avoidance)
  - Artemis lander for site certification

FLO Navigation Architecture Concepts
Future Work

Near Term (≈ 1 month)

- Wrap-up work in defining the top-level architecture concepts from strawman
  - Infrastructure elements and onboard sensor requirements
  - Precursor knowledge and emplacement requirements

Long Term (2 year)

- Navigation analyses to support architecture concepts
  - Performance capability vs fuel penalty trade
  - Methodology trades
  - Knowledge requirements vs navigation mitigation techniques
- Complete description of navigation architecture concepts

The Charles Stark Draper Laboratory, Inc.

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Exploration Programs Office Support
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To: Navigation Team
From: Gregg Barton
Date: 4/8/92
Subj: First Lunar Outpost Navigation Design Detailed Guidelines & Assumptions - Rev 3.0

ESB-92-129B
LMI-92-010B

This note is the third revision of the FLO navigation design Detailed Guidelines and Assumptions (DG&A). The second revision was presented to the ExPO control board for review on March 20, 1992. That revision was accepted and is now under ExPO configuration control. Revision 3.0 currently reflects the configuration control format and the assignment of a reference tracking number. The objective of the navigation DG&A is to identify the design drivers for defining the navigation architecture concepts. Each DG & A should provide definition for the navigation architecture supporting the First Lunar Outpost study. The detailed implementation specific requirements and guidelines will be derived the navigation strawman concepts. The navigation architecture drivers from the FLO Requirements and Guidelines (FLORG) document are included in Appendix A for reference. Appendix B lists the referenced detailed guidelines and assumptions from the FLO segments to show the inter-relationships with navigation.

	MAFE	
The objective of the navigation design guidelines and assumptions is to identify the design drivers for defining the navigation architecture concepts. Each design G & A should provide definition for the navigation architecture supporting the First Lunar Outpost study. Appendix A lists, as reference, the FLO requirements and guidelines (4 March 1992) that were considered drivers for the navigation architecture. The numbers in parenthesis are the requirement tracking numbers. If a navigation G & A is related or derived, then the nav G & A includes the requirement tracking number. Appendix B lists the navigation related detail design guidelines and assumptions from the March 13, 1992 draft to show the inter-relationships between each segment and the nav. G. Barton 03/20/92	?PG	D309
Navigation Architecture Concepts	?PG	H310
The navigation architecture will include the Earth-based infrastructure, the lunar-based infrastructure, and the onboard (spacecraft and surface vehicles) sensors. G. Barton 03/09/92 Rationale: The purpose of this guideline is to define the components that comprise the navigation architecture.	?PG	311
All lunar-based infrastructure elements must address any emplacement, maintenance, health monitoring and life-time issues. G. Barton 03/09/92 Rationale: The above are major factors that must be considered before selecting an infrastructure element as a navigation option. FLORG mission guideline #252B states that the lifetime of the lunar surface elements should be maximized.	?PG	313
The integrated navigation architecture will be designed with synergism in the support of the flight segment (ETO to lunar landing & lunar ascent to Earth landing), the lunar surface segment, and any lunar-based infrastructure survey, ephemeris maintenance or health monitoring. G. Barton 03/19/92 Rationale: FLORG #478 states that the navigation function shall provide an overall integrated navigation capability sufficient to meet mission objectives during all mission phases. The purpose of this guideline is to define the integrated mission areas of support. The navigation architecture should be designed as an integrated capability where, for example, elements selected to support a particular mission phase may also provide a useful hierarchical function during other phases.	?PG	314

	MAFE	
The navigation architecture will place strong consideration on utilizing existing navigation technology and systems. G. Barton 03/19/92 Rationale: This guideline is based on the desire to utilize existing hardware components and systems (FLORG #125), to minimize dependence on developing new technology (especially for technology sake) and to minimize dependence on other program's schedule and technical risk that is in the critical path for FLO mission success (FLORG #305A).	?PG	315
The navigation architecture will be designed for evolution. G. Barton 03/09/92 Rationale: This guideline is a flowdown from the FLORG system requirement #154 and based on the desire to allow evolutionary expansion of the navigation infrastructure. FLORG #253A states that both development cost and life cycle cost should be minimized to the degree practical. As a guideline on the extent of infrastructure utilization, assume a minimum of 4 flights per year. (Reference FLORG 307A & 464)	?PG	316
The navigation architecture will have the appropriate compatibility and synergism with telecommunications. G. Barton 03/09/92 Rationale: FLORG #335A states that an overall integrated communications, information management, and navigation for the terrestrial, lunar and in-flight system shall exist. The navigation concepts and options will be navigation driven but the integrated navigation architecture will factor in the mission communication objectives.	?PG	317
The navigation architecture will be required to support abort during all mission phases. G. Barton 03/19/92 Rationale: This guideline is directly derivable from FLORG #383.	?PG	473
Strong consideration shall be placed on in situ nav. G. Barton 03/11/92 Rationale: This guideline is based on the desire to minimize the required ground support. FLORG #372A states that surface operations should transition from ground-based to autonomous systems.	?PG	320
Precursors	?PG	H321

	MAFE	
The navigation architecture will require gravity knowledge sufficient to keep state vector propagation errors under TBD km after one orbital revolution for all orbital inclinations. G. Barton 03/09/92 Rationale: Navigation studies have shown that the driving factor in propagating the state vector is the uncertainty in the gravity knowledge. The ability to land within a 2 km (3 sigma) of the nominal landing point for the cargo missions (FLORG #153B) and the sortie missions (FLORG #382) may depend on a higher degree of gravity knowledge. The above guideline assumes that navigation will place accuracy requirements on gravity knowledge.	?PG	325
The navigation architecture will require pre-mission terrain maps of TBD accuracy of the landing site and the final TBD km of groundtrack. G. Barton 03/09/92 Rationale: The above guideline assumes that nav will require terrain models for mission planning, simulation and onboard data base in support of the altimeter and terminal landing site redesignation. DG&A #162 assumes stereo images of the landing site will have a 15 m or better resolution. DG&A #164 assumes that site maps should be tied to a common photographic data base.	?PG	326
Assume the gravity and terrain mapping precursor missions have preceded the FLO. G. Barton 03/09/92 Rationale: FLORG #244A states that precursor missions will have been performed to sufficiently characterize the landing site prior to outpost delivery. Any specific knowledge requirements from navigation should be detailed and levied back on the precursor missions.	?PG	323
Precursor missions will be utilized, whenever appropriate, for emplacement and calibration of the lunar-based infrastructure. G. Barton 03/11/92 Rationale: The navigation architecture may require a lunar-based infrastructure of which navigation satellites and/or surface beacons are required to be emplaced and calibrated. Any specific requirements should be detailed and levied back on the precursor missions.	?PG	322
Assume the Artemis lander can deliver and emplace (soft land) the lunar-based navsites that are required in the navigation architecture. G. Barton 03/09/92 Rationale: The Artemis lander is required to provide global access and is a prime candidate for delivering any navigation surface beacons that may be required in the navigation architecture. Any specific emplacement requirements should be detailed and levied back on the precursor mission.	?PG	324
ETO	?PG	H327

	MAFE	
The navigation architecture will be required to support launch escape of the crew from the launch vehicle in case of an abort. G. Barton 03/19/92 Rationale: FLORG #423 states that the crew will be safely propelled away from the launch vehicle during an on-pad or launch abort below TBD km altitude. FLORG #426 states that the crew module will be at a proper attitude for a safe landing after abort.	?PG	474
The design of the FLO mission does not require LEO rendezvous and docking navigation. G. Barton 03/19/92 Rationale: FLORG mission guideline #246A assumes that a single Earth launch vehicle will be utilized for each flight to the Moon. FLORG #248B states that the crew will be launched with the crew module.	?PG	475
The navigation architecture will support a launch insertion orbit of 100 nmi for up to a maximum stay time of 8 hours to TLI. G. Barton 03/09/92 Rationale: This guideline was derived from DG&A #46 and #405. FLORG #417 and DG&A #88 state that the launch avionics will be on the TLI stage. Navigation should call out any minimum stay time required due to Earth-based tracking constraints.	?PG	328
TLI	?PG	H329
The navigation architecture will provide a post-TLI navigation state update to within TBD accuracy (3 sigma) in support of the first mid-course burn. G. Barton 03/09/92 Rationale: The above guideline addresses one of the critical mission phases which will drive the design of the navigation architecture. The assumption now is to perform the first midcourse burn as soon as the navigation state update can support it.	?PG	330
The navigation architecture will support disposal of the TLI stage after the stage is separated from the transportation system. G. Barton 03/20/92 Rationale: FLORG #412 states that the lander will be separated from the TLI stage. FLORG #391 states that the TLI stage must be safely disposed. The above guideline addresses one of the safety issues which must be considered in the design of the navigation architecture. Consideration includes the desire to avoid creating hazardous space debris in orbit around the Earth or Moon, and to avoid impacting the Earth in a populated area.	?PG	331
Trans-Lunar / trans-Earth Coast	?PG	H332
Lunar Approach / LOI	?PG	H333

	MAFE	
The navigation architecture will provide a state precision accuracy at LOI such that the uncertainty in the navigation state (3 sigma) is within the allowable LOI corridor limits. G. Barton 03/09/92 Rationale: The lander must have inserted into an orbit that allows landing within the pre-designated landing site. Large navigation errors can drive the delta-V requirements higher than the allowed fuel budget. The above guideline addresses one of the critical mission phases which will drive the design of the navigation architecture.	?PG	334
The navigation architecture will provide the capability to support any approach orbital inclination. G. Barton 03/09/92 Rationale: FLORG #149A states that the lander will be required to access any lunar latitude / longitude.	?PG	335
Orbital Operations	?PG	H336
The navigation architecture will define for the transportation segment the minimum number of orbits between LOI and the de-orbit burn required to acquire a navigation state update. G. Barton 03/09/92 Rationale: FLORG #376 states that the crew module time in lunar orbit should be minimized. The transportation segment has identified a desire to reduce the loitering time in lunar orbit to minimize transportation consumables. DG&A #415 states that the lander will remain in orbit only long enough to acquire a state update to achieve the required landing accuracy.	?PG	337
The current implementation of the FLO requirements does not require LLO rendezvous and docking navigation. G. Barton 03/19/92 Rationale: This guideline is based on the current access (FLORG #149A) and any time return (FLORG #383 and 148A). Other implementations that meet those requirements may exist which will require this support.	?PG	476
Descent / Landing	?PG	H338
The navigation architecture will be required to provide navigation accuracy during powered descent such that state updates do not have a significant impact on fuel usage. G. Barton 03/20/92 Rationale: Large navigation corrections late in the powered descent can lead to guidance course corrections which significantly increase fuel usage.	?PG	488

	MAFE	
The navigation architecture will be required to support an autonomous landing of a cargo vehicle and a sortie mission to within 2 km (3 sigma) of the nominal landing point. G. Barton 03/19/92 Rationale: DG&A #62 assumes the lander has the capability to land autonomously. FLORG mission guideline #153B, #382 and DG&A #169 assume that cargo/sortie missions will have the capability to land within 2 km (3 sigma) of the designated landing site. There, however, currently is no requirement for accuracy. Instead, a 2 km capability was given to the mission definition lead as a placeholder for site certification planning. The navigation performance capability guideline should be based on the following: - Assume no surface or orbital navaids. - Balance between 1) Gravity knowledge (call out required knowledge). 2) State update accuracy of TBD km (3 sigma) TBD minutes before de-orbit burn. 3) Map tie errors (call out required knowledge).	?PG	339
The navigation architecture will be required to support an autonomous landing of a piloted vehicle with a precision landing performance capability of 100 (E) meter (3 sigma) of the designated landing point. G. Barton 03/20/92 Rationale: The above nav guideline was derived from the following. DG&A #62 assumes the lander has the capability to land autonomously. FLORG mission guideline #381 assumes that the piloted landers have the capability to land within 100 m (3 sigma) of the designated landing site. However, there currently is no requirement for precision landing. FLORG mission guideline #380 states that the piloted lander should land at least 1 km (E) away from the critical elements on the lunar surface to minimize blast ejecta damage. FLORG #385 states that the piloted lander shall land within safe walking distance of the habitat, less than or equal to TBD km (E).	?PG	340
The navigation architecture will be required to support landing site target redesignation for cargo and piloted landers. G. Barton 03/19/92 Rationale: FLORG #415 states that the lander will have the capability to redesignate the landing point until TBD seconds prior to touchdown due to landing hazards that could not be identified from orbit or the large landing site that must be certified due to the navigation inaccuracies. DG&A #408 states that the lander can tolerate hazards up to 0.5 meters in height. DG&A #409 states that the lander can tolerate total tilt up to 12 degrees. DG&A #122 assumes that the habitat landing will occur within 3 days of lunar sunrise. FLORG mission guideline #495 assumes that the piloted landings will occur within 4 days following lunar sunrise. Both cargo and piloted landings will have low sun angles to aid in the identification of surface hazards.	?*R	477

	MAFE	
The navigation architecture will be required to land the cargo lander with the solar arrays aligned within +/- 15 degrees of north/south. G. Barton 03/19/92 Rationale: DG&A #103 states that the habitat will be landed with the solar arrays aligned within +15 degrees of north/south to allow tracking of the sun by the arrays with very little cosine loss.	?PG	478
Surface	?PG	H343
The navigation architecture will be required to support manned roving distances of TBD km from the outpost with TBD meter accuracy (3 sigma). G. Barton 03/19/92 Rationale: The purpose of this guideline is to give an idea of the range (and accuracy) of the rover in which the navigation architecture must support. EVA period is 8 hours. (Reference FLORG 202C.) DG&A #215 states that a manned roving capability of up to 10 km away from the habitat should exist. DG&A #165 assumes that preplanned traverses will be indicated on site maps. DG&A #218 states that manned rovers will have a capability for direct Earth link communications.	?PG	344
The navigation architecture will be required to support robotic roving distances of 100 km from the habitat with TBD meter accuracy (3 sigma). G. Barton 03/09/92 Rationale: The purpose of this science design guideline is to give an idea of the range (and accuracy) of the telerobotic rover in which the navigation architecture must support. (Reference DG&A #201)	?PG	345
The navigation architecture support for EVA will be TBD. G. Barton 03/09/92 Rationale: At this point in time, the guideline is unknown. FLORG #441 states there will be 34 (E) EVAs exclusive of contingency.	?PG	346
Ascent	?PG	H347
The navigation architecture will support a launch insertion orbit of 100 km for up to a maximum stay time of TBD hours before TEI. G. Barton 03/09/92 Rationale: The above guideline addresses one of the critical mission phases which will drive the design of the navigation architecture. The navigation design will need to consider the inclination range.	?PG	348
TEI	?PG	H349
The navigation architecture will provide a post-TEI navigation state update to within TBD accuracy (3 sigma) in support of the first mid-course burn. G. Barton 03/09/92 Rationale: The above guideline addresses one of the critical mission phases which will drive the design of the navigation architecture. The assumption now is to perform the first midcourse burn as soon as the navigation state update can support it.	?PG	350

	MAFE	
Earth Approach	?PG	H351
The navigation architecture will provide sufficient state accuracy to place the vehicle within the flyable entry corridor. G. Barton 03/09/92 Rationale: The return module must fly within the entry corridor. If the flight angle is too steep, the vehicle may exceed g-limits or thermal constraints. If the flight angle is too shallow, the vehicle may skip out. The above guideline addresses one of the critical mission phases which will drive the design of the navigation architecture.	?PG	352
The navigation architecture will support disposal of the return stage after the stage is separated from the crew module. G. Barton 03/20/92 Rationale: FLORG #416 states that the crew module will be separated from the return stage. FLORG #419 states that the return stage must be safely disposed. The above guideline addresses one of the safety issues which must be considered in the design of the navigation architecture. Considerations include the desire to avoid creating hazardous space debris in orbit around the Earth or Moon, and to avoid impacting the Earth in a populated area.	?PG	479
Direct Descent	?PG	H353
The navigation architecture must provide accuracy during atmospheric entry such that the vehicle heating, loads, and flight path are within design limits. G. Barton 03/20/92 Rationale: Large uncertainties in navigated state during re-entry can impact vehicle design requirements for thermal protection, loads, and lift/drag ratio.	?PG	489
The navigation architecture will be required to support direct descent for a landing on water and land of TBD accuracy (3 sigma). G. Barton 03/19/92 Rationale: The above guideline addresses one of the critical mission phases which will drive the design of the navigation architecture and is derived from the following requirements. FLORG #386 states that the crew module will have a direct Earth re-entry. FLORG #498 states that the Earth landings on land will be at TBD locations. FLORG #257A states that the nominal landing point will be on land. FLORG #373A and #422 states that a contingency landing will be on water. FLORG #256A states that landings will be performed under any lighting constraints.	?PG	354

APPENDIX A

?PG H469

The following are the FLO Requirements and Guidelines (FLORG) that are considered drivers for the navigation architecture. The numbers in parenthesis are the requirement tracking numbers. ?PG D470

4.0 FIRST LUNAR OUTPOST DESCRIPTION

4.2 Design Guidelines and Assumptions

(253A) To the degree practical, minimize both development cost and life cycle cost.

(305A) To the degree practical, minimize dependence on other programs.

(125) To the degree practical, utilize existing hardware components and systems.

(376) To the degree practical, minimize crew module time in orbit.

(377) To the degree practical, minimize orbital operations such as rendezvous, docking, and assembly for piloted and cargo vehicles.

(252B) To the degree practical, maximize lifetimes of all lunar surface elements.

(246A) Assume a single Earth launch vehicle shall be utilized for each flight to the Moon.

(244A) Assume the necessary precursor missions will have been performed to sufficiently characterize the landing site prior to outpost delivery.

(495) Assume piloted landings on the lunar surface will occur within 4 days following lunar sunrise.

(380) Assume that flight elements shall land at least 1 km (E) away from critical elements on the lunar surface to minimize blast ejecta damage.

(381) For nominal missions and revisit missions, assume piloted landers shall provide the capability to land within 100 m (3 sigma) of the designated lunar landing site.

(153B) Assume cargo landers shall provide the capability to land within 2 km (3 sigma) of the designated lunar landing site.

(382) For sortie missions, assume piloted landers shall provide the capability to land within 2 km (3 sigma) of the designated landing site.

5.0 TECHNICAL REQUIREMENTS

5.2 System Requirements

(154) The system shall be designed for evolution.

(307A) The system shall provide the capability to support a minimum of four (4) flights per year.

(149A) The system shall provide the capability to access any lunar latitude/longitude.

(387) Sortie option lunar surface stays shall be restricted to lunar daylight.

(383) The system shall provide the capability for abort during all mission phases.

(385) For nominal missions, piloted elements shall land within safe walking distance of the habitat, less than or equal to TBD km (E).

(386) The system shall provide the capability for direct Earth re-entry of the crew module.

(498) The system shall provide the capability to perform Earth landings on land at TBD locations.

(373A) The system shall provide the capability to perform Earth water landings

(257A) Nominal Earth return landings shall take place on land.

(155B) The system shall only utilize technologies which have undergone system validation or engineering model demonstration in a relevant environment prior to commitment to development.

5.2.1 Flight Segment

(248B) The crew shall be launched with the space transfer vehicle crew module.

(256A) The flight segment shall provide the capability to perform Earth landings under any lighting conditions.

(168A) The flight segment shall provide the capability to revisit previous lunar landing sites.

5.2.1.2 Space Transit Sub-Segment

(414) The space transit sub-segment shall provide the capability for abort at all times during piloted lunar descent.

(415) The space transit sub-segment shall provide the capability for landing site aimpoint redesignation until TBD seconds prior to lunar touchdown.

(148A) The space transit sub-segment shall provide the capability for abort at all times following piloted lunar landing within 24 hrs. of abort declaration.

(412) The space transit sub-segment shall provide the capability for separation of the lander from the TLI stage.

(416) The space transit sub-segment shall provide the capability for separation of the crew module from the return stage.

5.2.1.2.1 TLI Stage Element

(417) The TLI stage element shall provide the capability for ascent guidance and control of the launch vehicle during launch from Earth.

(418) The TLI stage element shall provide the capability for attitude correction prior to TLI burn.

(391) The capability shall be provided for safe disposal of the TLI stage element.

5.2.1.2.3 Return Stage Element

(191B) The return stage element shall provide the capability for abort at all times following lunar landing within 24 hrs. of abort declaration.

(419) The capability shall be provided for safe disposal of the return stage element.

5.2.1.2.4 Crew Module Element

(489) The crew module element shall provide the capability to land on land when returning to Earth.

(422) The crew module element shall provide the capability to land on water when returning to Earth.

5.2.1.2.5 Launch Escape System Element

(423) The launch escape system element shall provide the capability to safely propel the crew module away from the launch vehicle during an on-pad or launch abort below TBD km altitude.

(426) The launch escape system element shall provide the capability to assure proper crew module attitude for a safe Earth landing after the abort.

5.2.2 Surface Segment

5.2.2.1 Habitation Sub-Segment

(441) The habitation sub-segment shall provide the capability to support 34 (E) EVAs exclusive of contingency.

5.2.2.2 Surface Mobility Sub-Segment

5.2.2.2.1 Rover Element

(202C) The rover element shall provide the capability to transport a crew of 2 (E) and a cargo of TBD mt a distance of TBD km from the outpost.

5.2.2.3.2 Navigation

(348A) The capability shall be provided to aid the navigation of elements in lunar orbit.

(349A) The capability shall be provided to aid the navigation of elements on the lunar surface.

(374A) The capability shall be provided to aid the navigation of piloted landers during lunar descent during the nominal mission.

(375A) The capability shall be provided to aid the navigation of ascent vehicles during lunar ascent.

5.2.3 Support Function Segment

(464) The Support Segment shall be able to support a rate of four FLO missions a year.

5.2.3.1 Mission Support and Control Sub-Segment

(466) The Mission Support and Control Sub-Segment shall maintain a capability during all mission phases to abort the mission and safely return the flight crew to Earth.

5.2.3.1.4 Surface Operations

(372A) The surface operations function shall provide for an operational transition from ground- based control to surface-based control and autonomy of lunar surface systems and elements.

5.2.3.4 TISN Sub-Segment

(335A) The TISN sub-segment shall provide an overall integrated capability for communication, information management, and navigation for the terrestrial, lunar, and in-flight systems involved in FLO missions.

(476) The TISN sub-segment shall facilitate fault detection, fault isolation, and recovery control for the terrestrial, lunar, and in-flight systems involved in FLO missions.

5.2.3.4.2 Navigation

(478) The navigation function shall provide an overall integrated navigation capability sufficient to meet mission objectives during all mission phases.

(479) The navigation function shall provide a capability for state vector updates to flight and surface vehicles.

(480) The navigation function shall provide data for post-flight trajectory analysis and reconstruction.

5.4 System Design Safety and Reliability

(485) All systems shall be "fail safe".

(486) Redundant systems shall provide a capability to verify their operational status to the fail safe level during ground turnaround and during the mission.

(487) The systems for the first lunar outpost mission shall not preclude a piloted mission from restoring full operational capability after a single functional failure.

G. Barton 03/20/92

APPENDIX B

?PG H471

?PG D472

The following are the mission/system detailed design guideline and assumptions for the First Lunar Outpost that are considered drivers for the navigation architecture. The numbers in parenthesis are the guideline tracking numbers.

3.1 LUNAR TRANSPORTATION SYSTEM (LTS) DETAILED ASSUMPTIONS

Space Transit Sub-Segment

(415) The space transit sub-segment shall provide the capability for the lander element to remain in lunar orbit only for that time required for the navigation state vector update to achieve the required landing accuracy.

Lunar Lander

(408) The lander element shall provide the capability to tolerate a landing hazard of less than or equal to 0.5 meters in height.

(409) The lander element shall provide the capability to operate with a total lander tilt of less than or equal to 12 degrees.

(62) The lander element shall provide the capability to automatically land on the lunar surface.

TLI Stage Design

(46) The TLI stage element shall provide the capability of a suborbital burn to achieve a launch insertion orbit of 100 nmi and maintain the orbit up to a maximum stay time of 8 hours prior to TLI burn.

Return Stage

(76) The capability shall be provided for the safe disposal of the return stage element.

3.2 HEAVY LIFT LAUNCH VEHICLE (HLLV) DETAILED ASSUMPTIONS

Earth to Orbit Sub-Segment

(405) The following orbits shall be used: 100 nm circular for lunar direct; 220 nm circular for Mars missions.

(88) All launch phase primary avionics (e.g., GN&C equipment, computers, telemetry equipment, etc.) shall be located on the TLI stage and shall be independent of the LTS avionics. Each stage shall be capable of sending stage sub-system status to the launch phase primary avionics.

3.3 HABITATION SUB-SEGMENT DETAILED ASSUMPTIONS

Habitat

(103) The habitat will be landed with the solar arrays aligned within +15 degrees of north/south.

(122) The habitat shall be landed between lunar dawn and lunar dawn plus 72 hours.

3.4 SCIENCE AND PAYLOADS DETAILED ASSUMPTIONS

Prelaunch

(162) Precursor data includes stereo image of [the landing site] area at 15 m or better resolution.

(163) The Landing site shall be chosen using precursor data and based on science criteria, resource criteria, and operational requirements.

(164) Site maps shall be produced on a photographic data base.

(165) Preplanned traverses will be indicated on site maps.

(166) Site topography and engineering data to be indicated on site maps.

Landing

(169) Landing accuracy for delivery of [the] campsite will be +/- 2 km (3 sigma).

Robotic Science Rover Operations

(201) The robotic science rover will have range sufficient to reach areas at least 100 km away from the habitat and return.

Manned Rover Element Design

(215) The manned rover will have the capability to land, transport, offload and deploy payloads of up to 400 kg at distances up to 10 km away from the habitat.

(218) The manned rover will have a capability for direct Earth link communications.

Robotic Science Rover Element Design

(225) Robotic science rover shall have stereo video image capability and capability for chemical analysis of geologic materials.

G. Barton 03/20/92

The Charles Stark Draper Laboratory, Inc.

EXPLORATION PROGRAMS OFFICE

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Telephone (713) 283-5470
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ESB-92-209
LMI-92-025

To: Mission Definition Leads
From: Gregg Barton
Date: 4/22/92
Subj: Response to Comments on Nav Detailed Guidelines & Assumptions

The following comments were received from the Space Transit Subsegment Subteam dated 4/1/92 on the Navigation Detailed Guidelines and Assumptions. The Response section will repeat the guideline under review and provide a recommended action.

Navigation

- *Omit all FLORG Repeat assumptions (#315, 316, 323, Appendix A)*

 # 477 - "...support landing site target redesignation..." - With what? How?

Recommendation: add more detail to the assumption

Response:

The navigation architecture will place strong consideration on utilizing existing navigation technology and systems. G. Barton 03/19/92 Rationale: This guideline is based on the desire to utilize existing hardware components and systems (FLORG #125), to minimize dependence on developing new technology (especially for technology sake) and to minimize dependence on other program's schedule and technical risk that is in the critical path for FLO mission success (FLORG #305A).	?PG	315
The navigation architecture will be designed for evolution. G. Barton 03/09/92 Rationale: This guideline is a flowdown from the FLORG system requirement #154 and based on the desire to allow evolutionary expansion of the navigation infrastructure. FLORG #253A states that both development cost and life cycle cost should be minimized to the degree practical. As a guideline on the extent of infrastructure utilization, assume a minimum of 4 flights per year. (Reference FLORG 307A & 464)	?PG	316

Nav DG&A #315 and 316 are not a repeat of the general FLORG requirements although the wording is similar. The objective is to provide specific guidance on how the navigation architecture should be defined. DG&A #315 addresses the multi-methodologies of navigation

options available to mission development. DG&A #316 addresses the top-down approach of evolving the navigation infrastructure instead of assuming a full-up operations at start-up.

Assume the gravity and terrain mapping precursor missions have preceded the FLO. G. Barton 03/09/92 Rationale: FLORG #244A states that precursor missions will have been performed to sufficiently characterize the landing site prior to outpost delivery. Any specific knowledge requirements from navigation should be detailed and levied back on the precursor missions.	?PG	323

Navigation DG&A #323 can be deleted as it is a repeat of FLORG #244A. The specific guidelines addressing gravity and terrain knowledge needs are found in Nav DG&A #325 and #326.

APPENDIX A ...

The Detailed Guidelines & Assumptions leads had originally requested that Appendix A be included due to the identification of the interfaces between the FLORG and the DG&A. At this point, it is unclear as to what direction the DG&A document will proceed. The decision to remove Appendix A will be left to the DG&A manager.

The navigation architecture will be required to support landing site target redesignation for cargo and piloted landers. G. Barton 03/19/92 Rationale: FLORG #415 states that the lander will have the capability to redesignate the landing point until TBD seconds prior to touchdown due to landing hazards that could not be identified from orbit or the large landing site that must be certified due to the navigation inaccuracies. DG&A #408 states that the lander can tolerate hazards up to 0.5 meters in height. DG&A #409 states that the lander can tolerate total tilt up to 12 degrees. DG&A #122 assumes that the habitat landing will occur within 3 days of lunar sunrise. FLORG mission guideline #495 assumes that the piloted landings will occur within 4 days following lunar sunrise. Both cargo and piloted landings will have low sun angles to aid in the identification of surface hazards.	?*R	477

Nav DG&A #477 was written as a place holder in response to FLORG #415 which stated that the lander would have target redesignation capability but did not identify which lander. It was assumed that the piloted lander will use target redesignation but it was not clear if this applied to the cargo vehicle as well. Navigation DG&A #477 should be written to reflect the correct implementation for both the piloted and cargo vehicles. However, before the specifics of #477 are written, the decision for the methodology for safe landing a cargo vehicle should be made.

The Charles Stark Draper Laboratory
555 Technology Square, MS2B, Cambridge, Massachusetts 02139 Telephone (617) 258-2282
Fax No. (617) 565-8674

TO: Navigation Team
FROM: Judy Miller
DATE: March 31, 1992

SUBJECT: Top Level and Second Tier Navigation Options "Menu"

Introduction

A menu of navigation options for the First Lunar Output has been completed and is described herein. The menu includes two elements: (1) the Top Level element starts with mission segments and graphically presents candidate navigation options for each, and (2) the Second Tier provides further details of the navigation options identified by the Top Level.

I. OPTIONS/[*MISSION SEGMENT*]

The top level menu relates navigation options to mission segments including both flight and surface operations. The purpose is to enable the user to graphically determine candidate sources of navigation data for a given segment of mission operations.

For a major mission segment of interest, the major factors influencing navigation are identified and the final right-hand side boxes indicate potential sources of navigation data. Only those sources which are compatible with the First Lunar Outpost Navigation Design Guidelines & Assumptions are included.

The vehicles considered were cargo and piloted flight vehicles, and manned and telerobotic surface vehicles. Mission operations segments considered were earth launch & low earth orbit (LEO), trans-lunar injection (TLI) and first midcourse burn, trans lunar coast, approach and lunar orbit insertion, lunar orbit, descent and landing on the lunar surface, surface operations, lunar launch, lunar orbit, trans earth injection (TEI) and first midcourse burn, trans earth coast, earth approach, and earth descent and landing. Note that all operations except for surface (which applies to manned and telerobotic rovers), and lunar launch through earth landing (which applies only to the piloted vehicle) apply to both cargo and piloted flight vehicles. The title indicates the applicability of vehicles for that phase. The top level is included as Attachment A.

II. OPTIONS/[NAVIGATION OPTIONS]

This second tier provides further details of the navigation options. Each option is categorized into: (a) earth-based, (b) lunar local-based navaids, or (c) astronomical/surface navigation types. Major features of each category are presented in tabular form with the contents of each table being tailored to the particular category. The second tier is included as Attachment B.

Attachment A

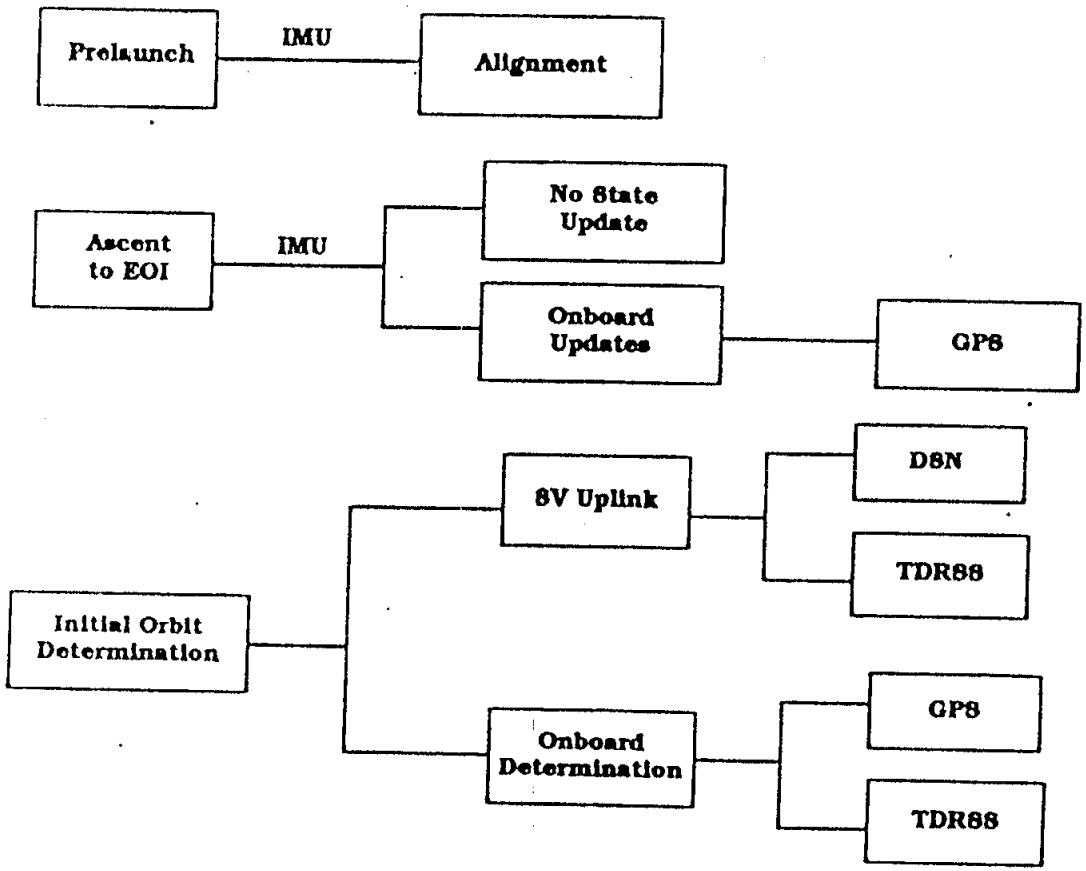
Menu Flight Phases

(03/27/92 Rev 2 .1)

1) Earth Launch & LEO
2) TLI & Midcourse Burn
3) Trans Coast
4) Approach & Lunar Orbit Insertion
5) Lunar Orbit
6) Descent & Landing
7) Surface
8) Lunar Launch
-) Lunar Orbit
9) TEI & Midcourse Burn
-) Trans Coast
10) Earth Approach
11) Earth Descent & Landing

PRELIMINARY
OPTIONS/Earth Launch & LEO
(Cargo/Piloted)

EXPLORATION PROGRAMS OFFICE



IMPLICATIONS

IMU mandatory/Alignment establishes initial accuracy

Inertial-only/Error propagation over 30 minutes

High dynamics during ascent/Added accuracy probably unnecessary

Arc length & Processing time /Coverage

Arc Length & Processing time

One-way range & Doppler/Favorable geometry/Rapid convergence/High accuracy

One-way Doppler (stable oscillator)/Convergence rate ?

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PRELIMINARY
OPTIONS/TLI & Midcourse Burn
(Cargo/Piloted)

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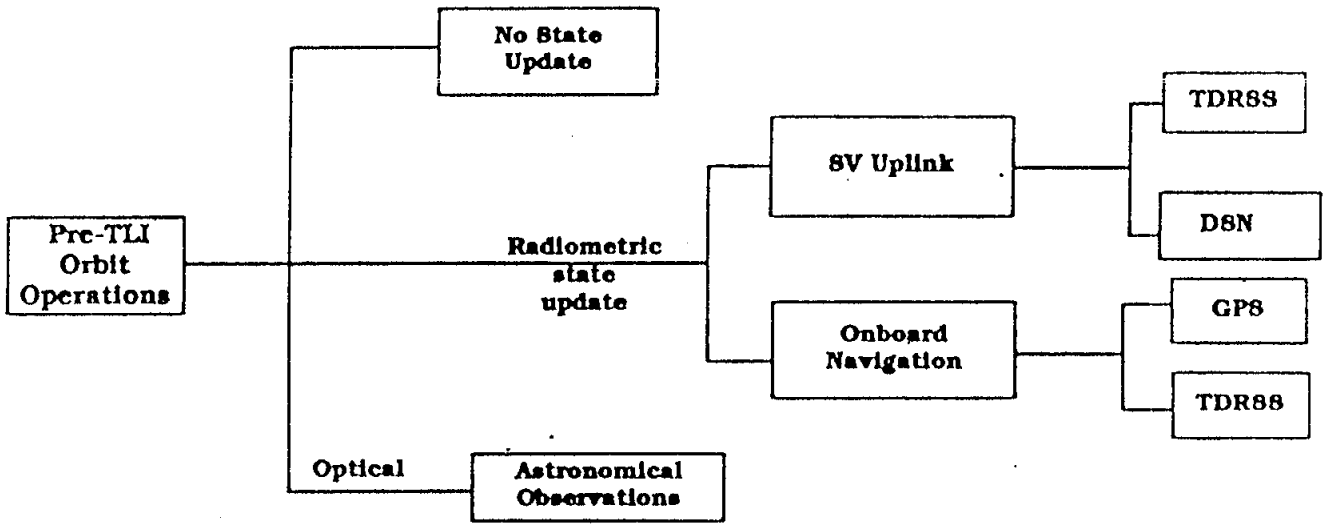

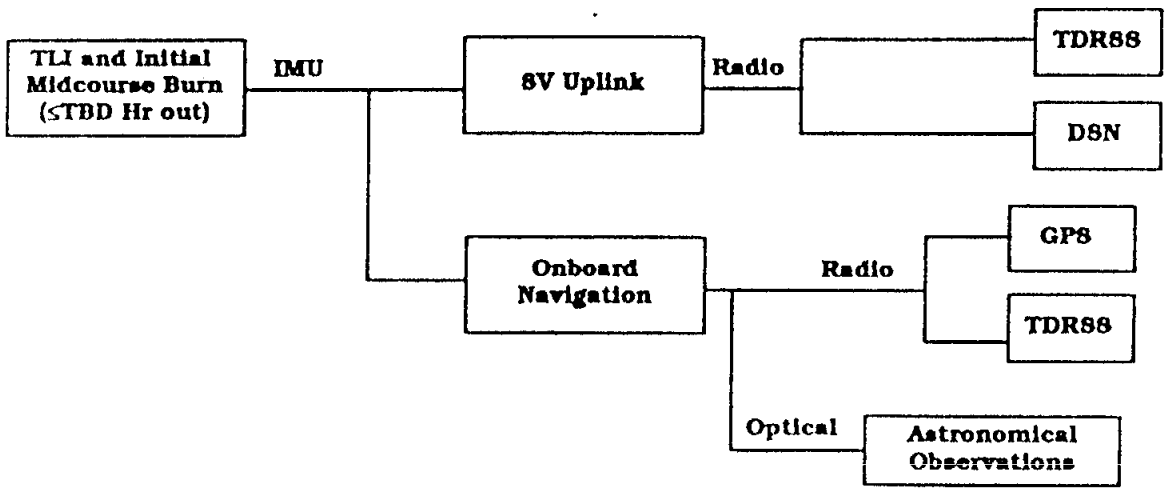

IMPLICATIONS

Error propagation over 4.5 hours (max)

Arc length & Processing time

Arc length & Processing time/Coverage

Good satellite visibility/Favorable geometry/<<100 meter accuracy (verify)

One-way Doppler(stable oscillator)

Autonomous/Attitude constraints

Arc length & Processing time

Arc length & Processing time

Navigation through thrust/Good geometry below GPS altitude

One-way Doppler(stable oscillator)

Autonomous/Attitude constraints

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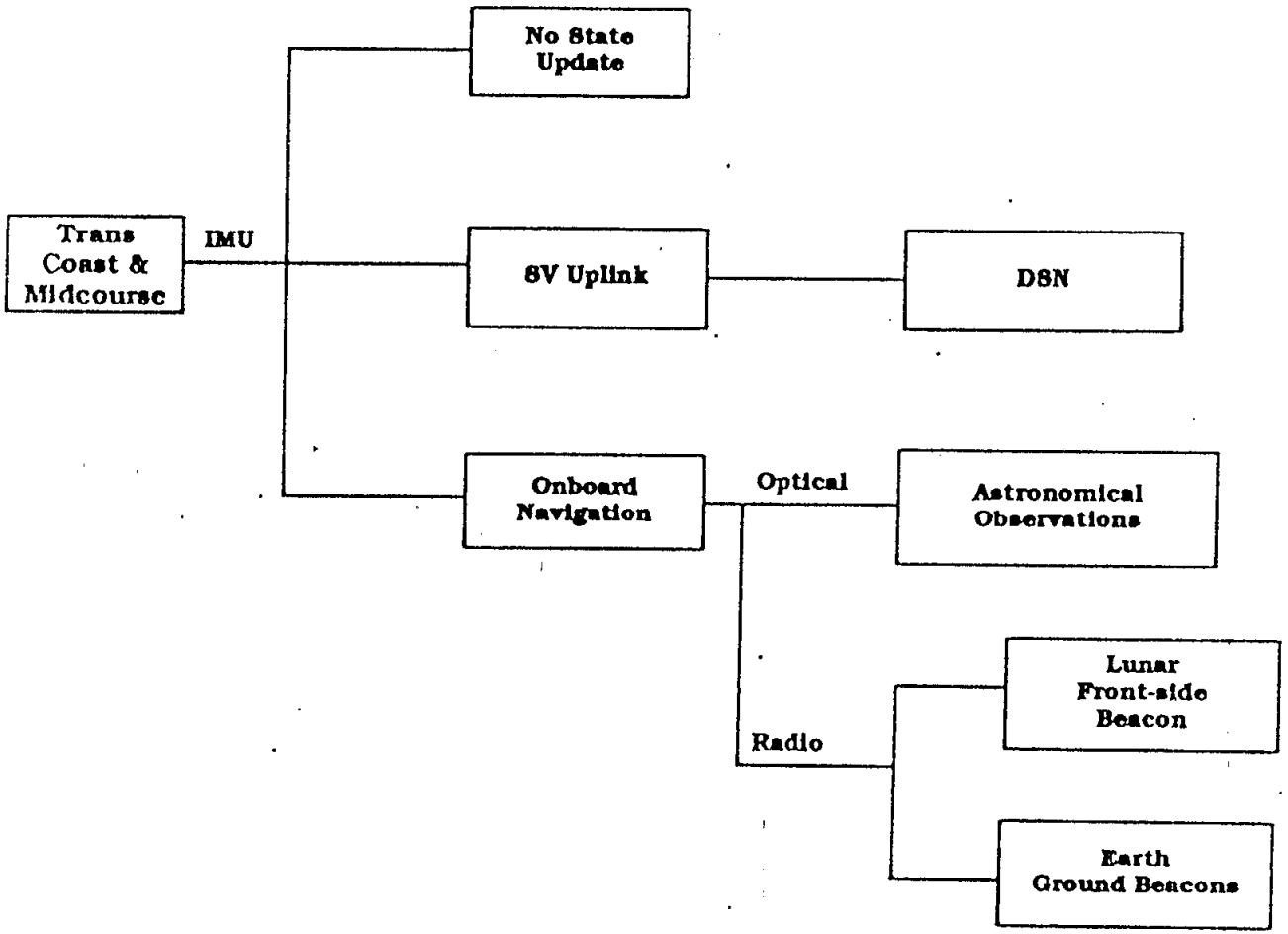
PRELIMINARY
OPTIONS/Trans Coast (Lunar & Earth)
(Cargo*/Piloted)



IMPLICATIONS

Error propagation over 4 to 5 days

Arc length & Processing time/Update rate

Autonomous/Attitude constraints

Signal strength/Emplacement

Signal strength

* Cargo is Trans-Lunar only.

03/26/02 Rev. 2.1

PRELIMINARY
OPTIONS/Lunar Approach & Orbit Insertion
(Cargo/Piloted)

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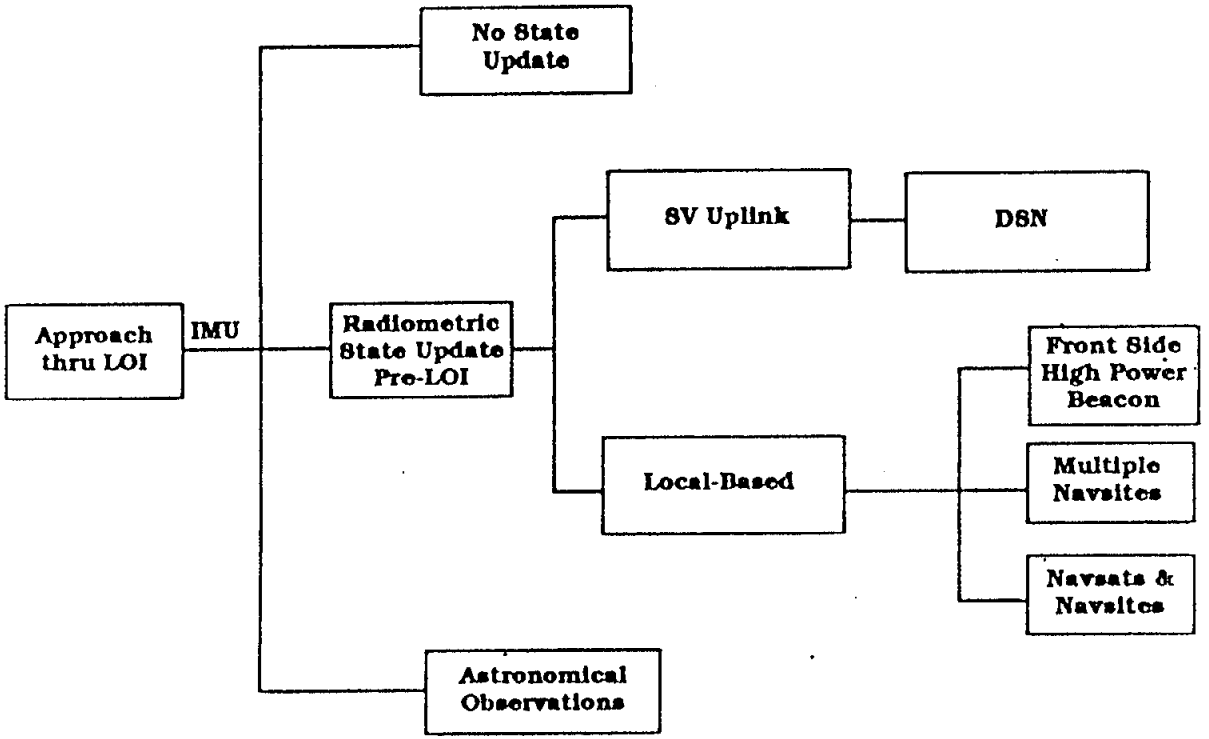

IMPLICATIONS

No State Update	Error dependent upon coast phase navigation
SV Uplink → DSN	Arc length & Processing time
Front Side High Power Beacon	Acquisition range
Multiple Navsites	Acquisition range/Coverage
Navsats & Navsites	Coverage
Astronomical Observations	Autonomous/Attitude constraints

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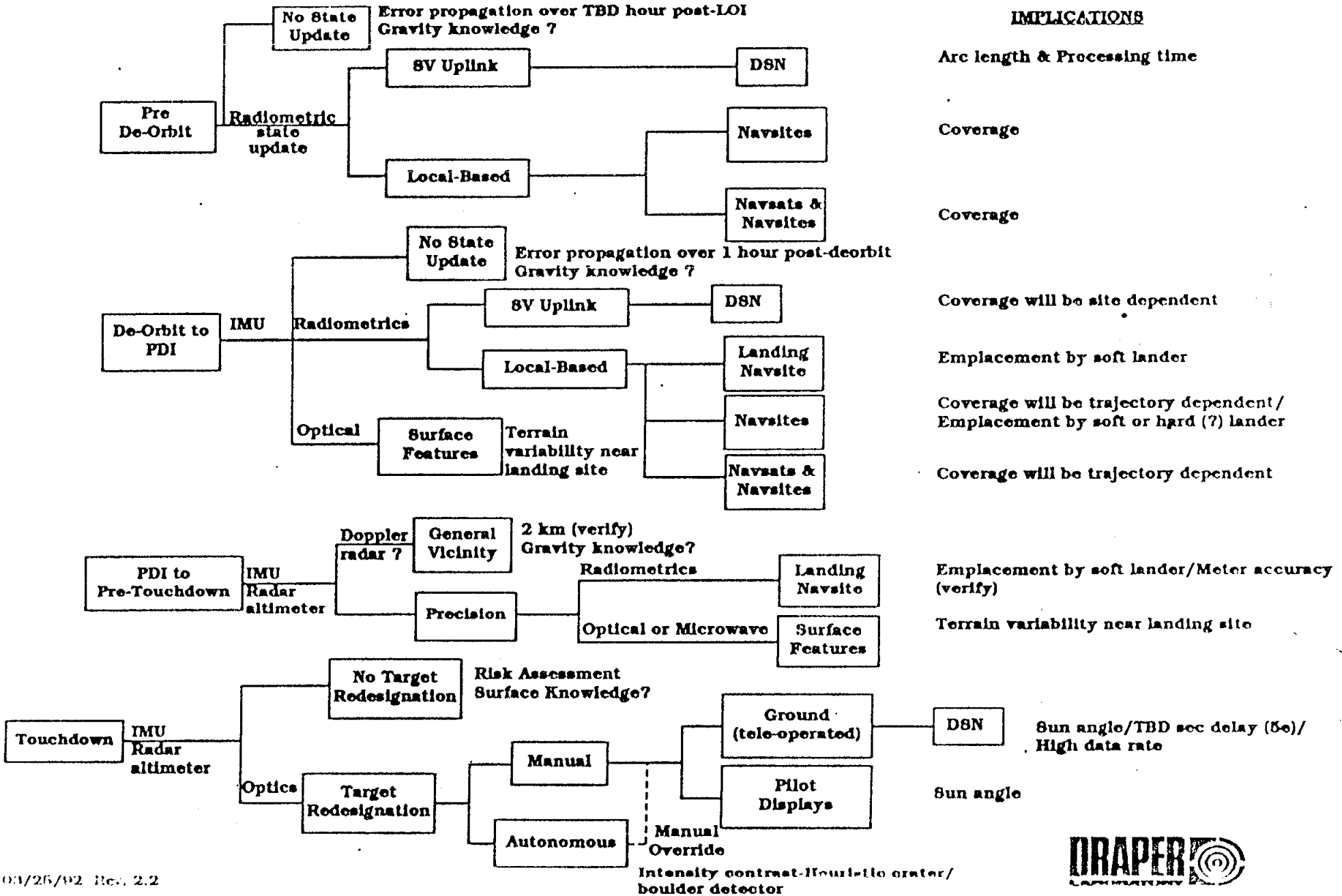
PRELIMINARY
OPTIONS/Descent & Landing
(Cargo/Piloted)

EXPLORATION PROGRAMS OFFICE



03/26/92 Rev. 2.2

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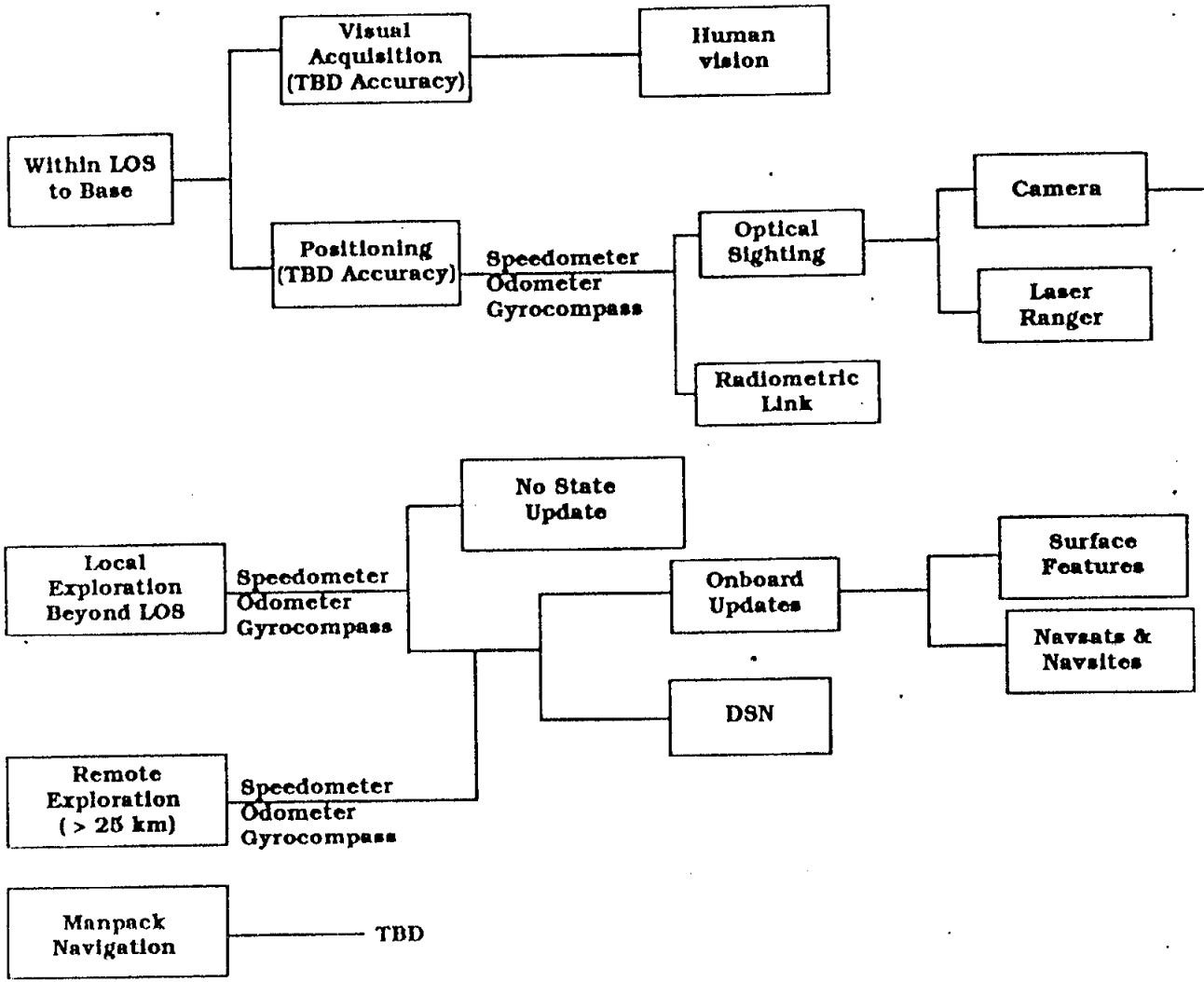
PRELIMINARY OPTIONS/Surface (Manned/Telerobotic)

EXPLORATION PROGRAMS OFFICE



IMPLICATIONS

Unique to manned rover/Limited accuracy

Use if onboard for another purpose

Error propagation over TBD hours

Surface knowledge/Feature extraction/Correlation/Automation

Coverage/Navsats only

Front side only coverage/Arc length & Processing time

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PRELI[illegible]
OPTIONS/Lunar Launch
(Piloted)

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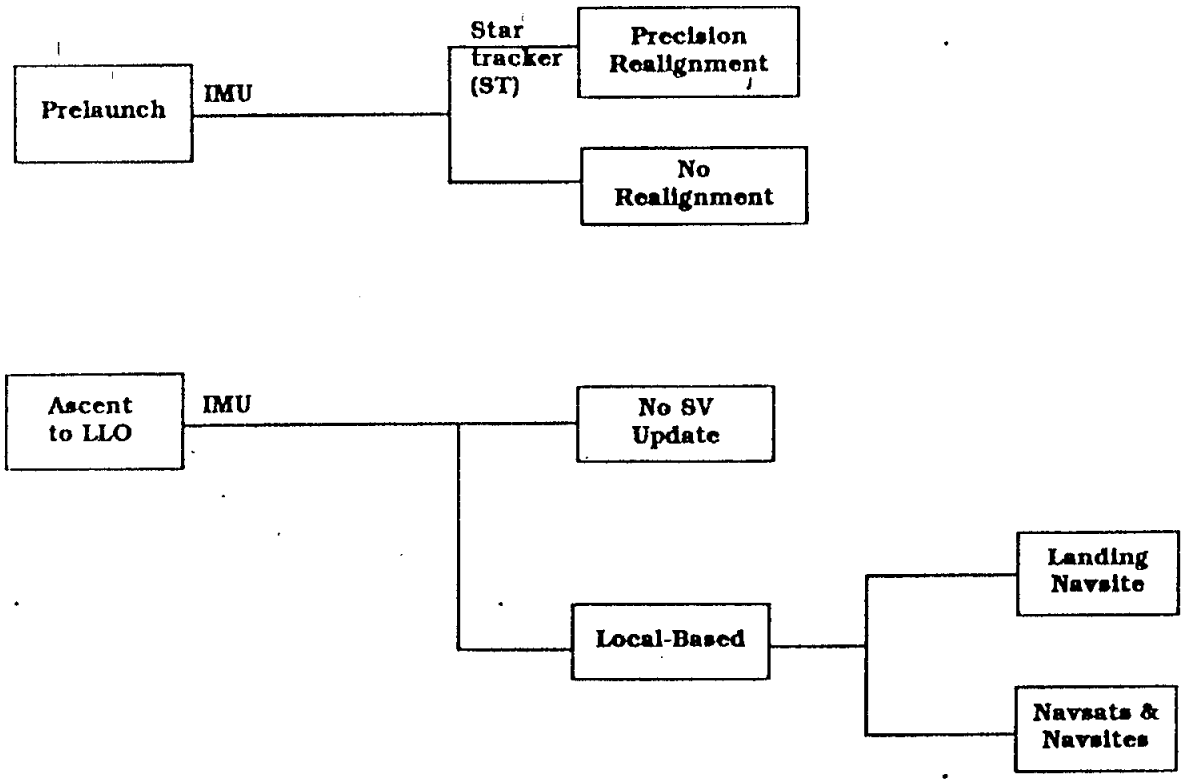

IMPLICATIONS

Time/Emergency launch

Accuracy/Alignment stability

Inertial-only/Error propagation over 10 minutes

Relative geometry will be trajectory dependent

Coverage/Relative dynamics may be high

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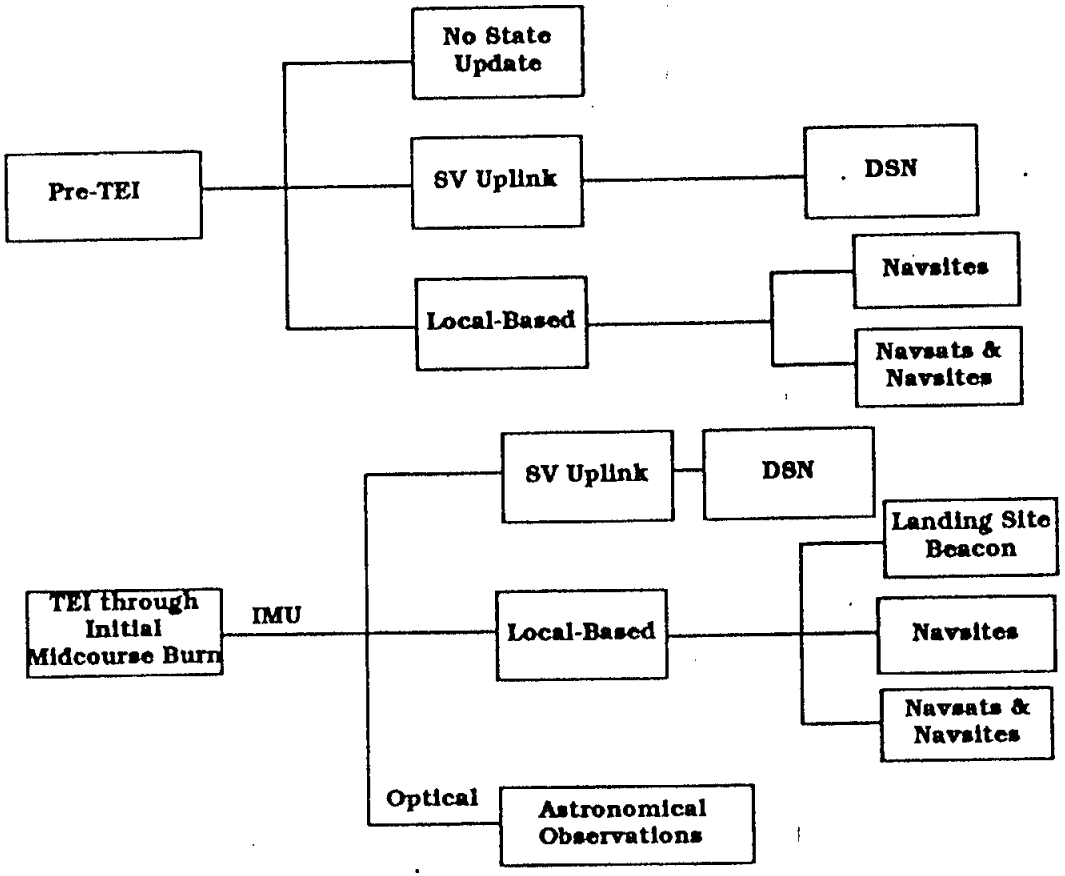
PRELI

OPTIONS/TEI and Midcourse Burn (Piloted)

EXPLORATION PROGRAMS OFFICE



IMPLICATIONS

Error propagation up to 12 hours

Arc length & Processing time

Coverage

Coverage

Arc length & Processing time

Coverage

Coverage/Acquisition range

Coverage

Autonomous/Attitude constraints

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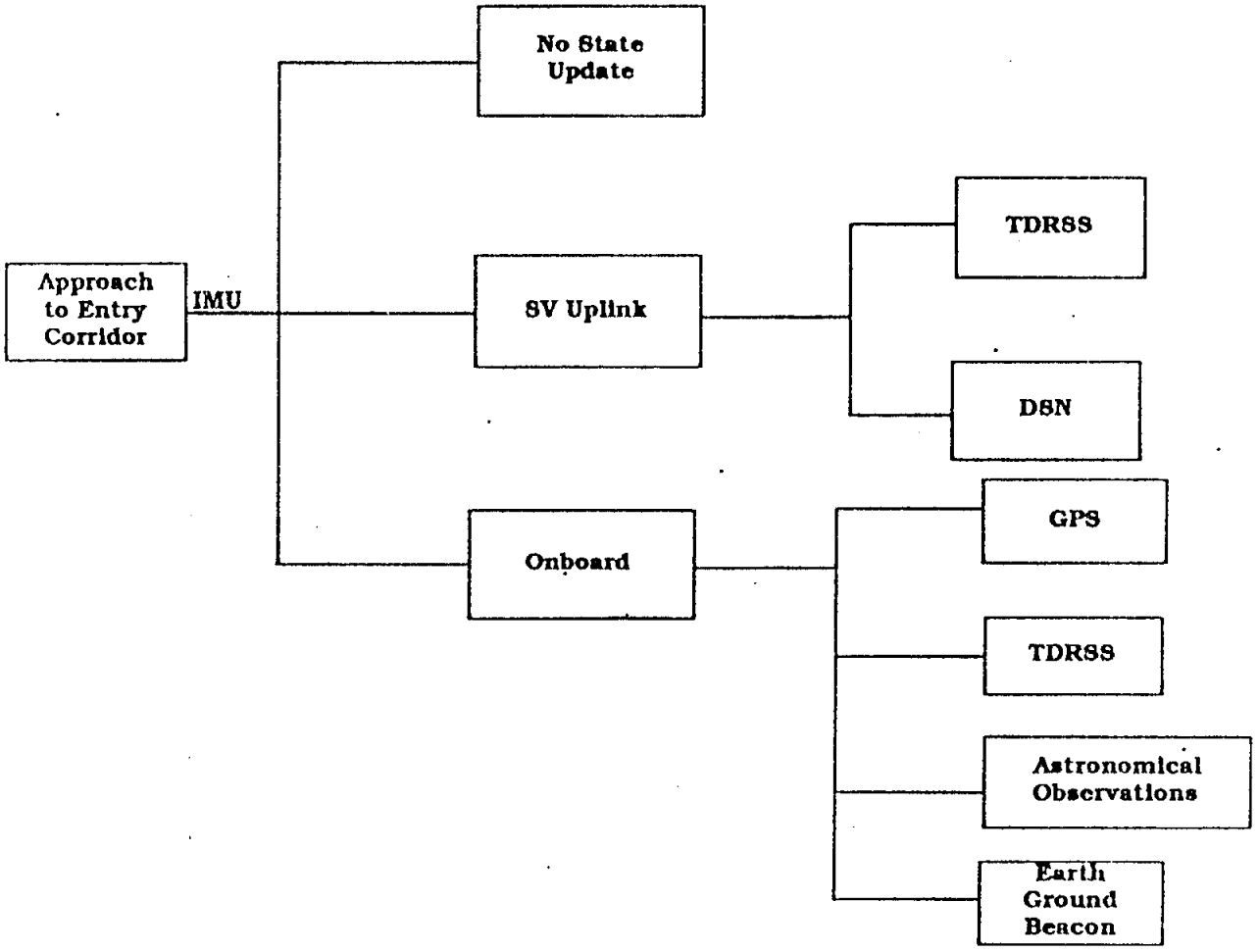
PRELIMINARY
OPTIONS/Earth Approach
(Piloted)

EXPLORATION PROGRAMS OFFICE



IMPLICATIONS

Error propagation dependent upon coast phase navigation

Acquisition range/Arc length & Processing time

Coverage/Arc length & Processing time

Acquisition range/Coverage

One way Doppler (stable oscillator)TDRS Antenna orientation (gimbal angles)/ Acquisition range

Autonomous/Attitude constraints

Signal strength/Coverage

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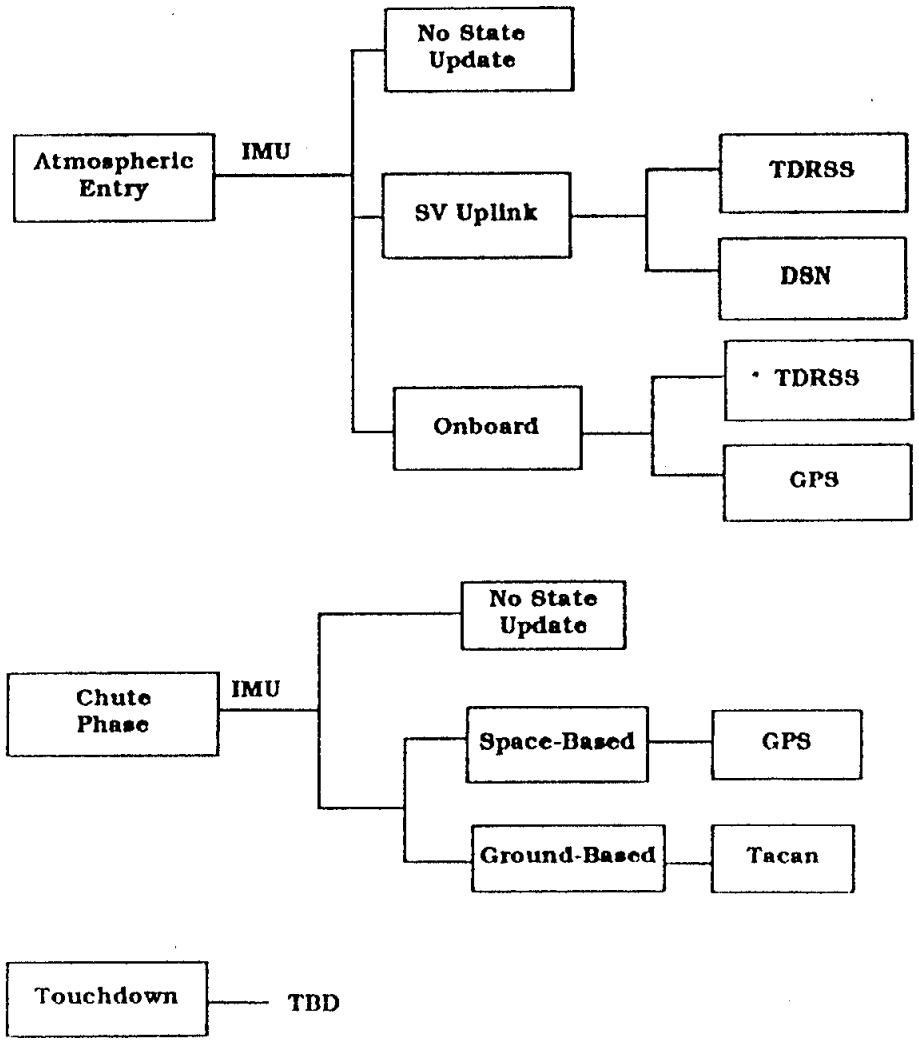
PRELIMINARY

OPTIONS/Earth Descent & Landing (Piloted)

EXPLORATION PROGRAMS OFFICE



IMPLICATIONS

Error propagation over TBD minutes

Coverage will be trajectory dependent/Blackout

Coverage will be trajectory dependent/Blackout

One way Doppler (stable oscillator)/ Coverage/State observability

Blackout

Error propagation for TBD minutes

Good accuracy

Coverage will be trajectory dependent

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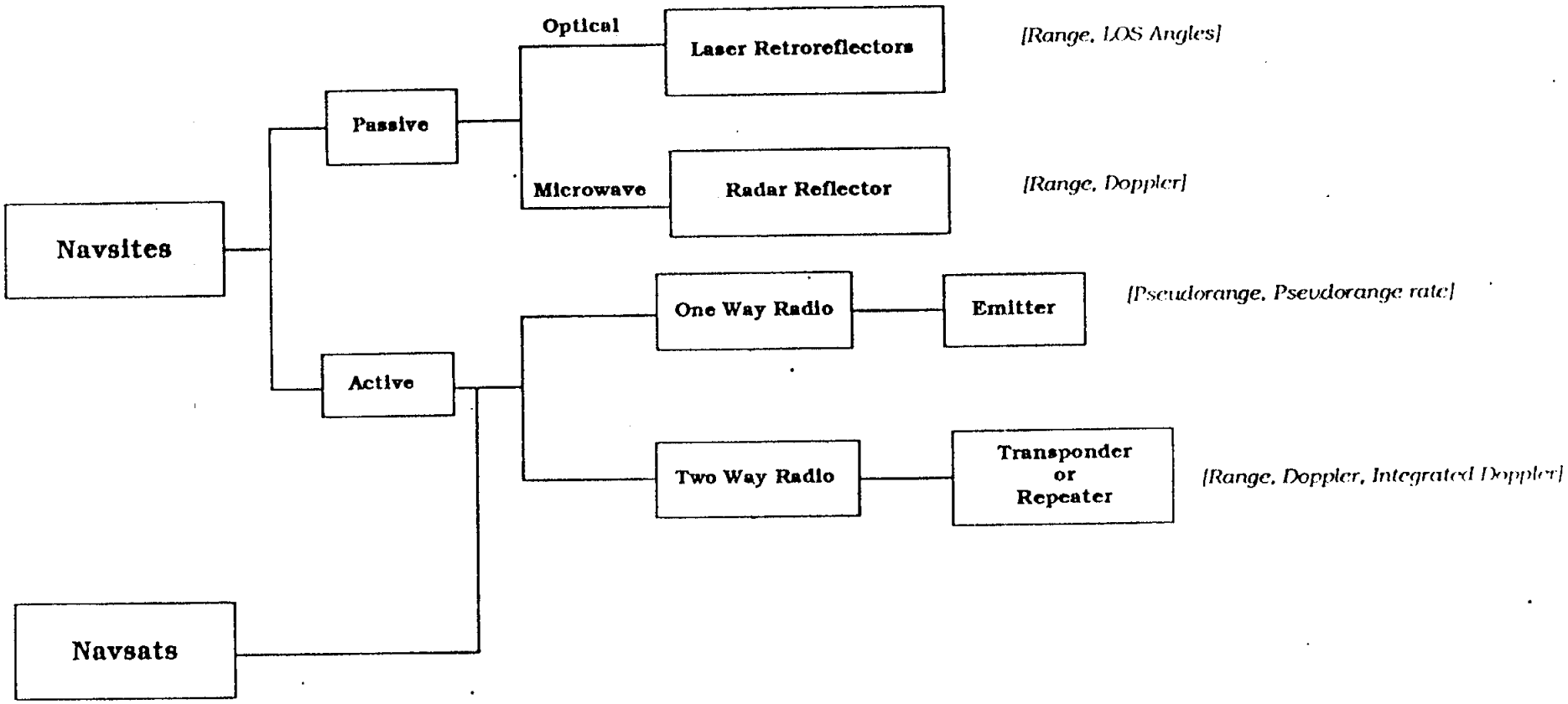
Attachment B

PRELIMINARY
OPTIONS/LOCAL-BASED NAVAIDS



NAVSITE OPTIONS

TYPE	SITE HARDWARE	USER HARDWARE	COVERAGE	EMPLACEMENT	SURVEY	ADVANTAGES	DISADVANTAGES
Laser Retro reflectors	Size Number of surfaces	Laser Transmitter & Receiver	Number Location	Hard landers Artemis Lander Rovers	Earth-Based (front side only) Survey Satellite LTS	- Passive devices - Low cost - May be able to survey from Earth - Fixed sites	- User laser pointing & tracking/stability/power - Incompatible with navsats - Emplacement/survey
Radar Reflector	Size Number of surfaces	Antenna Radar	Number Location	Hard landers Artemis Lander Rovers	Survey Satellite LTS	- Passive devices - Low cost - Fixed sites	- User radar power - Incompatible with navsats - Emplacement/survey
Emitter	Power Frequency Ref Clock Antenna size Antenna type Omni Directional	Antenna Receiver Frequency Ref Clock	Number Location	Hard landers Artemis Lander Cargo/Piloted Landers Rovers	DSN (front side only) Survey Satellite LTS	- Unlimited number of users - Fixed sites	- Requires power source (solar array & battery) - Frequency/time bias monitor/data transfer - Emplacement/survey - Life expectancy
Transponder or Repeater	Power Frequency Ref Antenna size Antenna type Omni Directional	Antenna Transmitter Receiver Frequency Ref Clock	Number Location	Hard landers Artemis Lander Cargo/Piloted Landers Rovers	DSN (front side only) Survey Satellite LTS	- No data transfer necessary - Fixed sites	- Requires power source (solar array & battery) - Noise - Emplacement/survey - Life expectancy

NAVSAT OPTIONS

TYPE	NAVSAT HARDWARE	USER HARDWARE	COVERAGE	MISSION	EPHEMERIS MAINTENANCE	ADVANTAGES	DISADVANTAGES
Emitter	Power Frequency Ref Clock Antenna size Antenna type Omni Directional	Antenna Receiver Frequency Ref Clock	Constellation size Orbit altitude/ inclination	Dedicated nav Shared with communications	DSN Navsites	- Unlimited number of users - Provides surface coverage	- Navsat mission cost - Orbit maintenence (fuel) - Ephemeris mainten ance - Frequency/time offset monitor/data transfer - Data link for ephemeris unless coded on signal
Transponder or Repeater	Power Frequency Antenna size Antenna type Omni Directional	Antenna Transmitter Receiver Frequency Ref Clock	Constellation size Orbit altitude/ inclination	Dedicated nav Shared with communications	DSN Navsites	- Simple devices - Provides surface coverage	- Navsat mission cost - Orbit maintenence (fuel) - Ephemeris mainten ance - Noise amplication - Data link for ephemeris unless coded on signal

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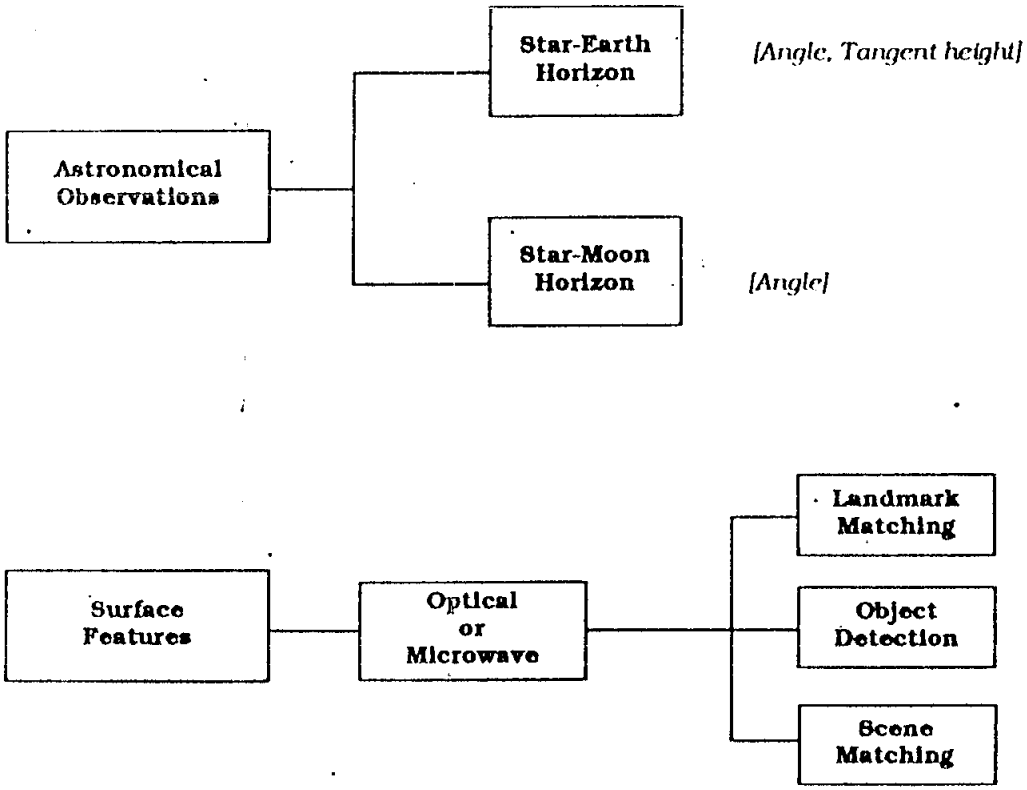
PRELIMINARY OPTIONS/ASTRONOMICAL & SURFACE



ASTRONOMICAL & SURFACE OPTIONS

TYPE	OBJECT	SECOND OBJECT	USER HARDWARE	ADVANTAGES	DISADVANTAGES
Star/ Earth horizon	Star	Earth horizon	Star sensor Horizon sensor	Autonomous	Vehicle attitude constraints
Star/ Earth horizon	Stellar Refraction		Star tracker Gyro	Autonomous	- Vehicle attitude constraints - New technology
Star/ Moon horizon	Star	Moon horizon	Star sensor Horizon sensor	Autonomous	- Vehicle attitude constraints - No lunar atmosphere
Landmark Matching	Surface Landmark	None	Optical (Passive or Active)/Microwave	- Autonomous - Provides surface-relative info	- Vehicle attitude constraints - Sun angle effects* - Landmark ident
Object Detection	Surface Area	None	Optical (Passive or Active)/Microwave	- Autonomous - Useful for onboard hazard avoidance	- Vehicle attitude constraints - Sun angle effects*
Scene Matching	Surface Area	None	Optical (Passive or Active)/Microwave	- Autonomous - Provides surface-relative info	- Sun angle effects* - Terrain variation - Terrain knowledge

* Passive optical only.

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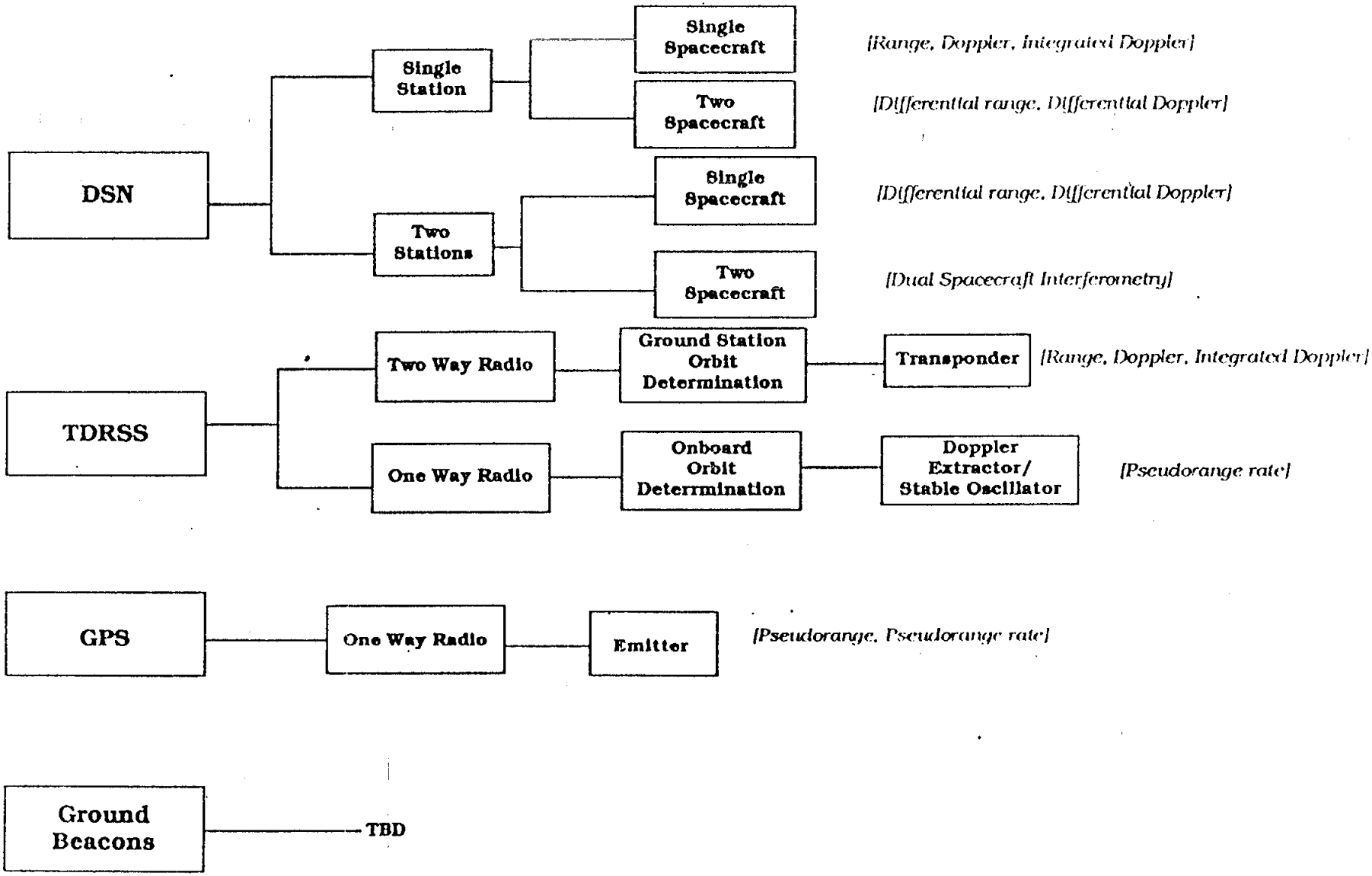
PRELIMIN...
OPTIONS/EARTH-BASED



EARTH-BASED OPTIONS

TYPE	GROUND EQUIPMENT	SPACE EQUIPMENT	USER HARDWARE	COVERAGE	RANGE/ FREQUENCY	ADVANTAGES	DISADVANTAGES
DSN	3 Antenna complexes located 120° apart in geodetic longitude/ JPL Orbit Determin	None	Antenna Transponder Data Link	Lunar front side only	TBD Minimum altitude/ S- & X-bands	- Provides data comm service - Proven technology	- Multi-use network - Saturable
TDRSS	WSGT Antennae/ Ground Transponders/ Flight Dyn Facility	TDRS Satellites (2-4 at GSO)	Antenna Transponder Data Link — — — — — Doppler Extractor/Osc/ Data Link	Limited by TDRS antenna gimbal angles ≥ 85% below 1200 km altitude	TBD Maximum altitude/ S- & Ku-bands	- Provides data comm service - Proven technology	- Multi-use network - Saturable — — — — — Undemonstrated technology
GPS	Monitor Stations Master Station	GPS Satellite Constellation (21 @ 20,000 km)	GPS Antenna/ Receiver/ Processor	Optimized for near earth users	TBD Maximum altitude/ L1 & L2-bands	- Non-saturable - Proven system for other applications	Rapid degradation of coverage/accuracy for backside GPS altitudes
Ground Beacons	Antenna Transponder	None	Antenna Transmitter Receiver	Limited by number/ location/range of beacons	TBD Maximum altitude/ TBD frequency	- Proven technology for other applications	Limited coverage

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FLO Navigation Architecture Concepts
Navigation Analyses Summary

EXPLORATION PROGRAMS OFFICE

1. Hoffman, David; "Surface Beacon Viewing Geometry From a Descending Spacecraft", JSC/SED, Jan 29,1992.
2. Munk, Michelle; "Lunar Surface Beacon Navigation Study - Summary of Results", JSC/SED, Feb 20, 1992.
3. Hoffman, David; "The Control of Periodic Trajectories about the Unstable Equilibrium Points of the Earth-Moon System - A Status Report), JSC/SED, Feb 21, 1992.
4. Cuthbert, Peter; "Lunar Approach with Lunar Navigation Satellite (5000 Km circ)", JSC/SED, March 12, 1992.
5. Kreimendahl, Frank; "Lunar Mission: Using GPS During Transfer", CSDL ESB-92-146, LMI-92-014, March 13,1992.
6. Brand, Tim; McClain, Wayne; "Preliminary Analysis of High Altitude Lunar Orbiter Lifetimes", CSDL ESB-92-150, LMI-92-012, March 16,1992.
7. Brand, Tim; Shepperd, Stan; "Habitat and Manned Landers Deorbit to Landing Navigation Study Updates", CSDL, March 19,1992.

SURFACE BEACON VIEWING GEOMETRY

- THE OBJECTIVE OF THIS QUICK, BACK OF THE ENVELOPE STUDY WAS TO DETERIMINE WHEN A SURFACE BEACON ON THE LUNAR SURFACE WOULD FIRST BECOME VISIBLE TO AN APPROACHING VEHICLE.

  ⇒ THE RESULT OF THIS STUDY WILL BE A DETERMINATION OF THE FRACTION OF THE DESCENT PROFILE THAT WOULD HAVE MEASUREMENTS RELATIVE TO A BEACON AT THE LANDING SITE.

- THE SURFACE RANGE (i.e. THE DOWNTRACK DISTANCE) FROM THE SPACECRAFT TO THE BEACON IS GIVEN BY THE FOLLOWING EQUATION:

$$s = -R\varepsilon + R\cos^{-1}\left[\left(\frac{R}{R+h}\right)\sin(\varepsilon + \tfrac{\pi}{2})\right]$$

Where,

R	—	The Lunar Radius
ε	—	The Minimum Horizon Clearance Angle
h	—	The Altitude of the Spacecraft
s	—	The Surface Range to the Beacon

- THE GEOMETRY OF THE APPROACH AND THE RESULTS OF THIS STUDY ARE SHOWN IN THE FOLLOWING ILLUSTRATIONS:

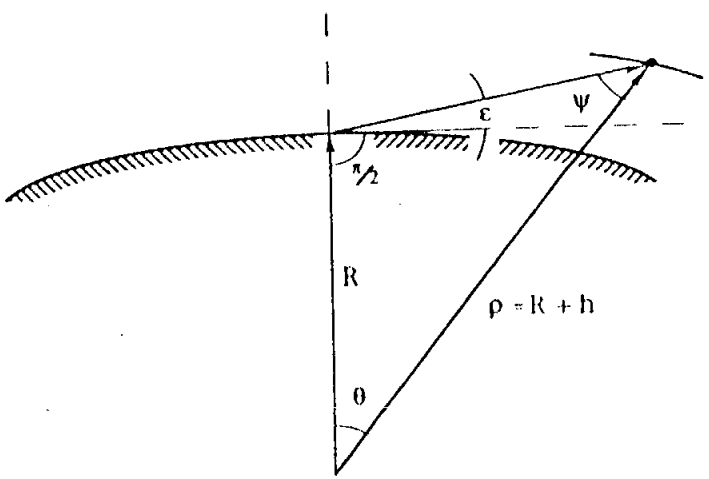
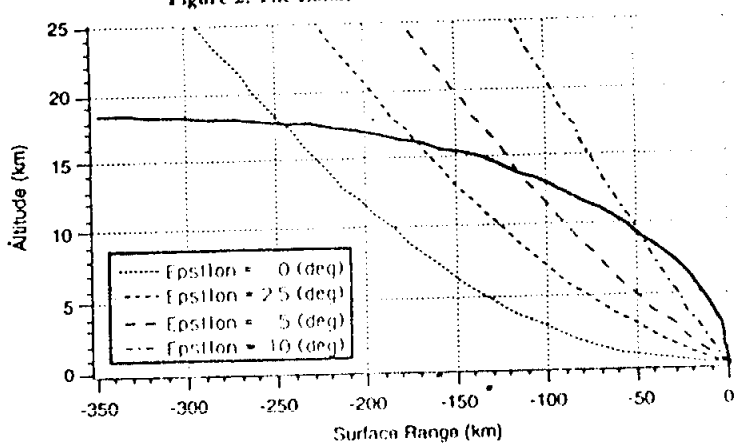


Figure 2: The Lunar Lander Descent Profile



APPROACH NAVIGATION USING NAVSITES: SUMMARY

- MISSION DESCRIPTION
  - -- Hyperbolic approach to the Moon
  - -- Extreme directions of asymptote derived from Earth/Moon transfer times and relative geometries
  - -- Simulation run from Sphere of Influence to LOI
  - -- Three navsites located on surface at (-90,72), (-90,-64) and (-27,0) (long,lat)

- INSTRUMENTATION
  - -- 2-way range and integrated Doppler to navsites (3 deg above horizon)

Results:

Approach Type	Filter RSS Errors at LOI Pos. (m)	Vel. (m/s)
DIRECT		
Low incl	170.8	.1406
45 deg incl	128.7	.1313
Polar	156.4	.1570
RETROGRADE		
Low incl	106.2	.1017
45 deg incl	135.6	.1305
Polar	170.4	.1625

Initial errors:
10 km position
.01 m/s velocity
50 m beacon location

delta t = 100 sec

Error growth bias:
(m^2/s^3)
1e-07 >500 km
5e-07 <500 km

- CONCLUSIONS
  - -- Three beacons are sufficient to navigate any approach to any landing site
  - -- Errors are reduced immediately (at SOI) so midcourse is supported

PARKING ORBIT NAVIGATION USING NAVSITES: SUMMARY

- MISSION DESCRIPTION
  - -- 100 km parking orbit, continued from the approach study trajectories
  - -- Unmanned and Manned missions (approach beacons only vs approach + site)

- INSTRUMENTATION
  - -- 2-way range and integrated Doppler to navsites (3 deg above horizon)

- INITIAL STATE ERRORS
  - -- Initial covariance matrix from perilune of corresponding approach

Results:

	Case	Beacon long, lat	Filter RSS Errors at first deorbit opportunity Pos. (m)	Vel. (m/s)
DIRECT UNMANNED	Low incl	-90,72 -90,-64 -27,0	880	.92
	45 deg incl	"	900	.86
	Polar	"	135	.32
DIRECT MANNED	Low incl	85.8,1.7 -27,0	350	.56
	45 deg incl	85.8,1.7	373	.71
	Polar	-90,72 -90,-64 63,0	185	.35
RETROGRADE MANNED	Low incl	85.8,1.7 -27,0	120	.23
	Polar	-90,72 -90,-64 63,0	125	.29

- CONCLUSIONS
  - -- It is sometimes preferable to go retrograde, depending on deorbit location
  - -- The availability of the three approach beacons varies for each orbit, but the maximum deorbit error for the cargo mission will be less than 1 rev of prop.
  - -- W ting to deorbit does not result in s nificant error reduction

THE CONTROL OF HALO ORBIT TRAJECTORIES: SUMMARY

- THE OBJECTIVE OF THIS STUDY WAS TO DETERMINE THE COST OF CONTROLLING A SATELLITE IN A HALO ORBIT WITH NAVIGATED STATE UNCERTAINTIES.

- THE METHOD USED WAS TO MODEL THE STATE UNCERTAINTIES AS ECRV's AND VARY THE TIME CONSTANT AND MAGNITUDES, AND TO USE LQR TO CONTROL THE SPACECRAFT.

- THE BASIC INDICATOR OF PERFORMANCE WAS TAKEN TO BE THE DELTA-V REQUIRED PER REV (WHICH HAS A PERIOD OF 14.68 DAYS).

- THIS STUDY INCLUDED TEN CASES. THE RESULTS OF THESE TEN RUNS ARE BEST SUMMARIZED IN THE FOLLOWING TABLE.

  ⇒ THE UNITS FOR THE TIME CONSTANT ARE MINUTES, ALL OTHER UNITS ARE IN METERS AND SECONDS. THE INTEGRATION TIME STEP FOR ALL CASES WAS 60 MINUTES.

Case	Time Constant	R_x	R_y	R_z	V_x	V_y	V_z	$\lvert\Delta \underline{V}\rvert$ Per Rev
NAV_05	600	1000	1000	1000	0.1	0.1	0.1	47.77
NAV_01	600	100	100	100	0.1	0.1	0.1	47.44
NAV_07	600	1000	1000	1000	0.01	0.01	0.01	6.65
NAV_03	600	100	100	100	0.01	0.01	0.01	5.01
NAV_06	60,000	1000	1000	1000	0.1	0.1	0.1	5.15
NAV_02	60,000	100	100	100	0.1	0.1	0.1	5.13
NAV_08	60,000	1000	1000	1000	0.01	0.01	0.01	1.34
NAV_04	60,000	100	100	100	0.01	0.01	0.01	1.26
NAV_10	60,000	1000	100	100	0.1	0.01	0.01	4.72
NAV_09	60,000	100	1000	1000	0.01	0.1	0.1	2.61

- THE IMPLICATION OF THIS DATA IS THAT THE PERFORMANCE IS THE MOST SENSITIVE TO THE ESTIMATE OF THE VELOCITY, PARTICULARLY ALONG THE EARTH-MOON AXIS.

NAVIGATION STUDY:
LUNAR APPROACH WITH NAVIGATION SATELLITE

- Mission description
  - Lunar vehicle: hyperbolic approach, inc=1.8, arg-periapse=60, long-asc-nod=14 (deg), V_∞=832 m/sec
  - Navsat: range of orbits tested, 1000, 5000 km circular orbits, polar and parallel to V_∞, polar and perpendicular to V_∞, navsat and lunar vehicle co-planar
- Measurements
  - Two-way doppler measurements of range and range rate
  - Range error: 4.0 meters Range rate error: 0.1 meter/sec (1sigma)
- Initial state errors (Simulation runs from SOI to LOI)
  - Lunar vehicle: diagonal covariance, errors at SOI 10 km in x, y, z; 0.01 m/sec in x, y, z
  - Navsat: typical DSN errors and correlations
- Conclusions
  - Polar orbits: filter works best (260 m, 0.40 m/sec final Lunar veh. errors)
  - Co-planar orbits: (1100 m, 2.00 m/sec final Lunar veh. errors) Out of plane errors large
  - Filter works best for 5000 km orbits (more measurement visibility)
  - Lunar vehicle velocity error estimates rise, position error estimates fall near LOI

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NAVIGATION ANALYSES SUMMARY
GPS for Earth/Moon Transfer Study

EXPLORATION PROGRAMS OFFICE

- GPS satellite transmissions are directed towards the Earth
- However, some spillover around the Earth allows occasional GPS satellite acquisition even when outside the GPS system shell
- For typical Earth/Moon transfer trajectories:
  - Craft spends only 1 hr inside the GPS shell
  - However, with a 9 dB gain horn antenna, the GPS system may be used within approximately 17 hrs of Earth (inbound or outbound)
- As might be expected radial error become highly correlated with clock bias error. Thus, the quality of the onboard clock becomes an issue.
- Nevertheless, this technique appears useful for:
  - Estimating state errors resulting from large lunar transfer orbit insertion burns
  - Estimating state errors on the return trajectory early enough to perform the final trajectory correction maneuver without major fuel penalties. It appears that perigee altitude errors of less than 1 km are possible several hours before reentry.

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NAVIGATION ANALYSES SUMMARY
Comm Orbiter Lifetime Study

EXPLORATION PROGRAMS OFFICE

- High altitude lunar orbiters are loosely held by the Moon
- Earth and Sun gravitation perturbations are significant
- However, stable orbits do exist at 5000 and 10,000 km altitudes
  - Stable is defined as 5 years without maneuvers to maintain orbit eccentricity
  - Trajectories tend to be stable for near-equatorial orbits (posigrade or retrograde)
  - Trajectories tend to be unstable for near-polar orbits (60 deg < inclination < 120 deg)
- If near-polar orbits are needed, fuel costs for orbit maintenance are modest. The following table shows approximate fuel costs (m/s) over a 10 year lifetime:

ALTITUDE (km)	ORBIT INCLINATION		
	60 deg	90 deg	120 deg
5000	25	45	35
10000	49	147	95

- Please note that this study is preliminary

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NAVIGATION ANALYSES SUMMARY
Habitat Lander Navigation Study

EXPLORATION PROGRAMS OFFICE

- Mission description
  - Hohmann deorbit from 100 km circular orbit
  - 5 min powered descent from 18 km altitude
- Baseline instrumentation
  - IMU (0.015 deg/hr (1s), 50 μg (1s))
  - Radar altimeter, and ground speed Doppler radar
- Initial state errors
  - Result from DSN update and 1 orbit coast
  - At deorbit: 667, 133, 136 m (DR, CT, VER) (1s)
- Landing-site-relative state errors at landing (1s)
  - Position errors: 437, 154, 0.55 m (DR, CT, ALT)
  - Velocity errors: 0.15, 0.27, 0.06 m/s (DR, CT, ALT)
- Conclusions
  - Performance largely driven by gravity uncertainties
  - Altimeter is essential for terminal altitude estimation
  - Doppler radar may be needed for terminal crosstrack velocity

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NAVIGATION ANALYSES SUMMARY
Manned Lander Navigation Study

EXPLORATION PROGRAMS OFFICE

- Mission description
  - Hohmann deorbit from 100 km circular orbit
  - 5 min powered descent from 18 km altitude
  - Habitat is 1 km crosstrack (optimal) wrt landing site
- Baseline instrumentation
  - IMU (0.015 deg/hr (1s), 50 μg (1s))
  - Radar altimeter, and ground speed Doppler radar.
  - 2-way range measurements to habitat (10 deg above horizon)
- Initial state errors
  - Result from DSN update and 1 orbit coast
  - At deorbit: 667, 133, 136 m (DR, CT, VER) (1s)
- Habitat-relative (horiz) and surface-relative (alt) at landing (1s)
  - Position errors: 5, 5, 0.55 m (DR, CT, ALT)
  - Velocity errors: 0.05, 0.10, 0.06 m/s (DR, CT, ALT)
- Conclusions
  - Radar altimeter is required because habitat not at landing site
  - Doppler radar may not be required depending on location of habitat wrt landing site

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FLO Navigation Architecture Concepts
Navigation Analyses (to 4/2/92)

EXPLORATION PROGRAMS OFFICE

1. Bordano A. , et.al., "Lunar Navigation Infrastructure (Concepts and Issues)", Presentation to Jay Greene, Jan 15, 1992.
2. Brand, Tim; Shepperd, Stan; "Lunar Navigation State Uncertainty Growth Rates Resulting from Higher Order Gravity Effects", CSDL, Jan 15, 1992.
3. Hoffman, David; "Surface Beacon Viewing Geometry From a Descending Spacecraft", JSC/SED, Jan 29,1992.
4. McClain, Wayne; Brand, Tim; "Impact of lunar Gravity Knowledge on Lunar Missions", CSDL ESB-92-050 LMI-92-004, January 31,1992.
5. Munk, Michelle; "Lunar Surface Beacon Navigation Study - Summary of Results", JSC/SED, Feb 20, 1992.
6. Hoffman, David; "The Control of Periodic Trajectories about the Unstable Equilibrium Points of the Earth-Moon System - A Status Report", JSC/SED, Feb 21, 1992.
7. Cuthbert, Peter; "Lunar Approach with Lunar Navigation Satellite (5000 Km circ)", JSC/SED, March 12, 1992.
8. Kreimendahl, Frank; "Lunar Mission: Using GPS During Transfer", CSDL ESB-92-146, LMI-92-014, March 13,1992.
9. Brand, Tim; McClain, Wayne; "Preliminary Analysis of High Altitude Lunar Orbiter Lifetimes", CSDL ESB-92-150, LMI-92-012, March 16,1992.
10. Brand, Tim; Shepperd, Stan; "Habitat Lander Deorbit to Landing Navigation Study", CSDL ESB-92-149, LMI-92-013, March 17, 1992.
11. Brand, Tim; Shepperd, Stan; "Manned Lander Deorbit to Landing Navigation Study - Preliminary", CSDL LMI-92-015, March 17, 1992.
12. Brand, Tim; Shepperd, Stan; "Habitat and Manned Landers Deorbit to Landing Navigation Study Updates", CSDL, March 19,1992.
13. Vytal, John, "Impact Resistance of Electronic Hardware for Lunar Drop", CSDL ETB-92-145, March 26, 1992.
14. Brand, Tim; Shepperd, Stan; "Habitat and Manned Landers Deorbit to Landing Navigation Study (Revision 1)", CSDL ESB-92-149/rev1, LMI-92-013/rev1, March 27, 1992.

LUNAR NAVIGATION INFRASTRUCTURE

(CONCEPTS AND ISSUES)

A. BORDANO, D. LONG, T. BRAND (CSDL)

J. BLUCKER, B. COCKRELL, M. TIGGES

JANUARY 15, 1992

OUTLINE

- APOLLO NAVIGATION
- LUNAR INFRASTRUCTURE DRIVERS
- LESSONS LEARNED (FROM MARS INFRASTRUCTURE STUDY)
- LUNAR OPTIONS (DEPENDS ON DRIVERS)
- POTENTIAL STUDY ISSUES

APOLLO NAVIGATION

INFRASTRUCTURE

GROUND

STDN (three 26 meter and eleven 9 meter S-Band stations)
During cis-lunar flight at least 3 stations could view vehicle
Stations measured range and 2-Way and 3-Way doppler (Average Range Rate)
Data transmitted to JSC MCC and state resolved
Used for all mission phases (prior to major events and when permission criteria violated)

ONBOARD

CSM: (Ground prime except during rendezvous or loss of Comm)
- Cis-lunar - Manual tracking and angle measurements
- Orbit nav - Manual landmark/star tracking
- Rendezvous nav - VHF range, Optical line-of-sight

LEM: (Primary and backup systems)
- Descent nav - inertial plus landing radar
- Rendezvous nav - Rendezvous radar
- Surface nav - Rendezvous radar tracking of CSM

NOTE: Ground and Onboard Nav systems were often used simultaneously.
Ground had a high speed processor for ascent and descent solution uplinks.
- -- Used as tiebreaker between primary and abort guidance
- -- Provided capability for landing target update

APOLLO NAVIGATION

INFRASTRUCTURE (CONT)

DESCENT NAV:

- PROBLEM
  - -- Propagation errors of 6-10 km after 1-2 revs
  - -- Significant map errors w.r.t. inertial coord. frame
  - -- Only nav aid landing radar
- SOLUTION (APOLLO 12)
  - -- Establish propagation trends over 8 orbits
  - -- Track landing site 2 revs prior to landing
  - -- Uplink new nav state and landing target bias after deorbit
- COMMENTS
  - -- Removed ~ 2 km error on Apollo 12
  - -- Procedurally complicated
  - -- Unworkable for global landing

LUNAR INFRASTRUCTURE DRIVERS

- REQUIREMENTS
  - -- FRONTSIDE VS GLOBAL OPERATIONS
  - -- REQUIRED LANDING PRECISION
    - -- FIRST VS REVISIT
- COST TRADE-OFFS
  - -- GROUND OPERATIONS
  - -- EMPLACEMENT (GROUND VS LOCAL)
  - -- RECURRING
  - -- AUTONOMY

LESSONS LEARNED

FROM MARS INFRASTRUCTURE STUDY

A simple system could go a long way

A Combination surface radio beacons (navsites) and orbiting satellites (navsats) could be a very effective nav system for diverse operations (lunar approach, orbital maintenance, landing, rovers)

NAVSITES[1]

- -- Act as transponder for space vehicles, navsats, and rovers
- -- Permanent anchor for a nav system
- -- Must be surveyed accurately (inertial)
- -- Earth based inertial survey of a navsite can be very accurate
- -- Navsite near landing site aides precision landing (requires surface relative survey

[1] Low Power/Omni antenna

LESSONS LEARNED

FROM MARS INFRASTRUCTURE STUDY

NAVSATS[2]

- -- Two way range/Doppler link to a navsite(s) for ephemeris maintenance (all onboard)
  - * "Moderate" navsat altitude preferable coverage
    - -- Low altitude → poor surface coverage, large propagation errors
    - -- High altitude → slow dynamics and Earth gravity effects
  - * Plane inclination and altitude optimized (dedicated orbit probably constrained)
- -- Backside Comm issues
  - * Backside Comm orbiter
  - * Multiple Navsats

[2] Directional antennas

LUNAR SYSTEM CONCEPTS

- <u>UPDATED APOLLO NAV</u>

  -- Clock improvements have significantly enhanced range & Doppler measurements

  -- Interferometry available (overkill?)

  -- Adequate for Apollo-type missions

  -- Not adequate if total planet operations coverage required (function of "drivers")

  -- East side landings near limb might be a problem

  -- Cost?

LUNAR SYSTEM CONCEPTS

Navsite/Navsat Infrastructure

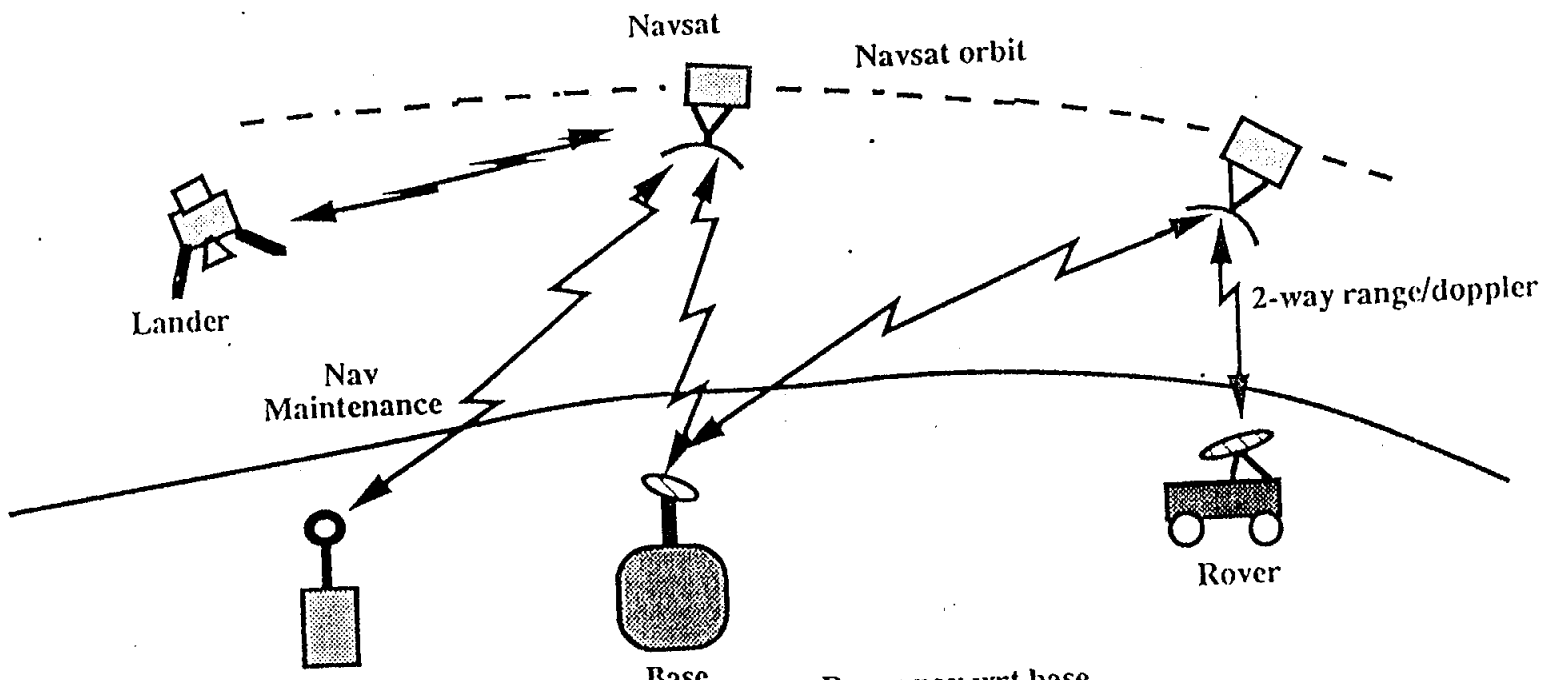
-- Provide global lunar surface access navigation infrastructure (2-4 navsats, initially 2 with up to 4 navsites)

-- Navsats provide autonomous user navigation for cis-lunar, trans-lunar, lunar base, descent, ascent, and rover.

-- No need for full "GPS Type" constellation, centralized ephemeris maintenance, simultaneous emitters for position triangulation and estimation of clock bias.

-- Cost/Reliability/Maintenance?

SURFACE & SCIENCE NAVIGATION



Rover nav wrt base
1) Dead reckoning (odometer, compass)
2) Periodic navsat updates (high precision wrt base)

POTENTIAL STUDY ISSUES

- Radiometry applied to lunar problem

} Ground Infrastructure

- Number, placement (geometry) of Navsites
- Number, altitude, and orbit of navsats
- Libration point nav (backside halo orbit comm link)
- Surface navigation concept

} Local Infrastructure

- Map tie (lunar surface cartography)
- Gravity uncertainty (backside anomalies

} Lunar Knowledge Base

LUNAR NAVIGATION

STATE UNCERTAINTY GROWTH RATES RESULTING FROM HIGHER ORDER GRAVITY EFFECTS

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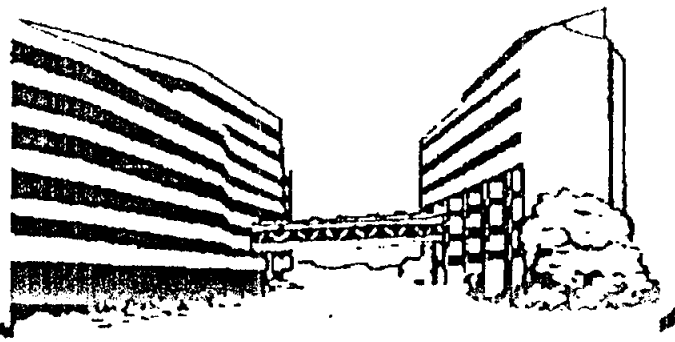
JANUARY 15, 1992

THE
CHARLES STARK DRAPER
LABORATORY, INC.



STUDY OVERVIEW

- Objective
  - For satellite in lunar orbit, find state uncertainty growth rates resulting from higher order gravity effects
- Assumptions
  - No navigation measurements
  - Spherical harmonic (16×16) gravity model*
  - Lunar orbits of varying altitude and inclination
- Method
  - Brute force Monte Carlo method
  - Assume 4×4 gravity model used onboard
  - Using many orbit (from different starting points around the moon), a statistical representation of expected state uncertainties resulting from neglect of higher order gravity is found

---

* Bill, B.G. and A.J. Ferrari, "A Harmonic Analysis of Lunar Gravity," Journal of Geophysical Research, Vol. 85, No. B2, Feb. 1980

DOWNRANGE UNCERTAINTY (M) AFTER 1 ORBIT

ALTITUDE (km) \ INCLINATION (deg)	0 deg	10 deg	20 deg	30 deg
50	975	1495	1328	1551
100	835	1227	1131	1317
150	720	1020	972	1128
200	625	856	841	973

VERTICAL UNCERTAINTY (M) AFTER 1 ORBIT

ALTITUDE (km)	INCLINATION (deg) 0 deg	10 deg	20 deg	30 deg
50	191	338	260	258
100	183	282	209	198
150	173	239	171	157
200	162	206	144	128

CROSSTRACK UNCERTAINTY (M) AFTER 1 ORBIT

ALTITUDE (km)	INCLINATION (deg			
	0 deg	10 deg	20 deg	30 deg
50	415	241	327	303
100	321	193	258	235
150	251	157	207	185
200	199	130	168	149

CONCLUSIONS

- Downrange uncertainties of 1 km or more per orbit (roughly 2 hours) are to be expected

- Equivalent numbers at Earth and Mars are
  - — Earth 200 m/orbit (approximately)
  - — Mars 600 m/orbit (approximately)

- Obviously, smaller bodies tend to less spherical and, thus, have more gravitational "noise"

- Caveat
  - — This may not be best way to model lunar gravity
  - — Significant mass concentrations (mascons) exist
  - — Accounting for these analytically would result in a spherical harmonic expansion model that falls off much more quickly (in other words, higher order gravity would have much less power)
  - — A complete evaluation of error growth must include the effects of uncertainty in the harmonic expansion coefficients

National Aeronautics and
Space Administration

Lyndon B. Johnson Space Center
Houston, Texas
77058

NASA

y to Attn of: ET4-92-04

January 29, 1992

TO: ET4/Chief, Performance Analysis Branch

FROM: ET4/Performance Analysis Branch

SUBJECT: Surface Beacon Viewing Geometry From a Descending Spacecraft

The enclosed material provides a discussion on the visibility of a beacon on the lunar surface as seen from an approaching spacecraft. This discussion focuses on the case where the spacecraft is making a powered descent to a landing site near the beacon. The first part of the enclosure develops an equation which relates the surface range at which the beacon becomes visible to the spacecraft's altitude. Then this equation is used to derive curves which can be laid over the trajectory profile (altitude vs. surface range) to show where the spacecraft could acquire the beacon's signal. Finally, a profile similar to the current manned lunar lander descent profile is used as an example to show that only a portion of the powered flight descent will be made while the beacon is visible.

David A. Hoffman

Enclosures

cc:
ET/C. A. Graves
G. A. Zupp
S. W. Wilson
B. F. Cockrell
ET2/C. Teixeira
J. P. Masciarelli
ET3/J. T. McNeely
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D. Weaver

CSDL/G. Barton
T. Brand

U.S. Gov't

The purpose of this note is to determine the visibility of a beacon on the lunar surface as seen from an approaching spacecraft. We are primarily interested case where the spacecraft is making a powered descent to a landing site near the beacon. In actual practice the spacecraft will be unable to acquire the signal until it is a few degrees above the horizon (*i.e.*, until its zenith angle is less than 90°) because of the uneven lunar surface. The point at which the beacon first becomes visible is primarily a function of the spacecraft's altitude, h, and surface range, s, from the beacon. When the spacecraft reaches the point where it can first acquire the beacon, we shall say that it has entered the cone of visibility. What we really want to determine is the fraction of the spacecraft's descent trajectory that lies inside the cone of visibility.

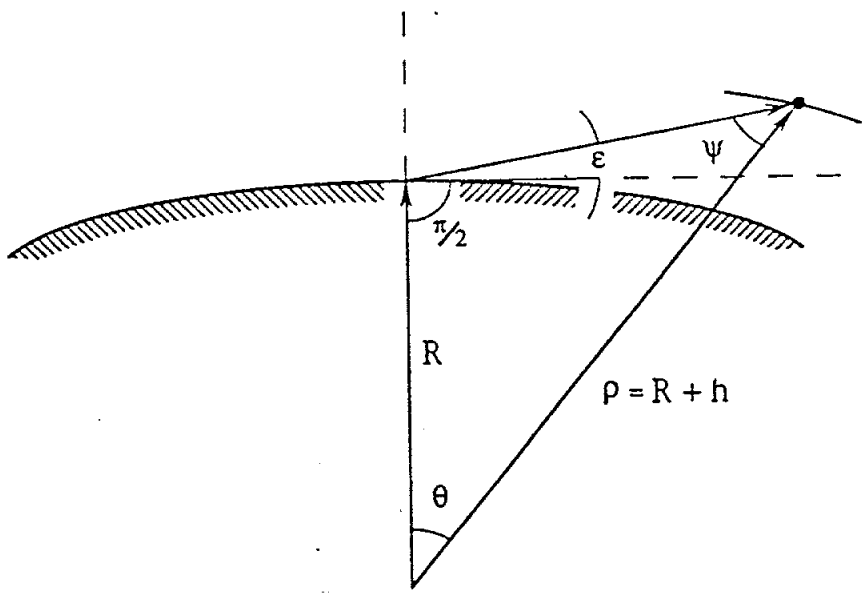


Figure 1: The Viewing Geometery.

Our first order of business will be to find the surface range at which the beacon will become visible for a given altitude. The geometry of the situation is shown in figure 1. This figure shows the relationship between the spacecraft and the beacon in a plane which includes the beacon, spacecraft, and the center of the moon. The surface range is given in terms of the lunar radius, R, as $s = R\theta$. The angle θ can be found from simple plane geometry. We have a plane triangle with one known interior angle and two known sides. We know that the sum of the interior angles must equal 180° so that

$$\psi = \frac{\pi}{2} - \varepsilon - \theta.$$

The law of sines states that

$$\frac{R}{sin\,\psi} = \frac{R + h}{sin(\varepsilon + \frac{\pi}{2})}.$$

Therefore,

$$sin\,\psi = \left(\frac{R}{R + h}\right) sin(\varepsilon + \frac{\pi}{2}).$$

With the aid of the following two trigonometric identities,

$$sin(-\alpha) = -sin(\alpha)$$

$$-cos(\alpha) = sin(\alpha - \frac{\pi}{2})$$

we can rewrite the previous equation as

$$cos(\theta + \varepsilon) = \left(\frac{R}{R + h}\right) sin(\varepsilon + \frac{\pi}{2})$$

so that

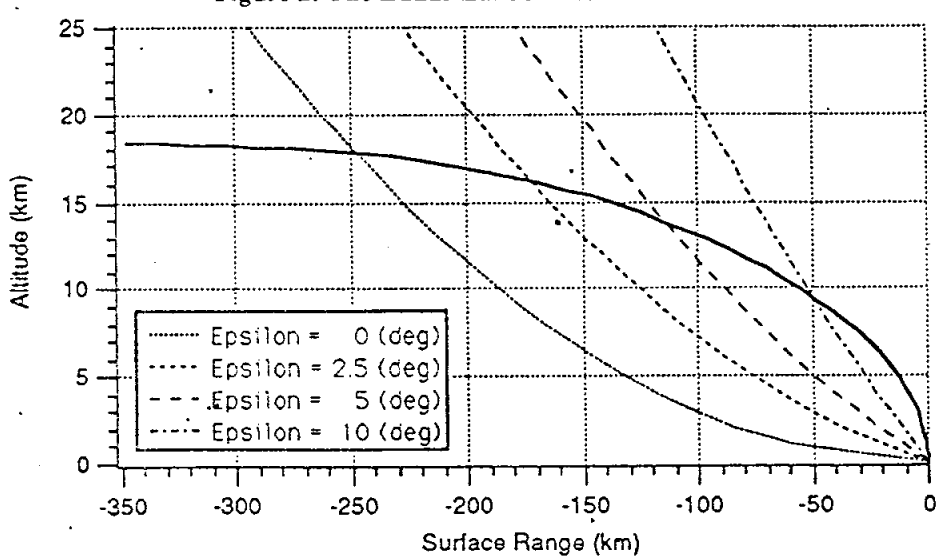
$$\theta = -\varepsilon + cos^{-1}\left[\left(\frac{R}{R + h}\right) sin(\varepsilon + \frac{\pi}{2})\right].$$

This equation was used to generate the data in the following table:

Altitude (km)	Surface Range (km) (ε = 0°)	(ε = 2.5°)	(ε = 5°)	(ε = 10°)
35	345.9	278.1	225.6	155.4
30	320.6	253.5	202.6	136.8
25	293.0	226.7	177.9	117.4
20	262.4	197.2	151.1	96.9
15	227.5	163.9	121.5	75.1
10	186.0	125.0	88.1	52.0
8	166.4	107.0	73.3	42.2
6	144.2	87.1	57.5	32.2
4	117.8	64.2	40.3	21.8
2	83.3	36.8	21.3	11.3

Now that we can determine the surface range at which the approaching spacecraft can acquire the beacon's signal, we can also determine the fraction of the descent trajectory which lies inside the cone of visibility. Figure 2 shows a typical descent trajectory for a lunar lander. This graph also shows the boundaries for three different cones of visibility, which correspond to three different elevation angles, as dashed lines. The points on the graph to the right of any of these lines will be within that particular cone of visibility.

Figure 2: The Lunar Lander Descent Profile



U.S. Gov't

The lunar lander typically follows a shallow descent profile because that is the fuel optimal approach. As a consequence only the latter part of the descent profile will be within the cone of visibility. The approach trajectory could be steepened so that more of the trajectory lies within the cone of visibility, at the expense of propellent consumption, but it appears to be unlikely that the entire powered descent profile can be brought inside it because of the steepness of the its boundary.

The fact that only the latter part of the descent trajectory lies within the cone of visibility may not be a serious problem. The orbit determination system should be able to estimate the incilination of the spacecraft's parking orbit accurately. This means that the crosstrack errors of the approaching spacecraft's navigation state should be relatively small. If the spacecraft carries a radar altimeter then the radial navigation error should also be fairly small (though it could still be significant). The principle error that would have to be removed by the surface beacon would be the downtrack error, and the first few range measurements should be able to reduce this error significantly.

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TO: D. Long (JSC), J. Blucker (JSC), G. Barton (Draper)

ESB-92-050
LMI-92-004

FROM: T.J. Brand

SUBJECT: Lunar Gravity Knowlege Requirements

DATE: January 31, 1992

CC: M. Johnston, W. McClain, J. Miller, J. Negro, G. Schmidt, N. Sears
JSC: A. Bordano, B. Cockrell, E. Jiongo, M. Tigges

At the January 15 meeting with Jay Greene, he asked for any input we could provide within two weeks on Lunar gravity knowledge requirements. The enclosed charts summarize out thoughts and recommendations. In a "data-poor" environment it is difficult to quantify the extent of the problem and its effect on various missions.

It should be mentioned that the method of characterizing the knowledge requirement must also be addressed. For example, specifying a requirement for a 30x30 harmonic model says nothing about the quality of the supporting data, the "goodness" of the fit, or the ability of this type of model to describe the field. A point mass model may be preferable. Similarly, characterizing the requirement in terms of error growth per rev may be useful for autonomous navigation studies but it does not address the needs of orbit lifetime studies.

Specific recommendations regarding the quality of the required gravity knowledge and alternate methods of obtaining that knowledge will require several months of study. If desired, we could draft a plan describing our approach to the overall problem.

/bb

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IMPACT OF LUNAR GRAVITY KNOWLEDGE ON LUNAR MISSIONS

W. McCLAIN
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(617) 258-2451

JANUARY 31, 1992

ESB-92-050
LMI-92-004

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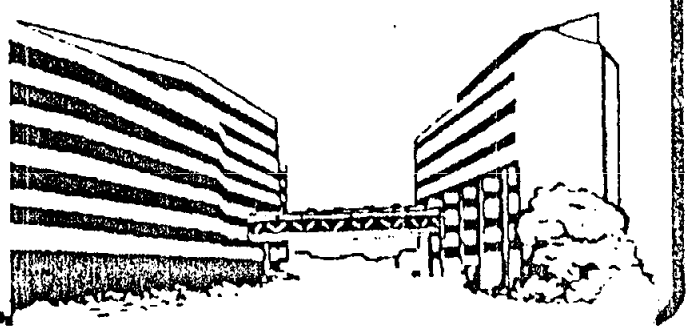

CURRENT STATE OF GRAVITY KNOWLEDGE

- Current knowledge is limited:
  - — A dozen or so gravity fields exist
  - — All based extensively on Apollo era data or earlier
    - Most are not independent (use same track data in part)
    - No farside tracking data available
  - — More recent models incorporate some laser ranging data of useful but limited value
  - — Large differences among performance of models (recent models considered better)
  - — No concensus on which recent model is best
    - No robust means for judging
    - New orbiter tracking data needed

GRAVITY MODELS (partial list)			
Sagitov et. al* (1986)	16 × 16	Ferrari (1975)	5 × 5
Tadzhidinov (1983)	16 × 16 plus masscons	Liu and Liang (1971)	15 × 8
Bills and Ferrari* (1980)	16 × 16	Akim and Vlasova	7 × 7
Ferrari (1977)	16 × 16		

*generally considered to be the better models

- Apollo anecdotal data suggest downtrack propagation errors due to gravity uncertainty of 6-10 km after 1-2 revs

POTENTIAL IMPACT OF GRAVITY KNOWLEDGE

- Mission objectives
  - Orbital mapping and science
  - Precision landing
  - Post flight trajectory reconstruction requirements

- Mission orbit design
  - Orbit lifetimes
  - Maneuver requirements (fuel budgets and maneuver schedules)

- Navigation performance
  - Determination of current state using local nav aids or ground-based (Earth) tracking
  - Forward prediction at spacecraft state (very sensitive to the gravity model accuracy)

IMPACT ON MISSION OBJECTIVES

- Mapping accuracy depends on accurate knowledge of the spacecraft state
  - — Primarily a postflight trajectory reconstruction requirement
  - — Combining imaging data with radiometric tracking may aid trajectory reconstruction
  - — Needs further study

- Science
  - — Internal structure and composition of moon inferred from gravity field

- Precision landing
  - — Poor gravity knowledge can be a key contributor to state vector inaccuracy
    - Inability to accurately propagate a state vector forward is a major concern
    - Local aids can alleviate this problem
  - — Apollo techniques (last minute tracking combined with state vectors and target updates) are not feasible at all sites
    - Not global solution
    - Operationally complex

IMPACT ON MISSION ORBIT

- Gravity field causes significant spacecraft altitude variations
  - Potentially ±40 km oscillation with 27 day period for a 100 km altitude orbiter
  - Longer term (secular) decay to impact

- Altitude variations caused predominantly by eccentricity variation
  - 27 day periodic variation in the eccentricity, due to lunar rotation
  - Longer term variations in the eccentricity, due to zonal harmonics, causes decrease in periapse altitude resulting in potential surface impact

- Accurate knowledge of gravity zonal harmonics necessary to establish possible existence of stable or near-stable mission orbits

- Improved gravity model yields improved knowledge of decay rate for unstable orbit
  - Orbit maneuver may be required to ensure sufficient lifetime
  - Frequency driven by rate of altitude decay
  - Fuel budget driven by lifetime requirement and decay rate

NAVIGATION PERFORMANCE

- Improved by several techniques
  - — Improved gravity model accuracy
  - — Navigation aid utilization
  - — Passive update techniques (optical)
  - — Ground-based tracking support
- Local navigation aids may provide viable alternative to improved gravity model
- Must be explored on a mission specific basis

GRAVITY KNOWLEDGE NEEDS/REQUIREMENTS

- Driven by:
  - Mapping and science orbit navigation and trajectory reconstruction needs
  - Orbit design (lifetime and maintenance costs)
  - Landing accuracy requirements and available nav aids

- Solutions
  - Improve knowledge
    1) Dedicated gravity mission
    2) Improvement over longer term using tracking data from other missions and local (backside) nav aids to enhance observability
  - Mitigation techniques
  - Robust navigation aids
    - Local infrastructure to (near) continuously observe and correct the onboard state vector
    - Limited aids, such as a radio transponder at the landing site

LOCAL NAVIGATION INFRASTRUCTURE CONCEPT

- Combination surface beacons (navsites) and navigation satellites (navsats)

- Navsites
  - Low power transponder (range and Doppler)
  - Distributed around front and backside of moon
    - Used to maintain accurate ephemeris on orbiting navigation satellites (and spacecraft)
    - Observation of spacecraft backside motion will aid gravity model determination
  - Emplaced at or near landing site
    - Alleviate need for accurate inertial state knowledge
    - Provide site-relative update 100-200 km uprange from landing
  - Navsite survey
    - Reference navsite (frontside) surveyed by ground based tracking
    - Other navsites surveyed relative to reference navsite by orbiter
    - Mars studies indicate this boostrap technique works very well

- Navsats
  - Moderate altitude orbit (≈5000 km)
  - Orbiter, landing craft, etc., use navsats as aid
  - Navsats use navsites to maintain ephemeris
  - Navsat can also function as a communications link (surface to Earth)
  - Number of navsats/navsites TBD (3 of each may be adequate for nav)

COMPENSATION TECHNIQUES

- Spacecraft overflies much the same territory on successive revs due to slow lunar rotation (for a 100 km orbit the spacecraft groundtrack moves 1 deg or 30 km per rev)
- May be possible to solve for unmodeled downtrack acceleration
- Requires beacon or navsat or Earth based tracking support
- Also assumes down-track error is secular and not due to possible periodic contributions by masscons

CONCLUSIONS

- Significant disagreement between gravity models
  - — Resolution needed on the extent of agreement between better models
  - — Also need a reliable measure of model uncertainty and its impact on navigation
- Improved gravity knowledge required for mission orbit stability and lifetime analyses
- Improved gravity may not be necessary for nav
  - — Requirements dependent
  - — Methods to alleviate problem for landing need study
- Emplacement of navigation aids may mitigate need for gravity improvement
- Performance and costs trades between gravity improvement mission and nav aid infrastructure deployment needed
- Results of imaging (mapping) orbiters might be combined with radiometric tracking (Earth based or local) to aid post-flight trajectory reconstruction
  - — Improved map accuracy (Mars Observer studies)
  - — Potential for improving gravity knowledge
- *SEVERAL MONTHS OF INTENSIVE STUDY ARE RECOMMENDED TO QUANTITATIVELY EVALUATE PROBLEMS AND SOLUTION OPTIONS*

LUNAR MISSION:
USING GPS DURING TRANSFER

Frank Kreimendahl
March 13, 1992
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LMI-92-014

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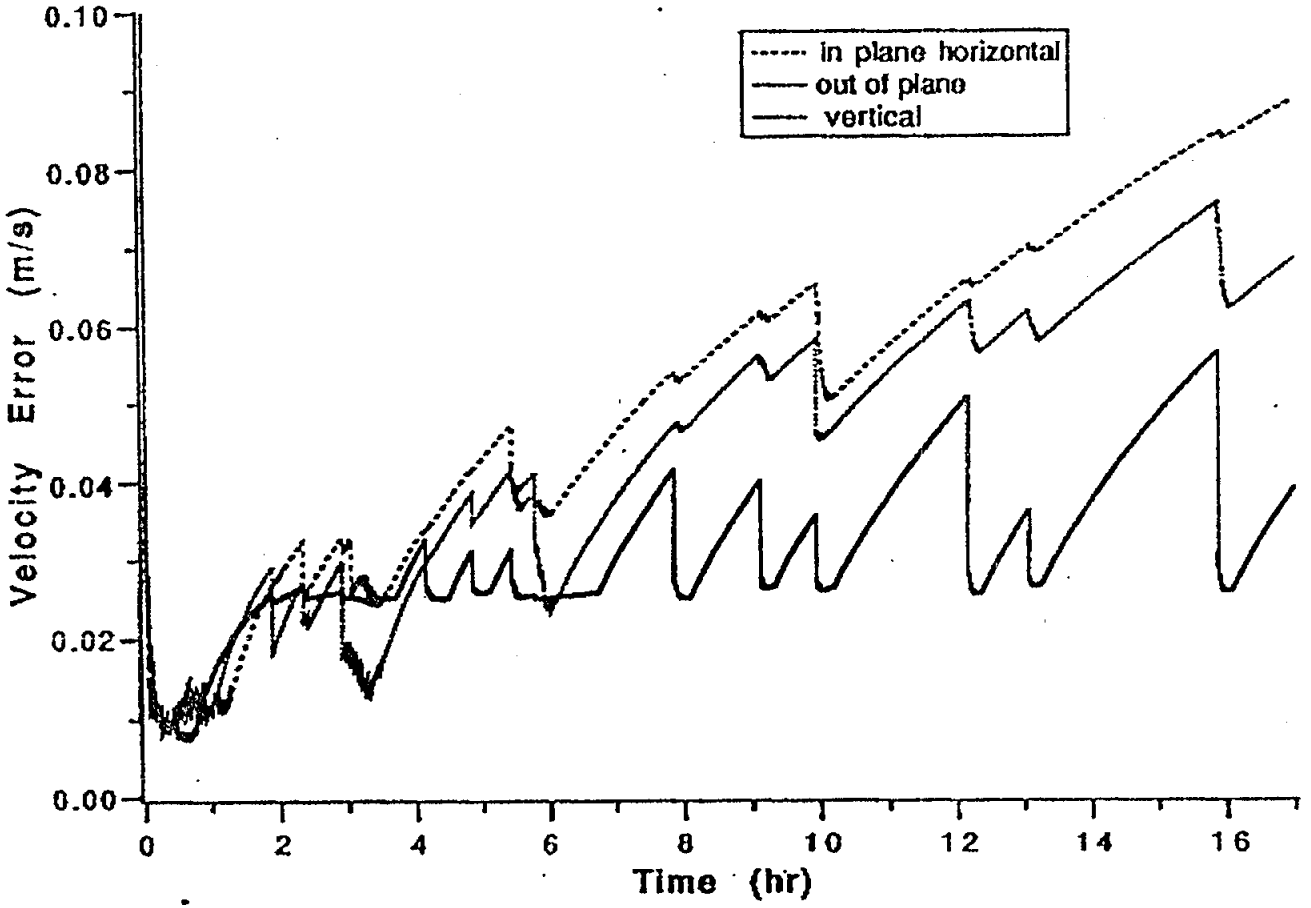
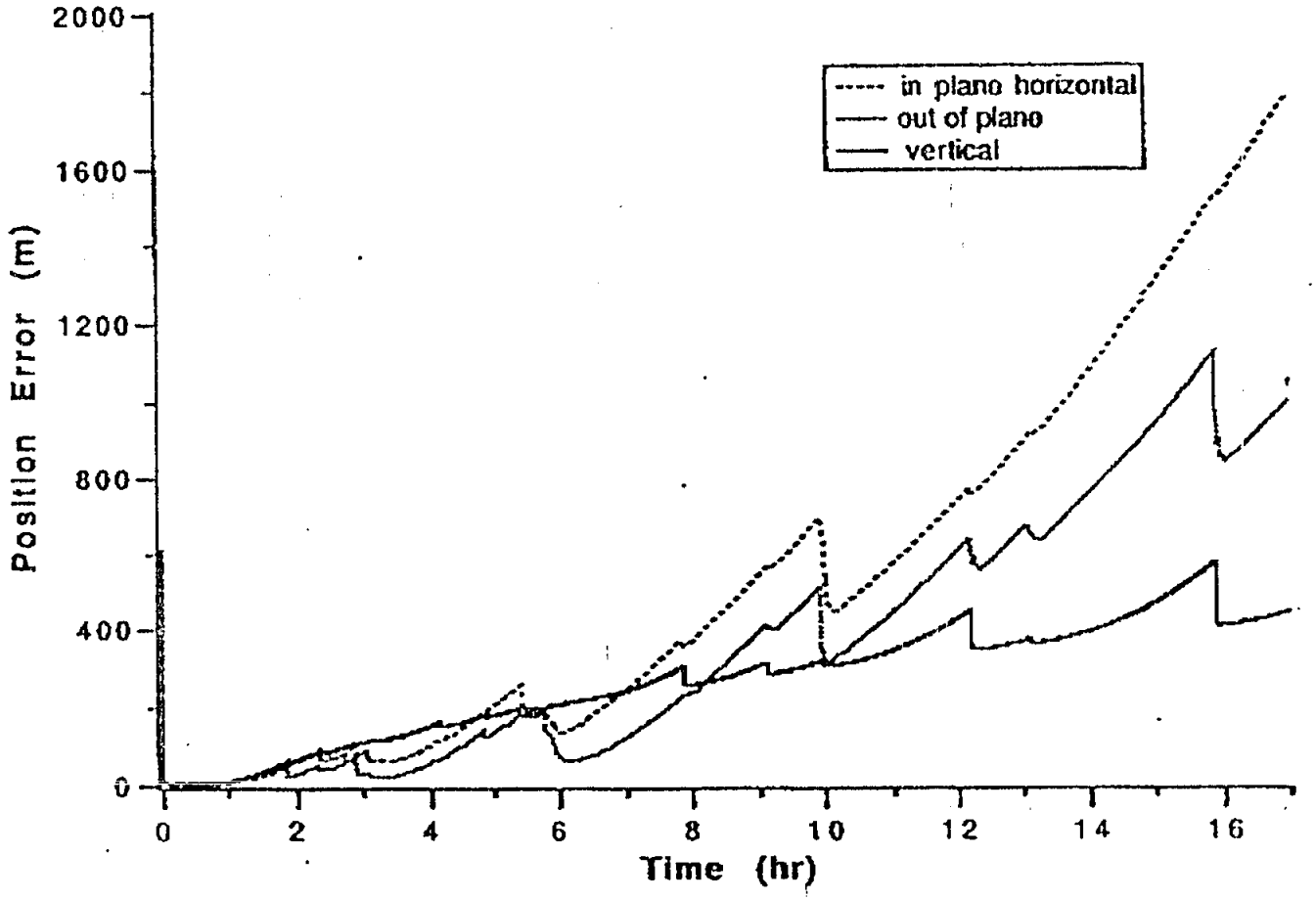
RMS VELOCITY ERRORS

0 hr Offset Time, 0.03 m/s/√hr Velocity Random-walk



RMS POSITION ERRORS

0 hr Offset Time, 0.03 m/s/√hr Velocity Random-walk



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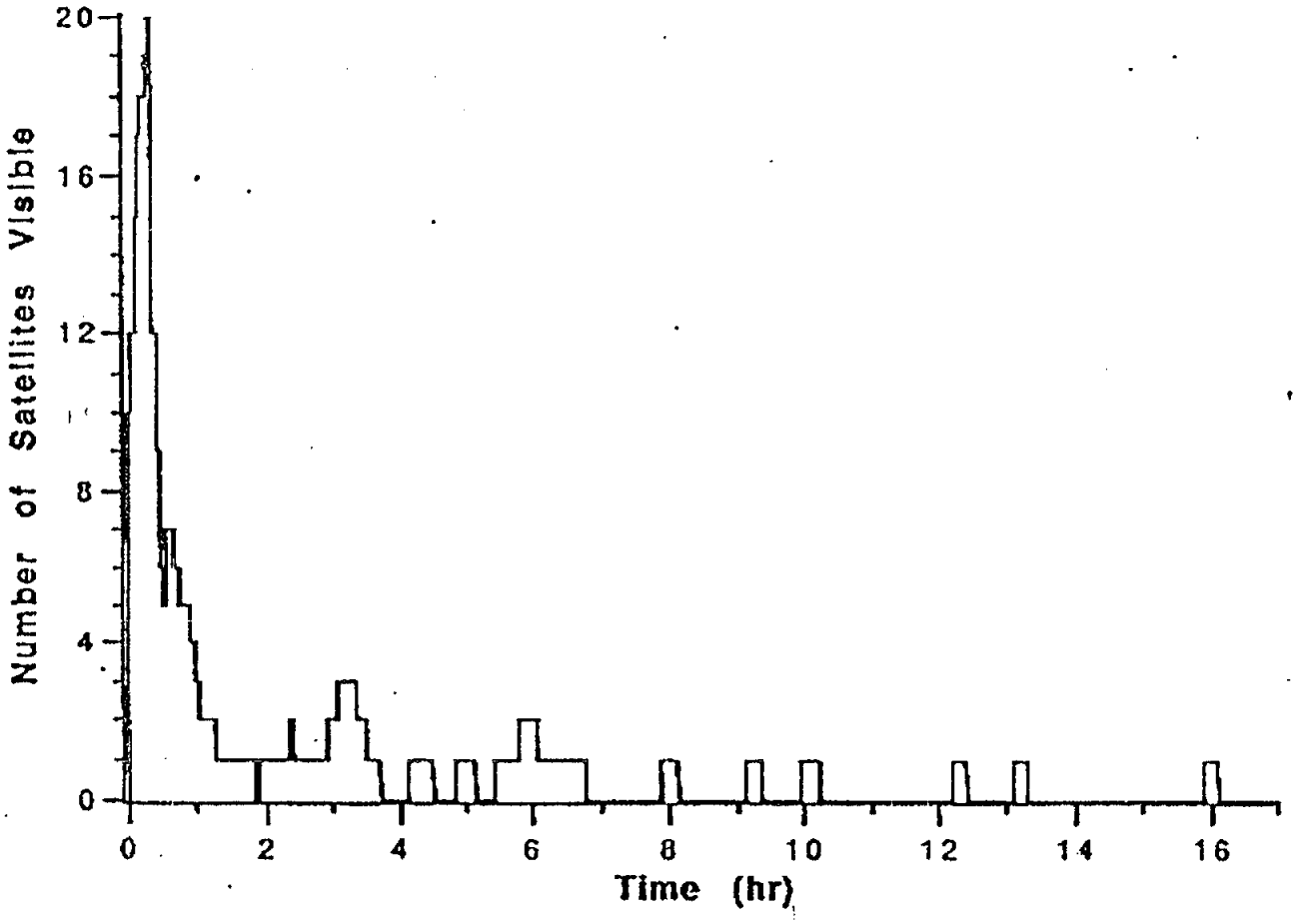
NUMBER OF GPS SATELLITES VISIBLE

Transfer to Moon, 0 Hour Offset Time (Better)



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NAVIGATION SIMULATION (cont'd)

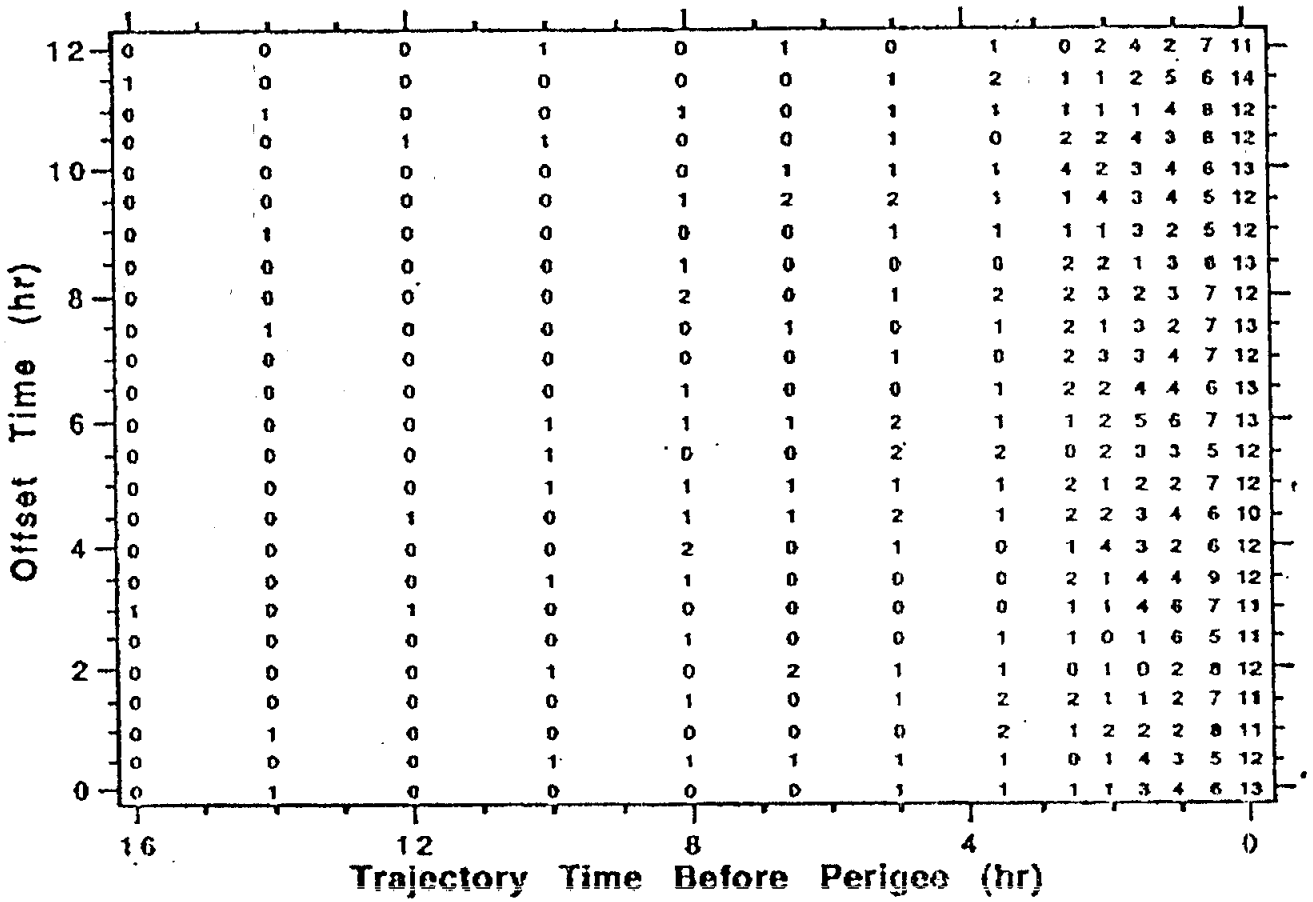
- GPS measurement errors(1σ): 4.6 m range bias, 3 m range noise, 0.015 m/s range-rate bias, 0.03 range-rate noise
- Random acceleration noise, causing velocity random-walk errors during the long coasting period, is used to account for possible venting, uncoupled RCS firings, and other disturbances
- Two different velocity random-walk values are used here; 0.03 m/s/$\sqrt{hr}$ for a smaller unmanned vehicle, 0.003m/s/$\sqrt{hr}$ for a larger manned vehicle with smaller and better balanced attitude control jets
- Results for two sample trajectories are shown, i.e. two different offset times, corresponding to better and worse GPS coverage near the earth (0 hr and 9.5 hr offset times from Page 7)

NAVIGATION SIMULATION - TRANSFER TO MOON

- Covariance analysis simulation with Kalman filter to process GPS range and range-rate measurements
- Navigation filter errors states include position, velocity, vehicle clock bias and drift, GPS range and range-rate biases (not estimated)
- GPS acquisition is assumed right after the transfer burn, i.e. at the start of the simulation of transfer to the moon
- The first 17 hours of the transfer trajectory to the moon are shown. This is about the time when the GPS signal becomes unavailable
- Initial rms position and velocity errors after the transfer burn are taken as 600 m and 1.6 m/s per axis (assuming about 3100m/s burn delta-v and 100 arcsec IMU alignment errors). These values are quickly reduced after the first GPS update due to good GPS coverage at low altitudes and are therefore not important to following navigation errors

NUMBER OF GPS SATELLITES AVAILABLE

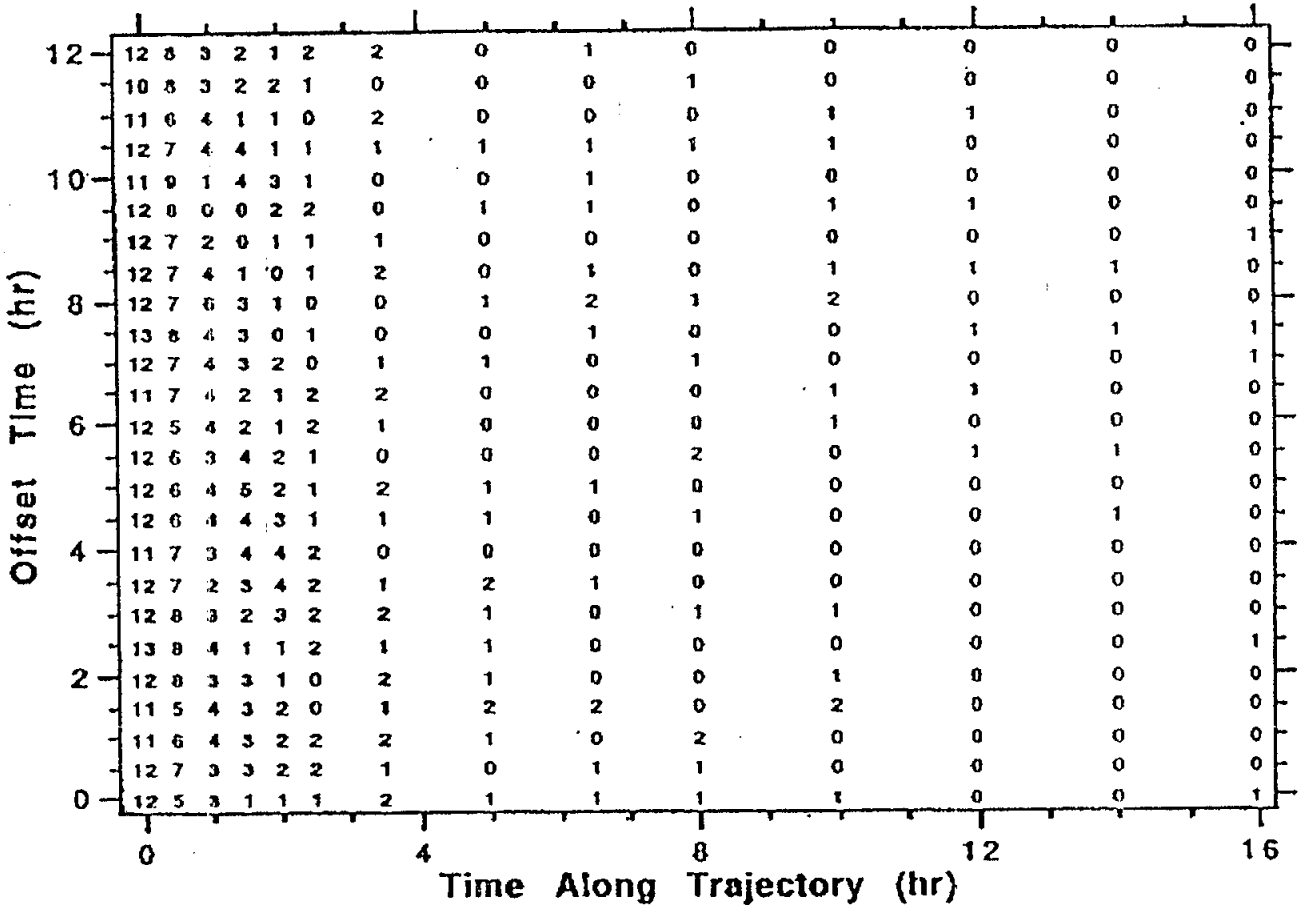
Transfer from Moon



Notes:
- Each row is for a particular trajectory with a different start (offset) time
- 24 satellite GPS constellation, L1-P code signal used
- 9dB antenna assumed on vehicle
- Vehicle goes below GPS altitude at about 1 hr out
- GPS signal distance limit reached at about 17 hr out

NUMBER OF GPS SATELLITES AVAILABLE

Transfer to Moon



12	8	3	2	1	2	2	0	1	0	0	0	0	0
10	8	3	2	2	1	0	0	0	1	0	0	0	0
11	6	4	1	1	0	2	0	0	0	1	1	0	0
12	7	4	4	1	1	1	1	1	1	1	0	0	0
11	9	1	4	3	1	0	0	1	0	0	0	0	0
12	8	0	0	2	2	0	1	1	0	1	1	0	0
12	7	2	0	1	1	1	0	0	0	0	0	0	1
12	7	4	1	0	1	2	0	1	0	1	1	1	0
12	7	6	3	1	0	0	1	2	1	2	0	0	0
13	8	4	3	0	1	0	0	1	0	0	1	1	1
12	7	4	3	2	0	1	1	0	1	0	0	0	1
11	7	4	2	1	2	2	0	0	0	1	1	0	0
12	5	4	2	1	2	1	0	0	0	1	0	0	0
12	6	3	4	2	1	0	0	0	2	0	1	1	0
12	6	4	5	2	1	2	1	1	0	0	0	0	0
12	6	4	4	3	1	1	1	0	1	0	0	1	0
11	7	3	4	4	2	0	0	0	0	0	0	0	0
12	7	2	3	4	2	1	2	1	0	0	0	0	0
12	8	3	2	3	2	2	1	0	1	1	0	0	0
13	8	4	1	1	2	1	1	0	0	0	0	0	1
12	8	3	3	1	0	2	1	0	0	1	0	0	0
11	5	4	3	2	0	1	2	2	0	2	0	0	0
11	6	4	3	2	2	2	1	0	2	0	0	0	0
12	7	3	3	2	2	1	0	1	1	0	0	0	0
12	5	3	1	1	1	2	1	1	1	1	0	0	1

Notes:
- Each row is for a particular trajectory with a different start (offset) time
- 24 satellite GPS constellation, L1-P code signal used
- 9dB antenna assumed on vehicle
- Vehicle reaches GPS altitude after about 1 hr
- GPS signal distance limit reached after about 17 hr

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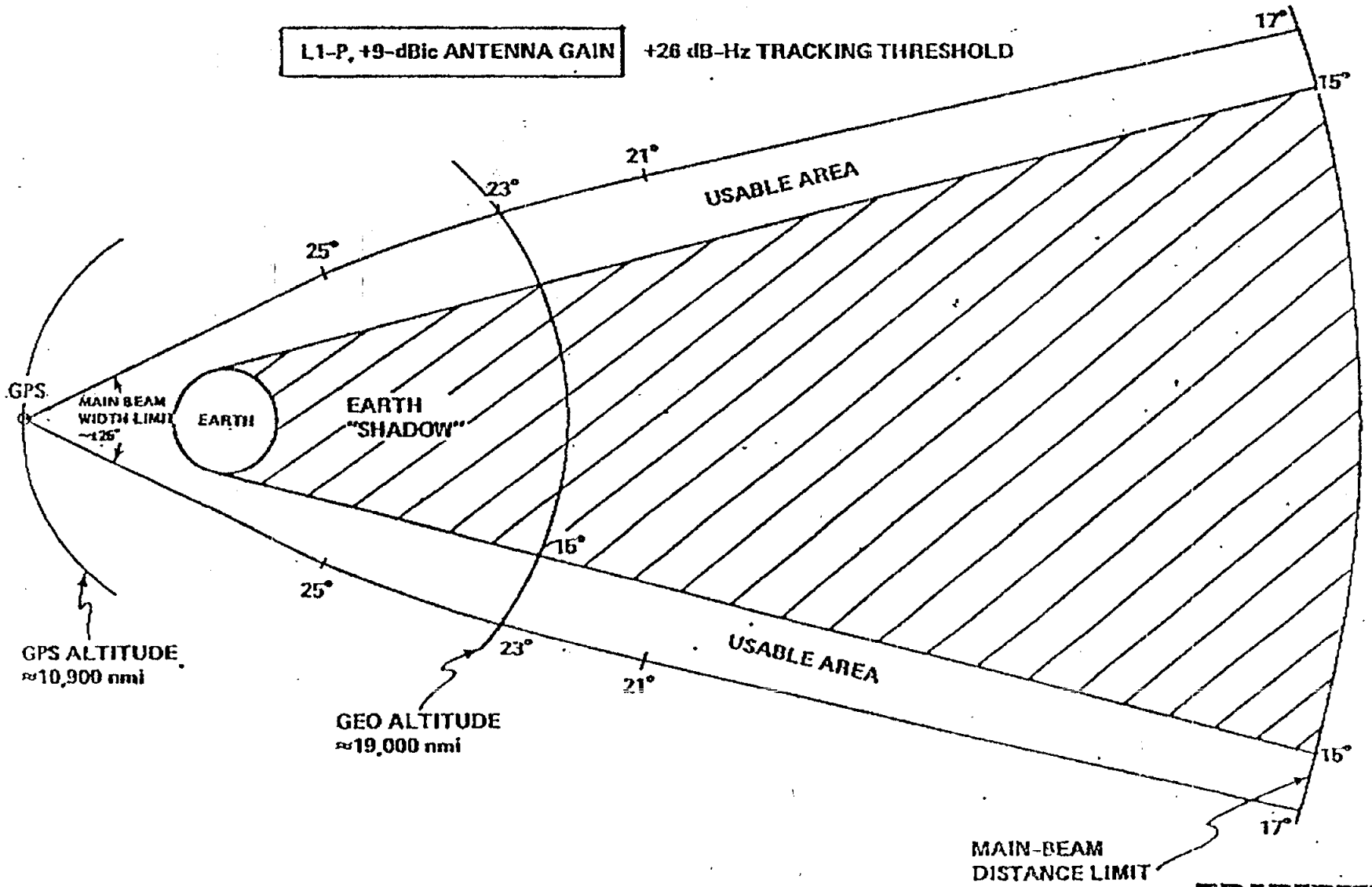
USABLE CROSS SECTION OF GPS TRANSMITTER ANTENNA PATTERN†



†GPS Signal and Link Information from "OTV GN&C Technology Requirements and System Design Study", CSDL-P-1798, Dec 1983

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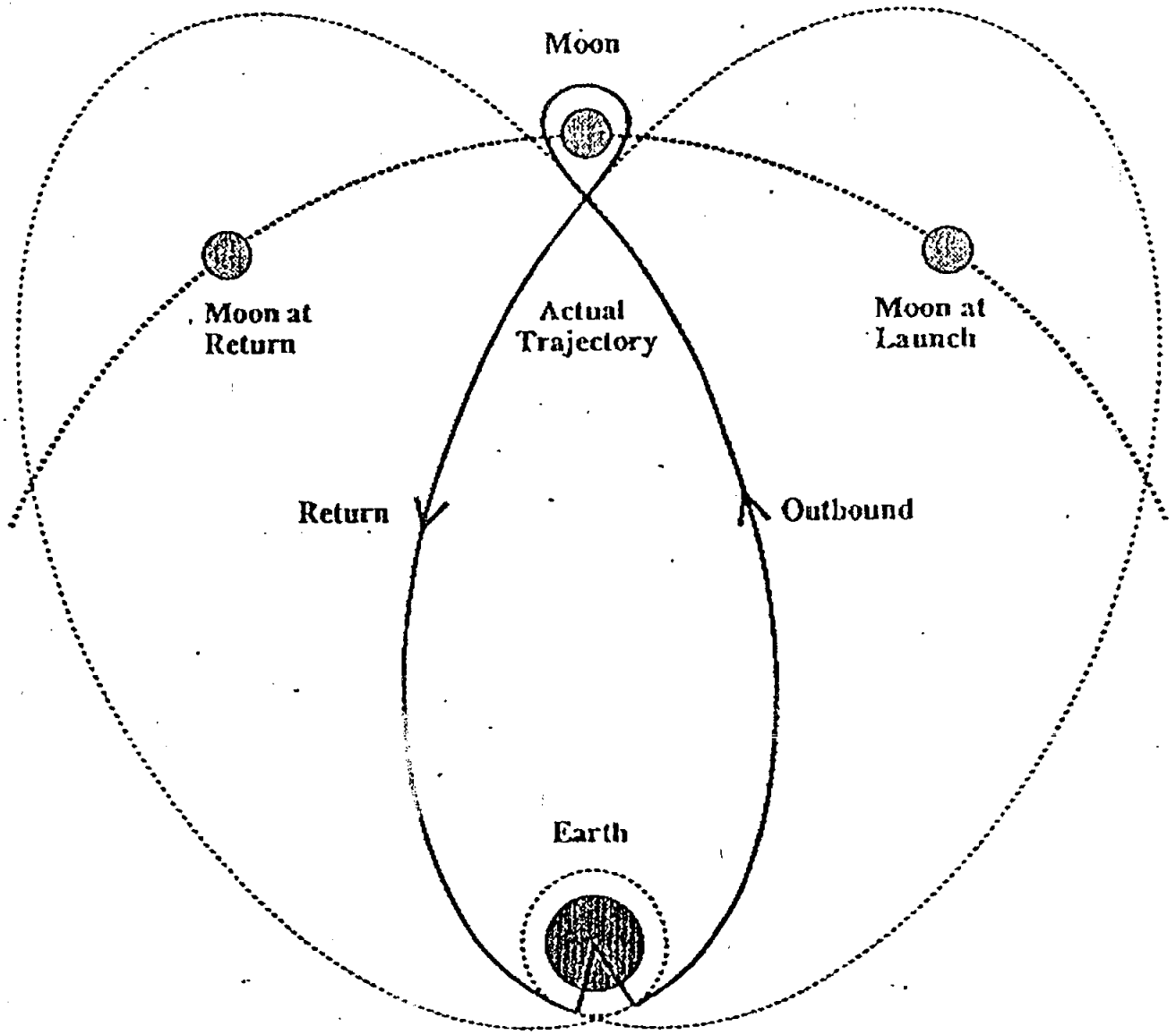
GLOBAL POSITIONING SYSTEM

- The GPS includes 24 satellites in 6 orbit planes with 4 satellites each at an altitude of about 20200 km (12 hr period)†
- The satellite transmission is directed toward earth. Any signal received from GPS when a vehicle is near or above GPS altitude comes from satellites on the far side of the earth. Only the main beam is used
- GPS signal strength decreases as range to satellites increase. A modest 9dB gain horn antenna is assumed on the vehicle. The antenna size is about 25x36x36 cm
- The L1-P code signal gives the longest usable range from a GPS transmitter and is used here. The L2-P code signal has a larger main beam width at ranges below about 92000 km

† Ref.: "System Specification for the Navstar Global Positioning System", 11 Dec., 1990

EARTH-RELATIVE CONICS AND TRAJECTORY

Lunar Insertion Δv = 3146 m/s, 6 day Free Return



LUNAR MISSION - TRANSFER TO/FROM MOON

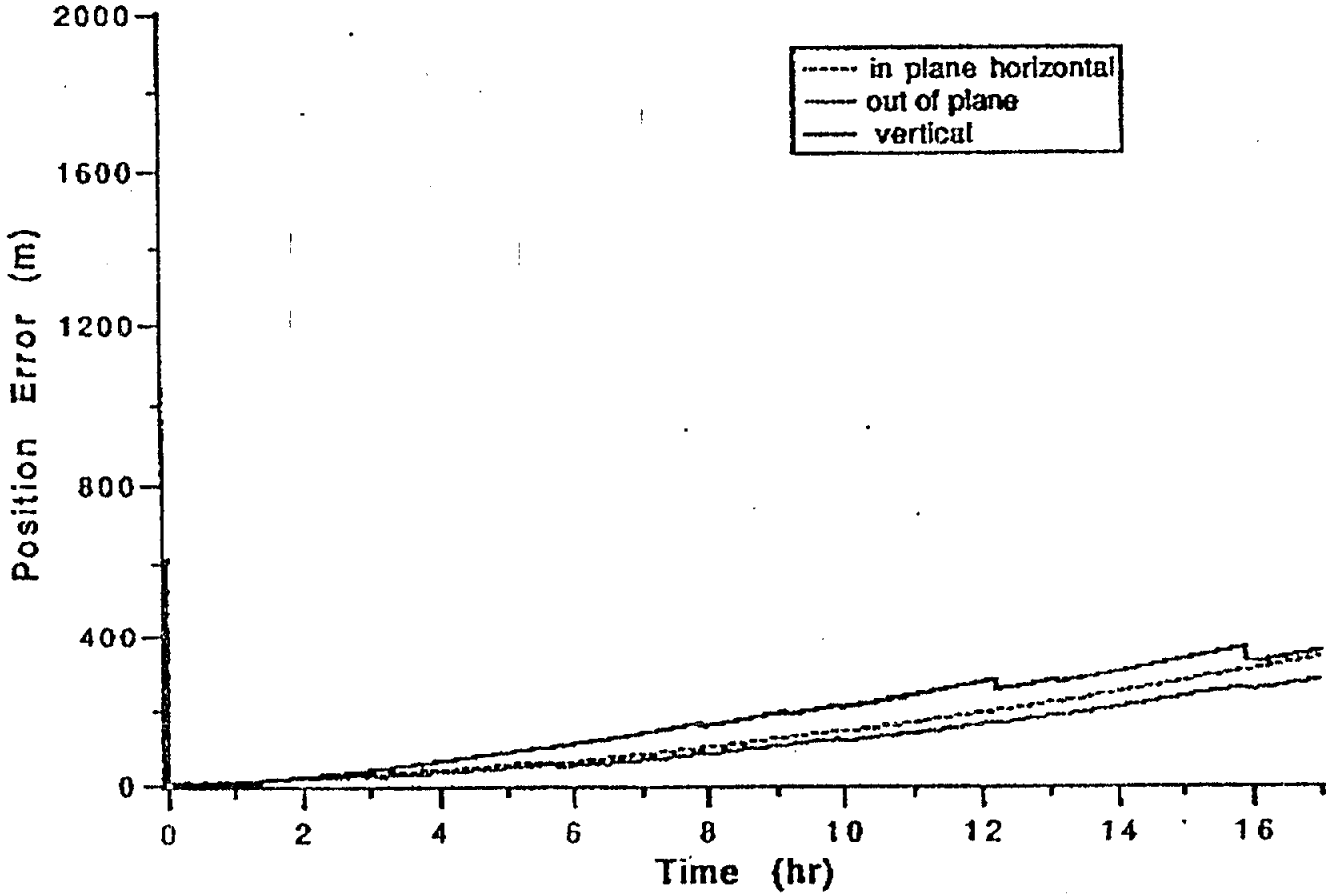
- The key question is if GPS is useful for navigation to or from the moon for a manned or unmanned vehicle
- The basic trajectory to and from the moon used in this study is a Hohmann-like transfer with a free return capability due to lunar gravity effects (see next page)
- The portion of interest during the transfer to the moon is after the transfer burn in low earth orbit to the time when GPS becomes unavailable at about 17 hours
- The particular trajectory used for navigation simulation starts at an altitude of 250 km and a speed of 10901 m/s. The assumed transfer orbit plane inclination is 23.5 deg, the average inclination of the lunar orbit plane
- The return transfer and portion of interest is basically the reverse of above; the last 17 hours of the trajectory aiming to a low earth orbit are simulated

TOPICS

- Lunar mission trajectory details
- Global Positioning System description, acquisition, and coverage
- Navigation study of transfer to moon with GPS
- Navigation study of transfer from moon with GPS
- Navigation study results of transfer to moon with added velocity error due to midcourse correction

RMS POSITION ERRORS - Smaller Noise

0 hr Offset Time, 0.003 m/s/√hr Velocity Random-walk



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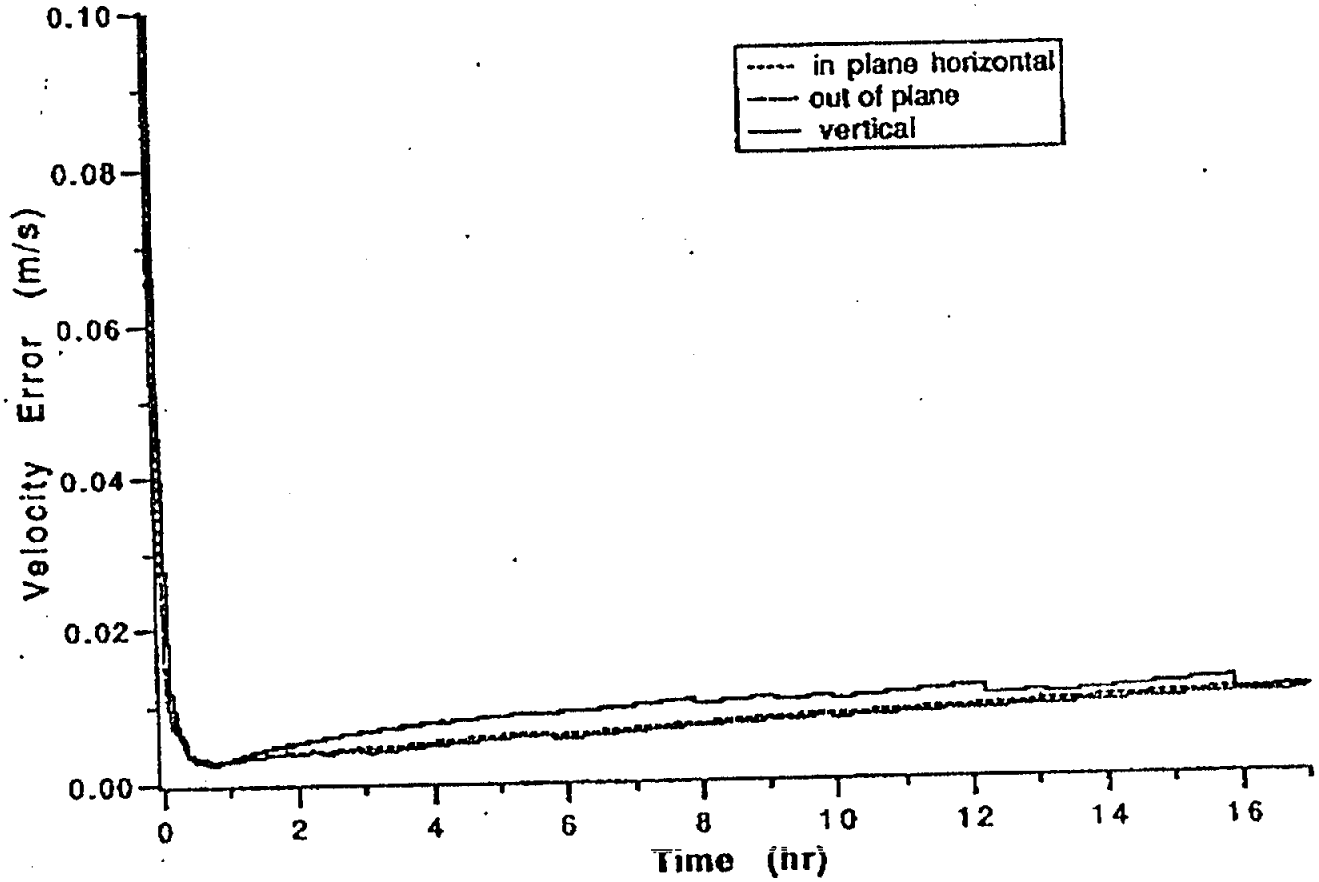
FAK 1

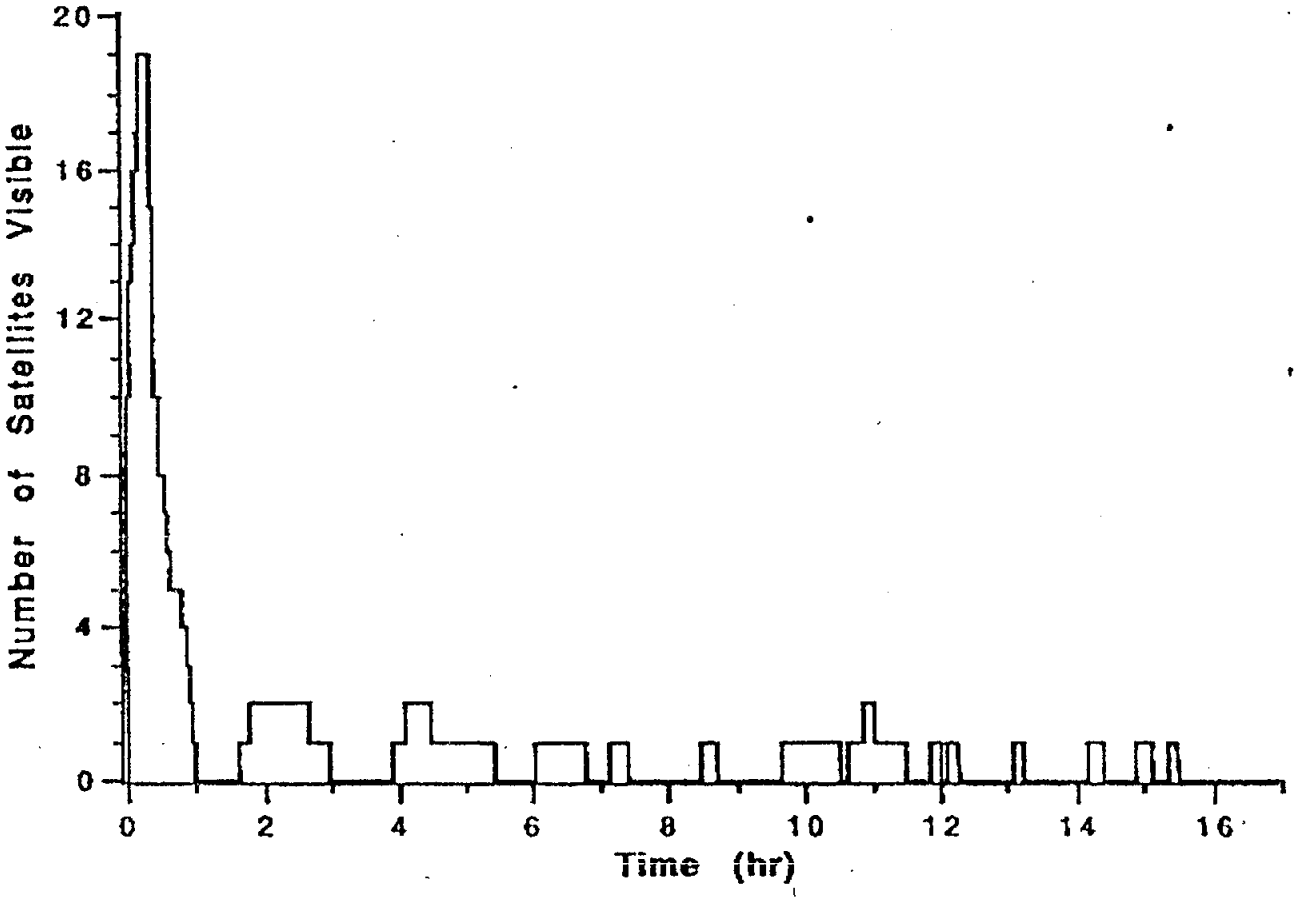
RMS VELOCITY ERRORS - Smaller Noise

0 hr Offset Time, 0.003 m/s/√hr Velocity Random-walk



NUMBER OF GPS SATELLITES VISIBLE

Transfer to Moon, 9.5 Hour Offset Time (Worse)



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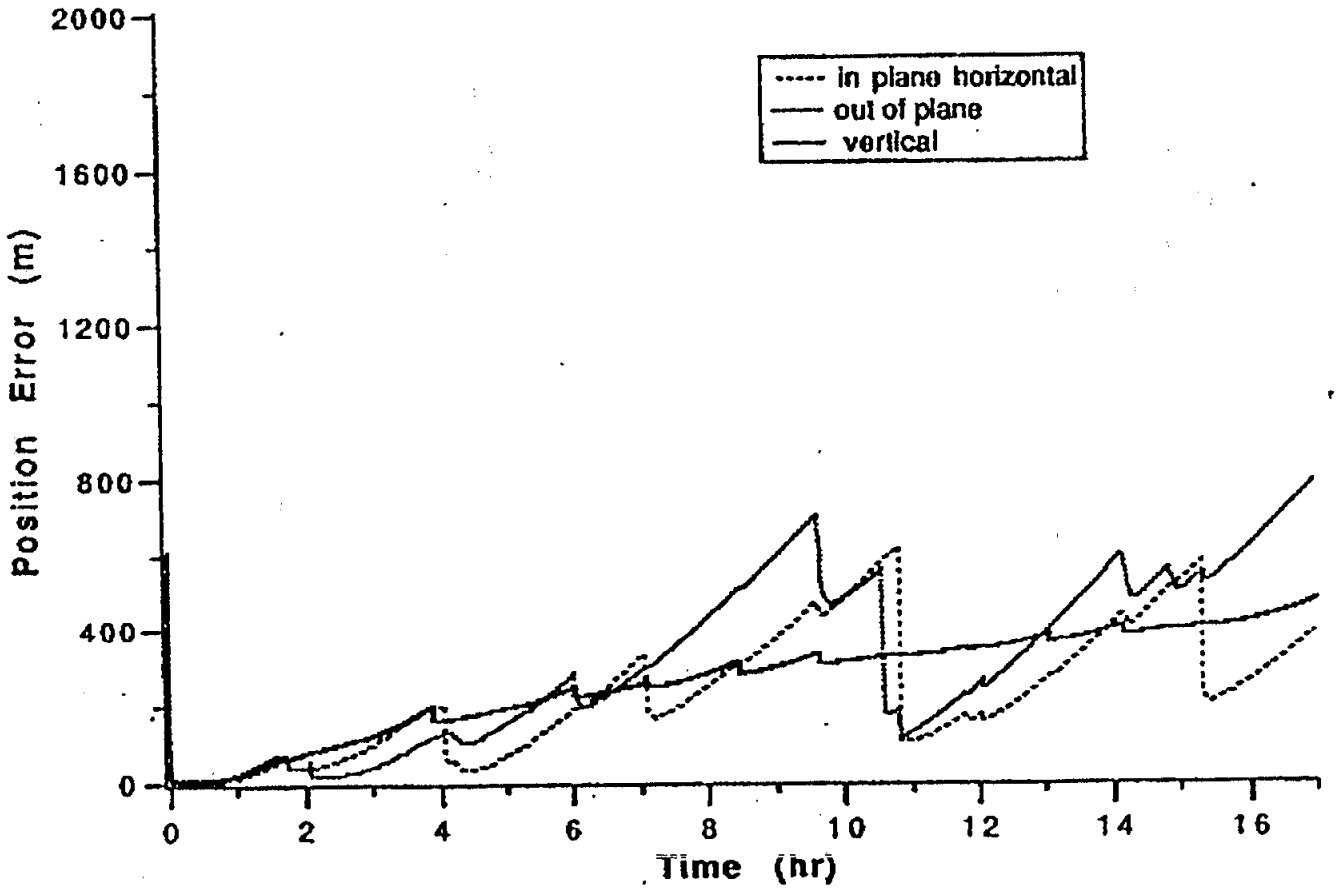
RMS POSITION ERRORS - Different Offset

9.5 hr Offset Time, 0.03 m/s/√hr Velocity Random-walk



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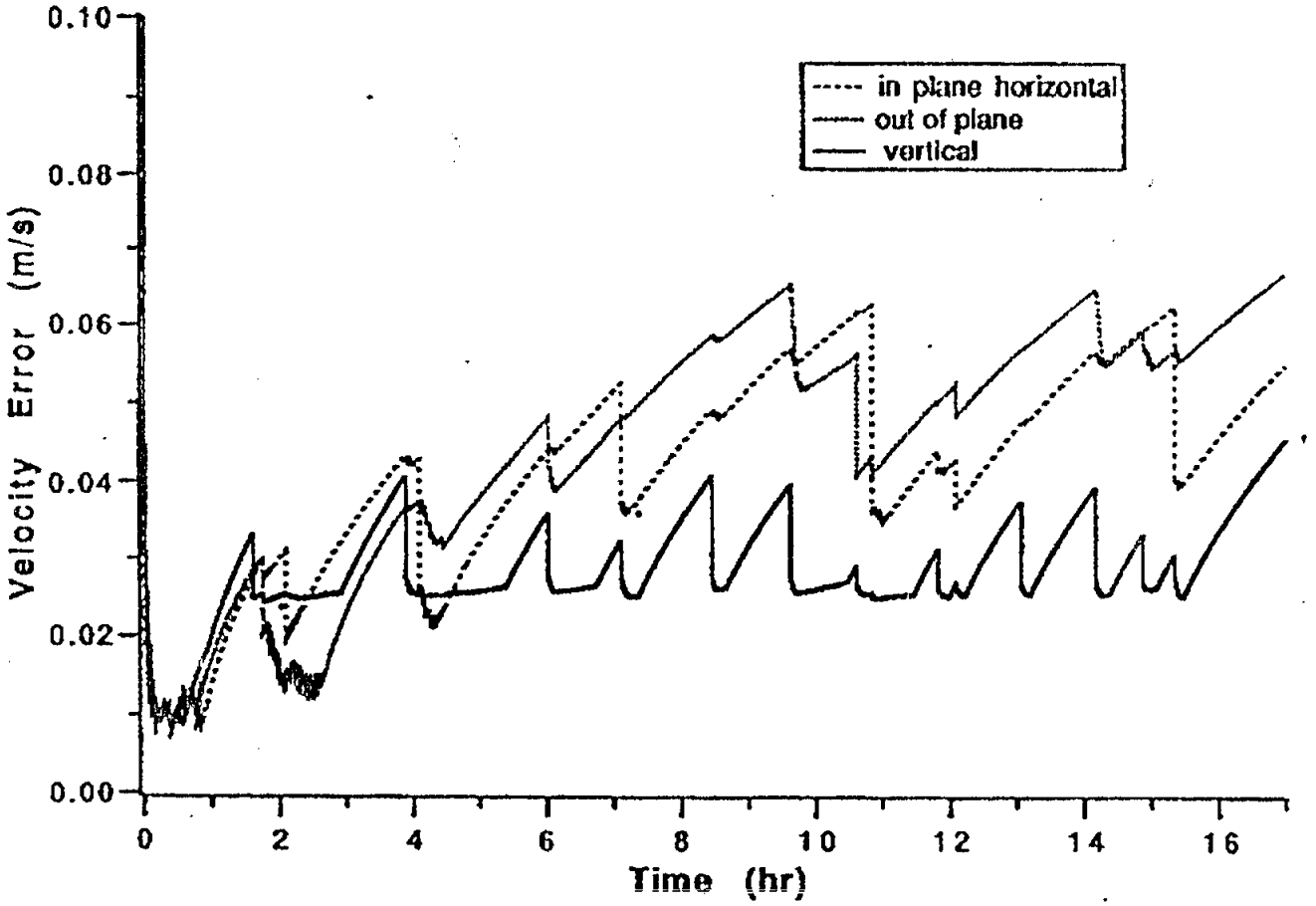
FAK 17

RMS VELOCITY ERRORS - Different Offset

9.5 hr Offset Time, 0.03 m/s/√hr Velocity Random-walk



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FAK 18

NAVIGATION RESULTS - TRANSFER TO MOON

- The rms position errors in the first (unmanned) case show general growth when fewer than three GPS satellites are acquired. The prime driver during the long coasting period is the velocity random-walk noise of 0.03 m/s/√hr
- The down range and cross track position errors show larger growth, since any GPS updates after about 1 hour are mostly in the vertical direction
- The vertical position error becomes correlated with the clock bias error, which is part of the error in range to GPS
- Decreases in rms errors after 6 hours show when one GPS satellite is acquired after a period with no satellites
- RMS velocity error performance, especially after about one hour, shows the growth due to the velocity random-walk noise. The evident floor value of vertical velocity after two hours is due to the clock drift error, modeled here as a Markov process

NAVIGATION RESULTS (cont'd)

- The rms position and velocity errors for the second case, with smaller velocity random-walk noise of 0.003 m/s/√hr, shows much less growth than the first case
- The limited observability from the later GPS updates does better with this case, because the position, velocity and clock error correlations are better maintained with smaller state noise
- A better clock would allow better performance, particularly in the vertical direction at later times. The first case would clearly benefit more from smaller clock errors

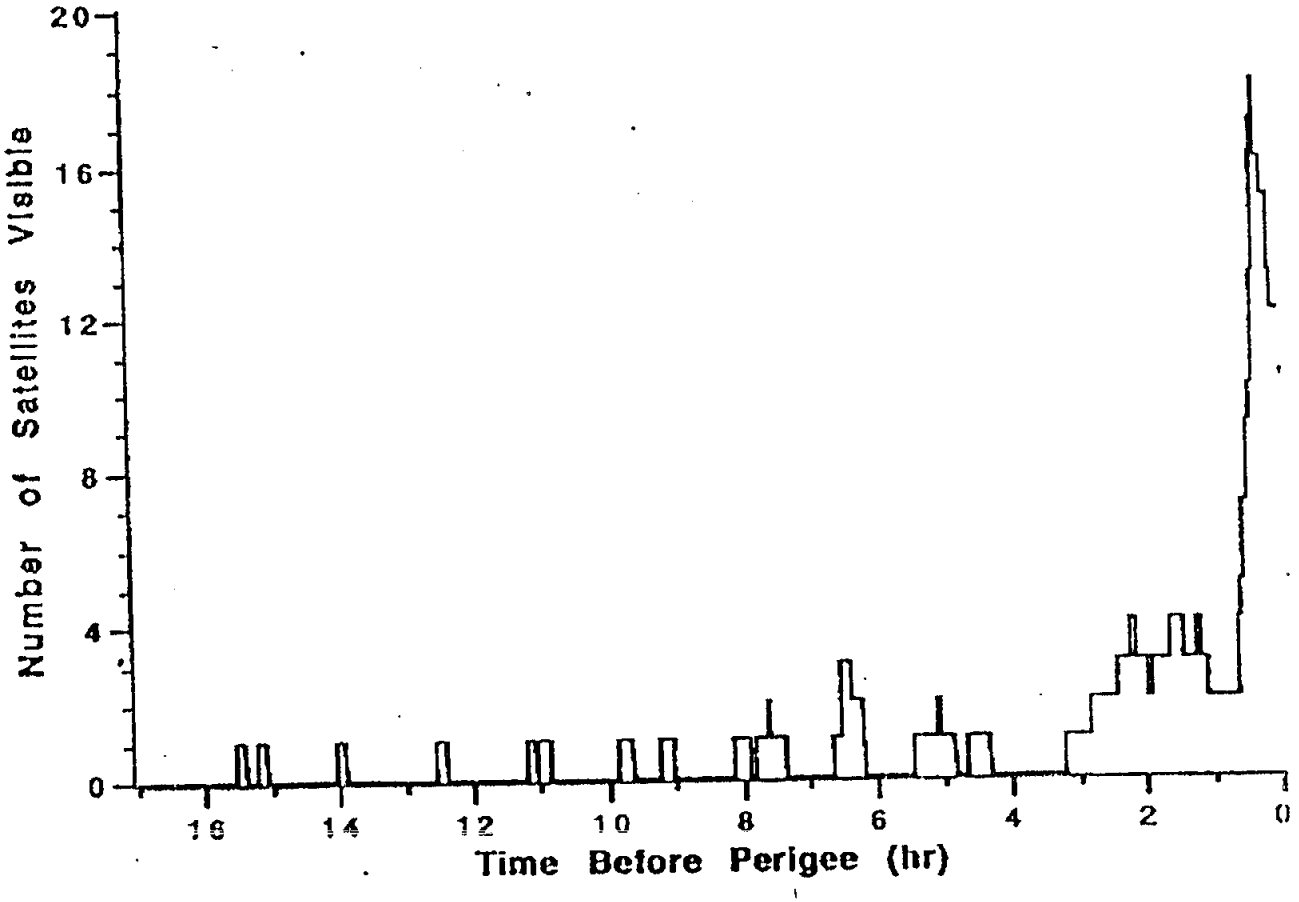
NAVIGATION SIMULATION - TRANSFER FROM MOON

- The last 17 hours of the transfer trajectory from the moon are shown. This is about the time when the vehicle goes below the maximum GPS signal distance
- Initial rms position and velocity errors assumed at the start of the 17 hours are 7 km and 1.6 m/s per axis (uncorrelated). The rms clock bias and drift errors assumed are 20000 nsec and 1 part in 10^{10}; respectively
- The sample trajectory is for 0 hours offset time as shown on Page 8. The GPS coverage is typical for the number of satellites available within the last 3 hours
- Two different velocity random-walk values are used here; 0.03 m/s/√hr for a smaller unmanned vehicle, 0.003m/s/√hr for a larger manned vehicle with smaller and better balanced attitude control jets
- The altitude error at perigee is presented to show the improvement provided by GPS updates as the vehicle approaches earth and whether it occurs early enough for any midcourse correction

DRAFT

NUMBER OF GPS SATELLITES VISIBLE

Transfer from Moon, 0 Hour Offset Time



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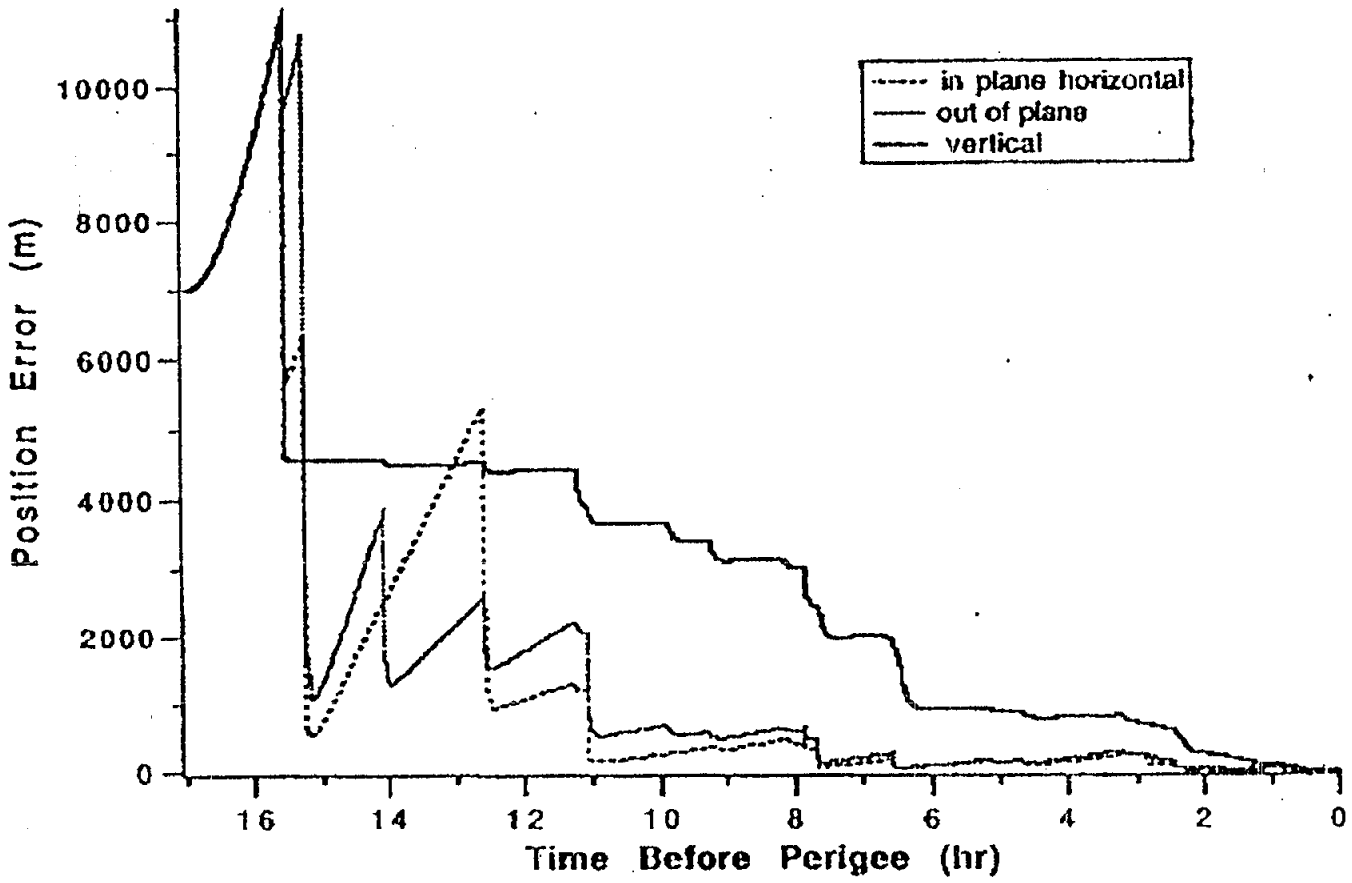
FAK 22

RMS POSITION ERRORS

0 hr Offset Time, 0.03 m/s/√hr Velocity Random-walk



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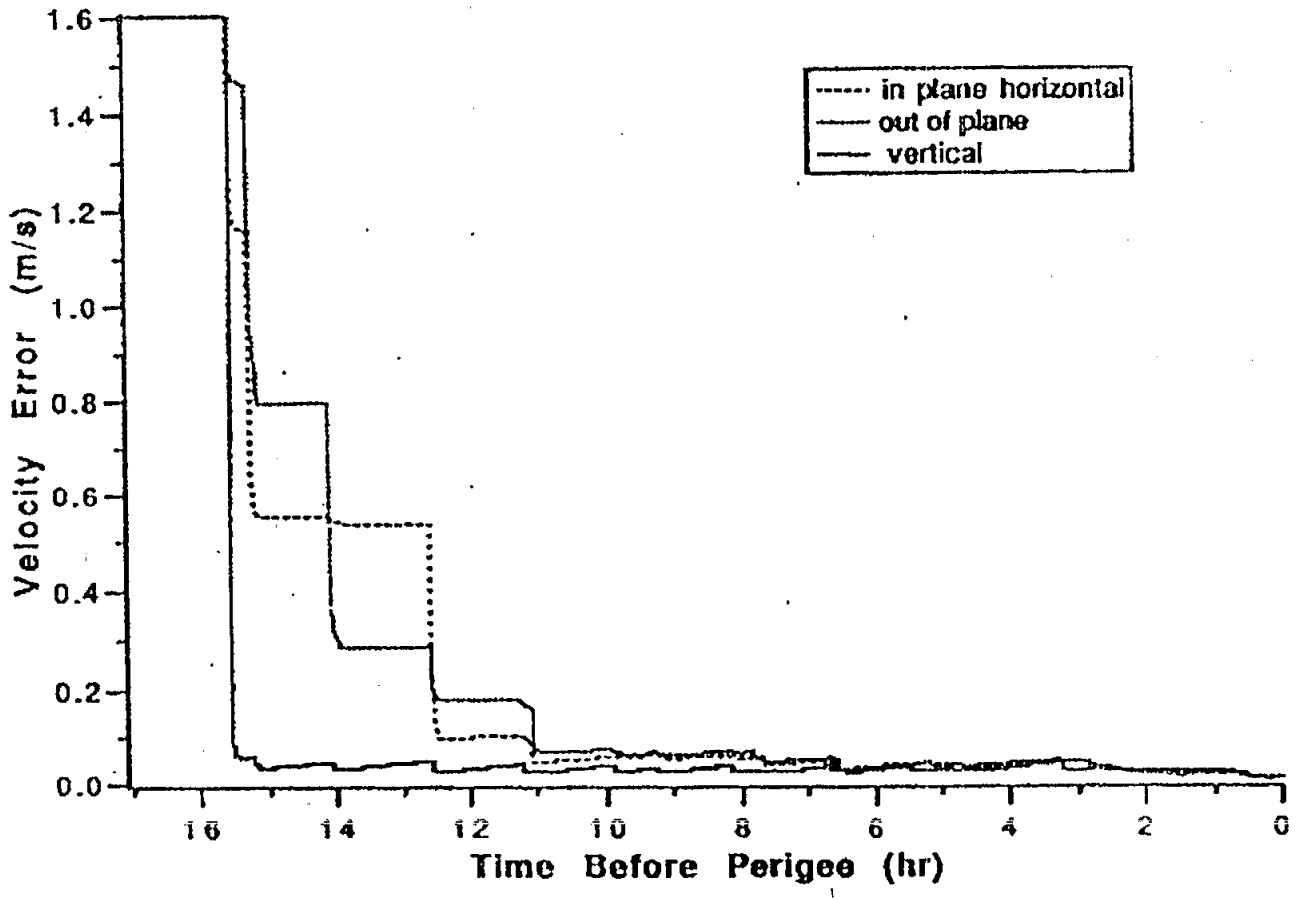
FAK 2

RMS VELOCITY ERRORS

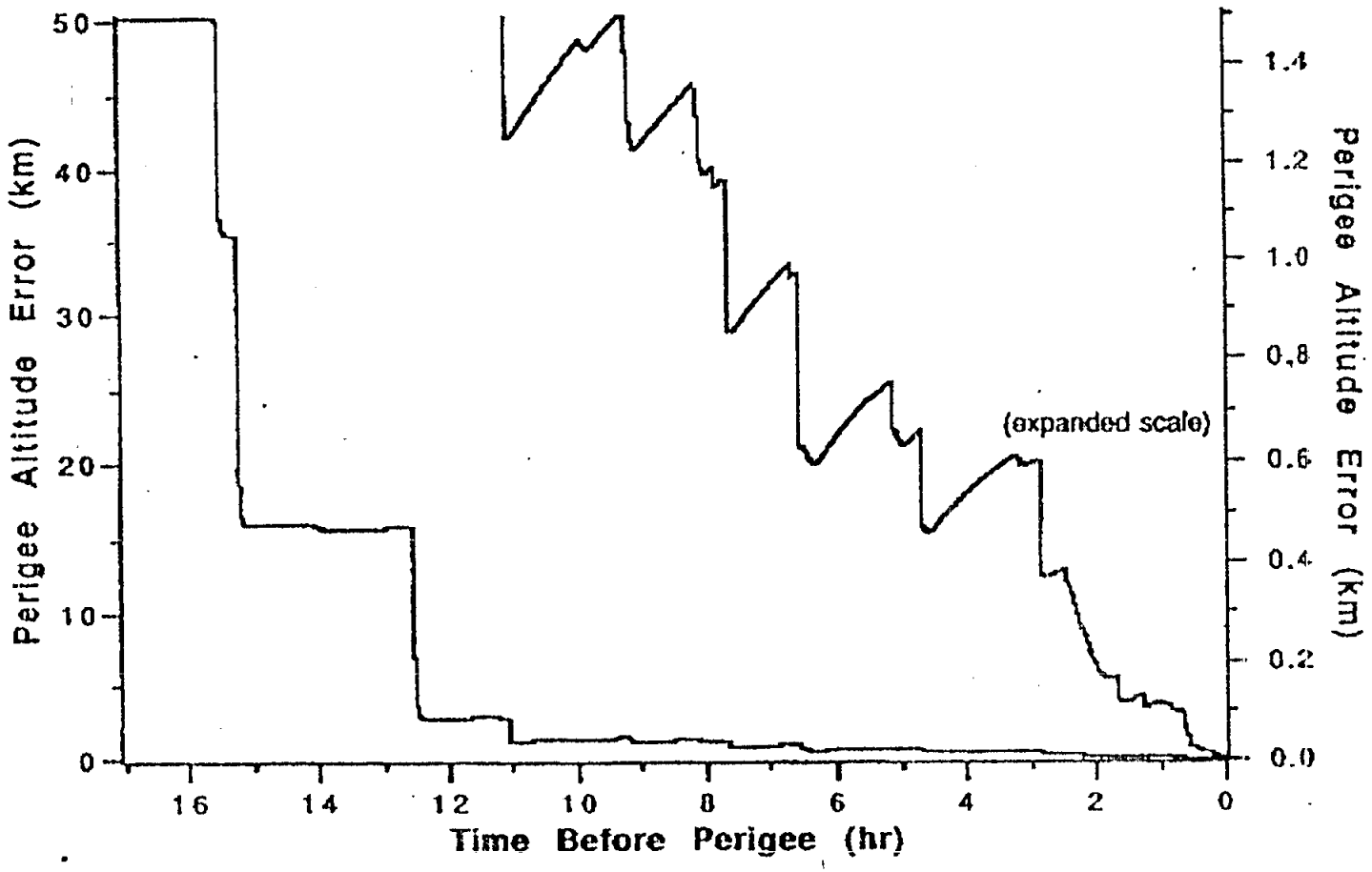
0 hr Offset Time, 0.03 m/s/√hr Velocity Random-walk



DRAPER LABORATORY

RMS PERIGEE ALTITUDE ERROR

0 hr Offset Time, 0.03 m/s/√hr Velocity Random-walk

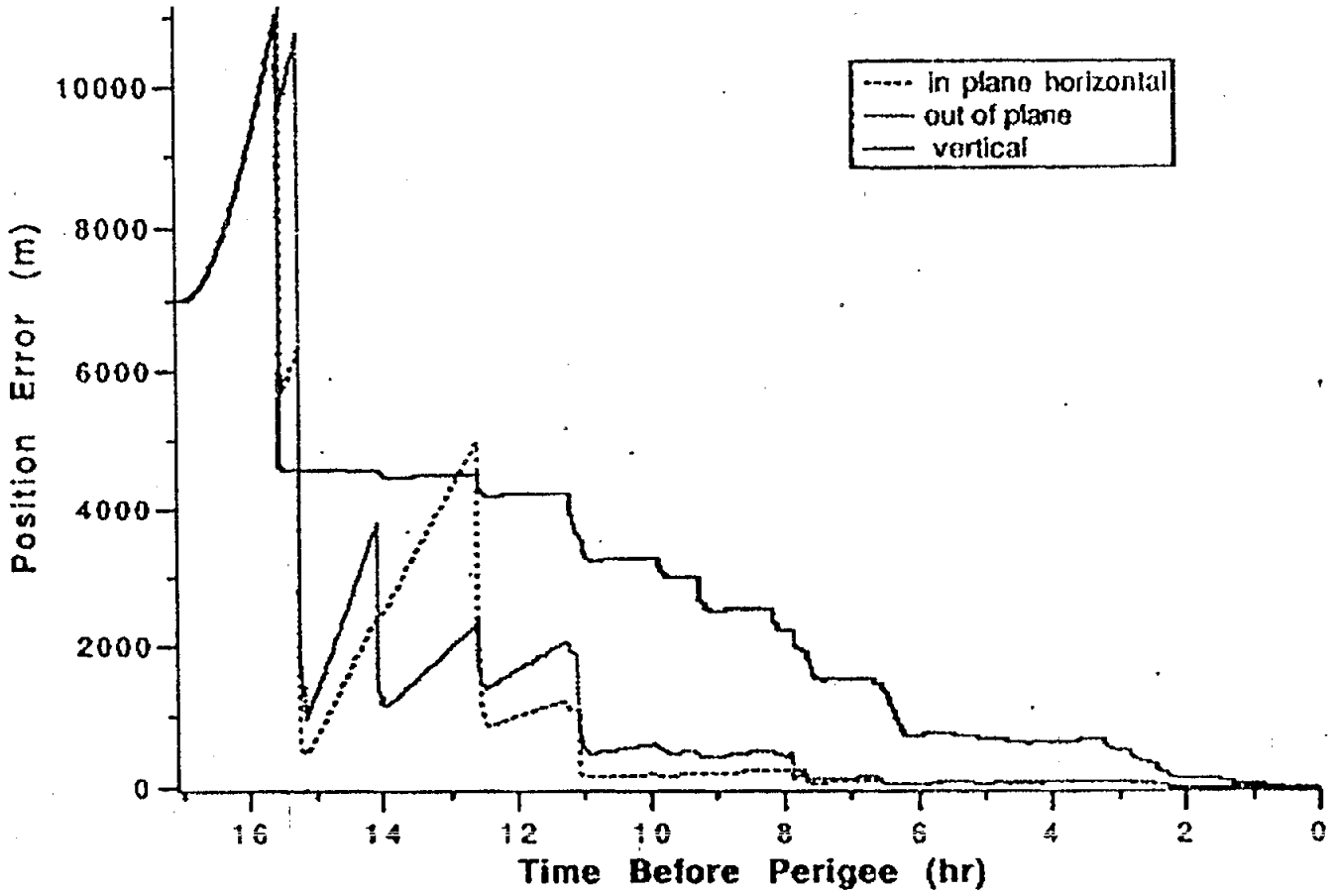


DRAPER LABORATORY

FAK

RMS POSITION ERRORS - Smaller Noise

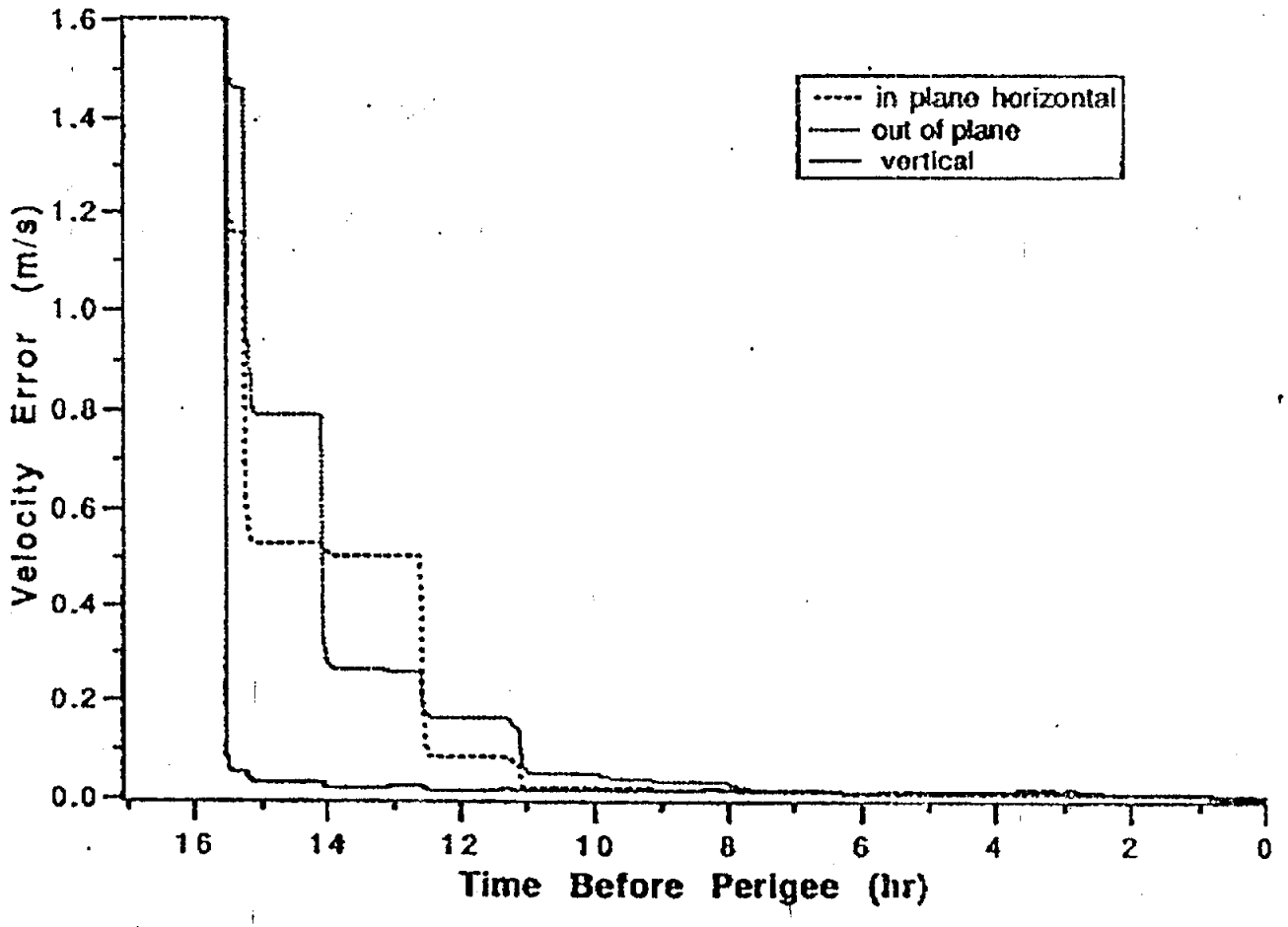
0 hr Offset Time, 0.003 m/s/√hr Velocity Random-walk



DRAPER LABORATORY

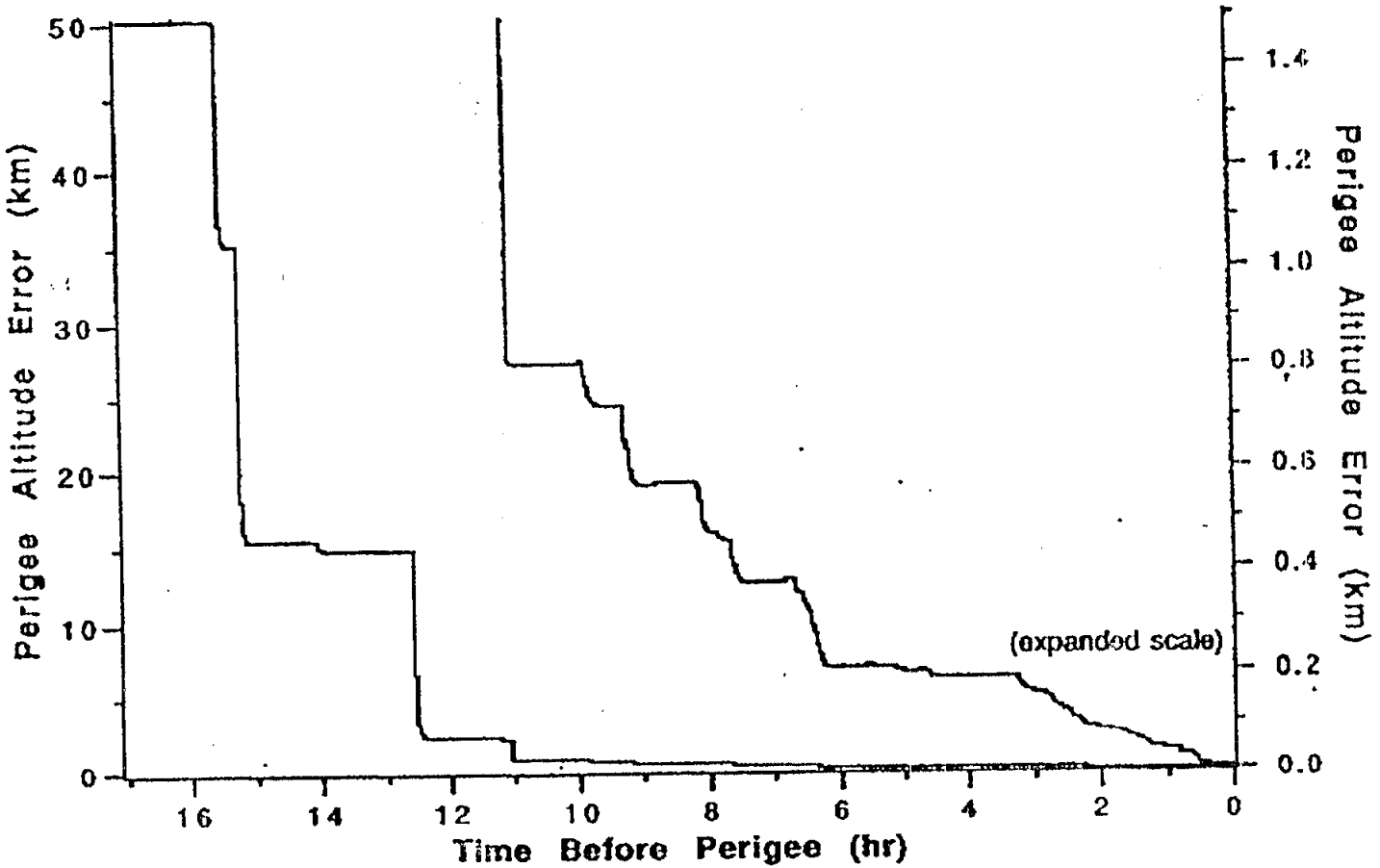
RMS VELOCITY ERRORS - Smaller Noise

0 hr Offset Time, 0.003 m/s/√hr Velocity Random-walk



RMS PERIGEE ALTITUDE ERROR

0 hr Offset Time, 0.003 m/s/√hr Velocity Random-walk



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NAVIGATION RESULTS - TRANSFER FROM MOON

- The GPS availability for this sample trajectory shows that four satellites become visible for periods between 1 and 2.5 hours before perigee
- The rms position and velocity errors, for both velocity random walk cases, show steady improvement as the vehicle approaches earth, even when single GPS satellites are visible. The case with smaller noise shows somewhat better performance than the larger noise case
- The user clock bias, part of the range error, keeps the rms vertical position error from being better estimated until late in the trajectory
- The rms perigee altitude error decreases below 10 km at over 12 hours before perigee for this sample trajectory. The difference in performance between the two cases is shown better with the expanded scale below 11 hours
- The perigee error is reduced below 1 km at about 7 hours for the larger velocity random walk case, at 11 hours for the smaller noise case

NAVIGATION SIMULATION WITH MIDCOURSE BURN

- Transfer to moon with a midcourse correction burn assumed at different times after the transfer burn
- A delta-v error in the midcourse burn of 0.1 m/s per axis is added to the rms velocity error at the time of the burn
- The rms velocity error graphs following show when the midcourse burn error is incurred; at 1, 2, 4 and 8 hours
- The particular trajectory used here has the better GPS coverage near earth, 0 hr offset time. The larger velocity random walk noise of 0.03m/s/√hr is used
- When the midcourse burn occurs at 1, 2 or 4 hours, the subsequent GPS measurements are able to remove the additional errors incurred (compare with rms position and velocity errors on p.12 and 13 without midcourse errors)
- When the midcourse burn occurs at 8 hours after the transfer burn, the sparse GPS updates cannot remove the additional errors

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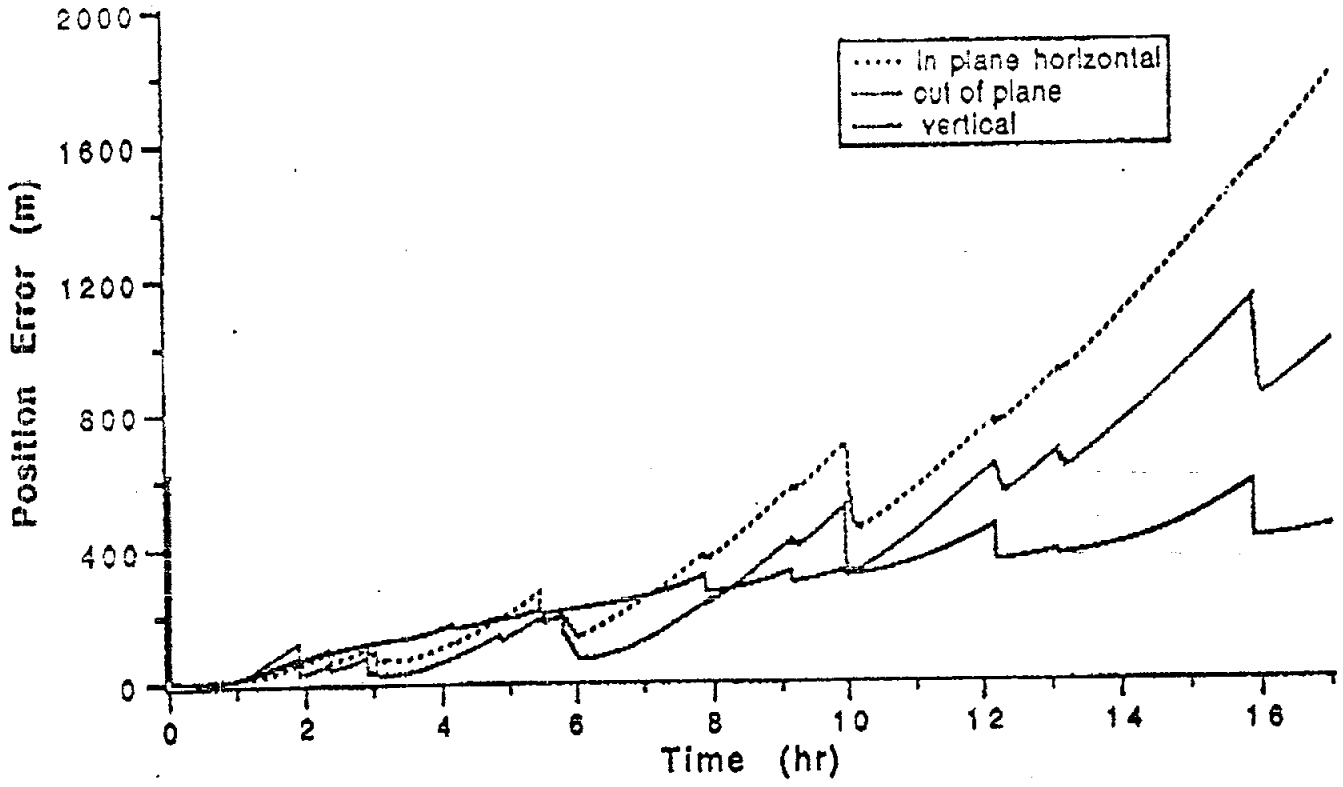
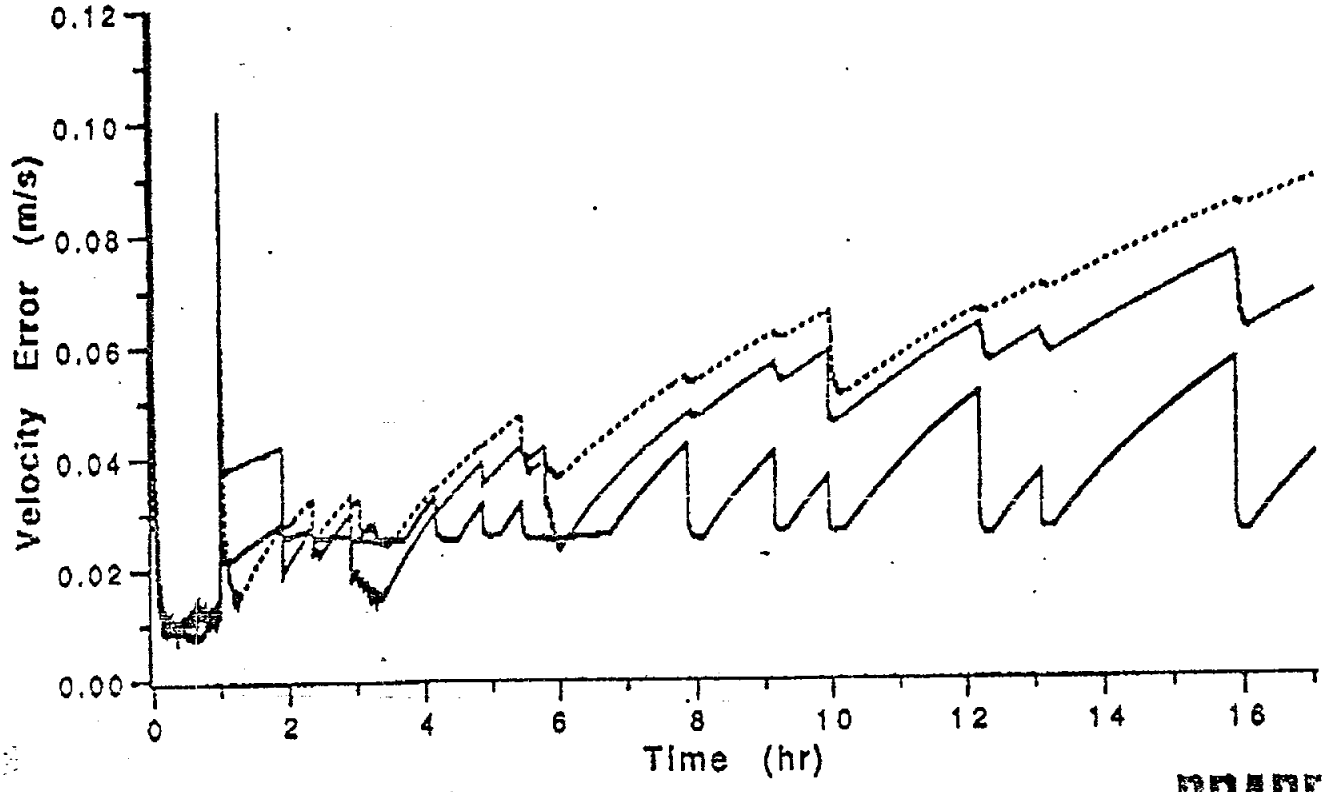
FAK

RMS POSITION & VELOCITY ERRORS

Midcourse at 1 hr, 0.03 m/s/√hr Velocity RW





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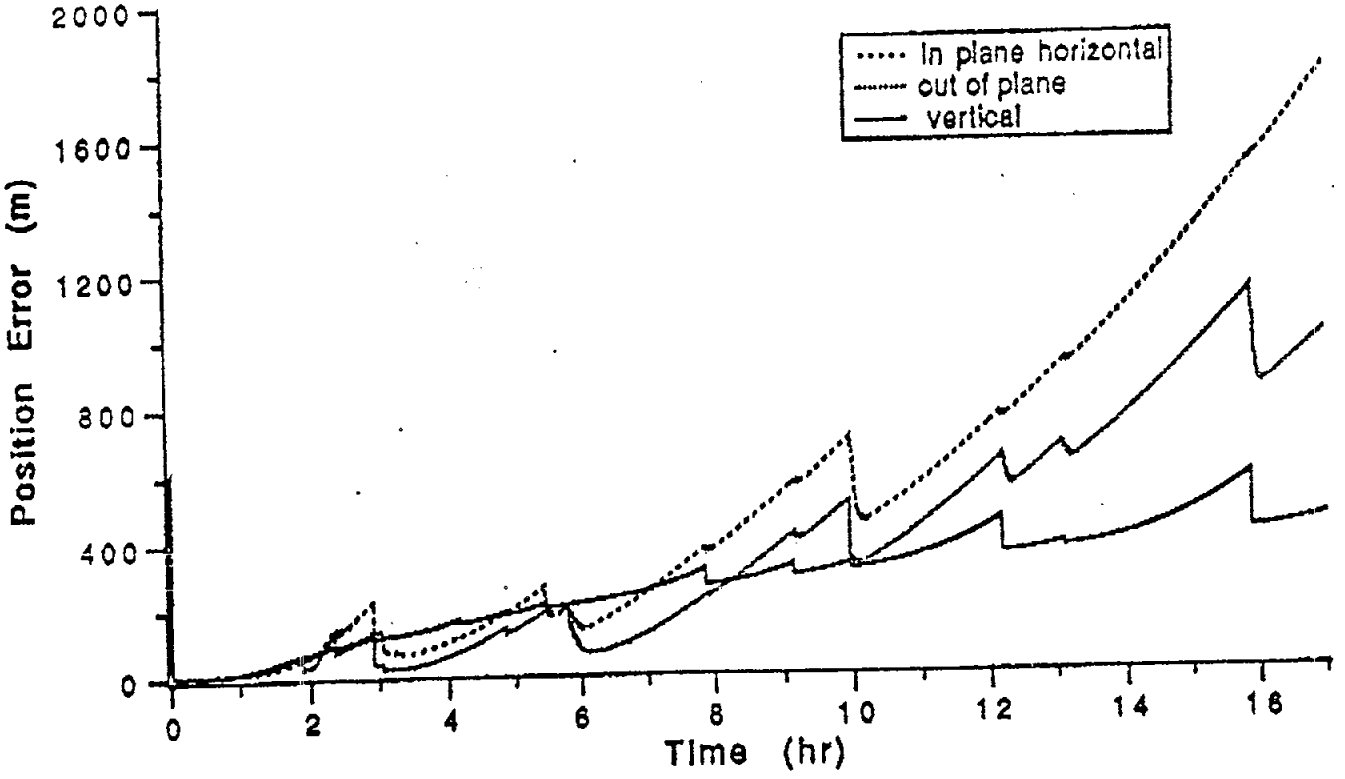
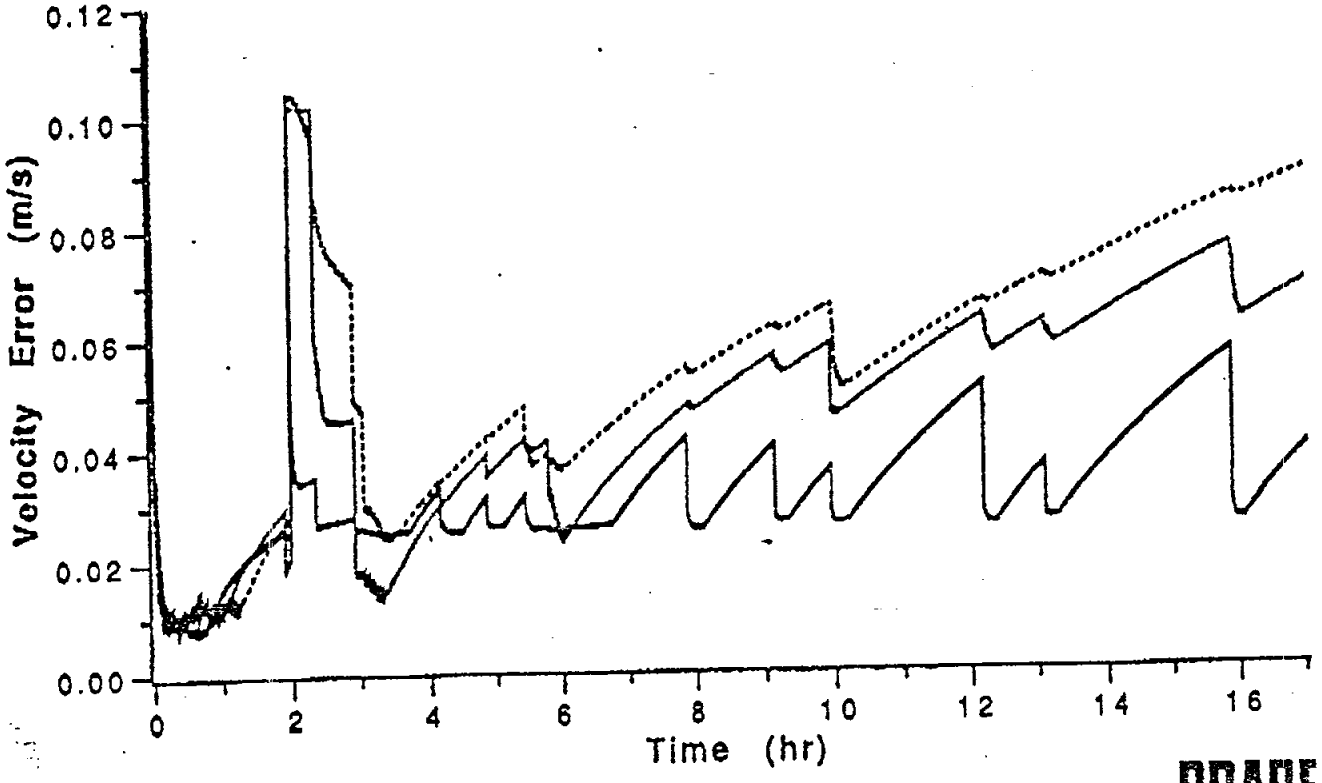
FAK 31

RMS POSITION & VELOCITY ERRORS

Midcourse at 2 hr, 0.03 m/s/√hr Velocity RW





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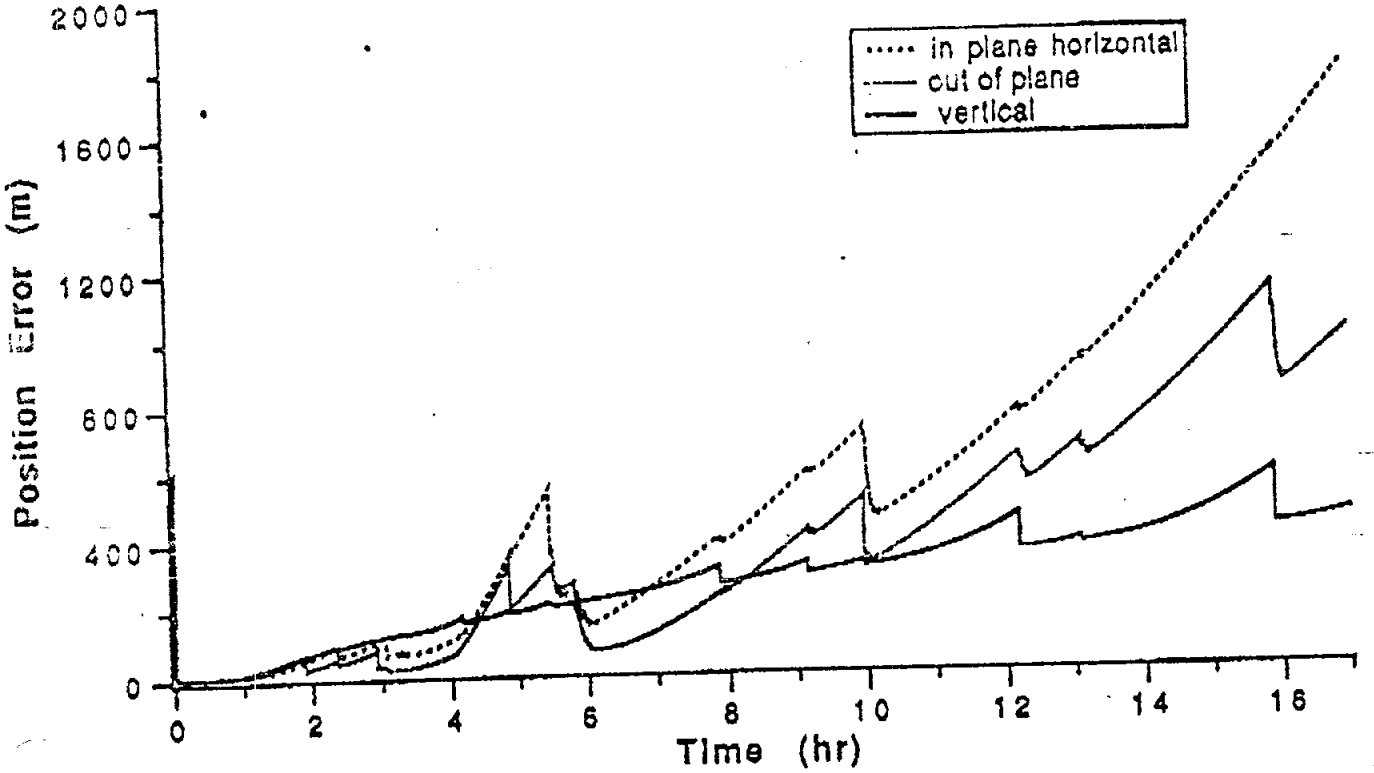
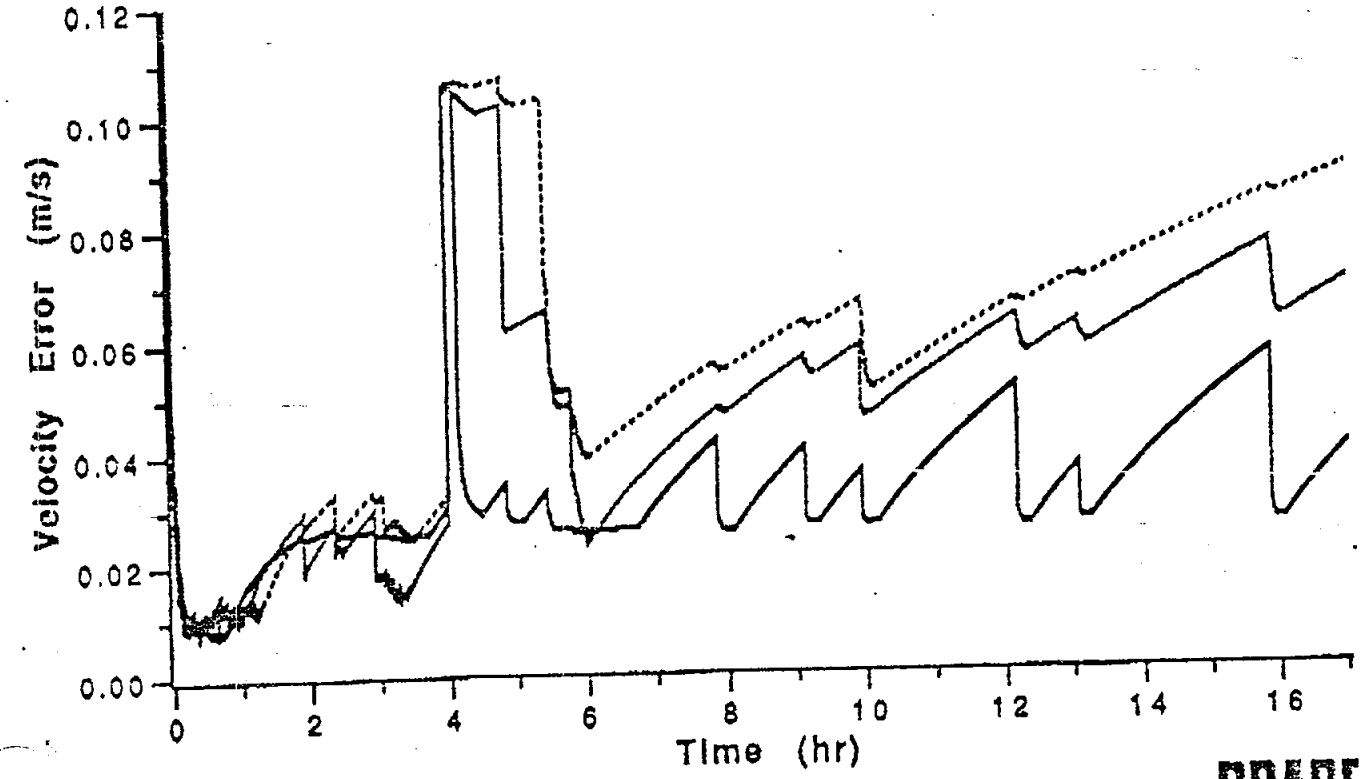
RMS POSITION & VELOCITY ERRORS

Midcourse at 4 hr, 0.03 m/s/√hr Velocity RW





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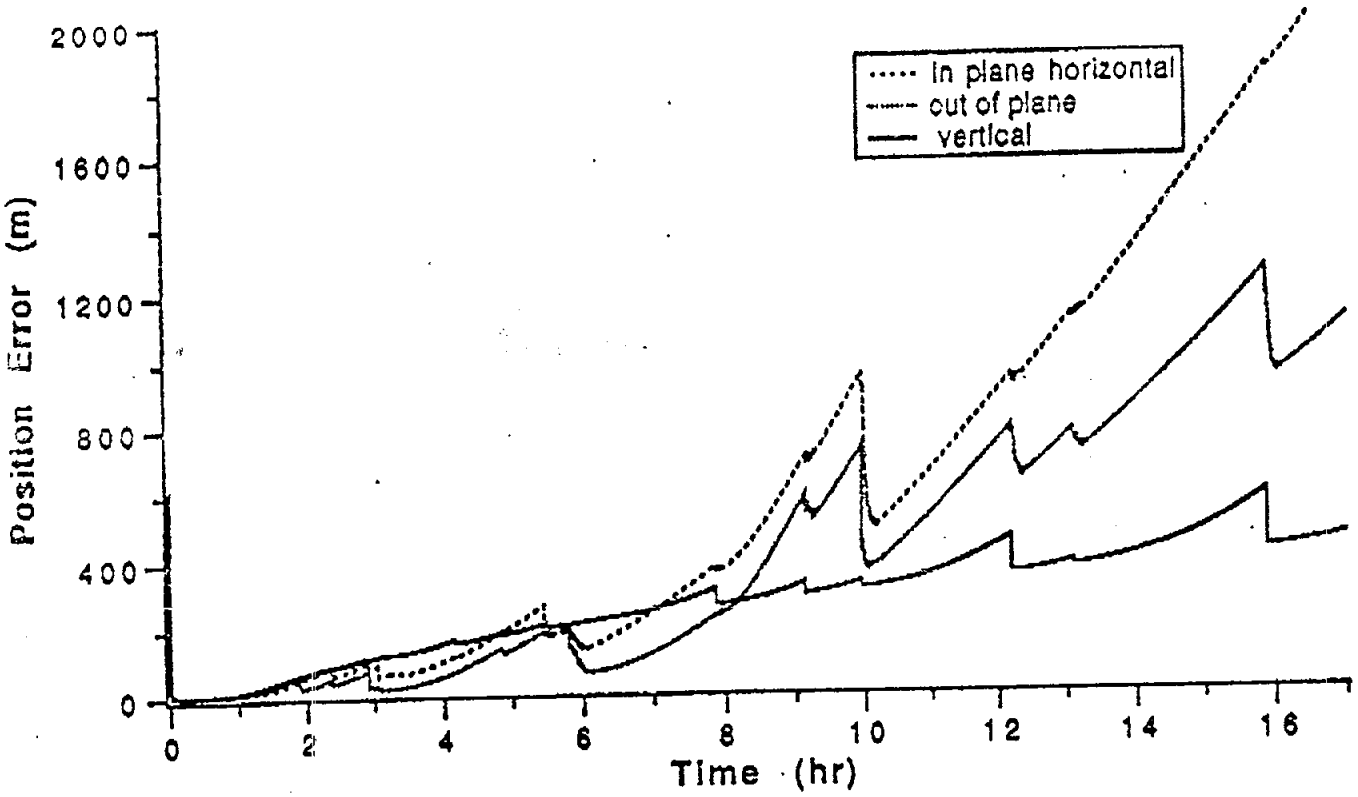
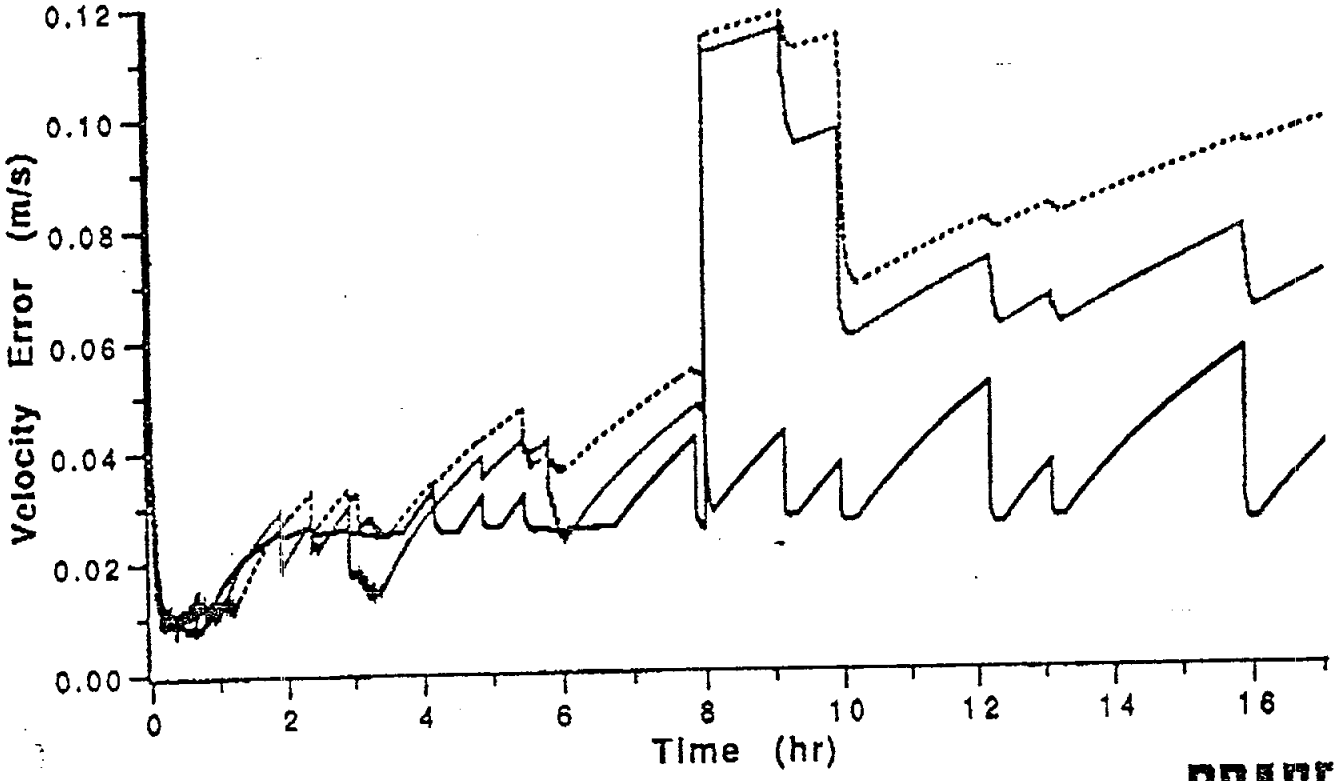
RMS POSITION & VELOCITY ERRORS

Midcourse at 8 hr, 0.03 m/s/√hr Velocity RW





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SUMMARY

- GPS measurements provide good position and velocity updates after a burn to transfer to the moon
- GPS updates, after a midcourse correction burn that occurs within several hours after the transfer burn, are able to remove midcourse delta-v errors
- Unmodeled or unmeasured accelerations during the long coast periods may be the prime error source for position and velocity growth when GPS is not available
- During a return transfer from the moon, results show that GPS updates are able to reduce perigee altitude errors below 1 km at least several hours before perigee

PRELIMINARY ANALYSIS OF HIGH ALTITUDE LUNAR ORBITER LIFETIMES

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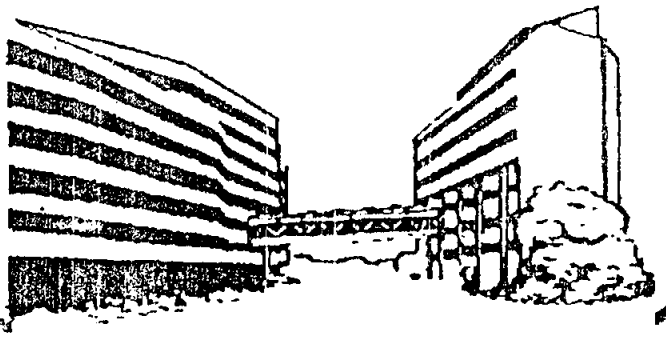
ESB-92-150
LMI-92-012

MARCH 16, 1992

THE
CHARLES STARK DRAPER
LABORATORY, INC.



HIGH ALTITUDE LUNAR ORBITER LIFETIMES

- Gravitational attraction of earth and sun significantly affect orbital motion
- Can produce significant variation in orbital lifetimes depending on specific orbit geometry
- Lifetimes dependent on:
  - — Initial altitude or semimajor axis
  - — Initial eccentricity
  - — Orbit inclination (selenographic frame)
  - — Ascending node (SI frame) maybe?

SI - selenocentric inertial

HIGH ALTITUDE LUNAR ORBITER LIFETIMES

LUNAR ORBITER ELEMENTS:

- TWO SEMIMAJOR AXIS REGIMES INVESTIGATED:
  a = 5000 km
  a = 10000 km

- INITIAL ECCENTRICITY -- e = 0.001

- SELENOGRAPHIC INCLINATIONS:
  i = 0, 30, 60, 90, 120,150, and 180 degrees

- SELENOGRAPHIC ASCENDING NODE = 158.0 degrees

- SELENOGRAPHIC ARGUMENT OF PERIAPSE = 60.0 degrees

- MEAN ANOMALY M = 0.0 degrees

HIGH ALTITUDE LUNAR ORBITER LIFETIMES

PARAMETERS INVESTIGATED

- Lifetime Without Burns
  - — Based on Periapse Altitude
- Interval Between Burns
  - — Based on Eccentricity Growth to $e = 0.01$
- Number of Burns
- Estimated ΔV to Support 10 Year Mission

  Based on ΔV to restore $\Delta e = 0.01 \times$ Number of Burns

  $\Delta V \approx 4.0$ m/sec per burn at 5,000 km

  $\Delta V \approx 3.5$ m/sec per burn at 10,000 km

HIGH ALTITUDE LUNAR ORBITER LIFETIME SUMMARY

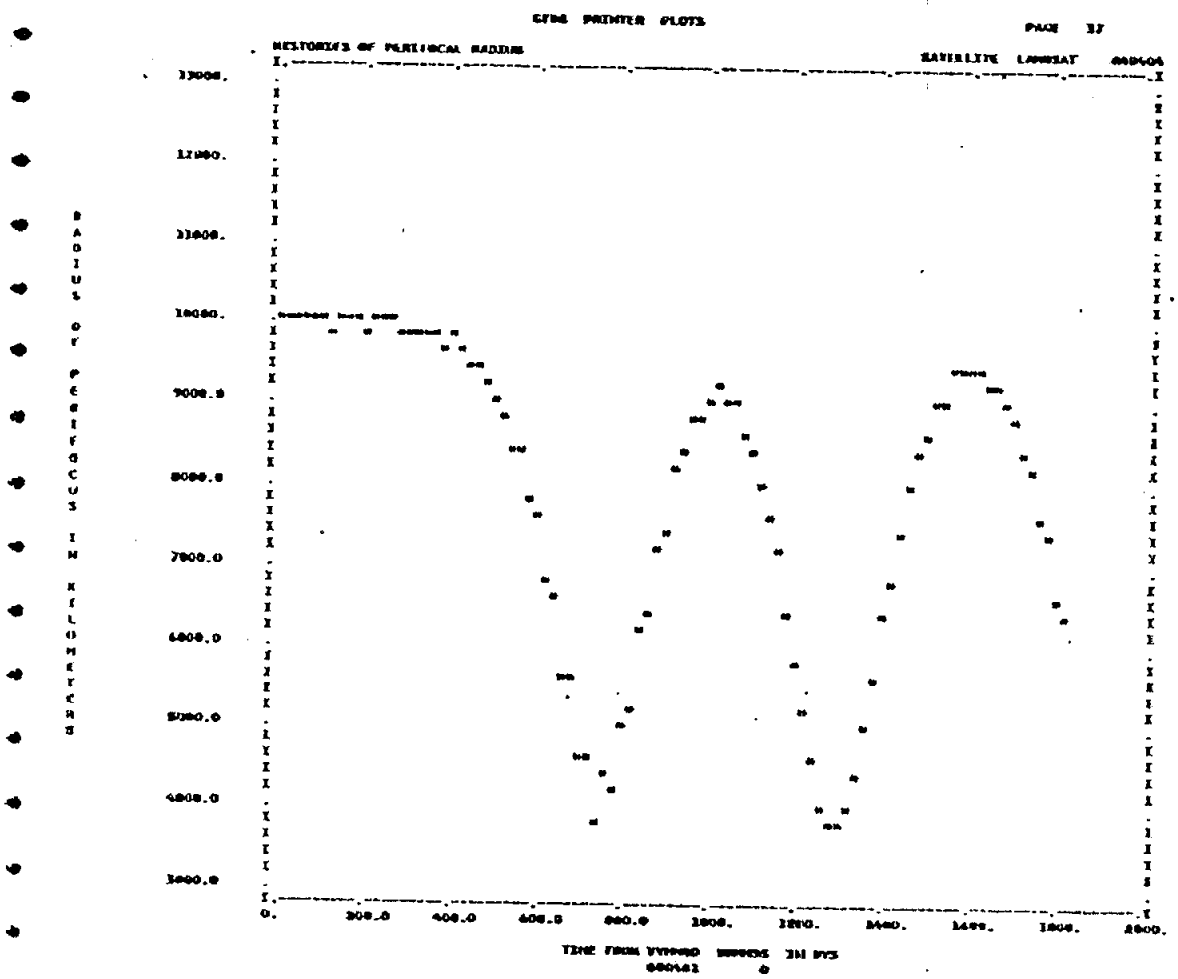
Case	a (km)	i (deg)	Lifetime (yrs) (w/o burns)	Burn Interval (days)	ΔV (m/s) (10 year life)	Number Burns
1	5000	0	> 5	>5 (years)	≈ 0†	≈ 0
2	"	30	> 5	>5 (years)	≈ 0†	≈ 0
3	"	60	> 5	§ 650	25	5
4	"	90	3	360	45	9
5	"	120	4	480	35	7
6	"	150	> 5	> 5 (years)	≈ 0†	≈ 0
7	"	180	> 5	> 5 (years)	≈ 0†	≈ 0
11	10000	0	> 5	>5 (years)	≈ 0†	≈ 0
12	"	30	> 5	>5 (years)	≈ 0†	≈ 0
13	"	60	> 5	*270	49	14
14	"	90	1	85	147	42
15	"	120	1.3	130	95	27
16	"	150	> 5	>5 (years)	≈ 0†	≈ 0
17	"	180	> 5	> 5 (years)	≈ 0†	≈ 0

e0 = 0.001 for all cases
w0 = 60.0
Ω0 = 158.0 degrees

§ secular altitude decay - impact just after end of 5th year
* 6000 km altitude oscillation
ΔV ≈ 4.0 m/s per burn at 5,000 km
ΔV ≈ 3.5 m/s per burn at 10,000 km
† based on five year prediction

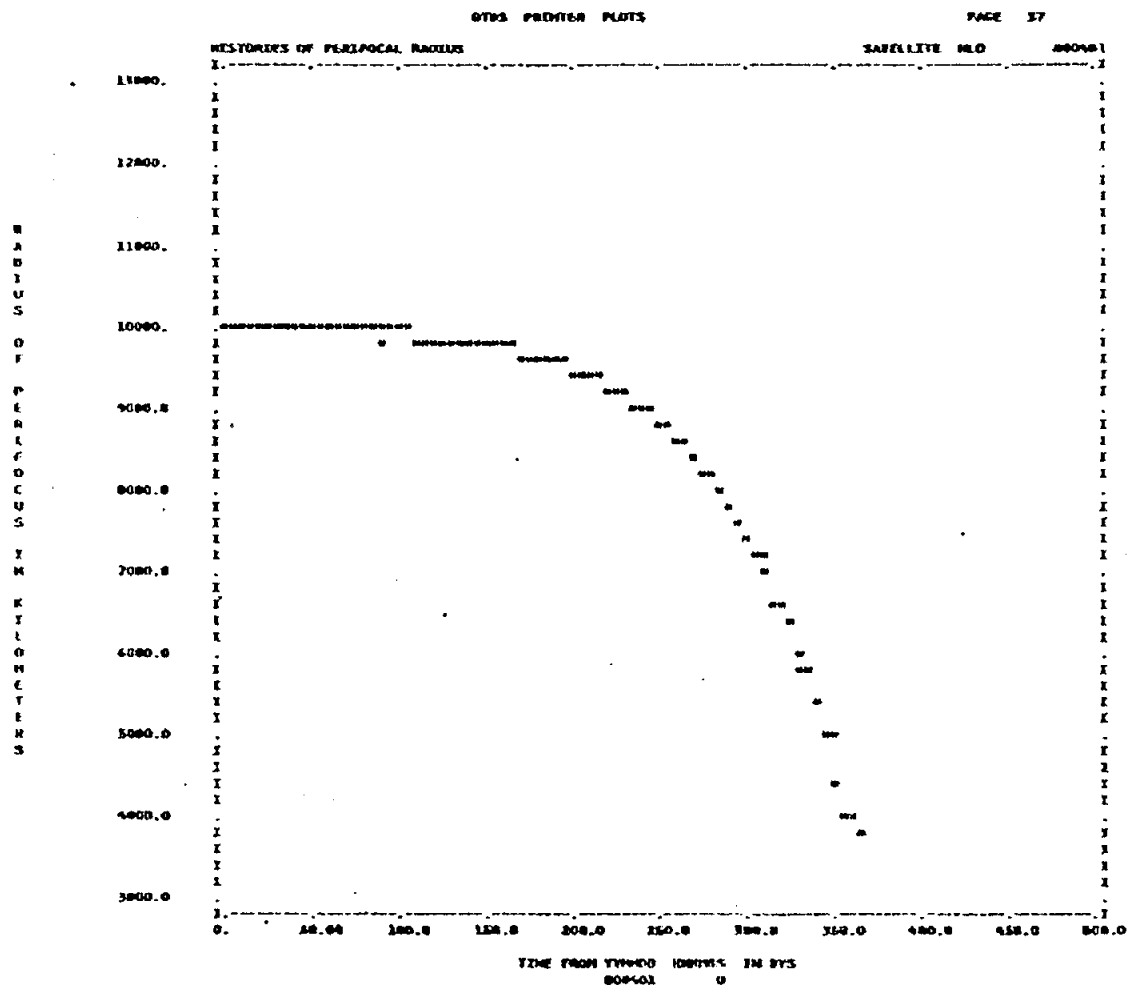
PERIAPSE HISTORY FOR CASE 13

a = 10,000 km i = 60 degrees



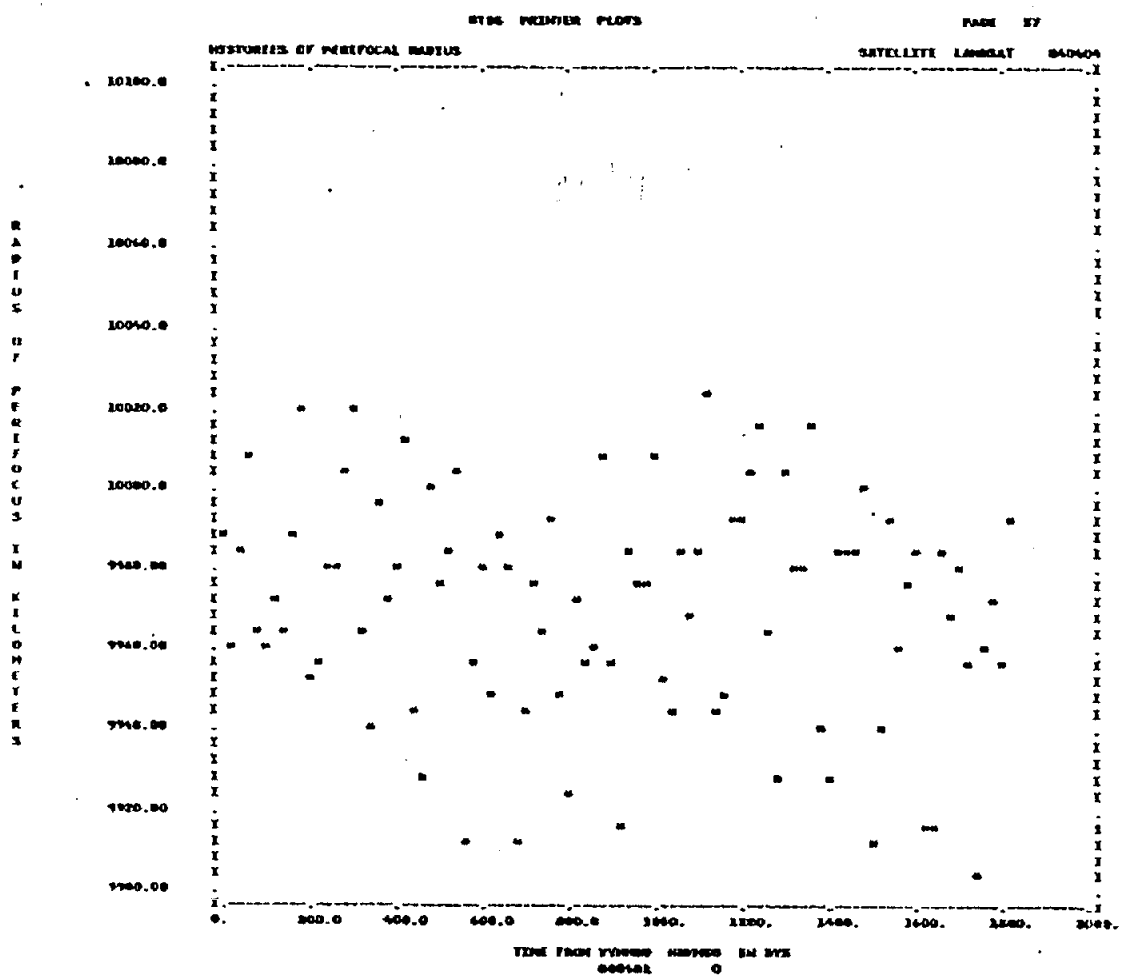
PERIAPSE HISTORY FOR CASE 14

a = 10,000 km i = 90 degrees



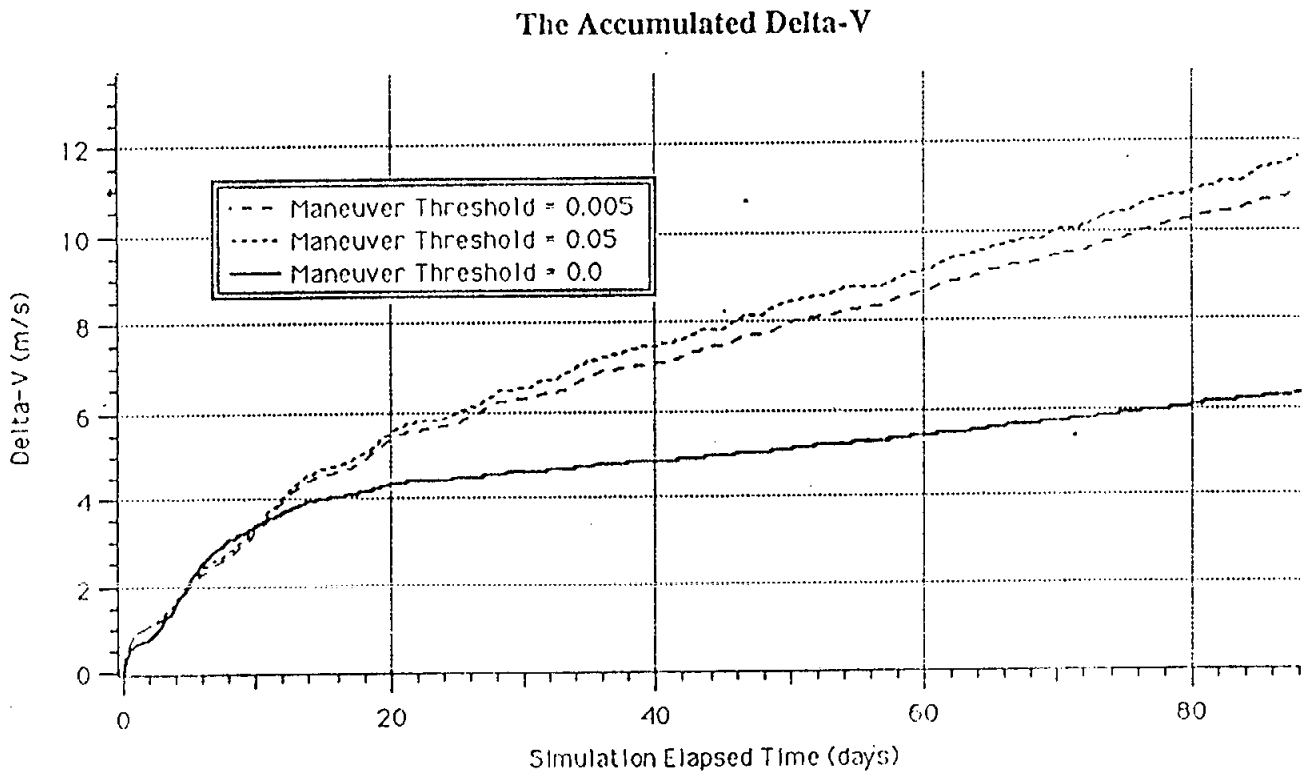
PERIAPSE HISTORY For CASE 17

a = 10,000 km i = 180 degrees



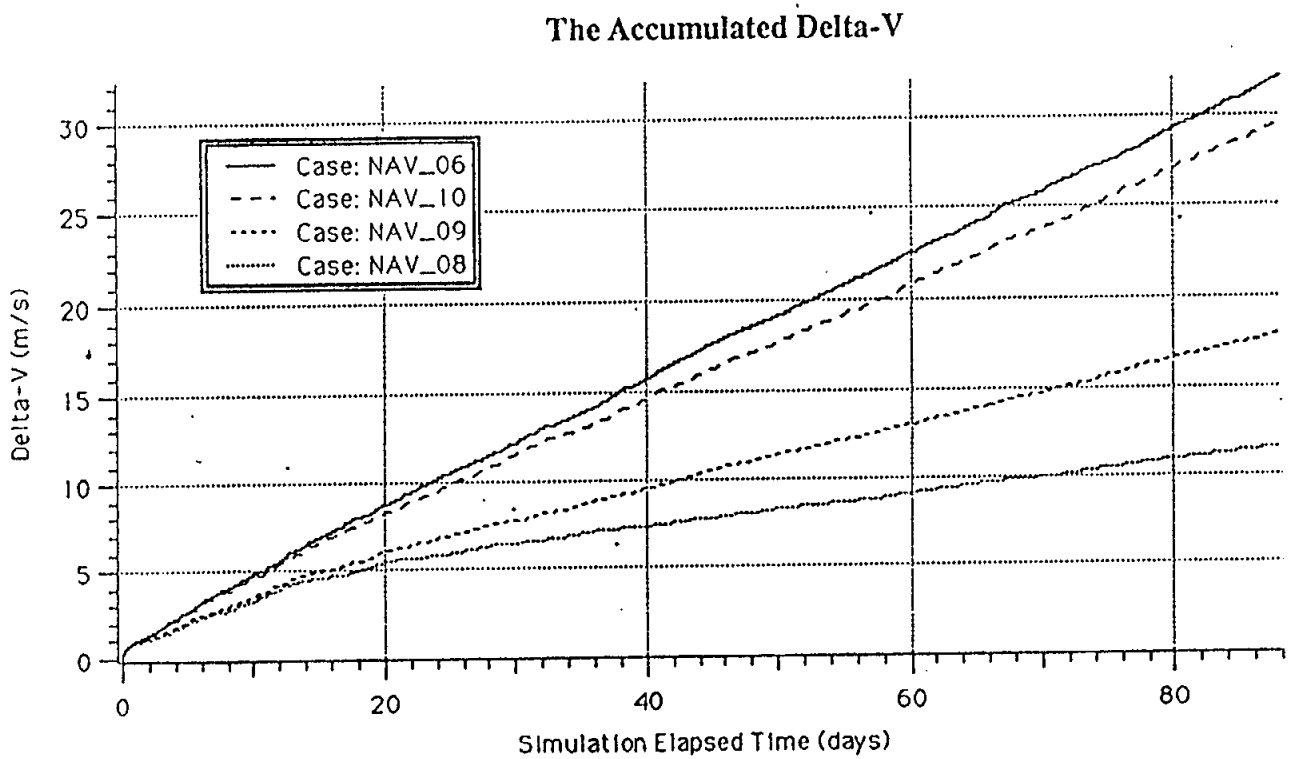
THE IMPACT OF DISCRETE MANEUVERS

- THE EFFECT OF DIFFERENT THRESHOLD VALUES ON THE CUMMULATIVE DELTA-V HISTORY IS SHOWN IN THE FOLLOWING PLOT.
  - ⇒ THE CASE WHERE THE THRESHOLD WAS SET TO 0.01 M/SEC WAS NOT SHOWN BECAUSE IT WAS VIRTUALLY INDISTINGUISHABLE FROM THE CASE WHERE THE THRESHOLD WAS SET TO 0.005 M/SEC.



THE COST OF NAVIGATION

- THE FOLLOWING GRAPH SHOWS THE CUMULATIVE DELTA-V HISTORY FOR SOME OF THE CASES ON THE PRECEDING PAGE.



THE COST OF NAVIGATION

- THE BASIC INDICATOR OF PERFORMANCE WAS TAKEN TO BE THE DELTA-V REQUIRED PER REV. (≈ 14 days)

- THIS STUDY INCLUDED TEN CASES. THE RESULTS OF THESE TEN RUNS ARE BEST SUMMARIZED IN THE FOLLOWING TABLE.

  ⇒ THE UNITS FOR THE TIME CONSTANT ARE MINUTES, ALL OTHER UNITS ARE IN METERS AND SECONDS. THE INTEGRATION TIME STEP FOR ALL CASES WAS 60 MINUTES.

CASE	TIME CONSTANT	R_x	R_y	R_z	R_x	R_x	R_x	$\lvert\Delta\underline{V}\rvert$ PER REV
NAV_05	600	1000	1000	1000	0.1	0.1	0.1	47.77
NAV_01	600	100	100	100	0.1	0.1	0.1	47.44
NAV_07	600	1000	1000	1000	0.01	0.01	0.01	6.65
NAV_03	600	100	100	100	0.01	0.01	0.01	5.01
NAV_06	60,000	1000	1000	1000	0.1	0.1	0.1	5.15
NAV_02	60,000	100	100	100	0.1	0.1	0.1	5.13
NAV_08	60,000	1000	1000	1000	0.01	0.01	0.01	1.34
NAV_04	60,000	100	100	100	0.01	0.01	0.01	1.26
NAV_10	60,000	1000	100	100	0.1	0.01	0.01	4.72
NAV_09	60,000	100	1000	1000	0.01	0.1	0.1	2.61

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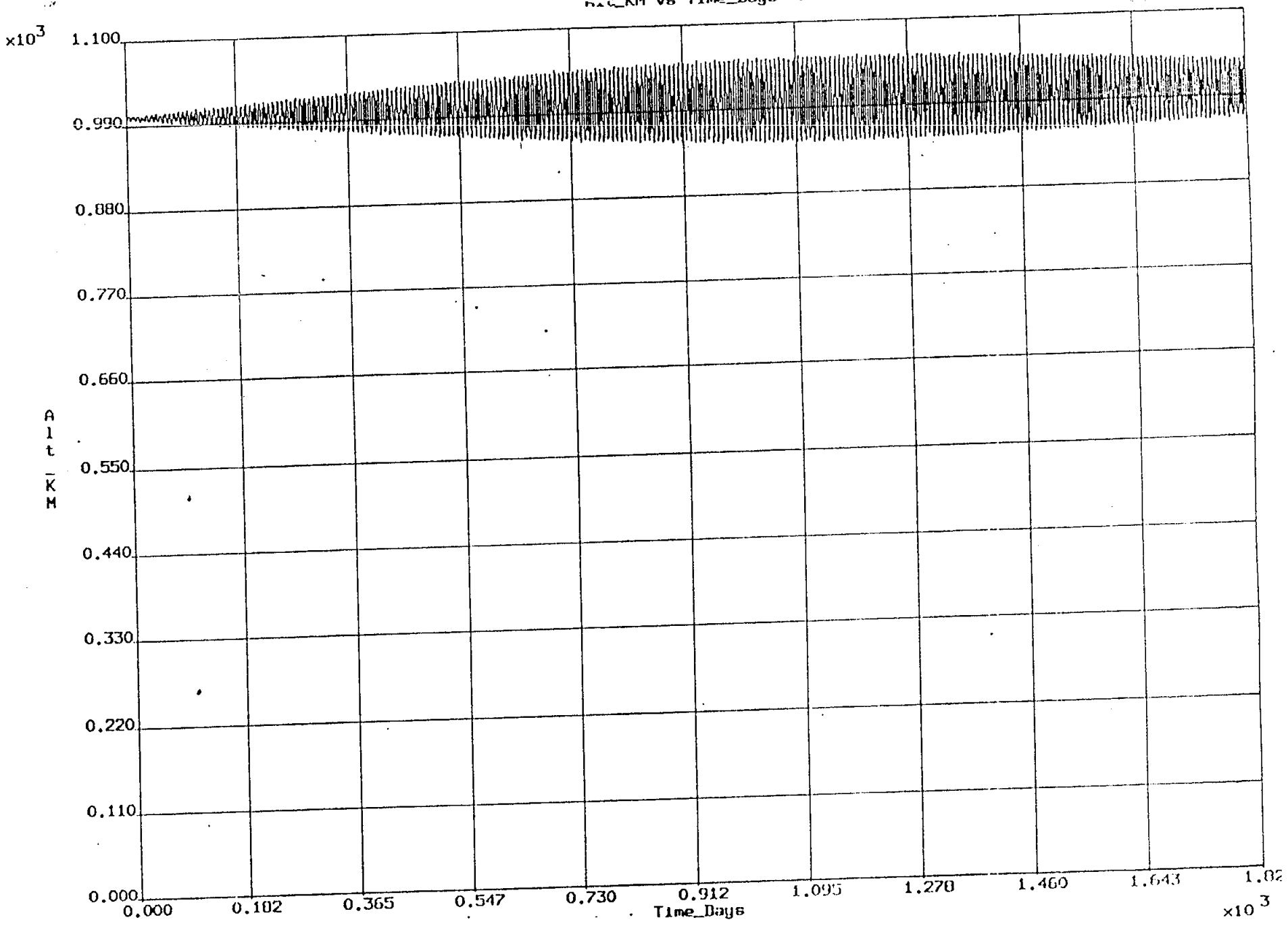
Circular, Polar Lunar Or 1000 KM (16x16 & Earth Pert)

Alt_KM vs Time_Days Bill + Ferrari



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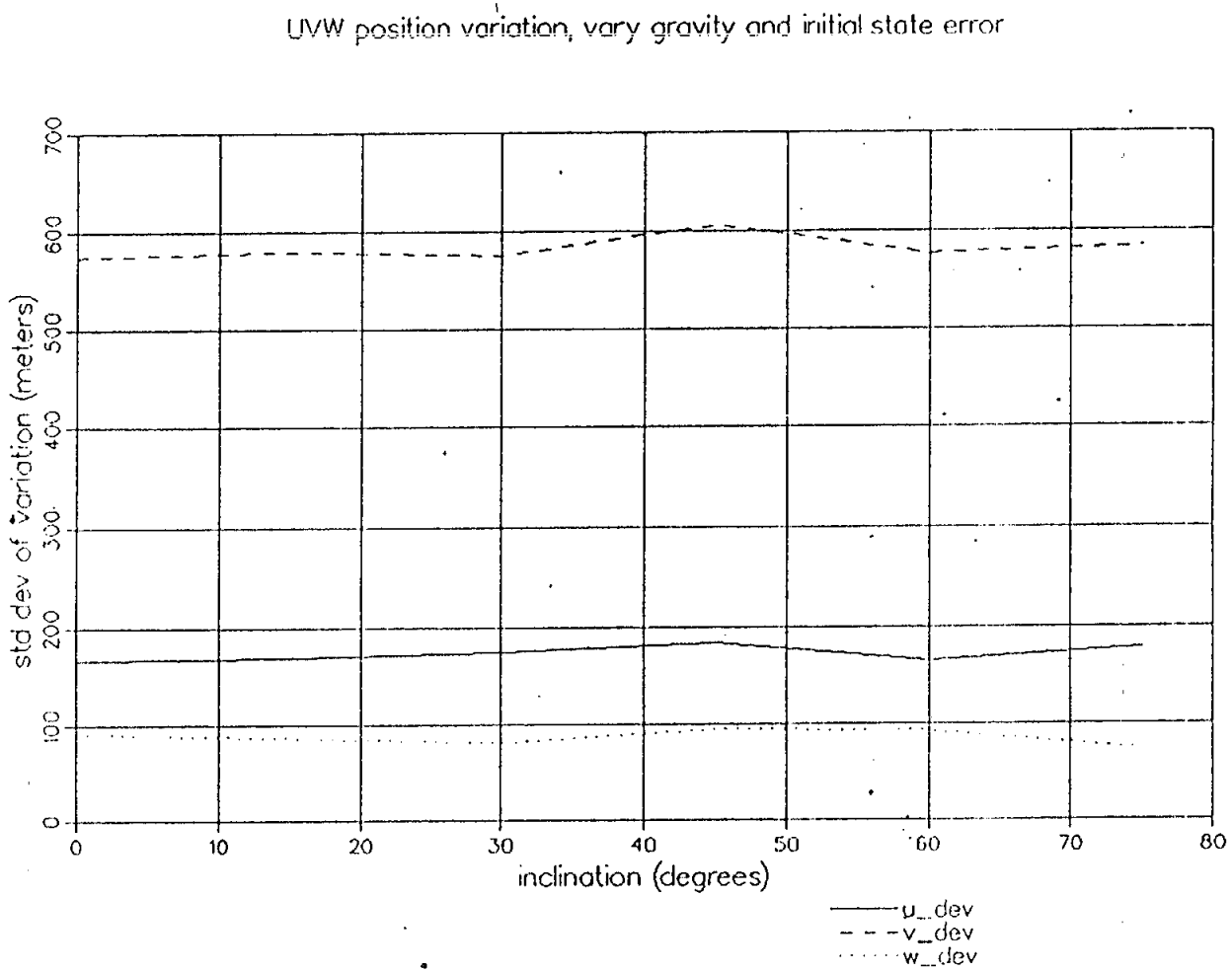
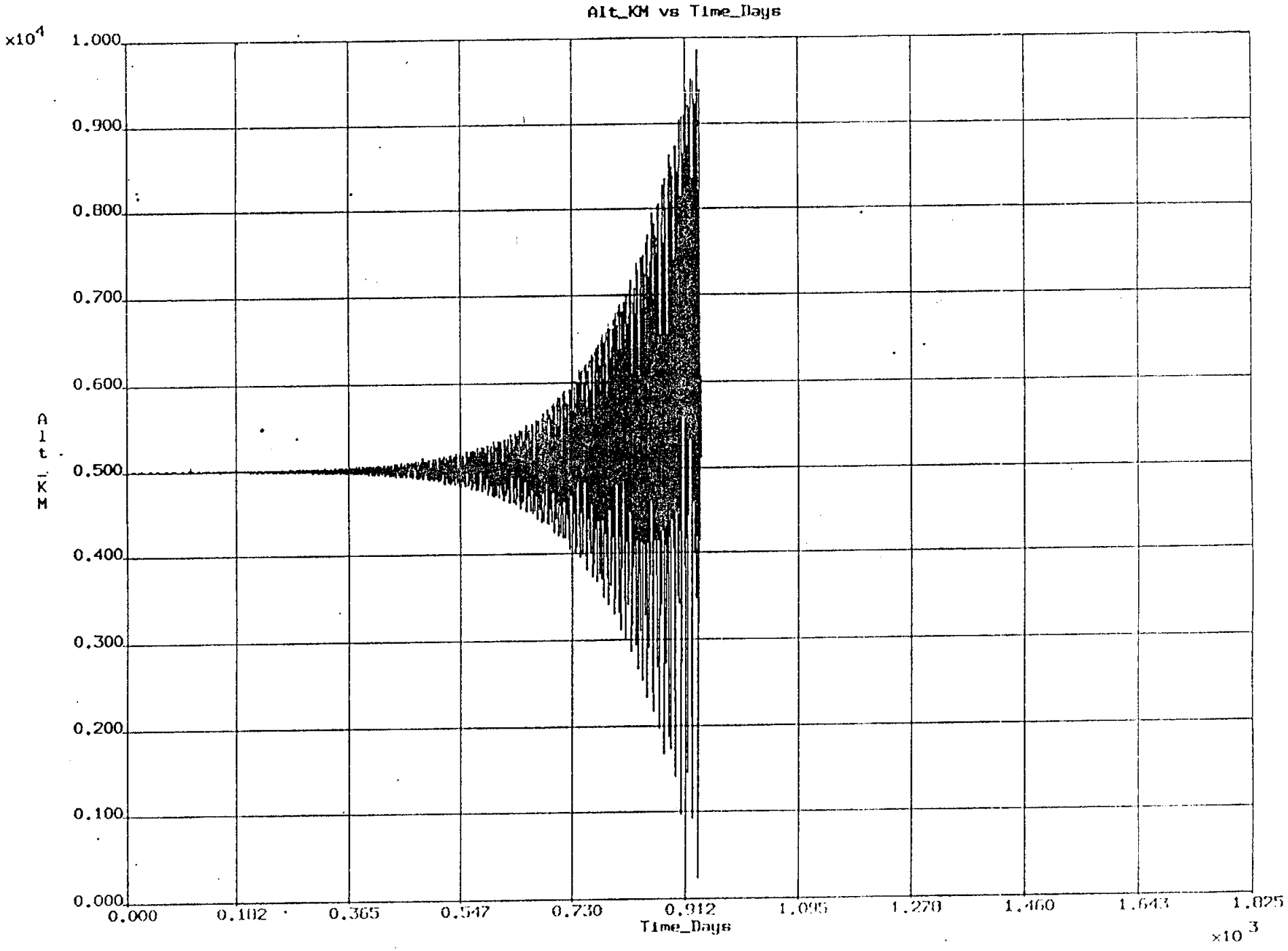
- AOS state uncertainty, varying gravity and initial state errors



Circular, Polar Lunar Orbit - 5000 KM (16x16 & Earth Pert)



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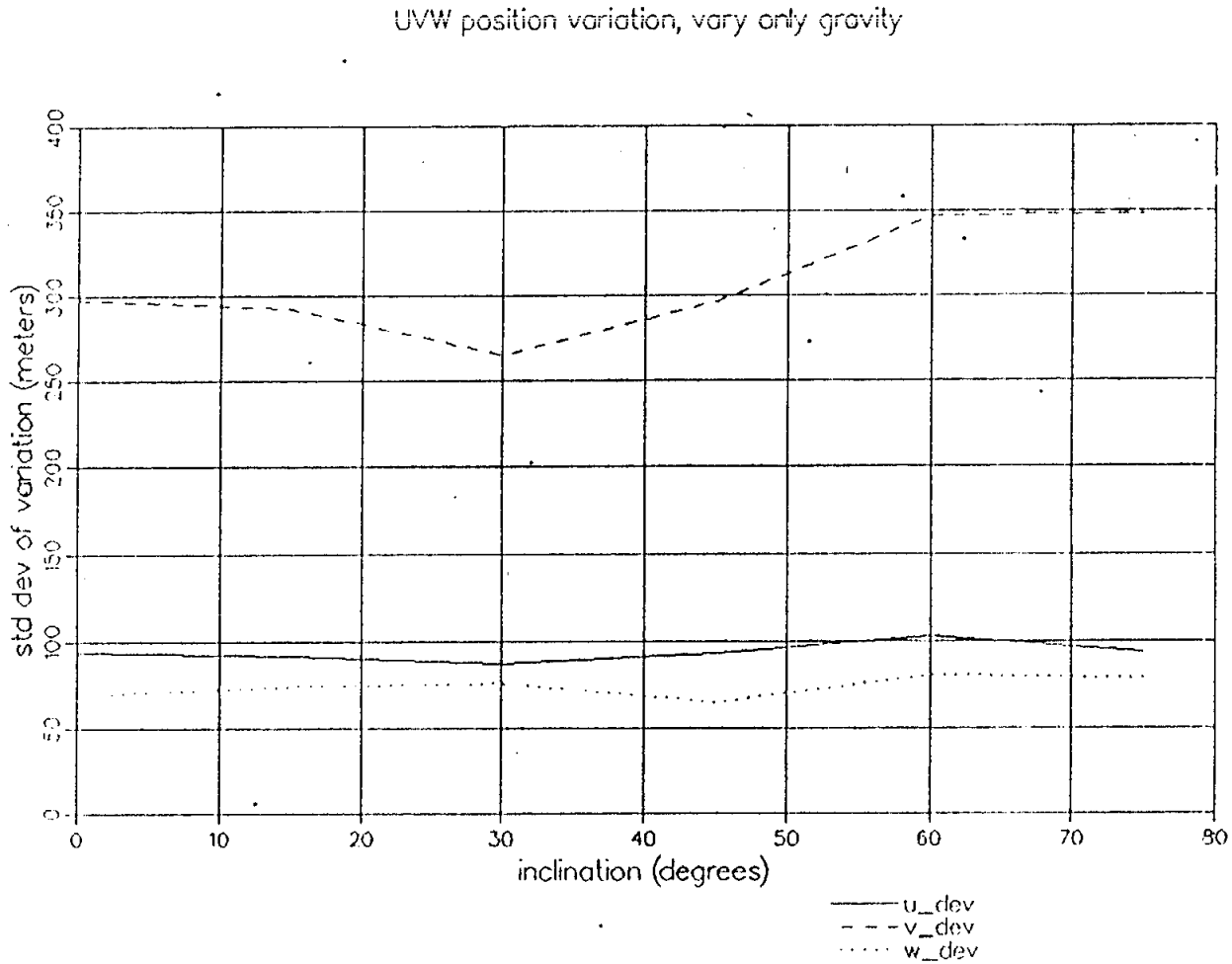
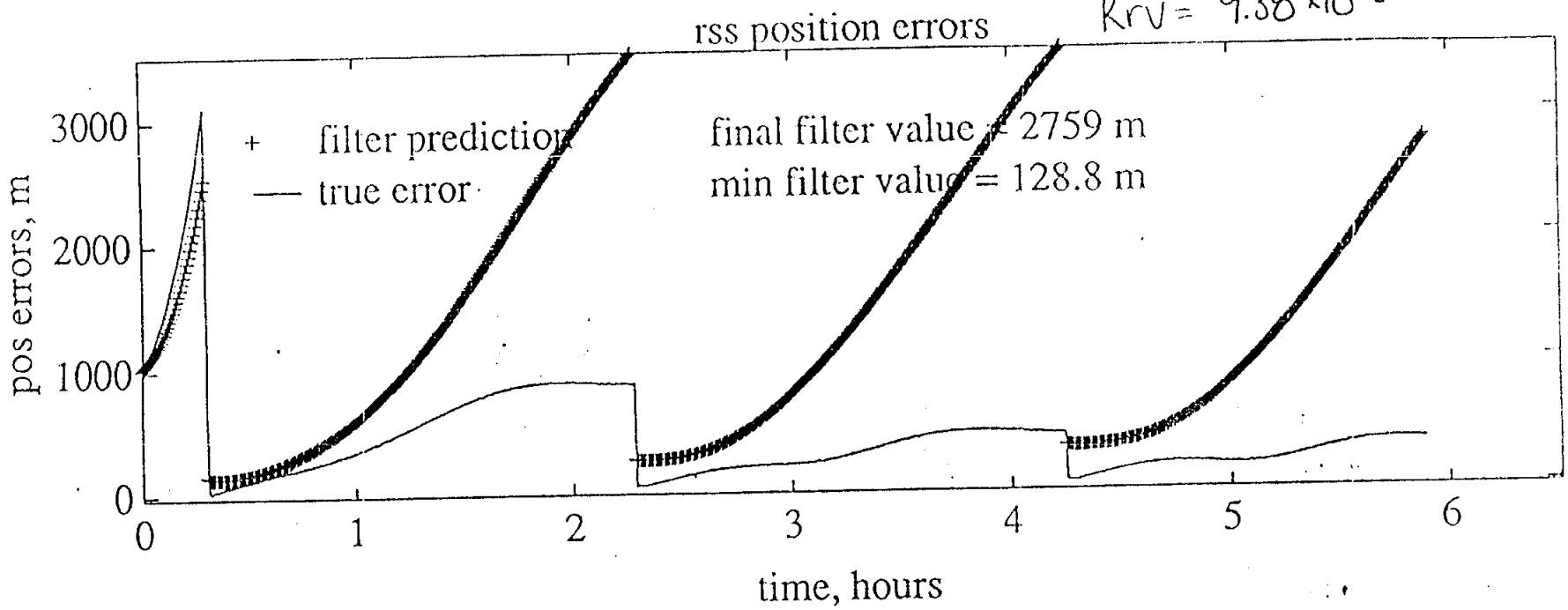
U.S. Gov't

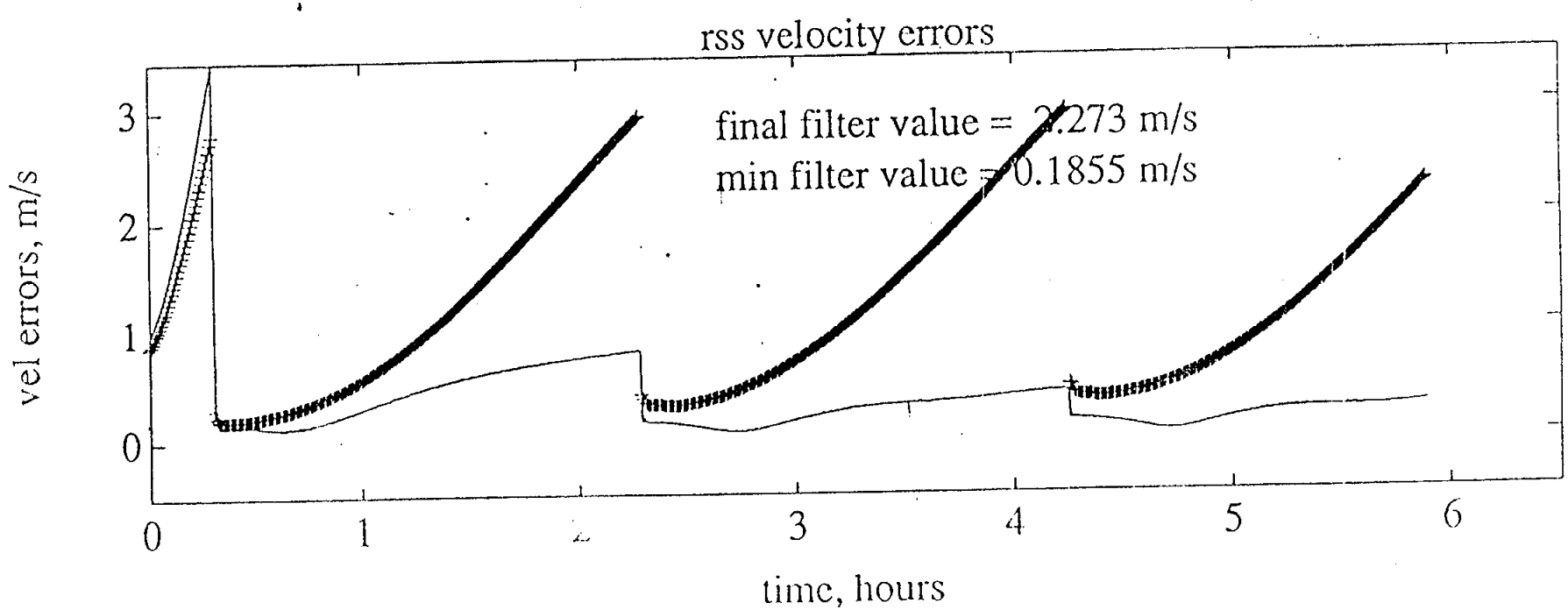
- AOS state uncertainty, varying only gravity



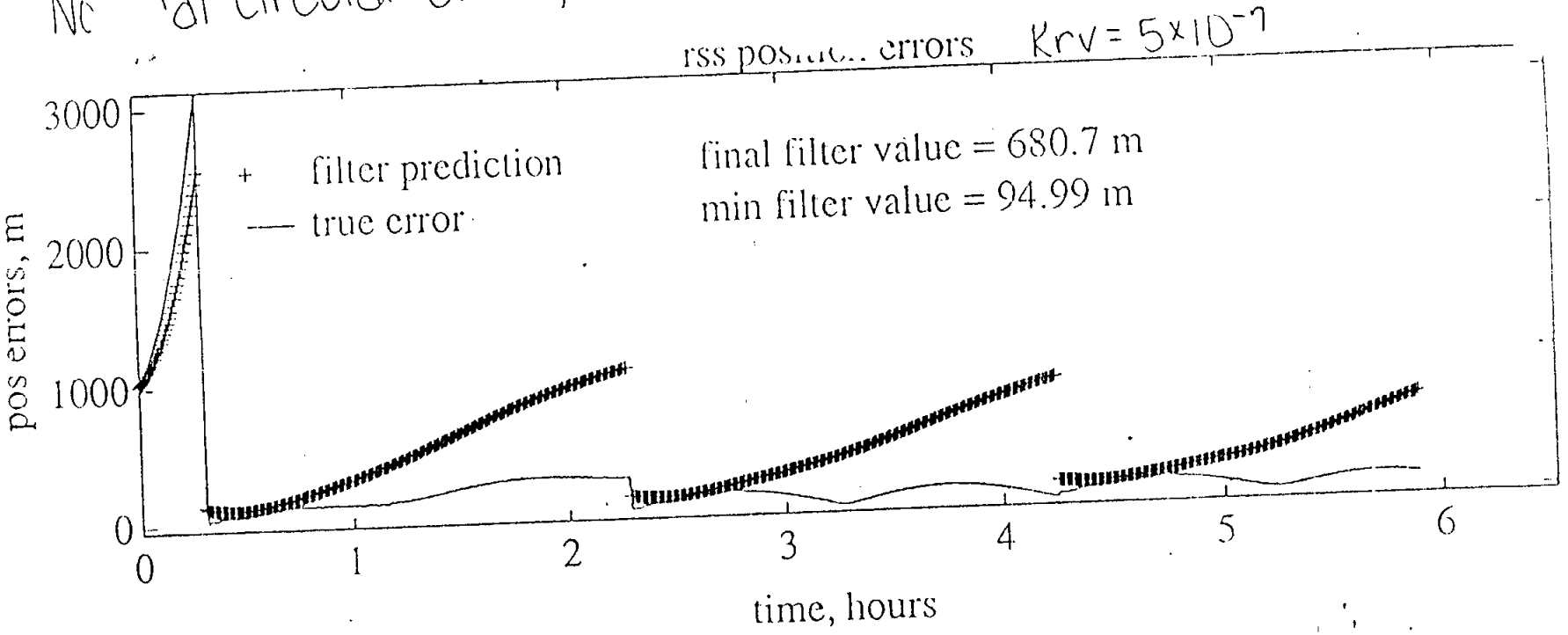
Nc. Tial Circular Orbit - DSN Covar[illegible] Init. - 1 Beacon at site

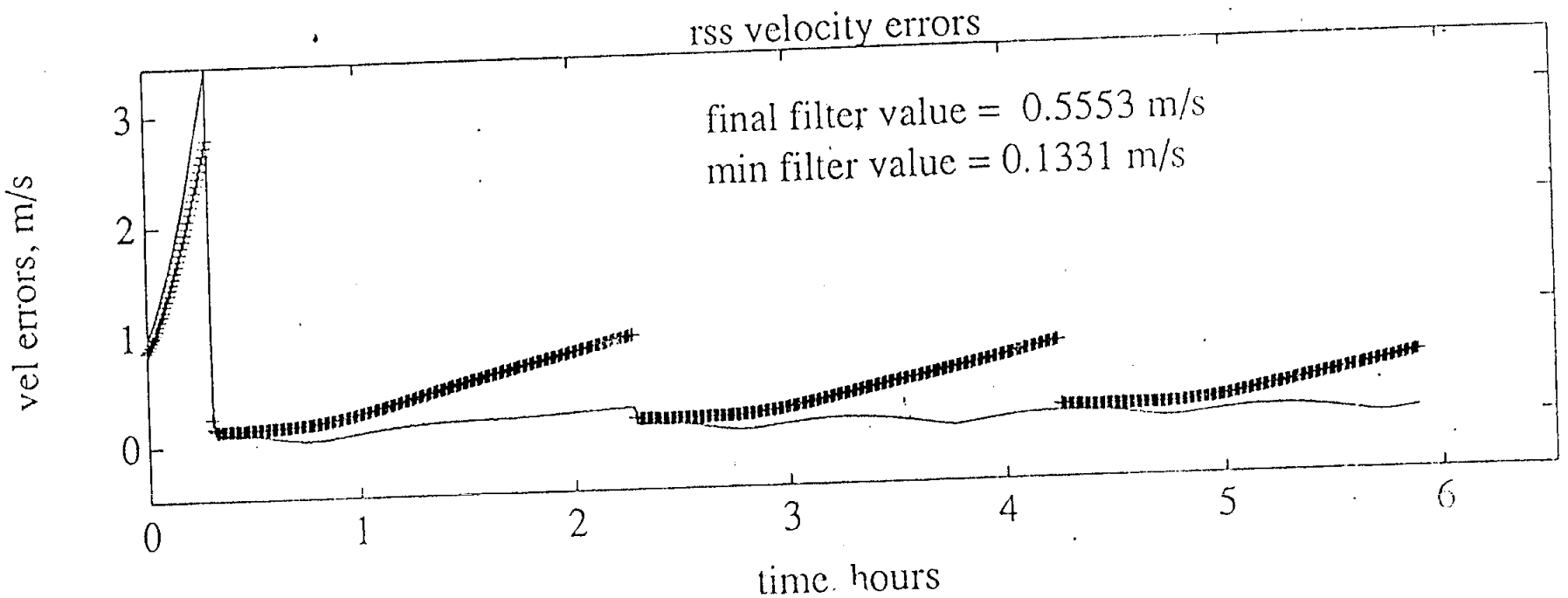
$K_{rv} = 9.38 \times 10^{-6}$





Nc ial Circular Orbit, DSN Covariance





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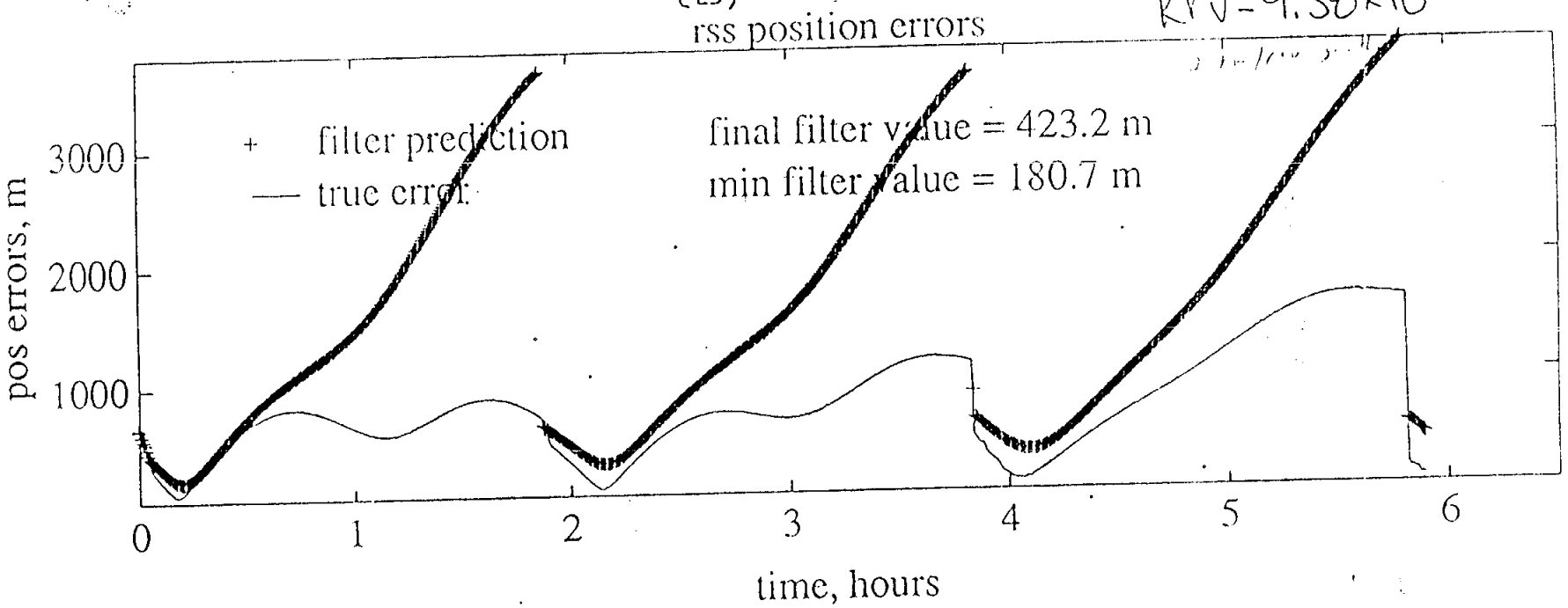
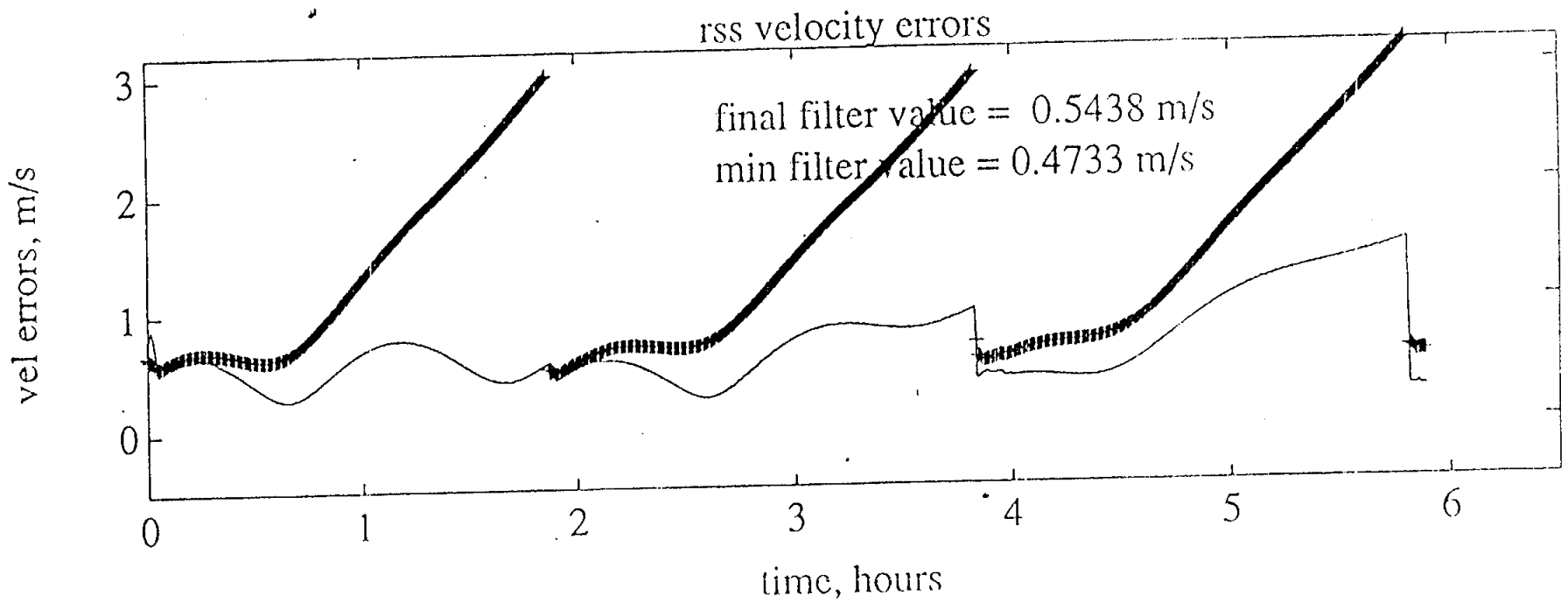
Final Circular Orbit - Clock (LS)

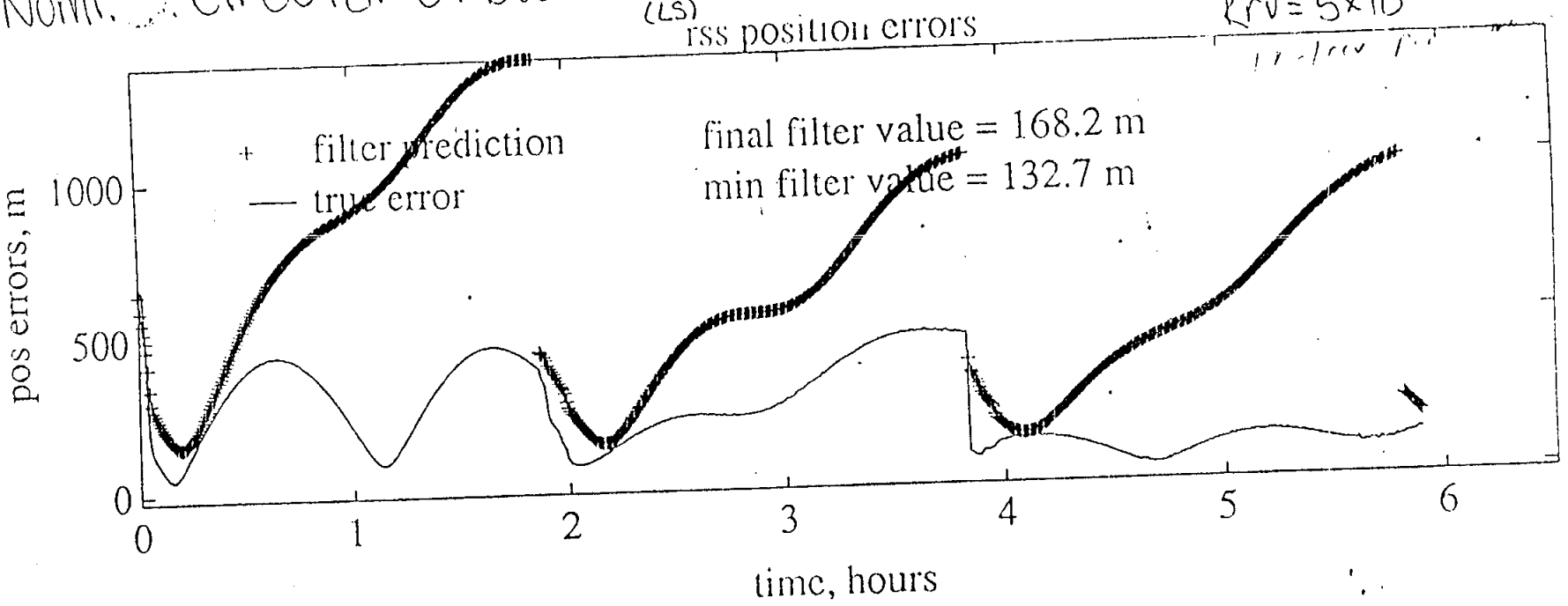
$K_{rv} = 9.38 \times 10^{-6}$

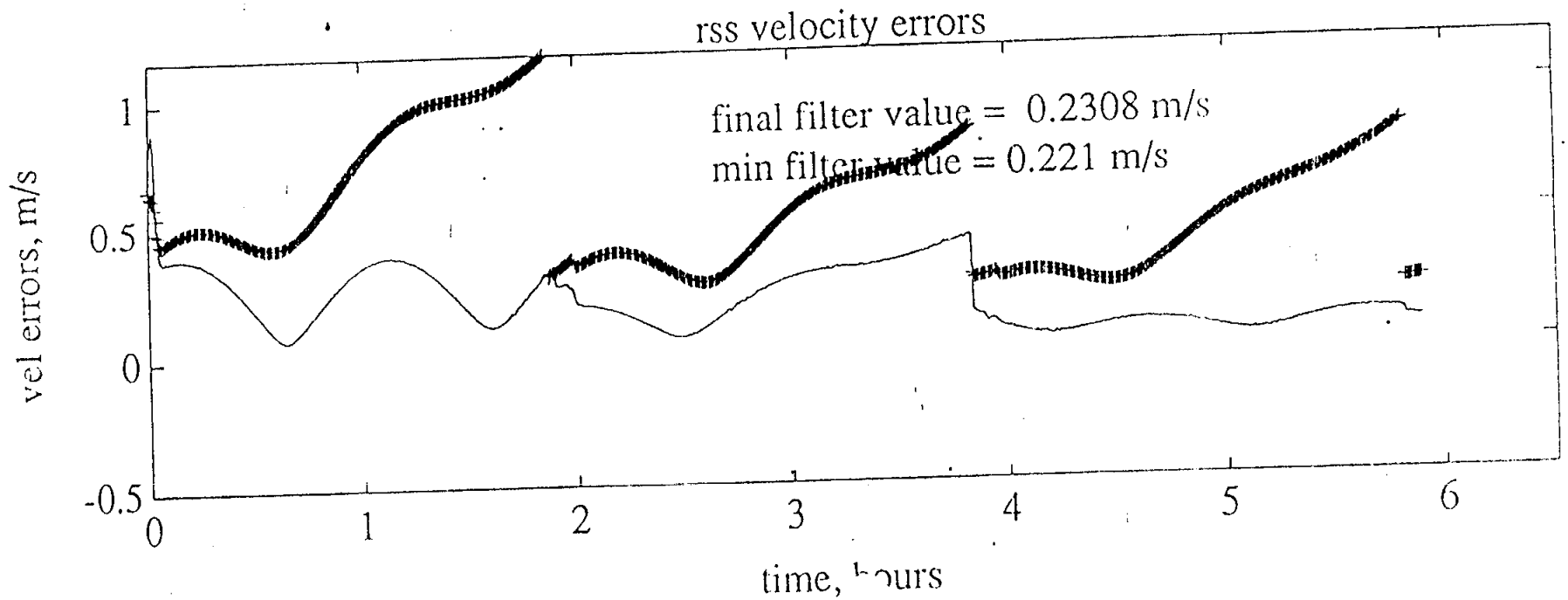




Nomi. Circular Orbit - Good B (LS) - 2 Init. - 1 Bounded

$K_{rv} = 5 \times 10^{-7}$





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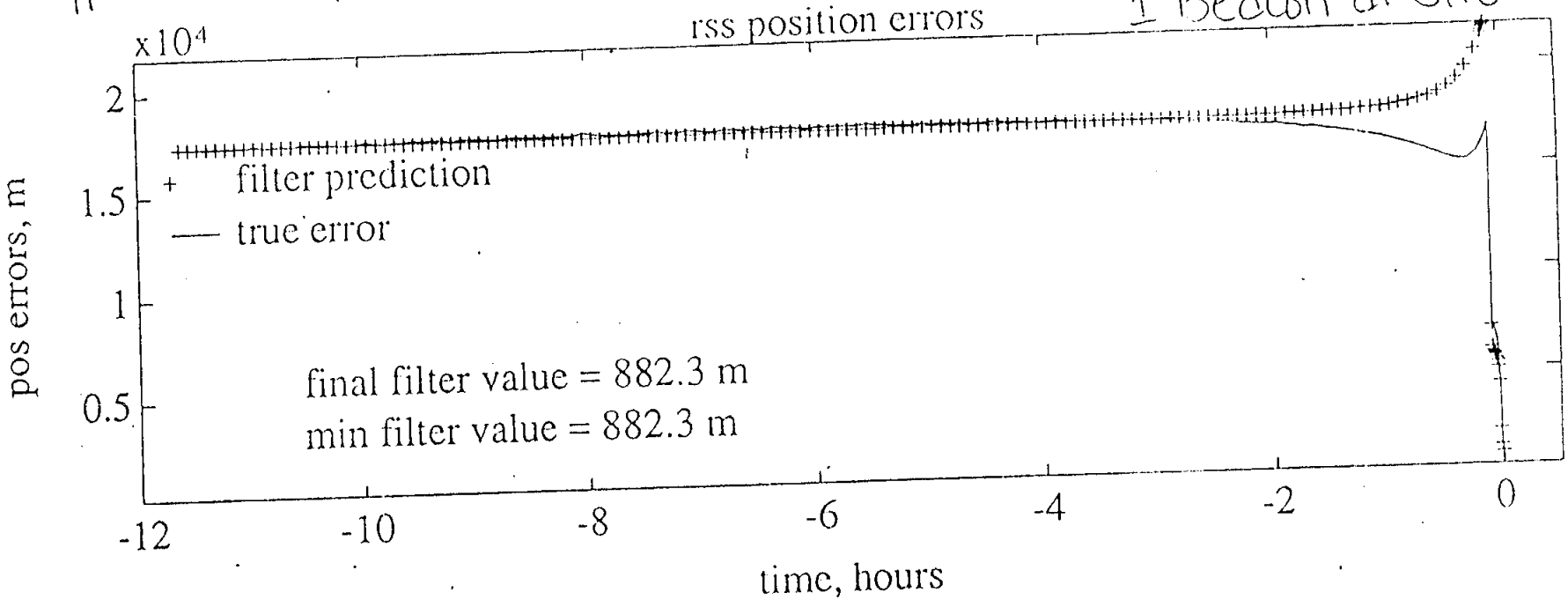
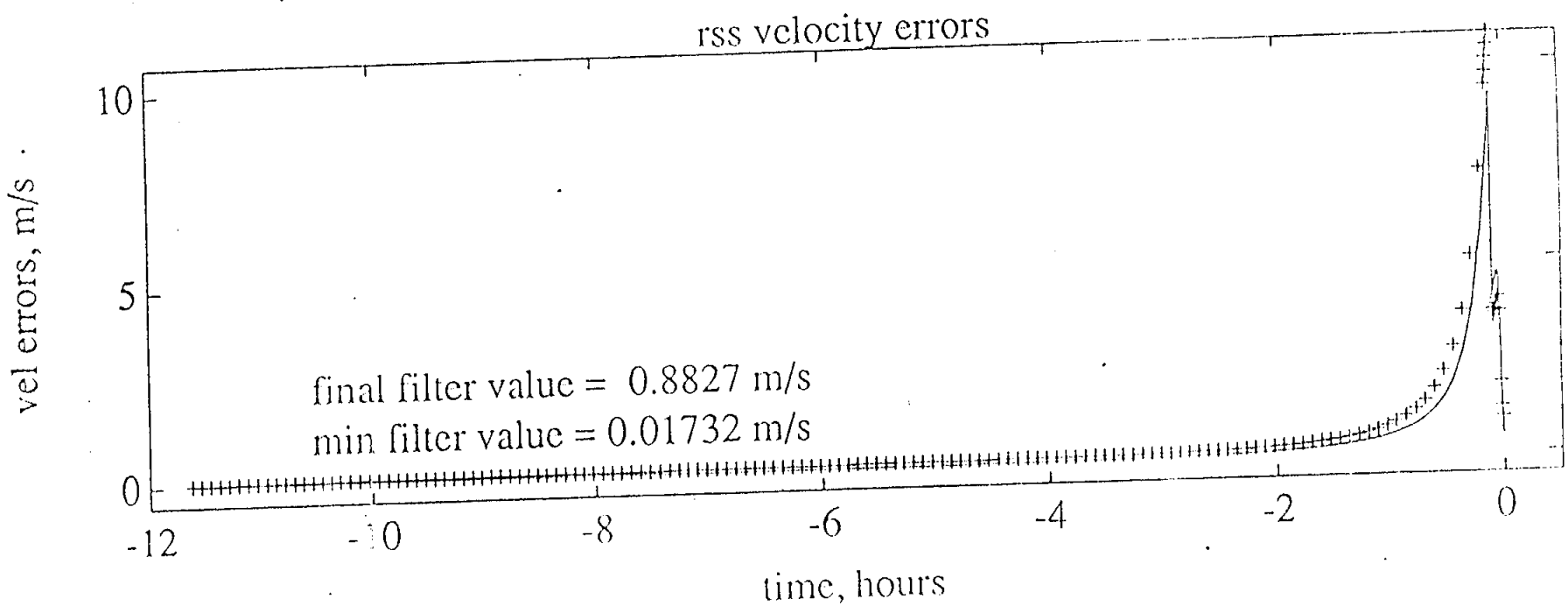
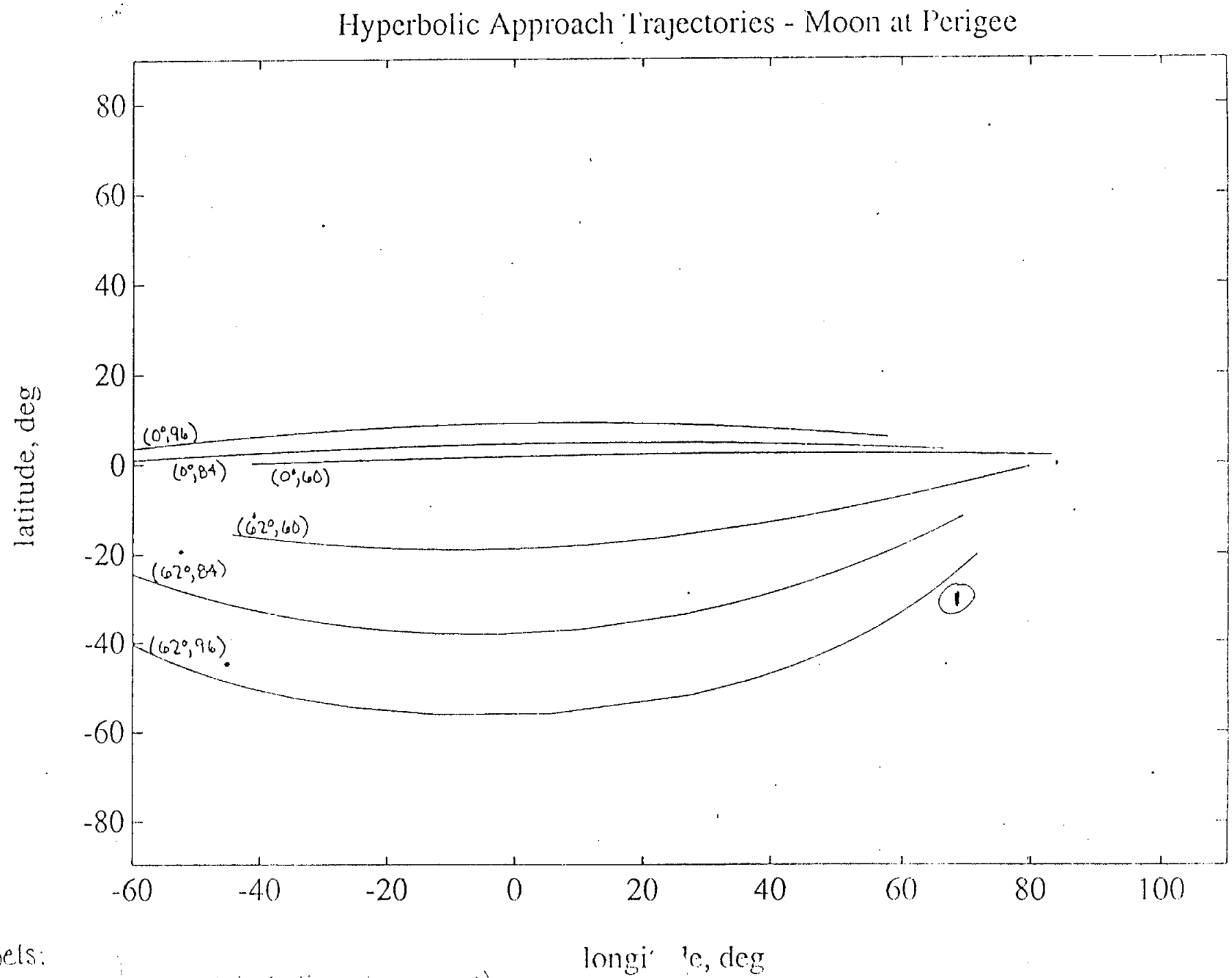
Hyperbolic Approach - Moon [illegible] rpogee, LTO inc = 62°, TOF = 6

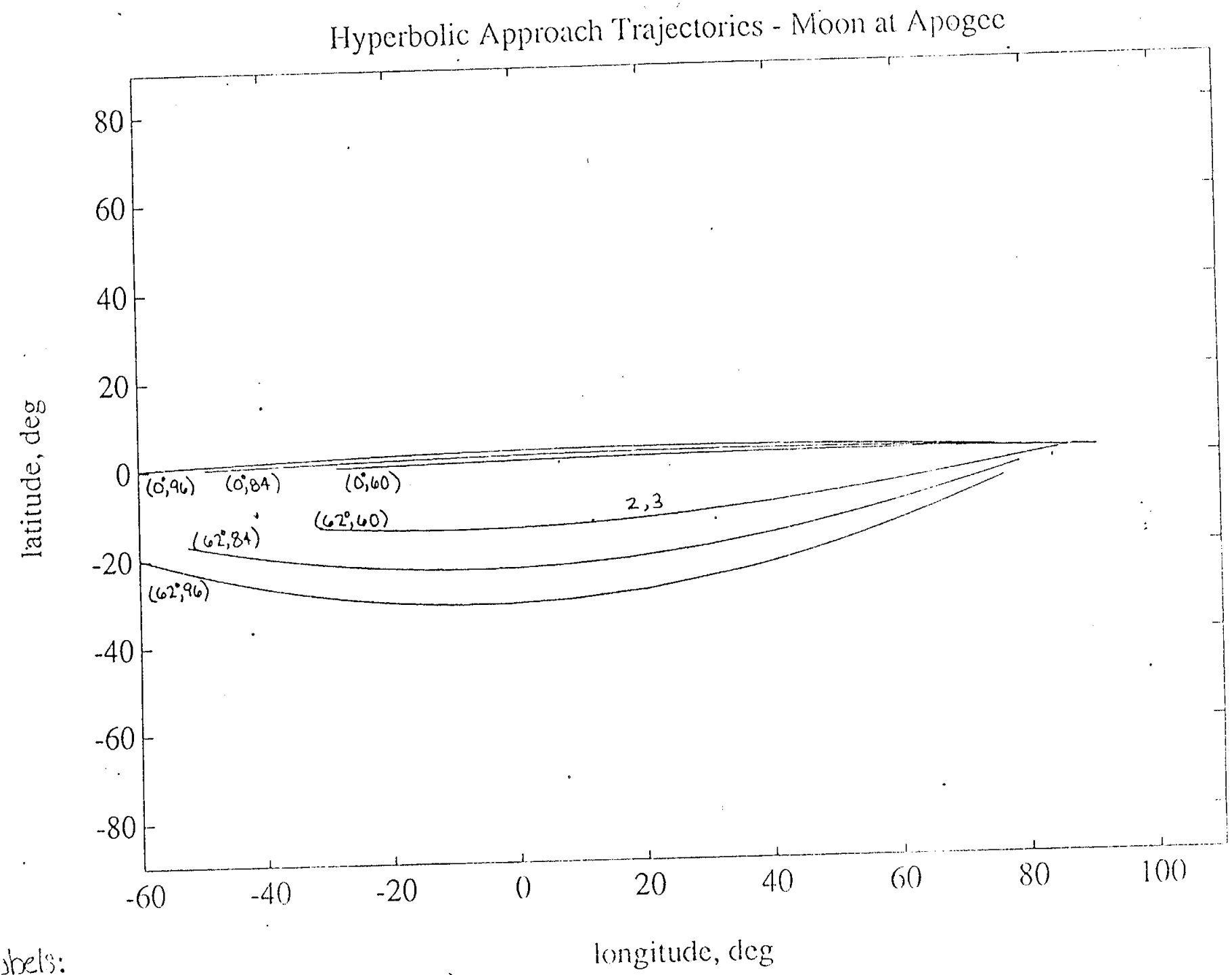
1 Beacon at Site





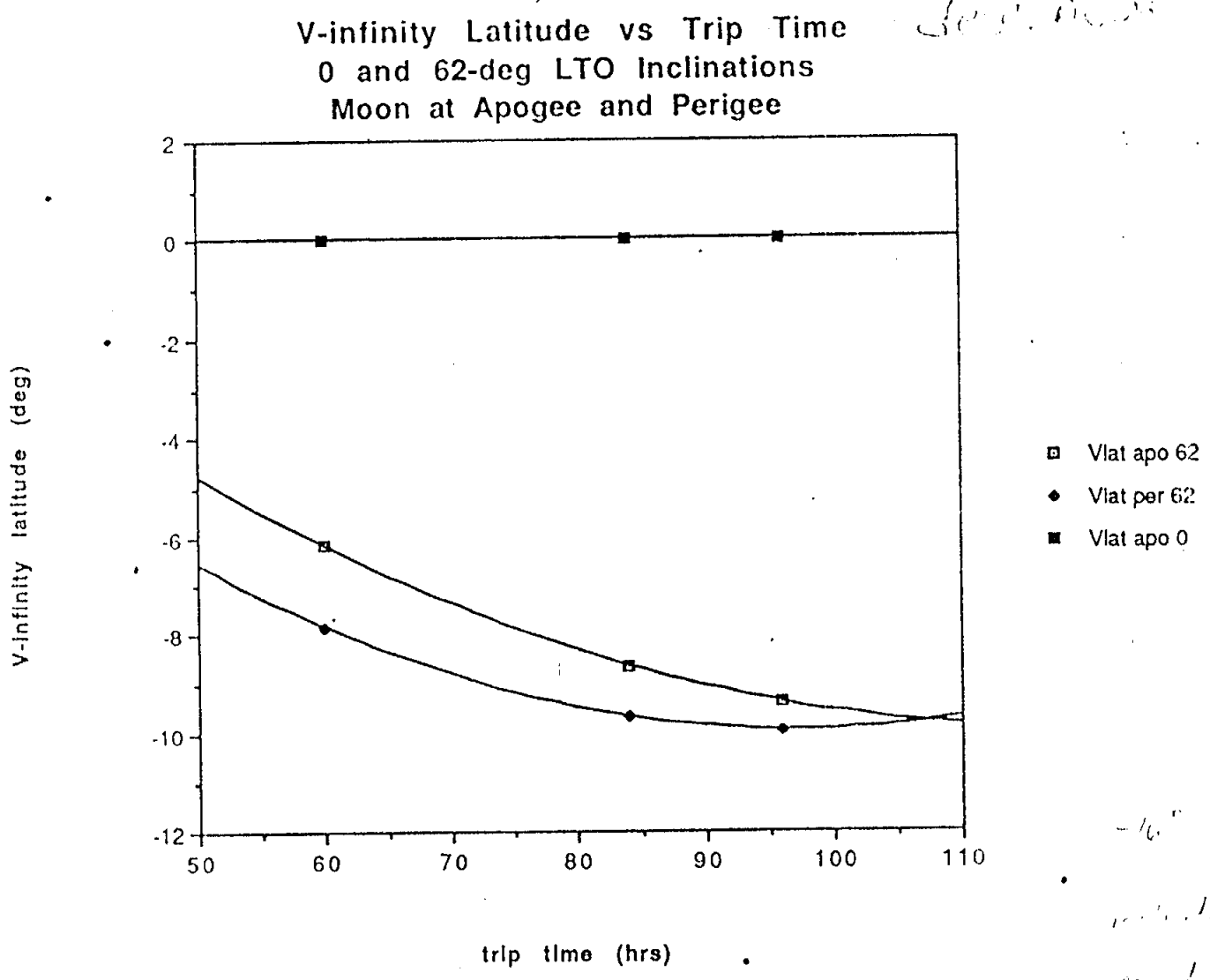


bels:
(,) = (transfer orbit inclination, trip time (hrs))

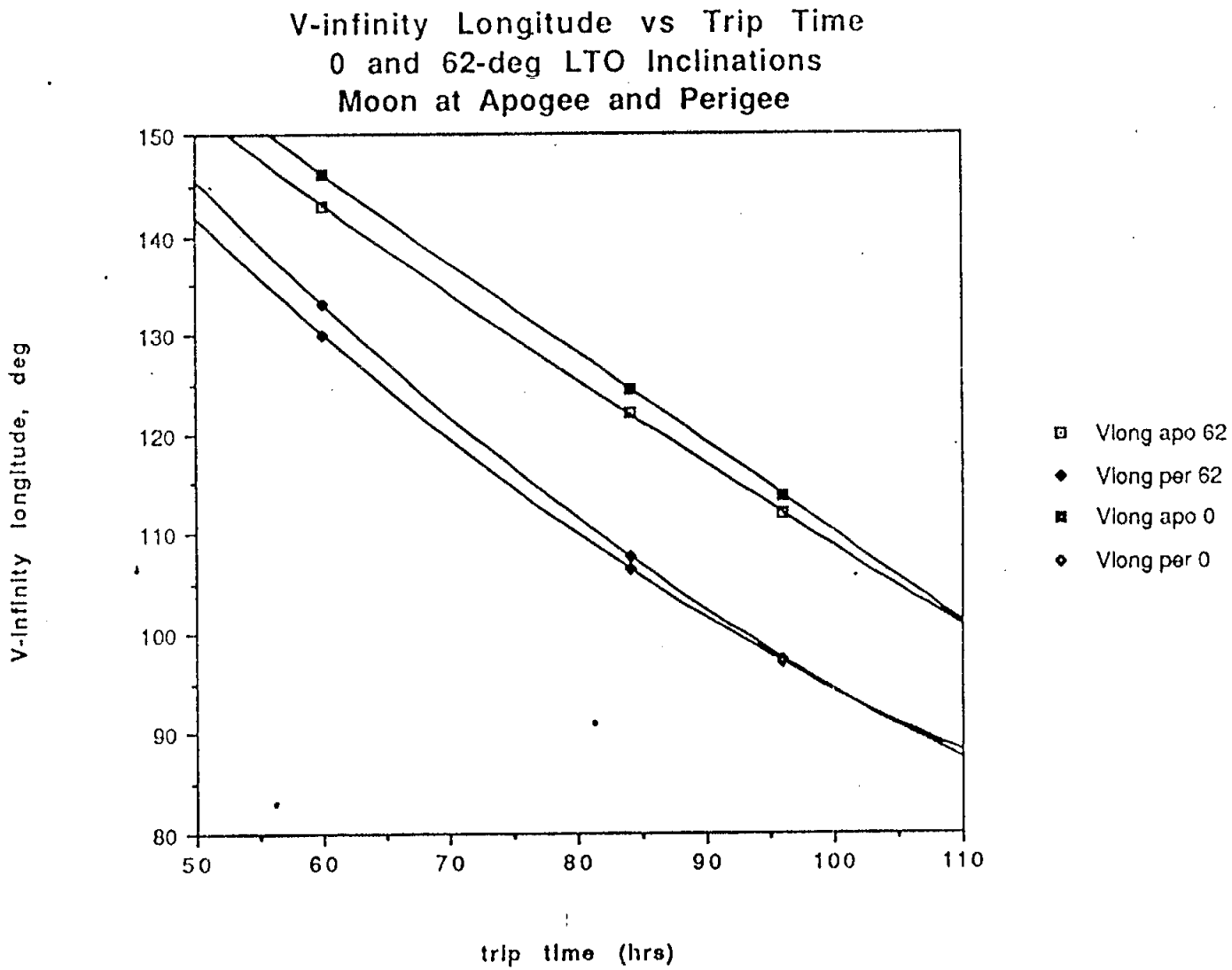


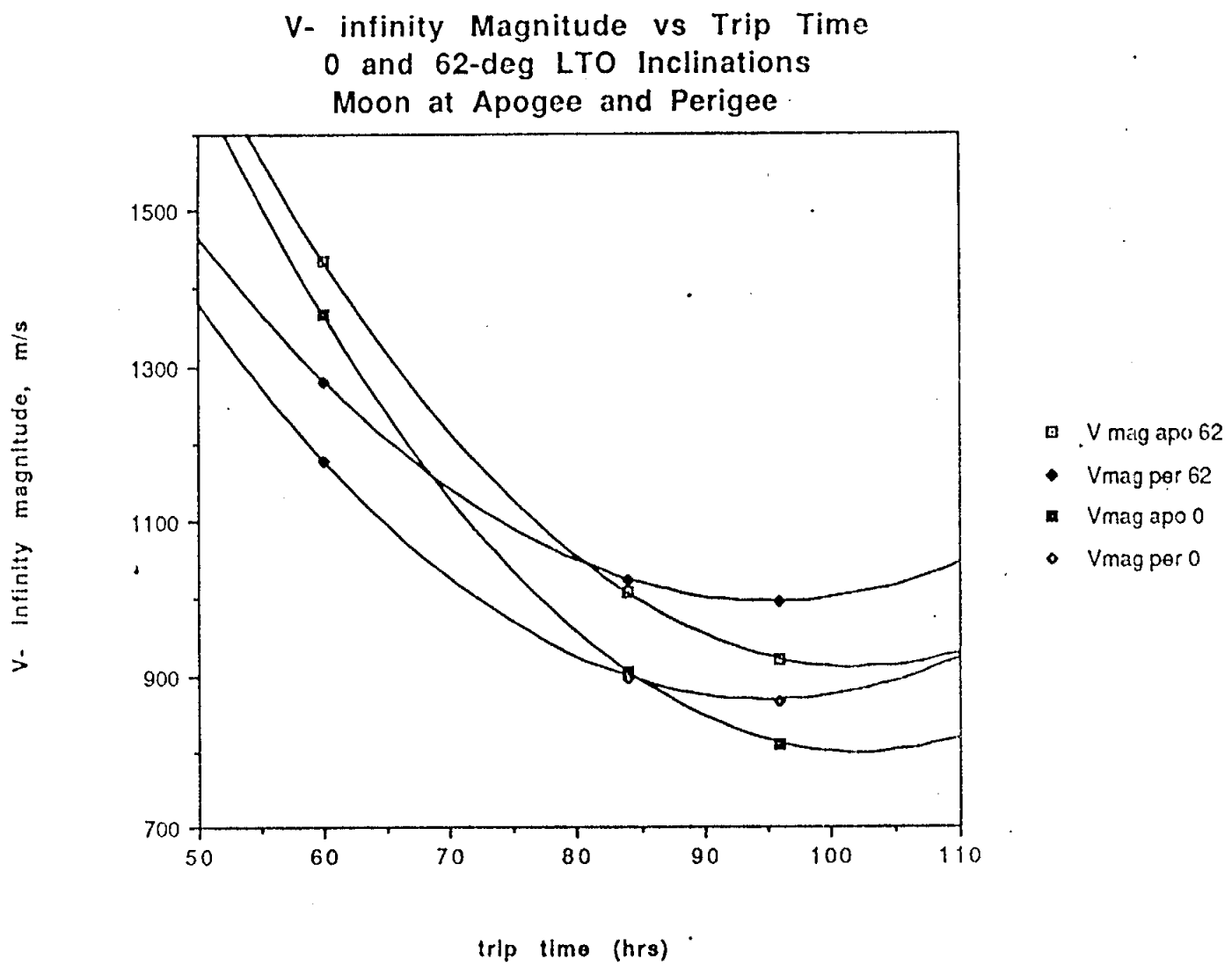
Labels:

(a,b) = (transfer orbit inclination, trip time (hrs))


V-infinity Latitude vs Trip Time
0 and 62-deg LTO Inclinations
Moon at Apogee and Perigee
V-infinity latitude (deg)
2
0
-2
-4
-6
-8
-10
-12
50
60
70
80
90
100
110
trip time (hrs)
Vlat apo 62
Vlat per 62
Vlat apo 0

S. Gov't


V-infinity Longitude vs Trip Time
0 and 62-deg LTO Inclinations
Moon at Apogee and Perigee
V-infinity longitude, deg
150
140
130
120
110
100
90
80
50
60
70
80
90
100
110
trip time (hrs)
Vlong apo 62
Vlong per 62
Vlong apo 0
Vlong per 0

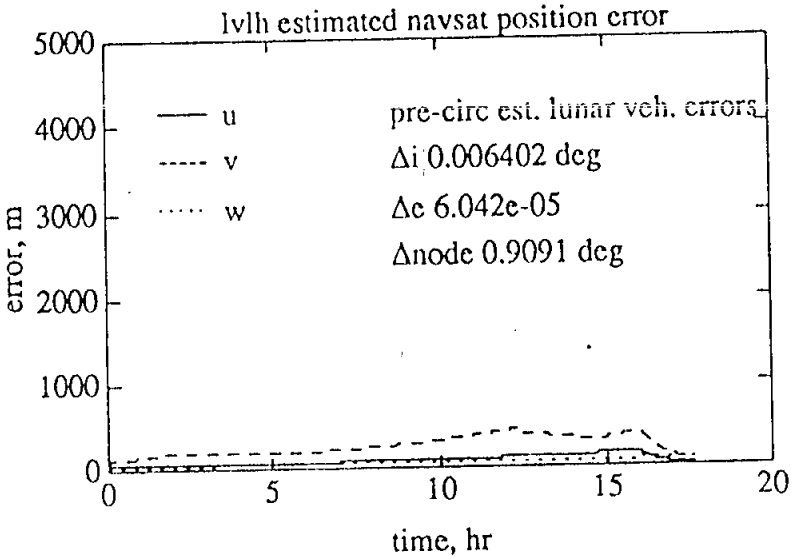

V- infinity Magnitude vs Trip Time
0 and 62-deg LTO Inclinations
Moon at Apogee and Perigee
V- infinity magnitude, m/s
1500
1300
1100
900
700
50
60
70
80
90
100
110
trip time (hrs)
V mag apo 62
Vmag per 62
Vmag apo 0
Vmag per 0

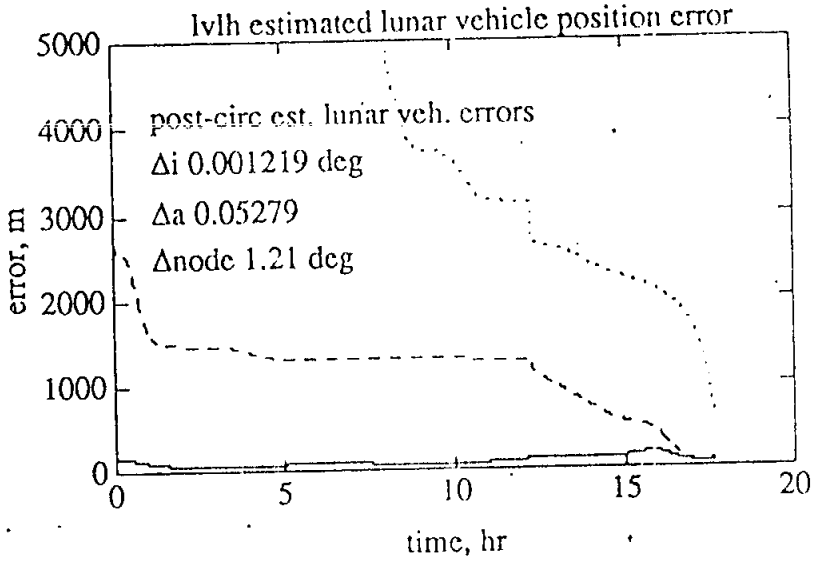
LUNAR SURFACE BEACON NAVIGATION STUDY

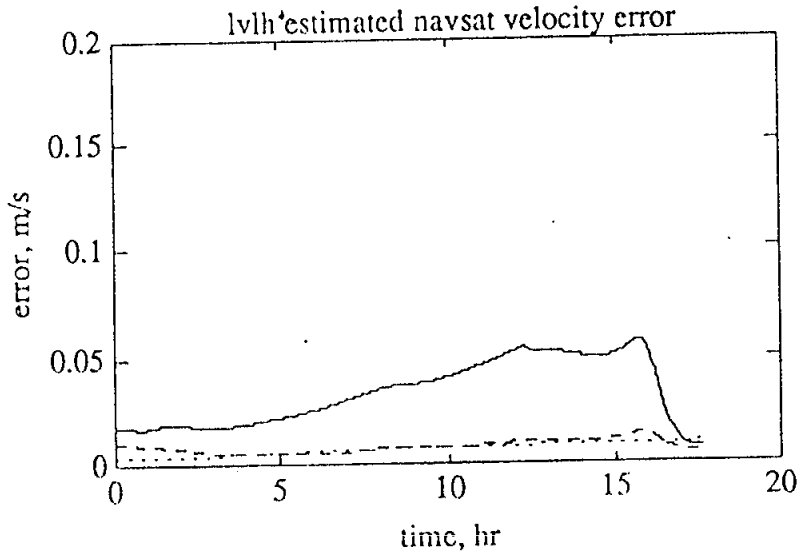
SED Technical Status Review

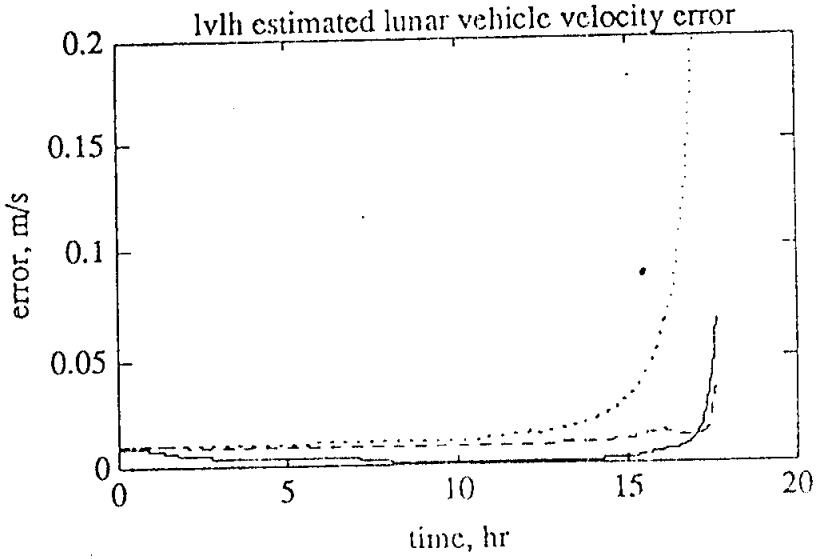
Michelle M. Munk - ET4
February 25, 1992

CASE 4









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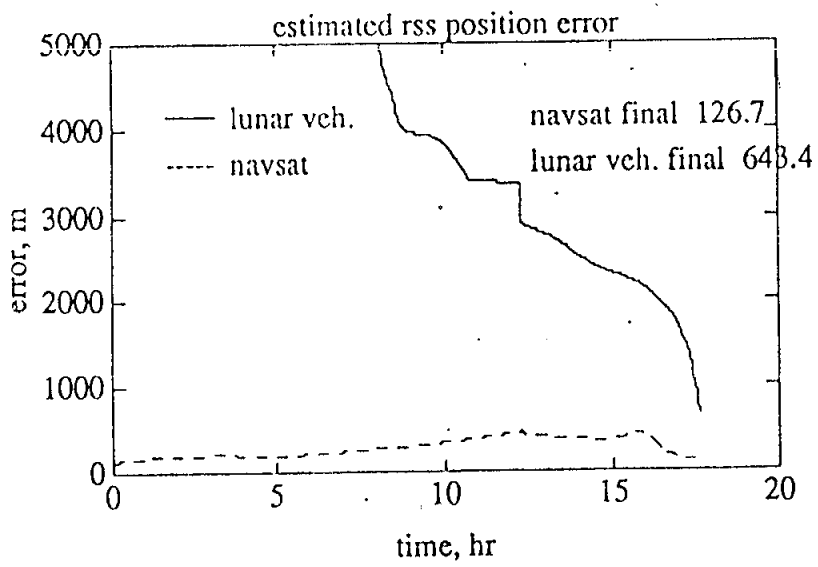
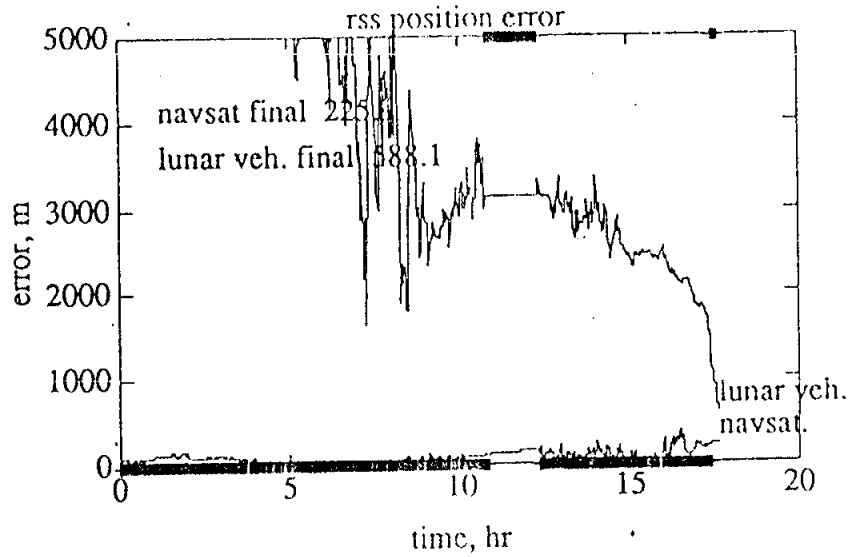
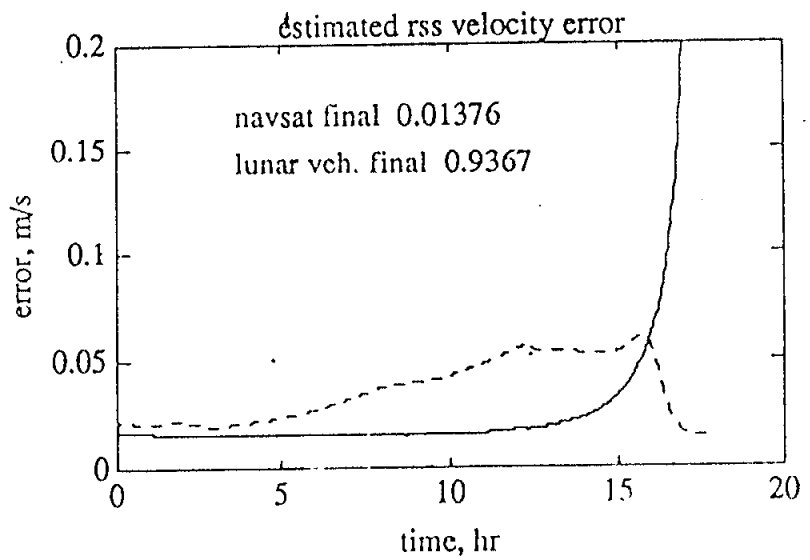
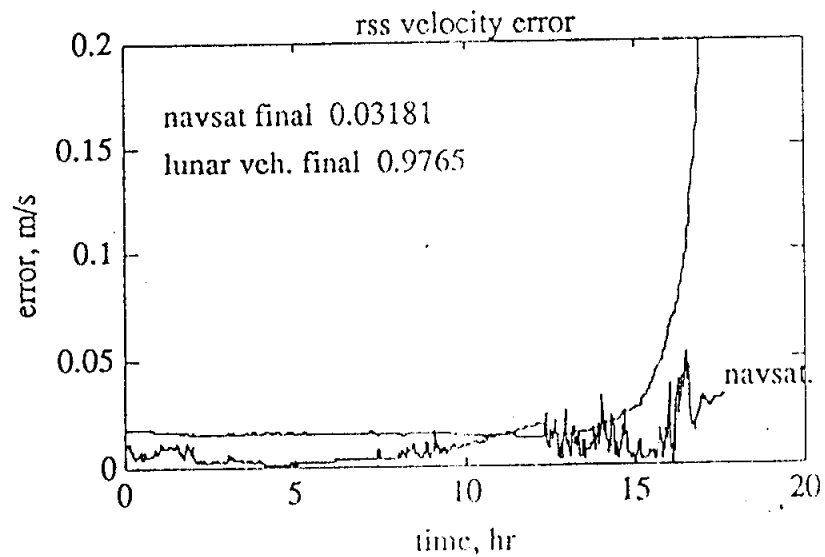
CASE 4

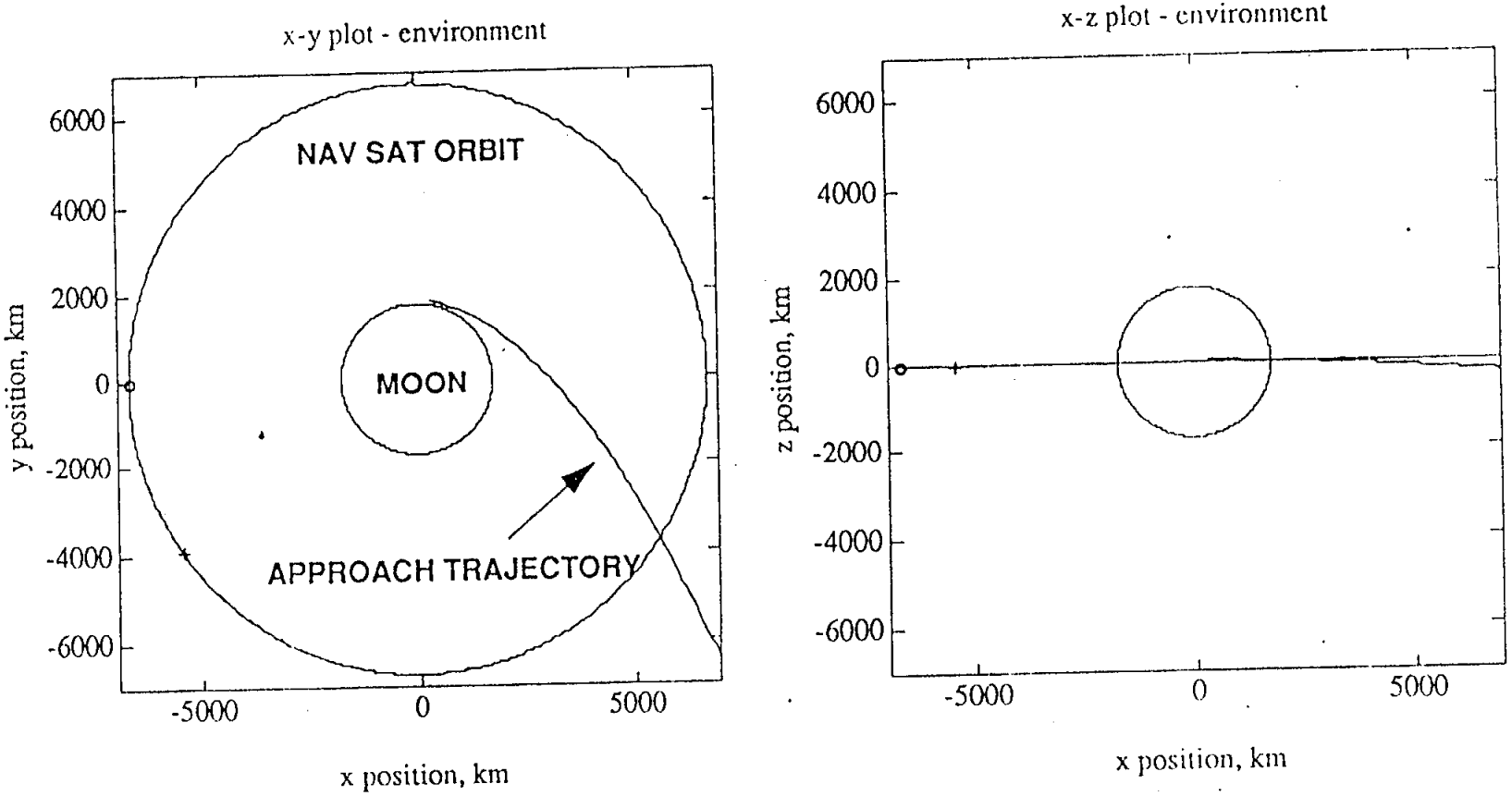

estimated rss position error
lunar veh.
navsat
navsat final 126.7
lunar veh. final 643.4
error, m
time, hr


rss position error
navsat final 225.4
lunar veh. final 588.1
lunar veh.
navsat
error, m
time, hr


estimated rss velocity error
navsat final 0.01376
lunar veh. final 0.9367
error, m/s
time, hr

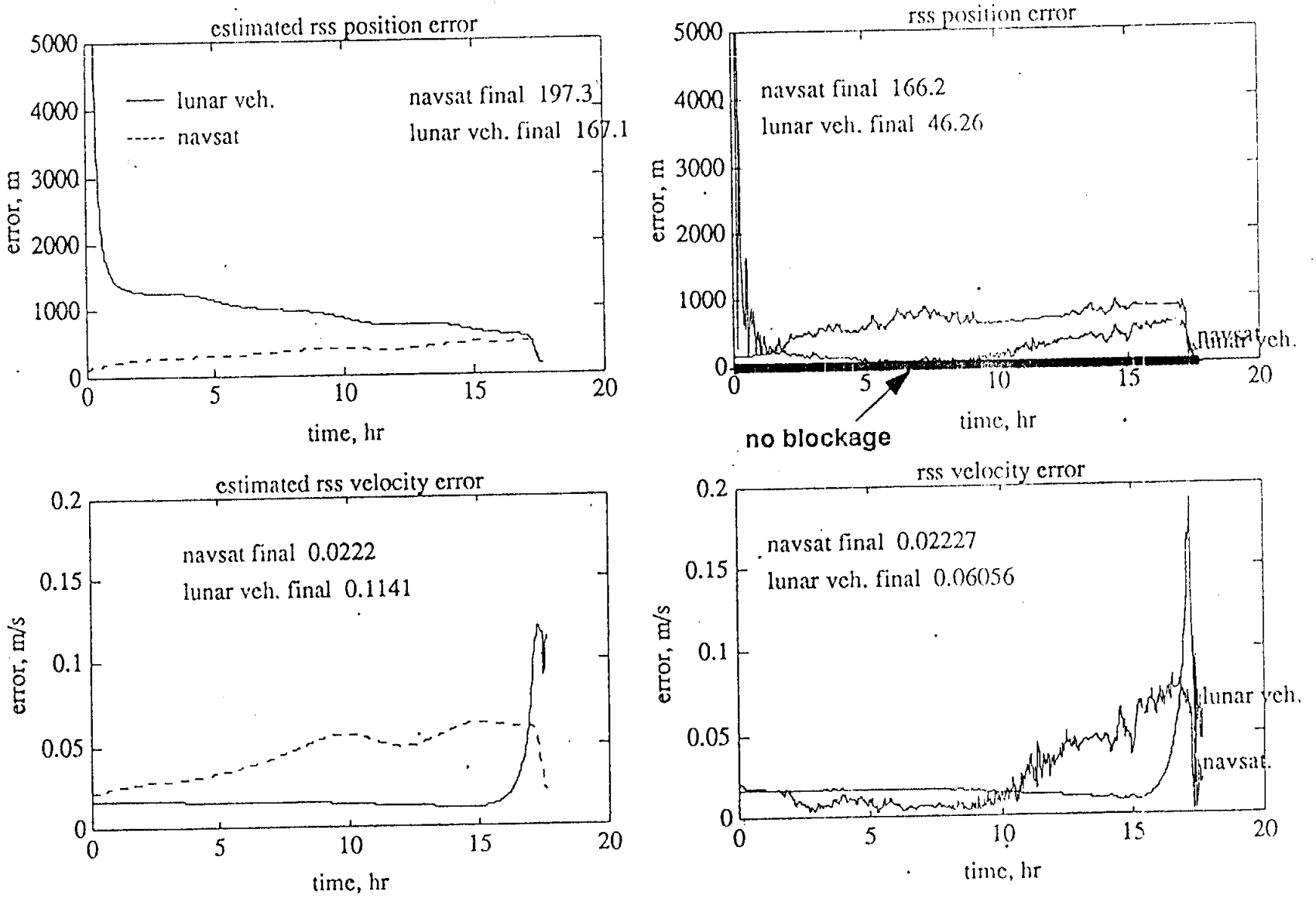

rss velocity error
navsat final 0.03181
lunar veh. final 0.9765
navsat
error, m/s
time, hr

CASE 4



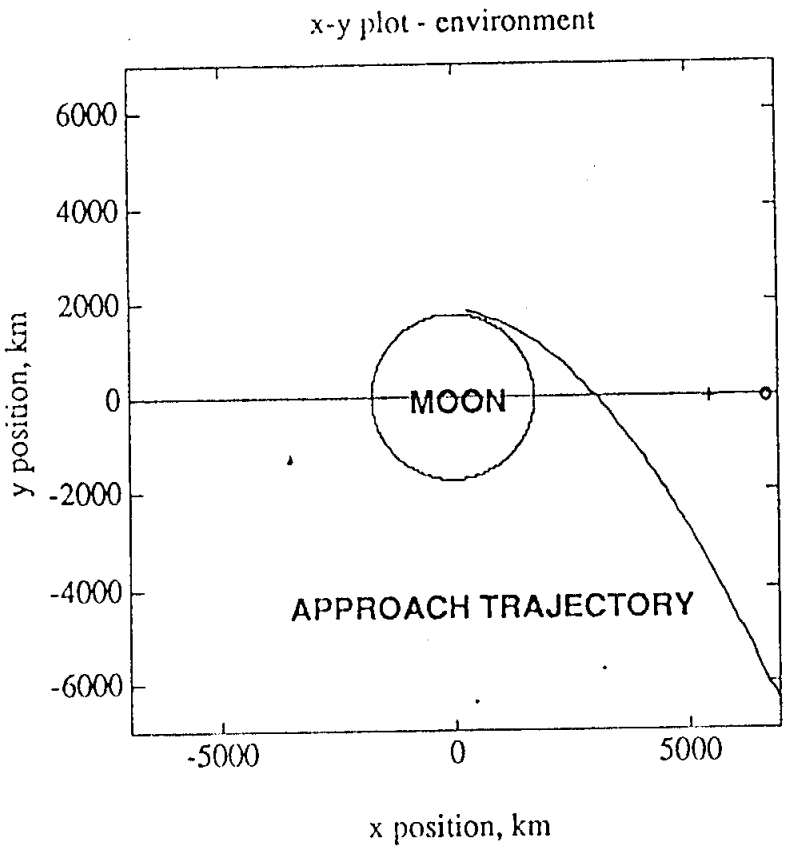
o indicates start, + indicates direction of travel

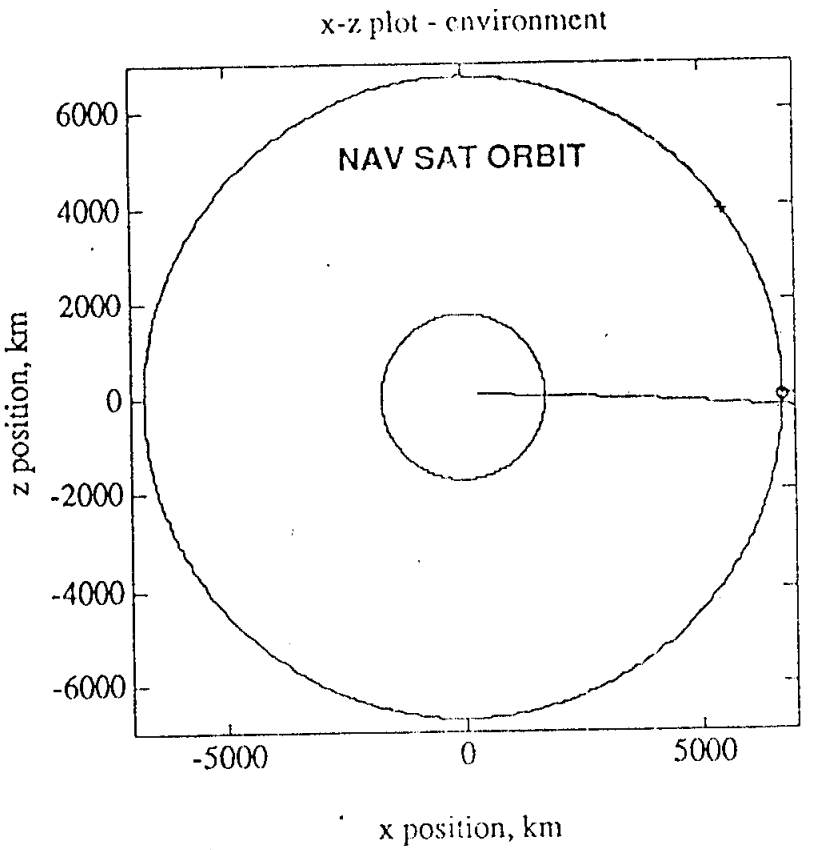
CASE 1



Broad band in position error plot indicates measurements blocked by Moon if at 1 , clear measurements if at 0

CASE 1





o indicates start, + indicates direction of travel

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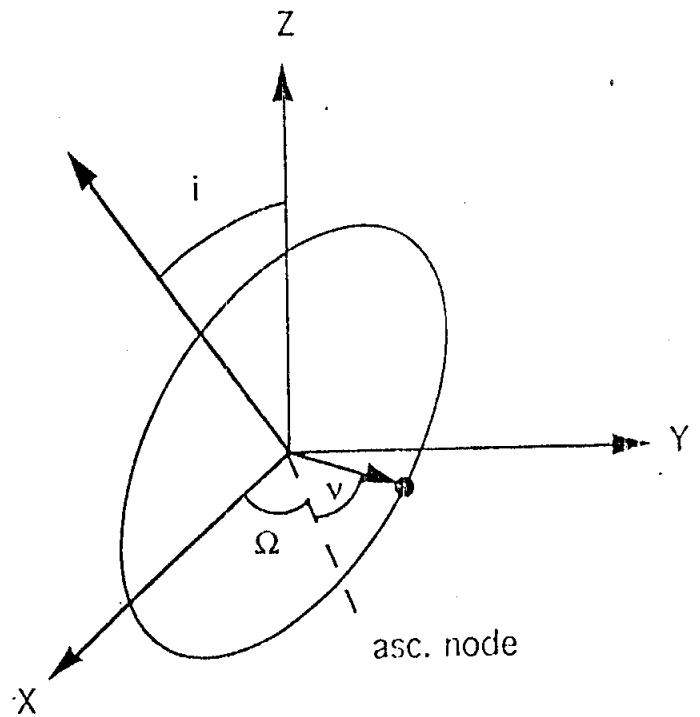
CASE 1

nav. sat. inclination (i) = 90 deg

nav. sat. long. asc. node (Ω) = 0 deg

nav. sat. altitude = 5000 km

nav. sat. initial phase angle (ν) = 0 deg



Other Error Sources

- Measurement Errors

  Range: 4 meters
  Range rate: .01 meter/sec

- State (process) noise (compensation for inaccuracies in Lunar gravity model) results in the following estimated error growth rate

NAVSAT: 89.5 m/hr (1234 m/rev) → $3 \times 10^{-9}\ m^2/s^3$

LUNAR VEHICLE: 3.1 m/hr ($R_{SOI} > R > 500$ km)

62.7 m/hr (R < 500 km) $K_{process} = 5 \times 10^{-7}\ m^2/s^3$

- Measurement rate = 1meas. / 100 sec ($R_{SOI} > R > 500$ km)
- Measurement Bias = 28 meters 1 meas./10 sec (R < 500 km)

U.S. Gov't

Initial Filter State Errors

Deep Space Tracking Network covariance matrix t = 0

inertial, km, km/sec

- NAVIGATION SATELLITE

(0.00465008000000	-0.00698569000000	-0.00006314990000	0.00000121498000	-0.00000071497600	-0.00000017476100	) e+0
(-0.00698569000000	0.01147540000000	0.00027373300000	-0.00000187562000	0.00000114767000	0.00000025837100	) e+0
(-0.06314990000000	0.27373300000000	0.36158800000000	-0.00002090570000	0.00003197810000	0.00002889860000	) e-3
(0.12149800000000	-0.18756200000000	-0.00209057000000	0.00003243130000	-0.00001891880000	-0.00000447221000	) e-5
(-0.07149760000000	0.11476700000000	0.00319781000000	-0.00001891880000	0.00001209110000	0.00000277628000	) e-5
(-0.17476100000000	0.25837100000000	0.02889860000000	-0.00004472210000	0.00002776280000	0.00001666780000	) e-6

Errors ($\delta u\ v\ w$) ?

Data Summary

Description	Case	Maximum Estimated Errors 0.5 hrs Before Periapse				
		Position, m		Velocity, m/sec		
		App	Com	App	Com	
Polar	1	2810	2204	00.689	00.265	
Bias	1a	2810	2204	00.700	00.265	
No Nav.	1b	32 k	10 k	20.300	01.300	
45 deg node	2	2757	3390	00.712	00.410	
-45 deg node	3	3167	996	00.697	00.133	*BEST*
Equatorial	5	4254	952	02..500	00.130	
Equat. 500 km *	5a	2401	250	00.401	00.146	

* Impossible Case

lunar vehicle Trajectory*

inclination (i) = 1.8 deg

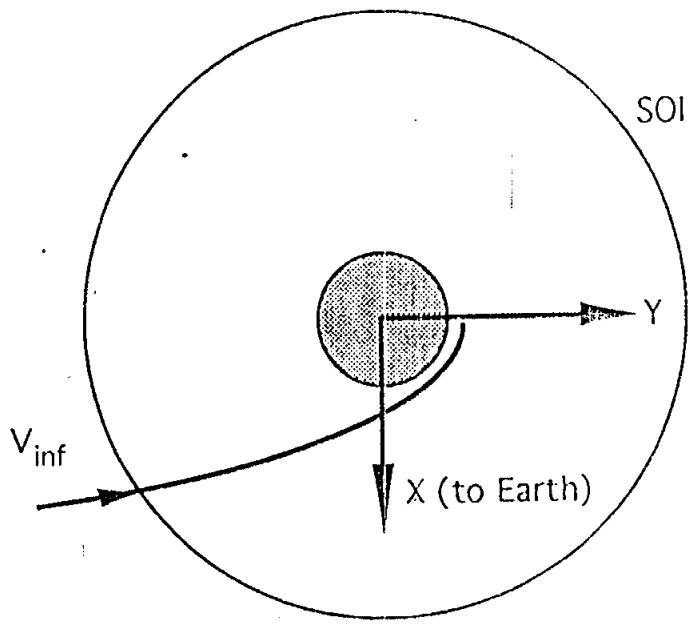
arg. of periapse (ω) = 60 deg

long. of asc. node (Ω) = 14 deg

periapse altitude = 100 km

trip time from SOI to periapse = 17 hr

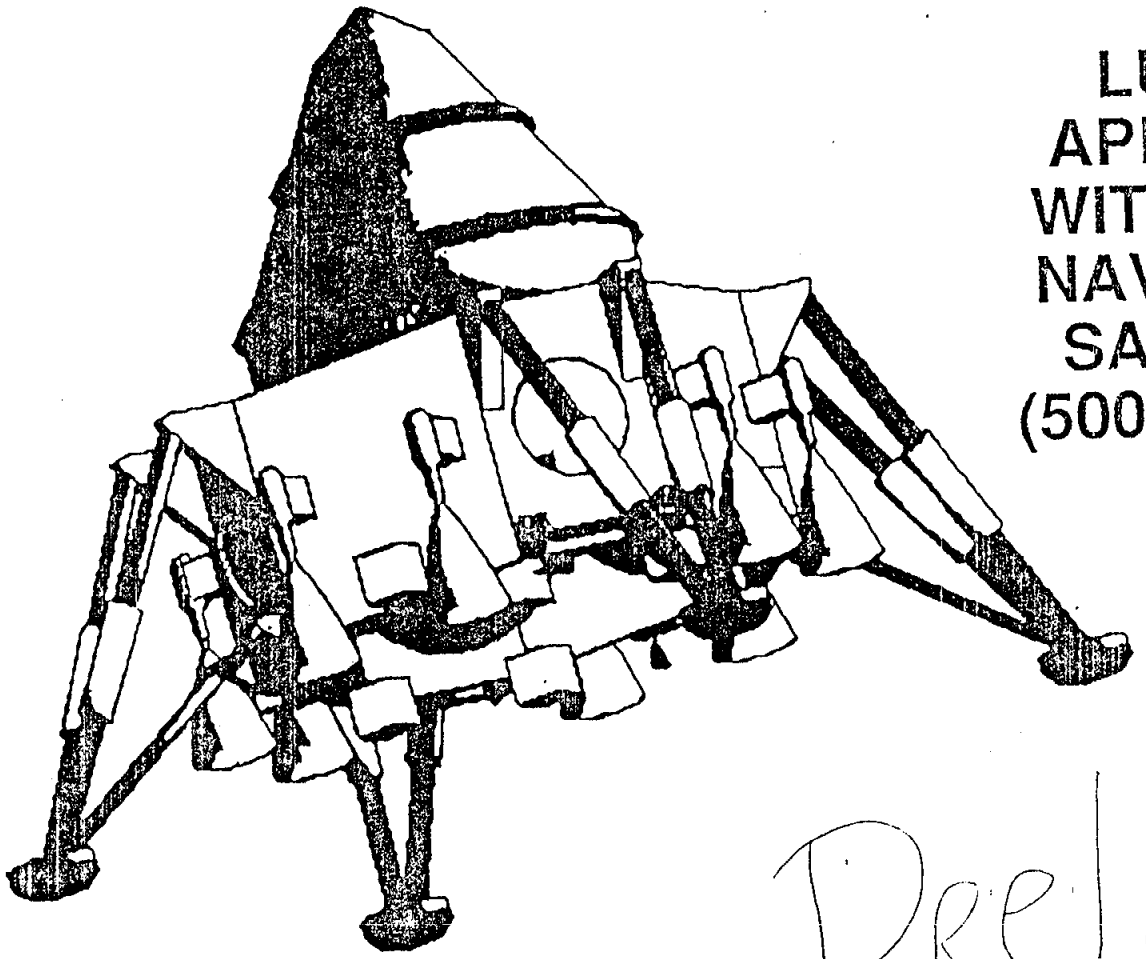
nav. sat. period = 13.7 hr



* From Steve Lewis/ET32

Johnson Space Center – Houston, Texas

NAVIGATION STUDY	Systems Engineering Division	
	PETER CUTHBERT / ET4	Mar. 12, 1992



LUNAR APPROACH WITH LUNAR NAVIGATION SATELLITE (5000 Km circ)

Preliminary

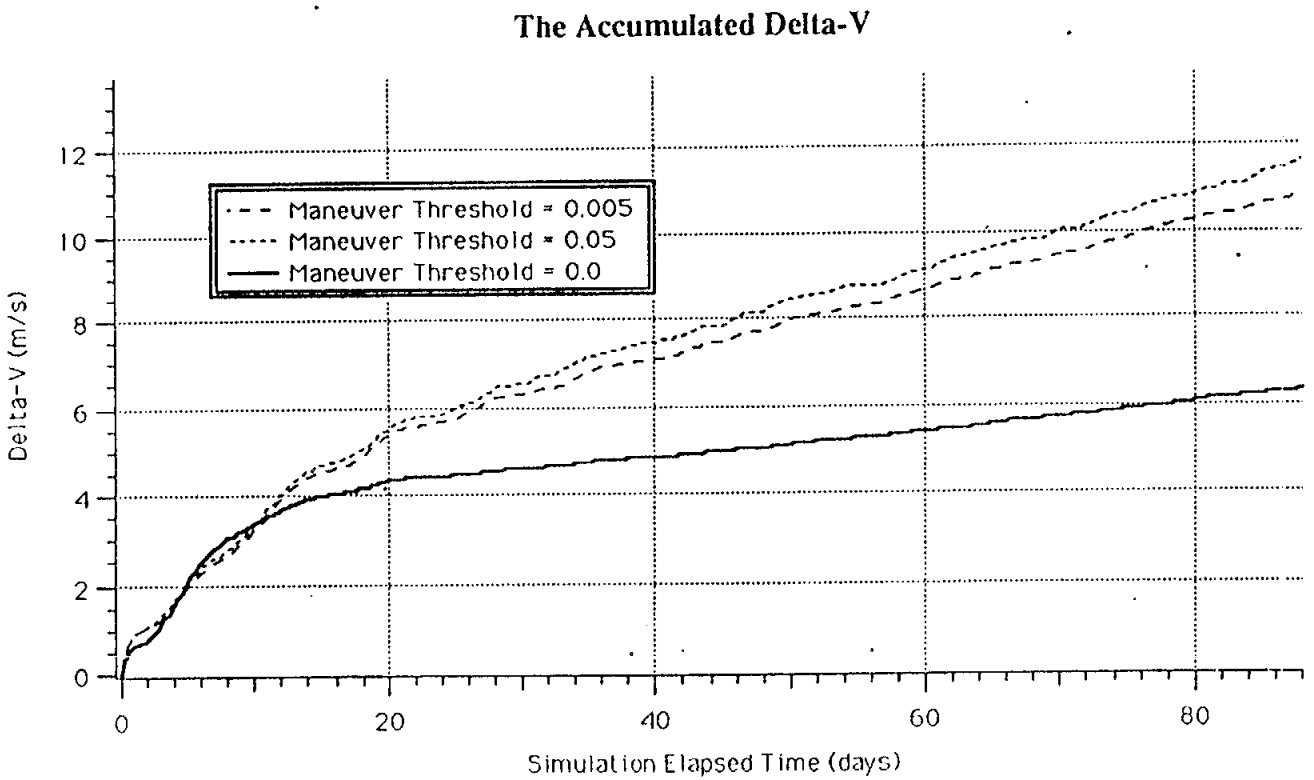
SUMMARY

- THE USE OF A COMMUNICATIONS SATELLITE STATIONED IN ORBIT AROUND THE TRANSLUNAR EQUILIBRIUM POINT OF THE EARTH MOON SYSTEM APPEARS TO BE A FEASIBLE OPTION FOR PROVIDING LUNAR FAR SIDE COMMUNICATIONS.
  - ⇒ THIS TYPE OF SYSTEM WOULD BE A LOT LESS EXPENSIVE TO PUT IN PLACE (AND TO MAINTAIN) THAN A NETWORK OF COMMUNICATIONS SATELLITES IN LOW LUNAR ORBIT.

- OUT PRELIMINARY PERFORMANCE STUDY LEADS TO THESE THREE CONCLUSIONS:
  1. THE SATELLITE CANNOT PROVIDE 100% COMMUNICATIONS COVERAGE WITHOUT A CONSIDERABLE PROPELLENT EXPENDITURE. HOWEVER, A MINOR REDUCTION IN THE COVERAGE YIELDS A SIGNIFICANT REDUCTION IN THE PROPELLENT CONSUMPTION.
  2. THE NAVIGATION ERROR IS THE BIGGEST FACTOR IN DETERMINING THE PROPELLENT CONSUMPTION. THIS COST CAN BE PROHIBITIVE, BUT IF SUFFICIENT ACCURACY IS OBTAINED THE COST MAY BE ACCEPTABLE.
  3. A PROPULSION SYSTEM WHICH CAN ONLY EXECUTE DISCRETE MANEUVERS RESULTS IN A (RELATIVELY) SMALL PROPELLENT COST.

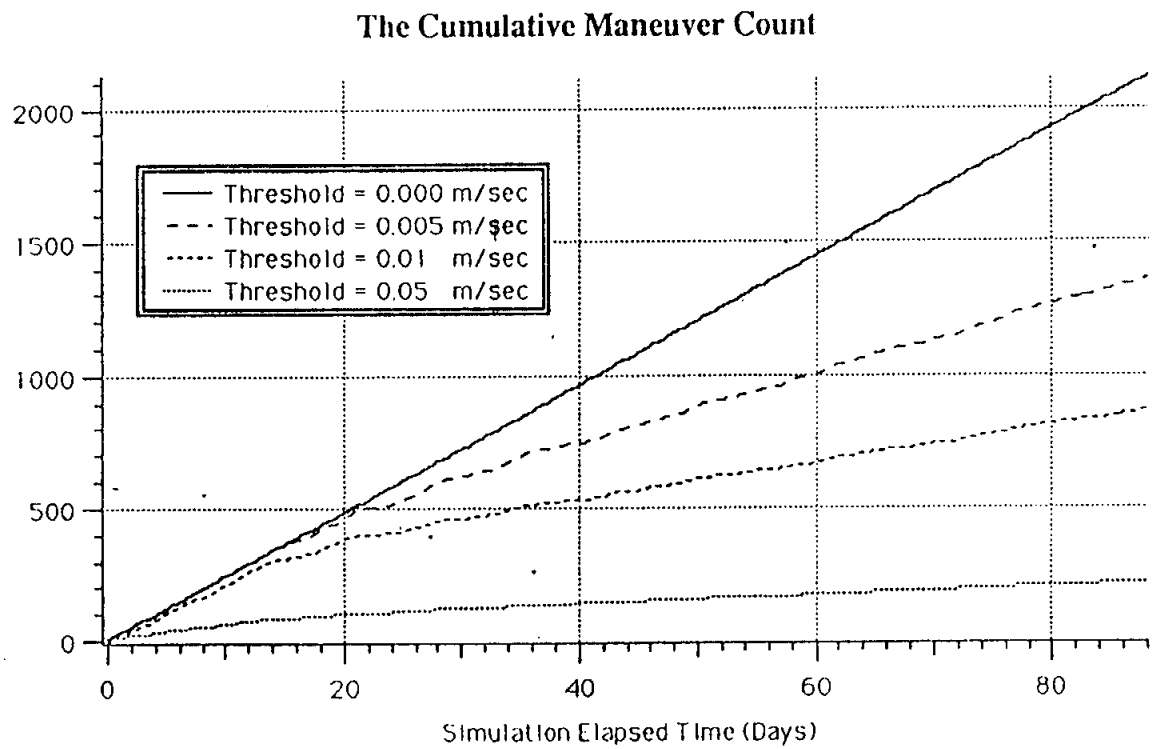
THE IMPACT OF DISCRETE MANEUVERS

- THE EFFECT OF DIFFERENT THRESHOLD VALUES ON THE CUMMULATIVE DELTA-V HISTORY IS SHOWN IN THE FOLLOWING PLOT.
  - ⇒ THE CASE WHERE THE THRESHOLD WAS SET TO 0.01 M/SEC WAS NOT SHOWN BECAUSE IT WAS VIRTUALLY INDISTINGUISHABLE FROM THE CASE WHERE THE THRESHOLD WAS SET TO 0.005 M/SEC.



THE IMPACT OF DISCRETE MANEUVERS

- THE EFFECT OF THREE DIFFERENT THRESHOLD VALUES ON THE NUMBER OF MANEUVERS EXECUTED IS SHOWN IN THE FOLLOWING PLOT.



THE IMPACT OF DISCRETE MANEUVERS

- THE HALO ORBIT CONTROLLER PRODUCES CONTINUOUS THRUST COMMANDS, BUT THE PROPULSION SYSTEM MAY NOT BE ABLE EXECUTE CONTINUOUS MANEUVERS.

- THERE ARE SOME CAVEATS TO THIS STATEMENT:

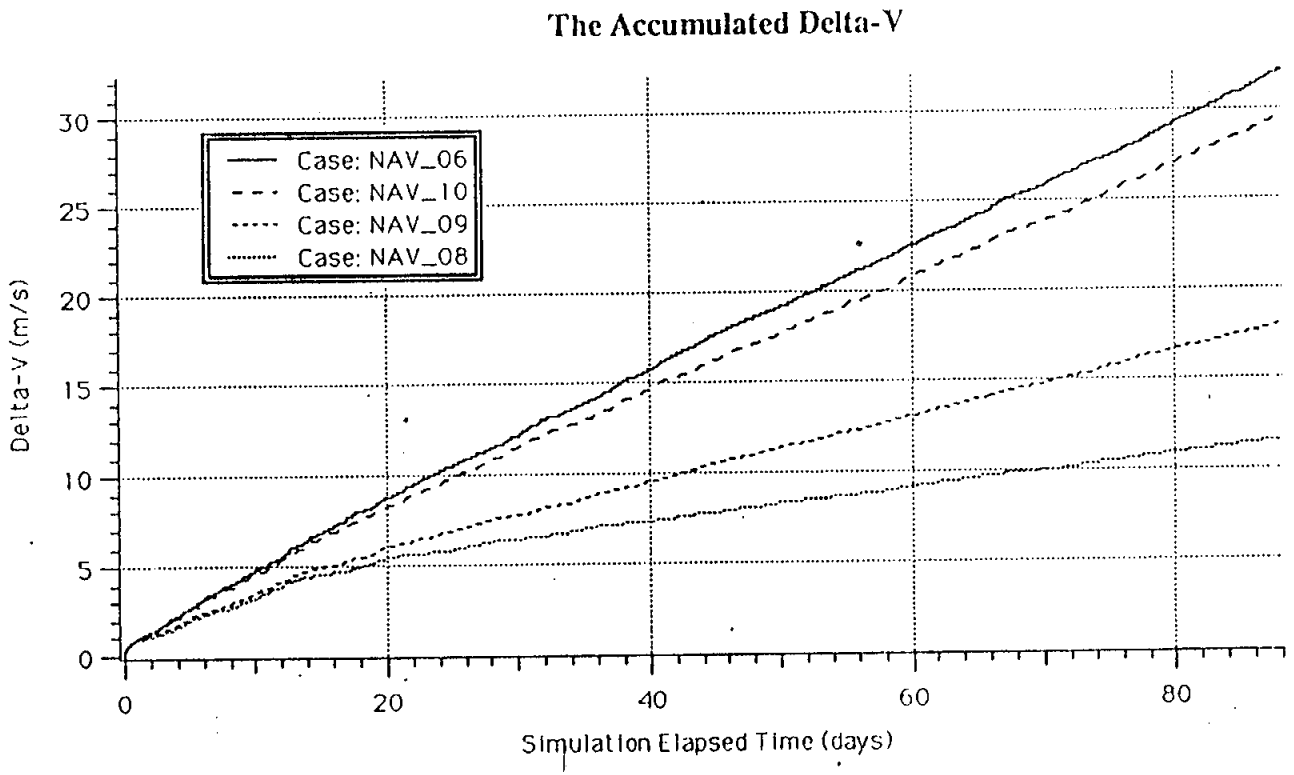
  1. AN PROPULSION SYSTEM WHICH USES ELECTRIC THRUSTERS MAY BE ABLE TO EXECUTE THE CONTINUOUS CONTROL LAW.

  2. THE MANEUVER COMMANDS WILL NOT BE CONTINUOUS IN PRACTICE BECAUSE THE VEHICLE'S STATE IS MAINTAINED BY A NUMERICAL INTEGRATION.

- THE IMPACT OF DISCRETE MANEUVERS WAS ASSESSED BY PUTTING A MINIMUM IMPLUSE THRESHOLD INTO THE SIMULATION (IF THE COMPUTED THRUST COMMAND WAS BELOW THIS THRESHOLD IT WAS NOT EXECUTED).

THE COST OF NAVIGATION

- THE FOLLOWING GRAPH SHOWS THE CUMULATIVE DELTA-V HISTORY FOR SOME OF THE CASES ON THE PRECEDING PAGE.



THE COST OF NAVIGATION

- THE BASIC INDICATOR OF PERFORMANCE WAS TAKEN TO BE THE DELTA-V REQUIRED PER REV.

- THIS STUDY INCLUDED TEN CASES. THE RESULTS OF THESE TEN RUNS ARE BEST SUMMARIZED IN THE FOLLOWING TABLE.

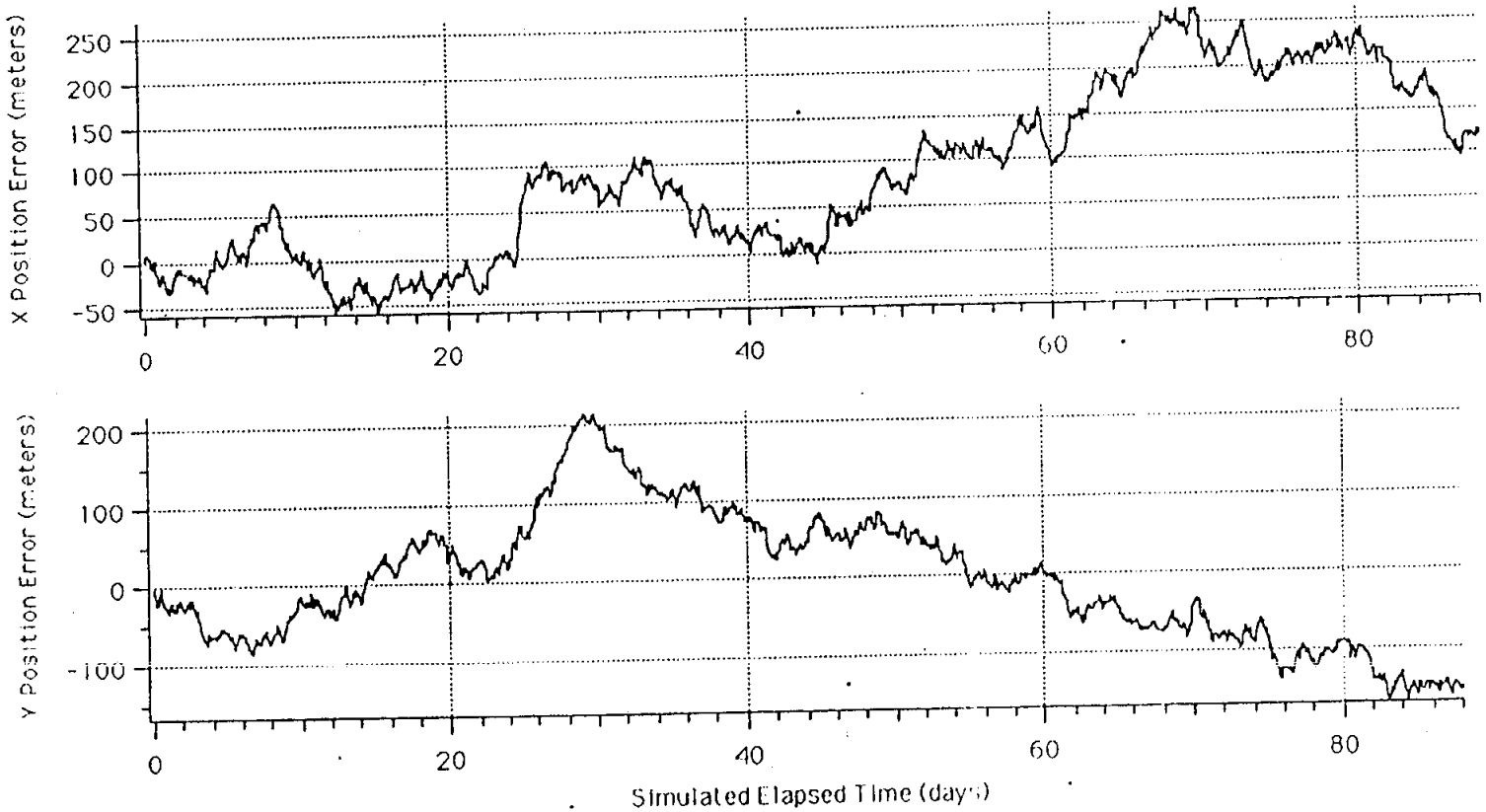
  ⇒ THE UNITS FOR THE TIME CONSTANT ARE MINUTES, ALL OTHER UNITS ARE IN METERS AND SECONDS. THE INTEGRATION TIME STEP FOR ALL CASES WAS 60 MINUTES.

CASE	TIME CONSTANT	R_x	R_y	R_z	R_x	R_x	R_x	$\lvert\Delta\underline{V}\rvert$ PER REV
NAV_05	600	1000	1000	1000	0.1	0.1	0.1	47.77
NAV_01	600	100	100	100	0.1	0.1	0.1	47.44
NAV_07	600	1000	1000	1000	0.01	0.01	0.01	6.65
NAV_03	600	100	100	100	0.01	0.01	0.01	5.01
NAV_06	60,000	1000	1000	1000	0.1	0.1	0.1	5.15
NAV_02	60,000	100	100	100	0.1	0.1	0.1	5.13
NAV_08	60,000	1000	1000	1000	0.01	0.01	0.01	1.34
NAV_04	60,000	100	100	100	0.01	0.01	0.01	1.26
NAV_10	60,000	1000	100	100	0.1	0.01	0.01	4.72
NAV_09	60,000	100	1000	1000	0.01	0.1	0.1	2.61

THE COST OF NAVIGATION

- THE FOLLOWING GRAPHS SHOW A SAMPLE OF WHAT THE NAVIGATION ERRORS LOOKED LIKE FOR A CASE WITH A LARGE ECRV TIME CONSTANT.

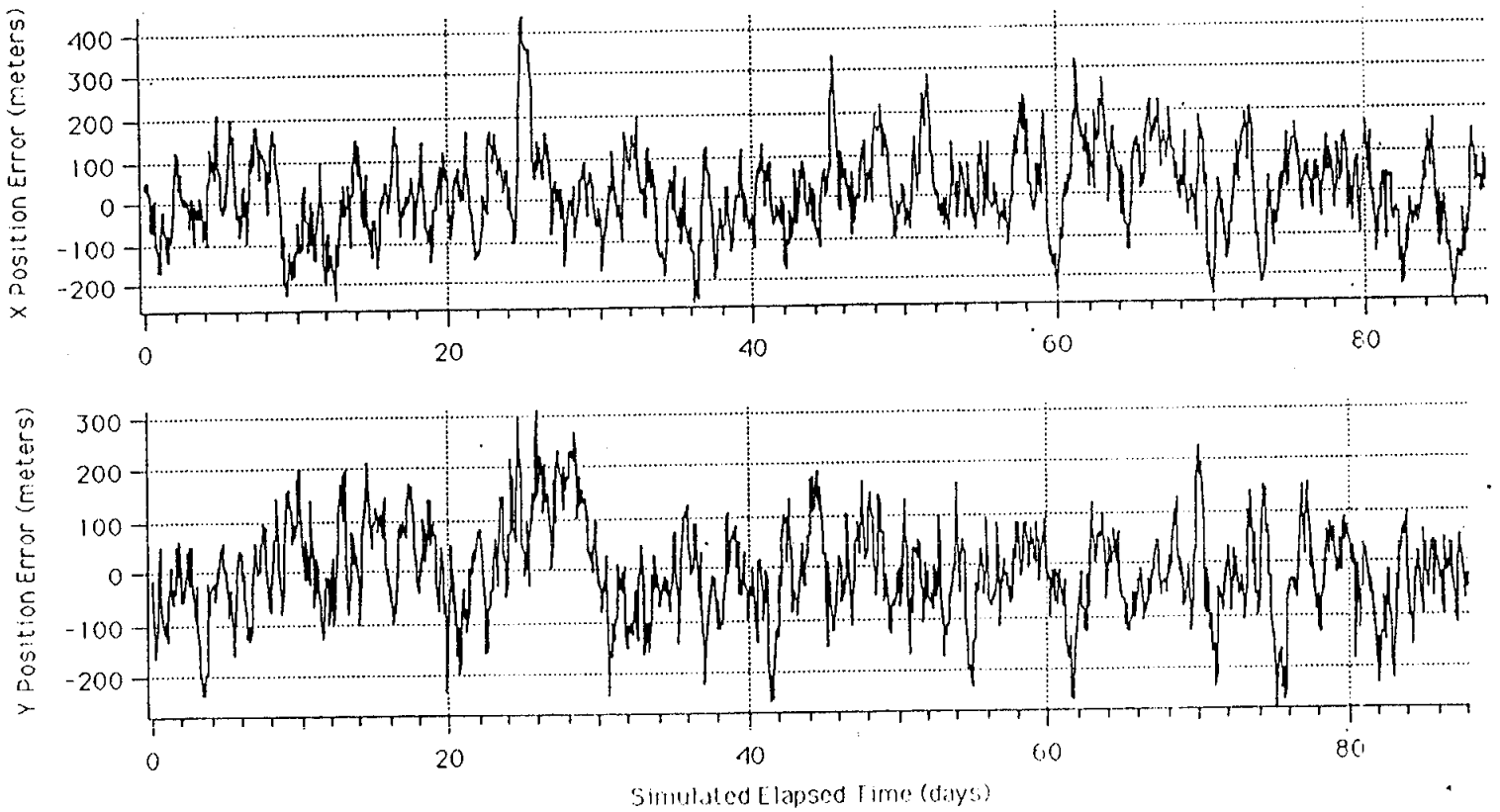
The Simulated In Plane Navigation Error



THE COST OF NAVIGATION

- THE FOLLOWING GRAPHS SHOW A SAMPLE OF WHAT THE NAVIGATION ERRORS LOOKED LIKE FOR A CASE WITH A SMALL ECRV TIME CONSTANT.

The Simulated In Plane Navigation Error



THE COST OF NAVIGATION

- THE ACCURACY OF THE NAVIGATION SOLUTION WILL, IN ALL LIKELYHOOD, PROVE TO BE THE BIGGEST SINGLE FACTOR IN DETERMINING THE AMOUNT FUEL CONSUMED FOR ORBIT MAINTENANCE.

- THE FULL IMPACT OF NAVIGATION ERRORS ON THE COST OF MAINTAINING A HALO ORBIT WILL NOT BE KNOWN UNTIL A (CONCEPTUAL) NAVIGATION SYSTEM HAS BEEN DEVELOPED AND ANALYZED.

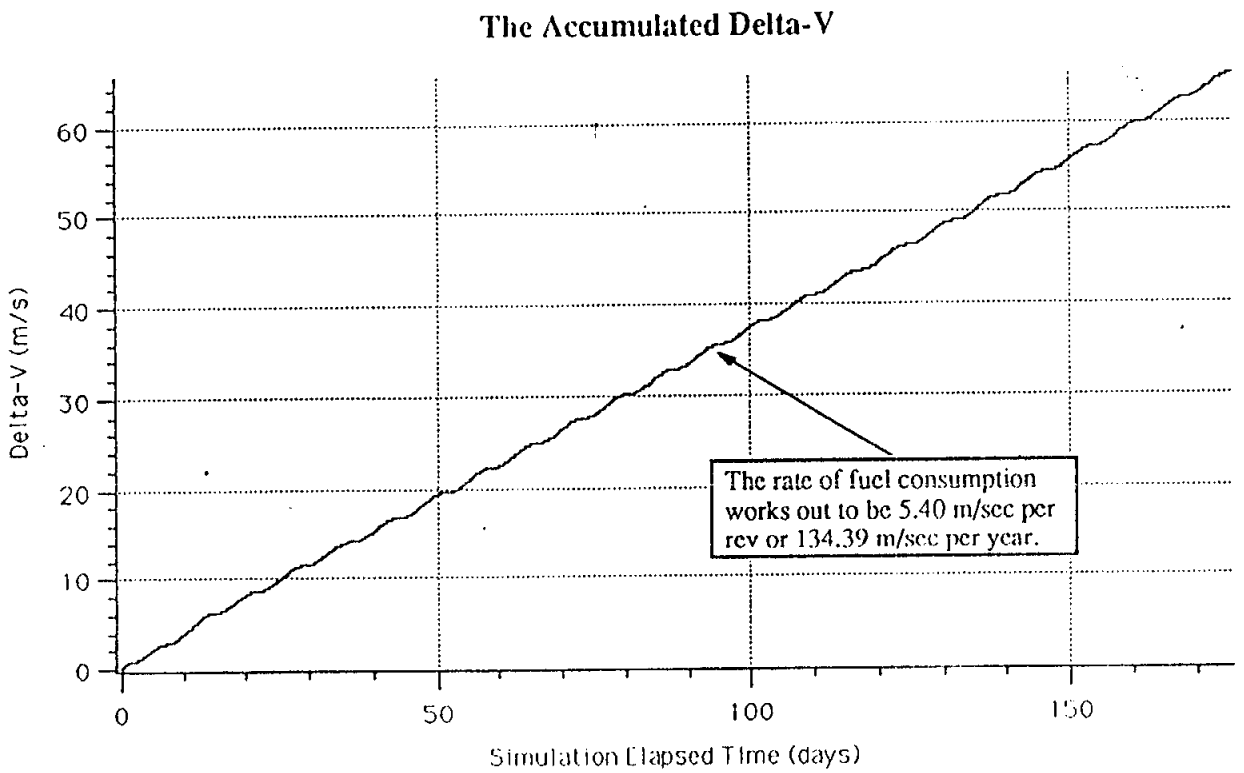
- A PRELIMINARY ASSESSMENT OF THE PERFORMANCE IMPACT OF NAVIGATION ERROR HAS BEEN CONDUCTED BY MEANS OF A PARAMETRIC STUDY.

  - ⇒ THE NAVIGATION ERROR HAS BEEN MODELLED AS AN EXPONENTIALLY CORRELATED RANDOM VARIABLE (ECRV) FOR EACH AXIS OF THE POSITION AND VELOCITY. NO CORRELATION BETWEEN ANY AXES OF THE POSITION AND/OR VELOCITY WAS CONSIDERED.

  - ⇒ THE TWO PARAMETERS WHICH WERE VARIED WERE THE VARIANCE OF THE NOISE AND THE ECRV TIME CONSTANT.

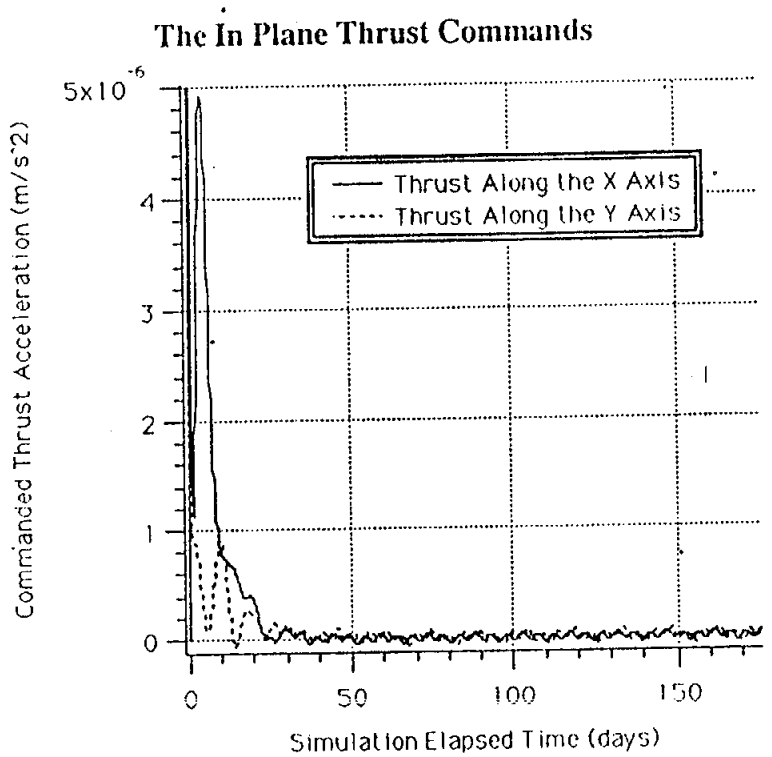
THE COST OF VISIBILITY

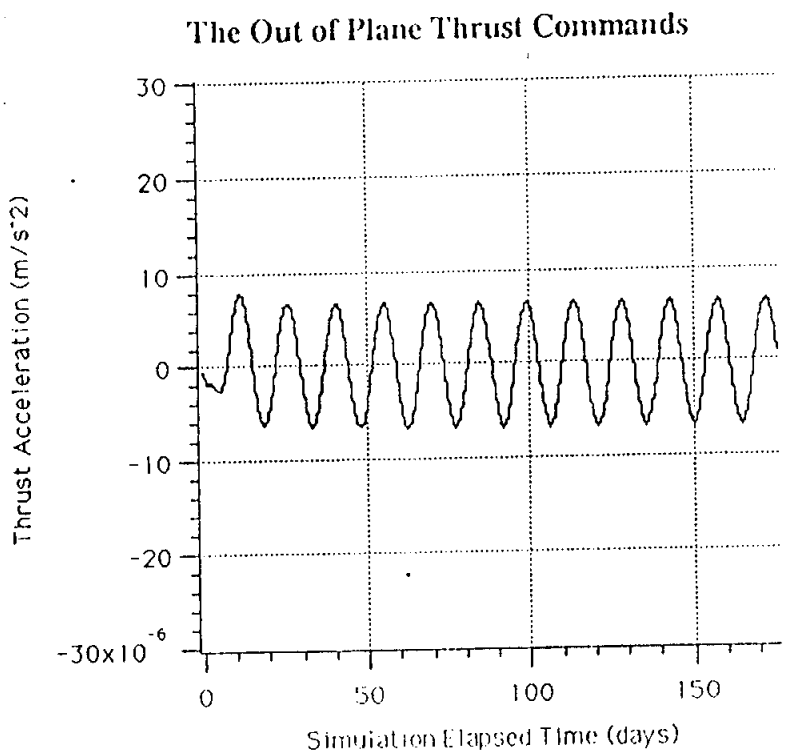
- THIS CONTINUAL OUT OF PLANE THRUSTING INCURS A SIGNIFICANT PERFORMANCE COST.



THE COST OF VISIBILITY

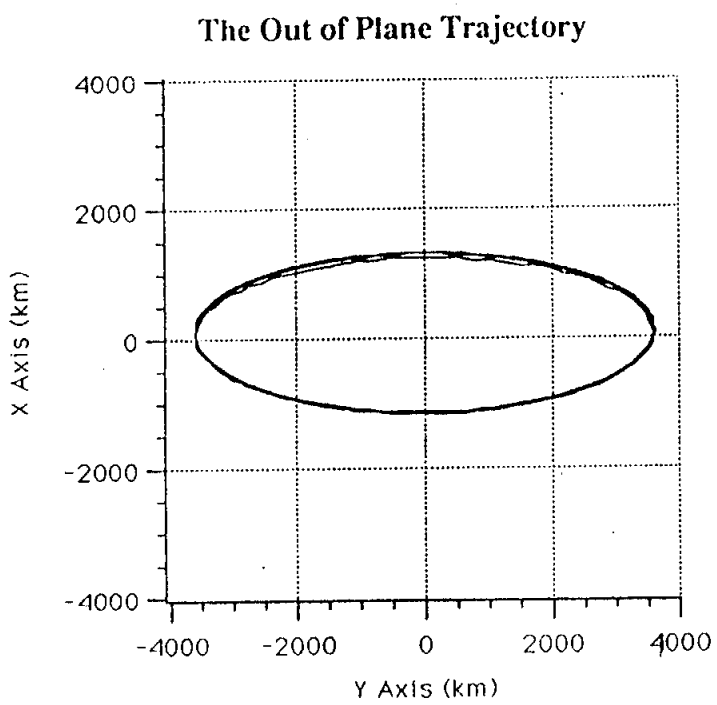
- A LOOK AT THE THRUST COMMANDS REVEALS THAT THE CONTROLLER IS CONTINUALLY THRUSTING ALONG THE Z AXIS TO IN ORDER TO CHANGE THE FREQUENCY OF THE OUT OF PLANE MOTION.

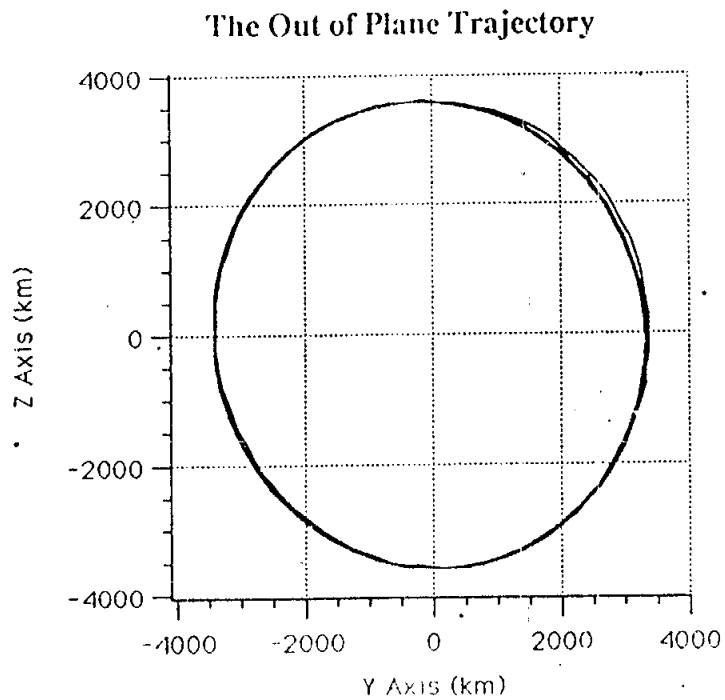




THE COST OF VISIBILITY

- THE TRAJECTORY SHOWN HERE HAS A MAXIMUM OFF-AXIS AMPLITUDE OF 3600 KM. IT WAS PROPAGATED FOR TWELVE REVS (APPROXIMATELY SIX MONTHS).





THE COST OF VISIBILITY

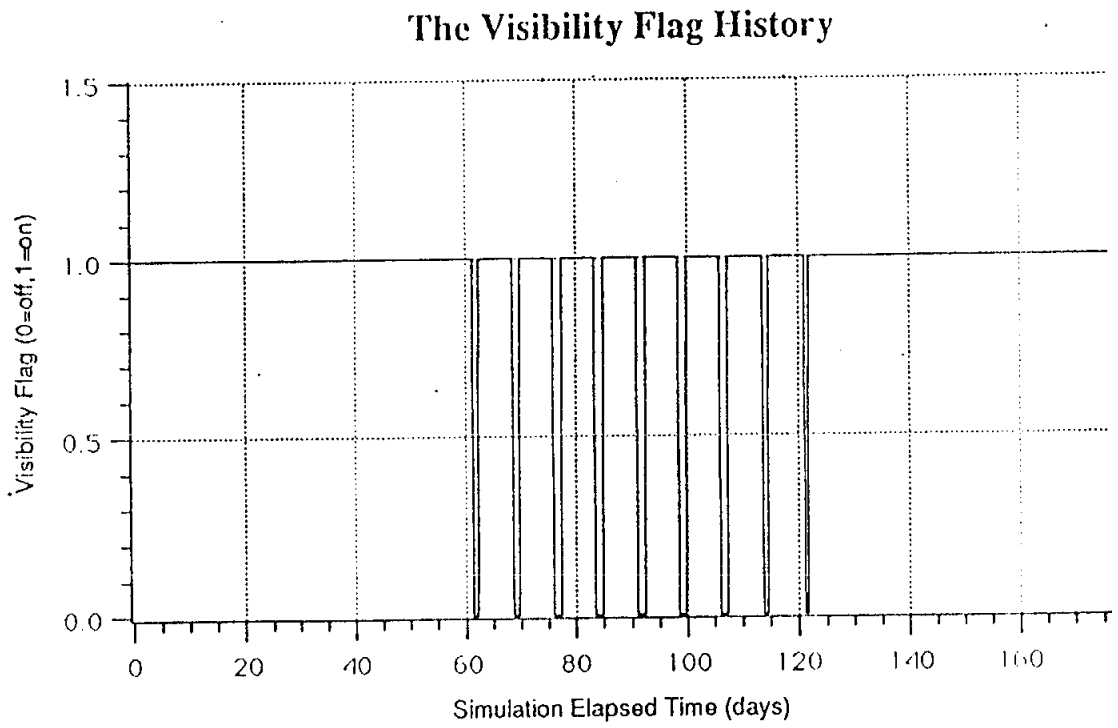
- THE TRADE OFF BETWEEN THE IMPROVED VISIBILITY AND THE GREATER PROPELLENT EXPENDITURE FOR LARGER ORBITS CAN BE SEEN IN THE FOLLOWING TABLE.

MAXIMUM OFF-AXIS AMPLITUDE (km)	FRACTION OF TIME VISIBLE (percent)	CUMMULATIVE DELTA-V (m/sec per year)
3600	69%	10.10
5400	88%	29.58
7200	94%	59.80

- RECALL THAT THERE IS ANOTHER APPROACH TO IMPROVING THE VISIBILITY OF THE SATELLITE: ADJUSTING THE RATIO OF THE FREQUENCIES OF THE OUT OF PLANE AND IN PLANE MOTIONS.
  - ⇒ OUR SIMULATION RESULTS INDICATED THAT IT DOESN'T MATTER WHICH FREQUENCY IS ADJUSTED.
  - ⇒ THE PERFORMANCE REALIZED BY THIS OPTION WILL BE SHOWN ON THE NEXT FEW CHARTS.

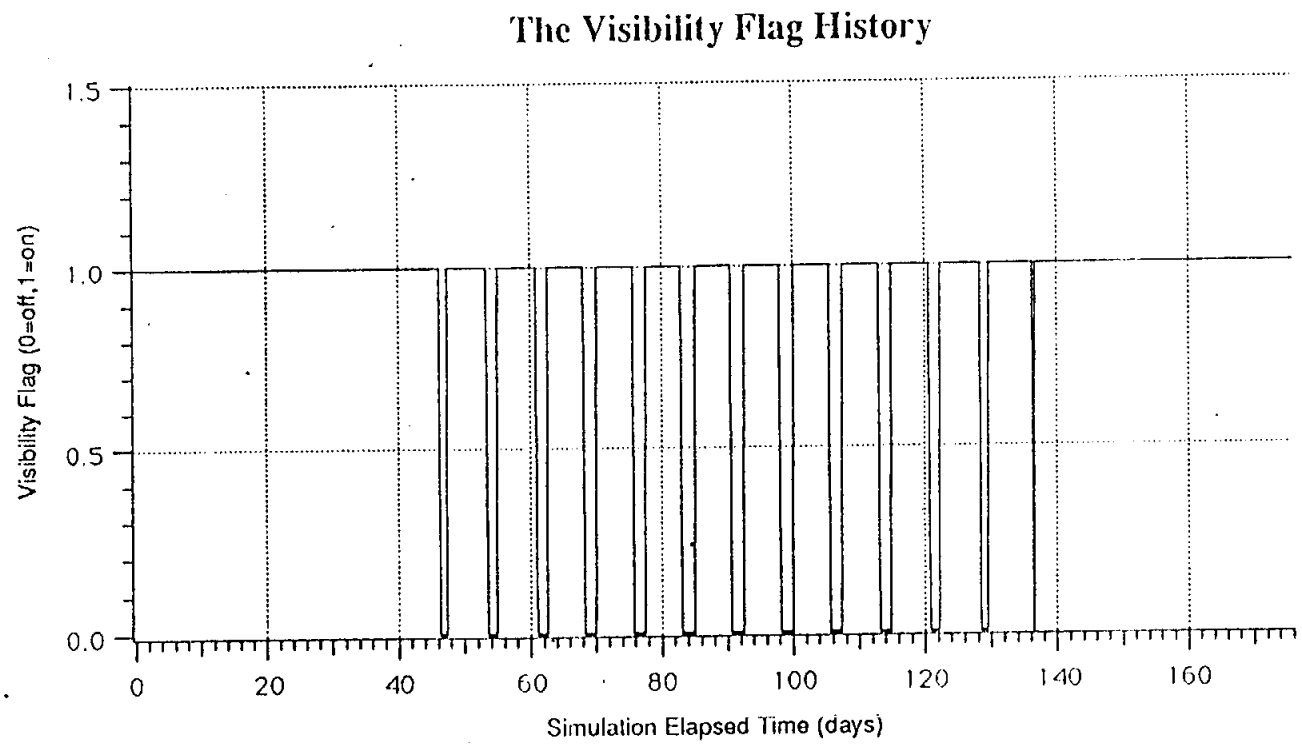
THE COST OF VISIBILITY

- THE VISIBLITY OF THE SPACECRAFT AS IT FOLLOWS THE 7200 KM LISSAJOUS ORBIT (FOR 12 REVS) IS SHOWN BELOW:



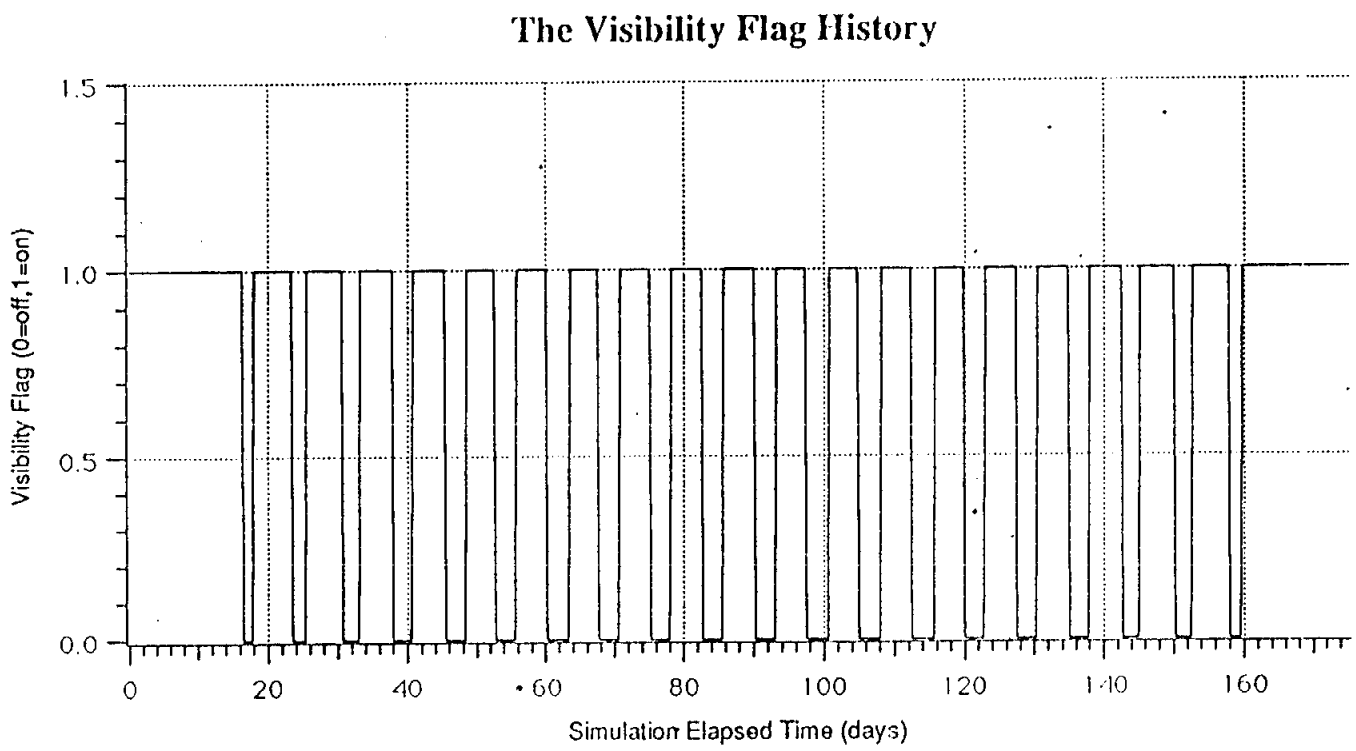
THE COST OF VISIBILITY

- THE VISIBLITY OF THE SPACECRAFT AS IT FOLLOWS THE 5400 KM LISSAJOUS ORBIT (FOR 12 REVS) IS SHOWN BELOW:



THE COST OF VISIBILITY

- THE VISIBLITY OF THE SPACECRAFT AS IT FOLLOWS THE 3600 KM LISSAJOUS ORBIT (FOR 12 REVS) IS SHOWN BELOW:



THE COST OF VISIBILITY

- OBVIOUSLY THE NATURAL MOTION OF THE SATELLITE WILL CAUSE IT TO BE OBSCURED AT TIMES.

- THERE ARE TWO POSSIBLE WAYS TO INCREASE THE AMOUNT OF TIME THAT THE SATELLITE IS VISIBLE FROM THE EARTH.

  1. TO STATION THE SATELLITE IN ORBITS WHICH HAVE LARGER OFF–AXIS APLITUDES. THIS TYPE OF ORBIT IS CALLED A LISSAJOUS ORBIT.

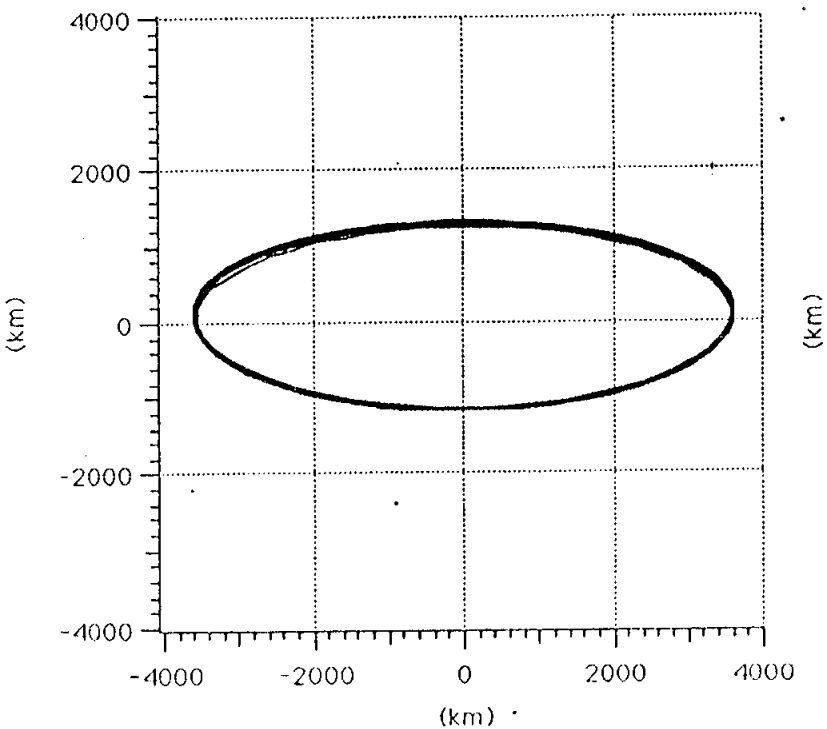
  2. TO ARTIFICIALLY ADJUST THE FREQUENCY RATIO OF THE IN PLANE AND OUT OF PLANE MOTIONS. THIS TYPE OF ORBIT IS CALLED A HALO ORBIT.

- THE RELATIVE PERFORMANCE OF THESE TWO OPTIONS WILL BE SHOWN IN THE NEXT FEW CHARTS.

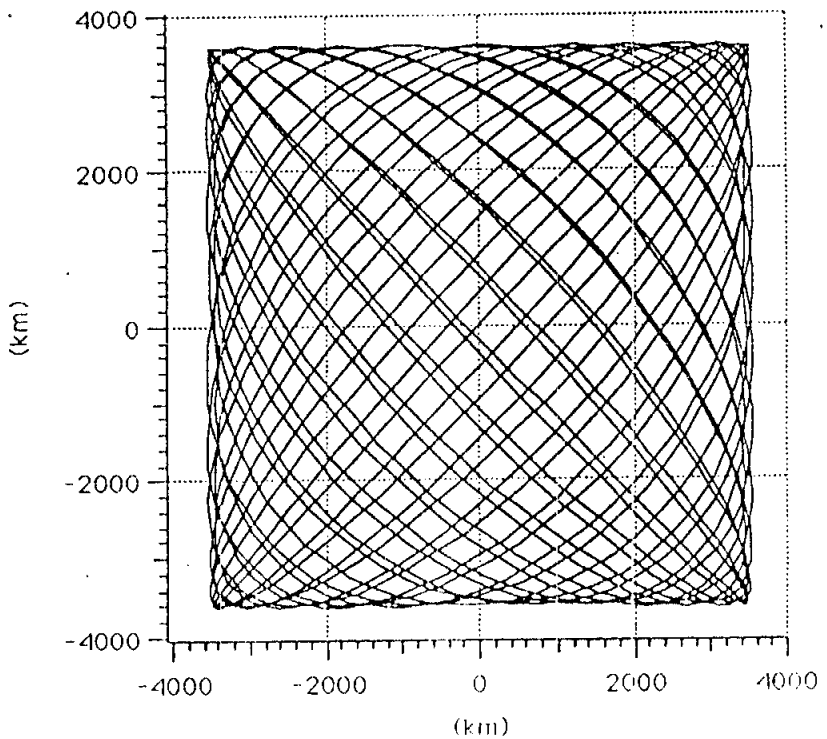
THE COST OF VISIBILITY

- THE TRAJECTORY SHOWN HERE HAS A MAXIMUM OFF-AXIS AMPLITUDE OF 3600 KM. IT WAS PROPAGATED FOR TWENTY FOUR REVS (APPROXIMATELY ONE YEAR).

The In Plane Trajectory



The Out of Plane Trajectory



THE COST OF VISIBILITY

- THE NATURAL MOTION OF THE SPACECRAFT FOLLOWS A THREE DIMENSIONAL TRAJECTORY ABOUT THE TRANSLUNAR EQUILIBRIUM POINT.
  - ⇒ THE MOTION IN THE PLANE OF THE EARTH–MOON ORBIT IS ELLIPTICAL, WHILE THE MOTION PERPENDICULAR TO THE LINE OF SIGHT OF THE EARTH–MOON AXIS FORMS A LISSAJOUS FIGURE (THE LISSAJOUS FIGURE RESULTS BECAUSE THE NATURAL FREQUENCIES OF THE IN PLANE AND OUT OF PLANE MOTION DIFFER).
  - ⇒ THIS MOTION IS ILLUSTRATED ON THE NEXT PAGE.

- IN ORDER FOR THE SATELLITE TO BE VISIBLE FROM THE EARTH IT MUST BE GREATER THAN 3100 KM FROM THE EARTH–MOON AXIS (OTHERWISE IT IS OBSCURED BY THE MOON).

INTRODUCTION

- THE PURPOSE OF THIS PRESENTATION IS TO PRESENT THE LATEST RESULTS OF THE HALO ORBIT CONTROL STUDY.

- THIS PROJECT HAS, TO DATE, BEEN FOCUSSED ON THE DEVELOPMENT AND VALIDATION OF A CONTROL ALGORITHM FOR HALO ORBIT MAINTENANCE.

  ⇒ THE LINEAR QUADRATIC REGULATOR ALGORITHM APPEARS TO PROVIDE AN EFFICIENT AND ROBUST CONTROL LAW FOR THIS APPLICATION.

- THE MOST RECENT WORK ON THIS PROJECT HAS BEEN A PRELIMINARY LOOK AT SOME OF THE PERFORMANCE ISSUES THAT WILL DETERMINE WHETHER OR NOT A HALO ORBITING COMMUNICATIONS SATELLITE IS FEASIBLE.

- THERE ARE THREE PERFORMANCE ISSUES THAT HAVE BEEN EXAMINED:

  1. THE PERFORMANCE COSTS OF MAINTAINING SATELLITE VISIBILITY.
  2. THE PERFORMANCE COSTS OF NAVIGATION ERROR (PRELIMINARY).
  3. THE IMPACT OF IMPLEMENTING A CONTINUOUS CONTROL INPUT WITH DISCRETE MANEUVERS.

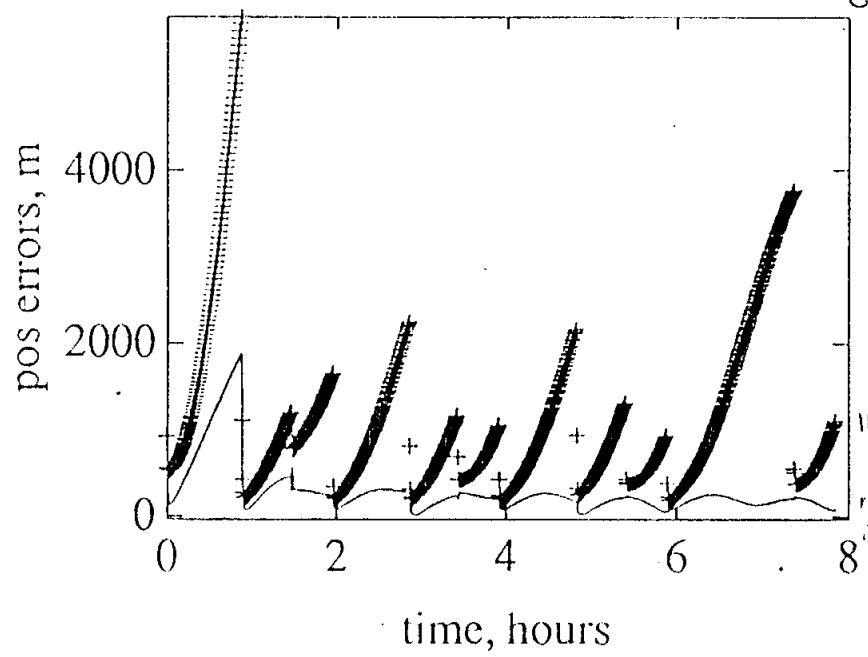
THE CONTROL OF PERIODIC TRAJECTORIES ABOUT THE UNSTABLE EQUILIBRIUM POINTS OF THE EARTH–MOON SYSTEM
(A Status Report)

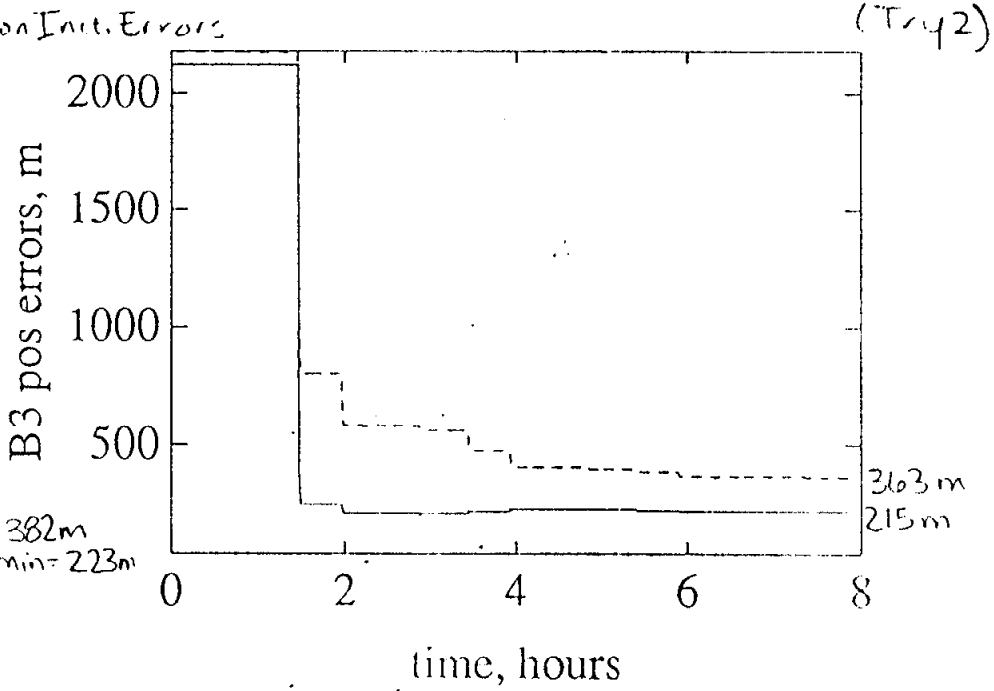
— *by* —

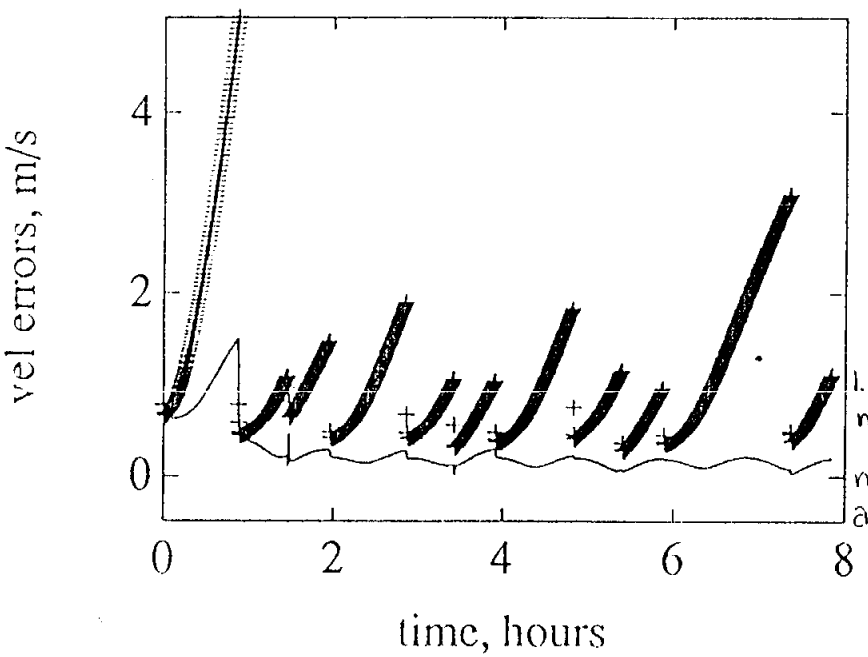
David Hoffman
Systems Engineering Division

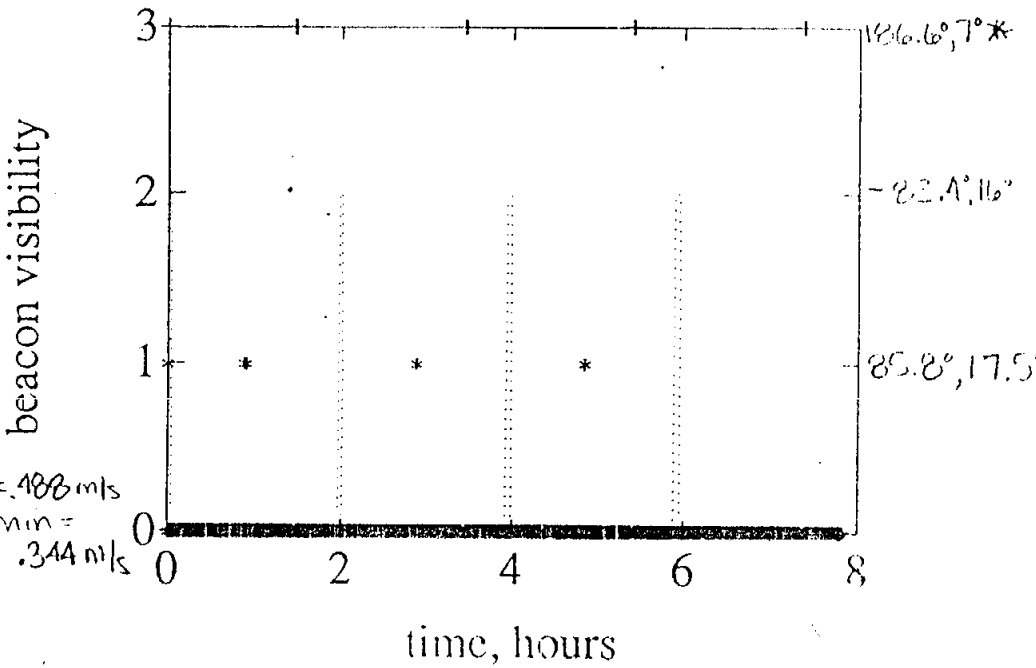
February 21, 1992

100 Km Circular Orbit. i- , -83.4° 3σ Beacon Locatio certain (Tr42)



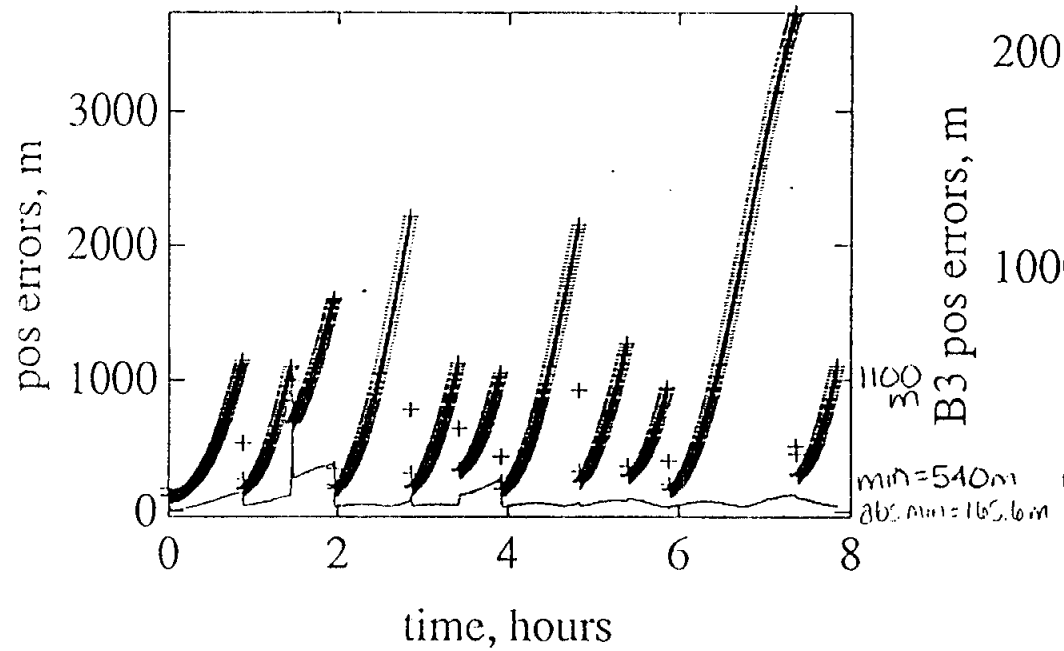


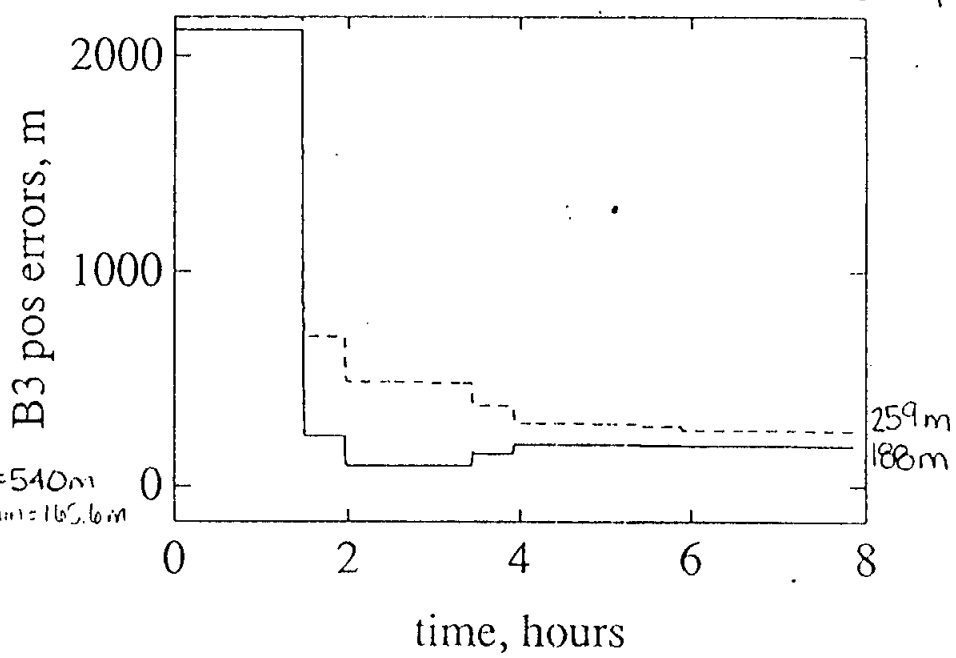


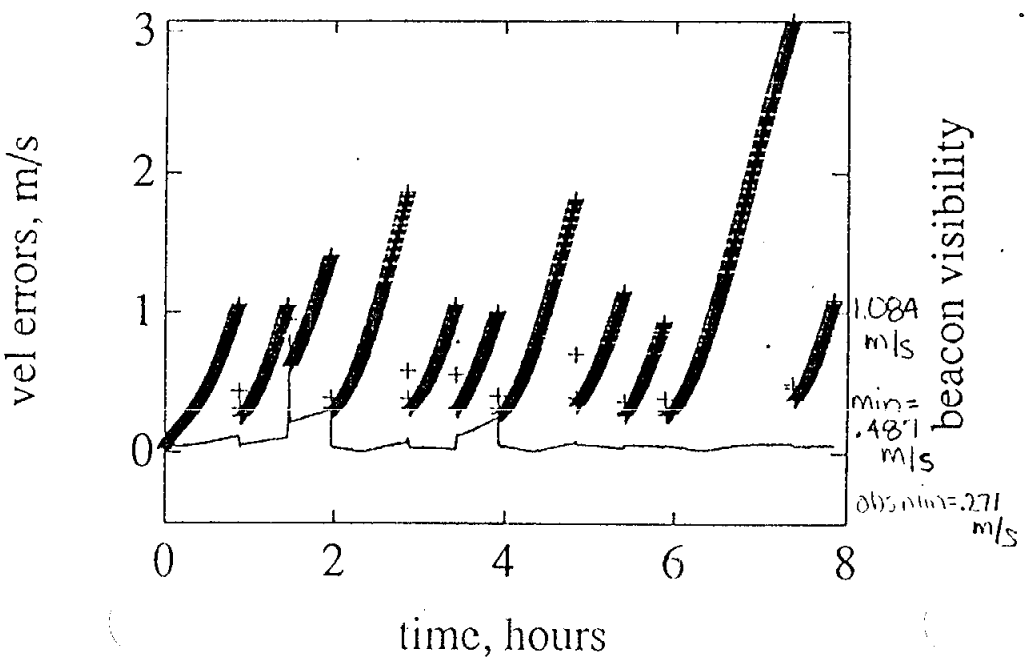


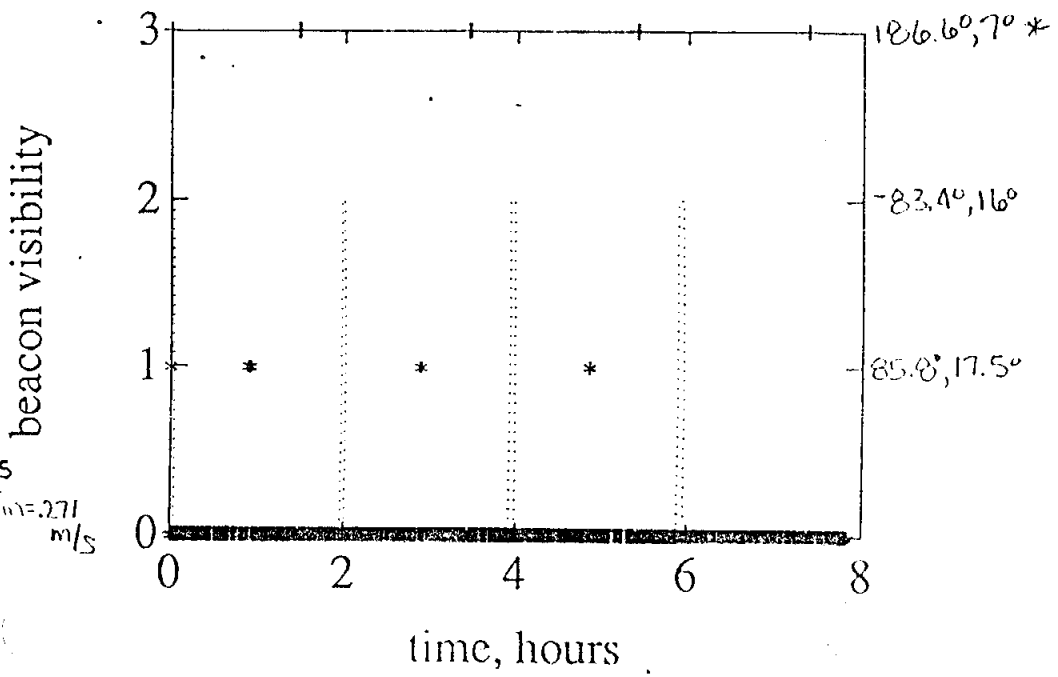
100 km Circular Orbit ; Ω = -83.4° : 3σ Beacon Locati ertai

DSN Init. Errors (Try 2)



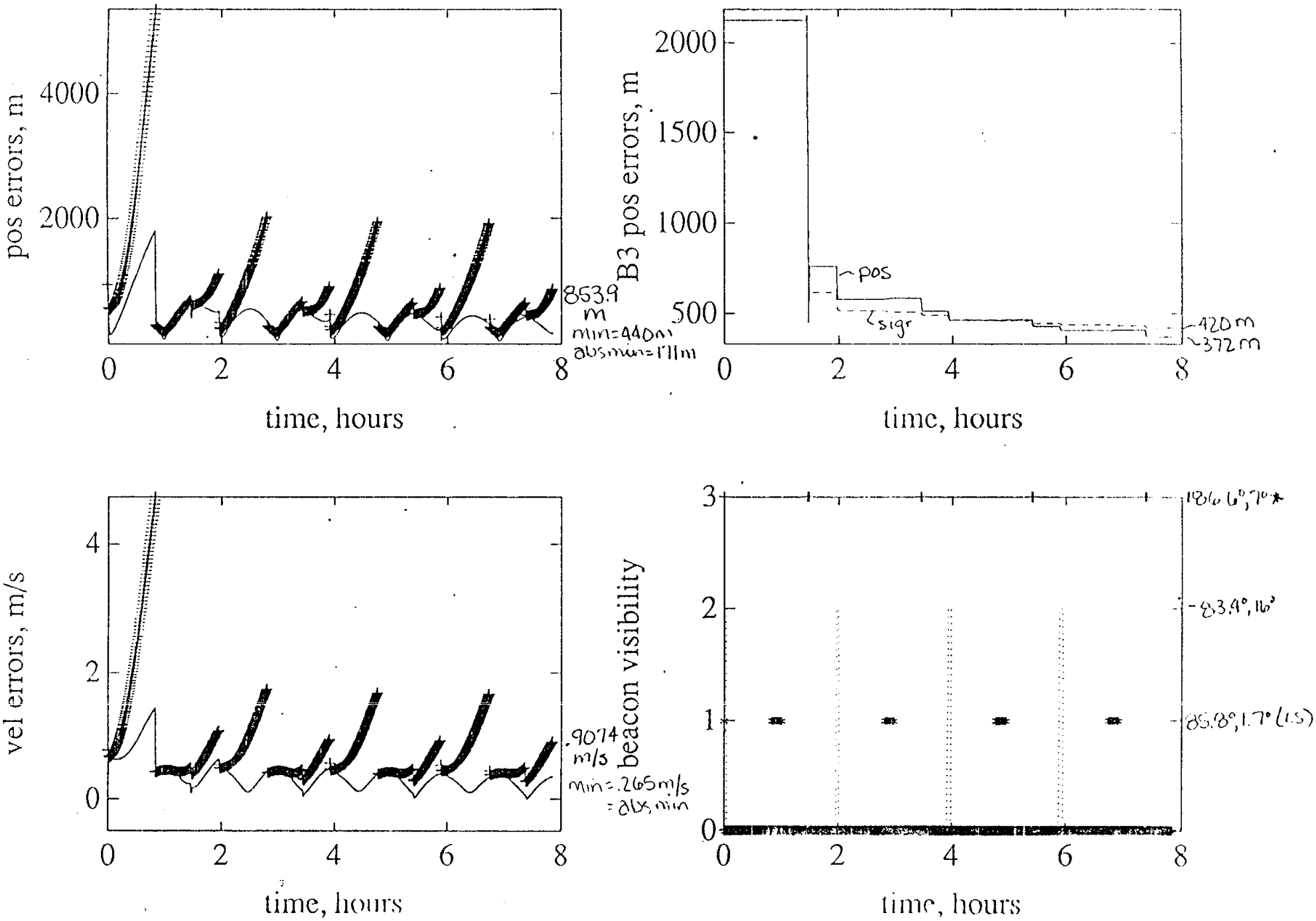




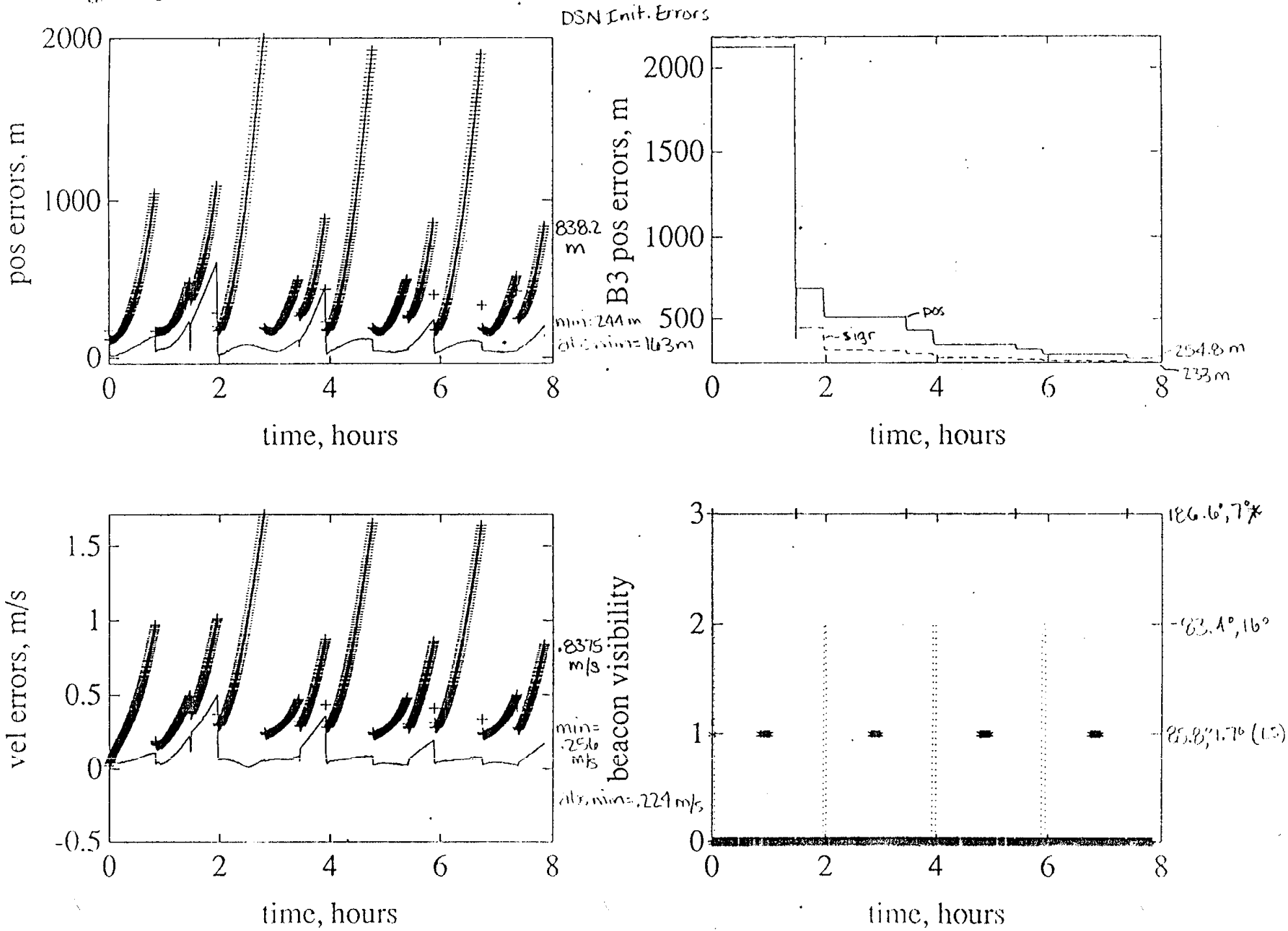


100 Km Circular Orbit; = -83.4° - 3rd Beacon Locat ... ncertair

BeaconInit.Errors

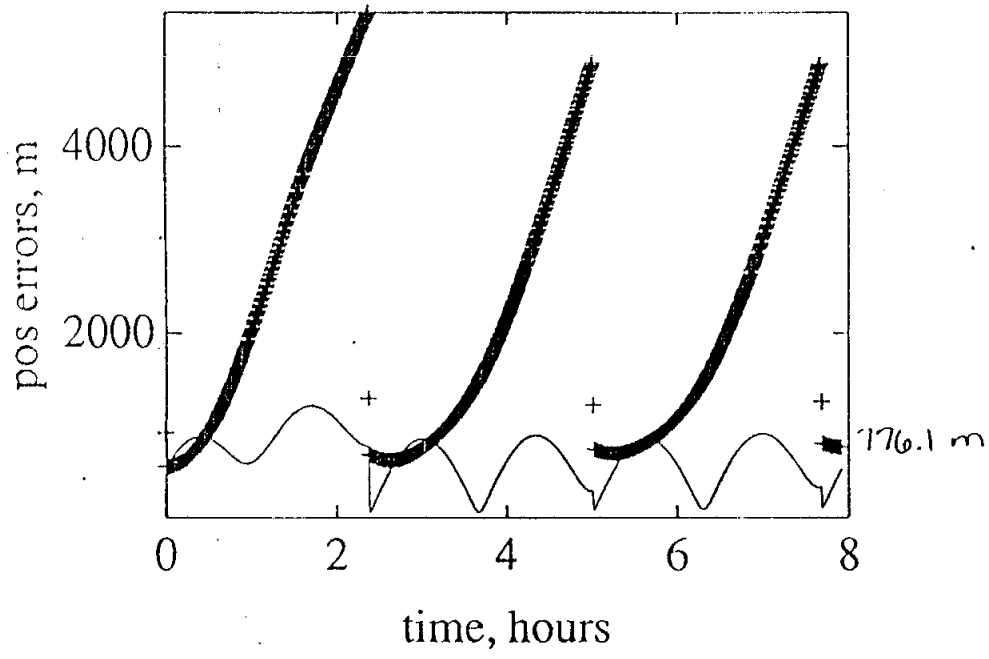


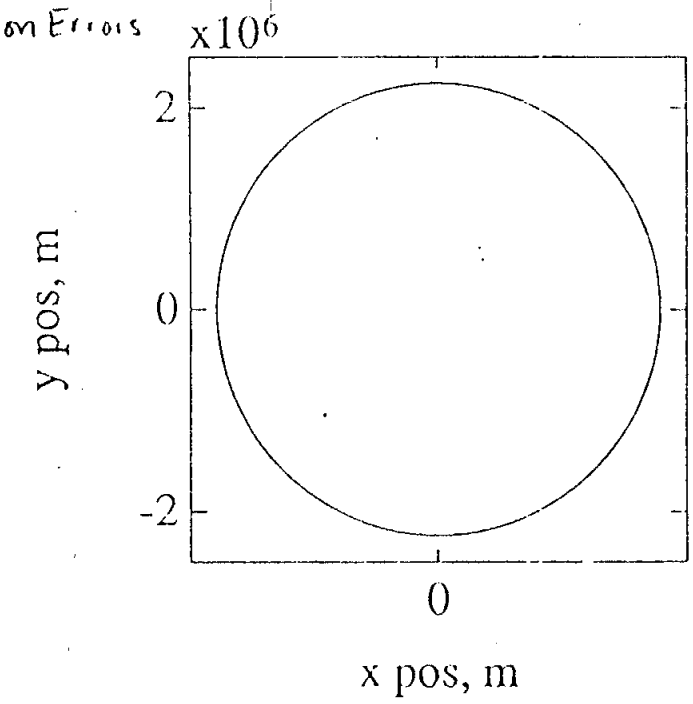
100 Km Circular Orbit, i-4°, : 3rd Beacon Location Unce

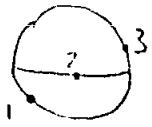


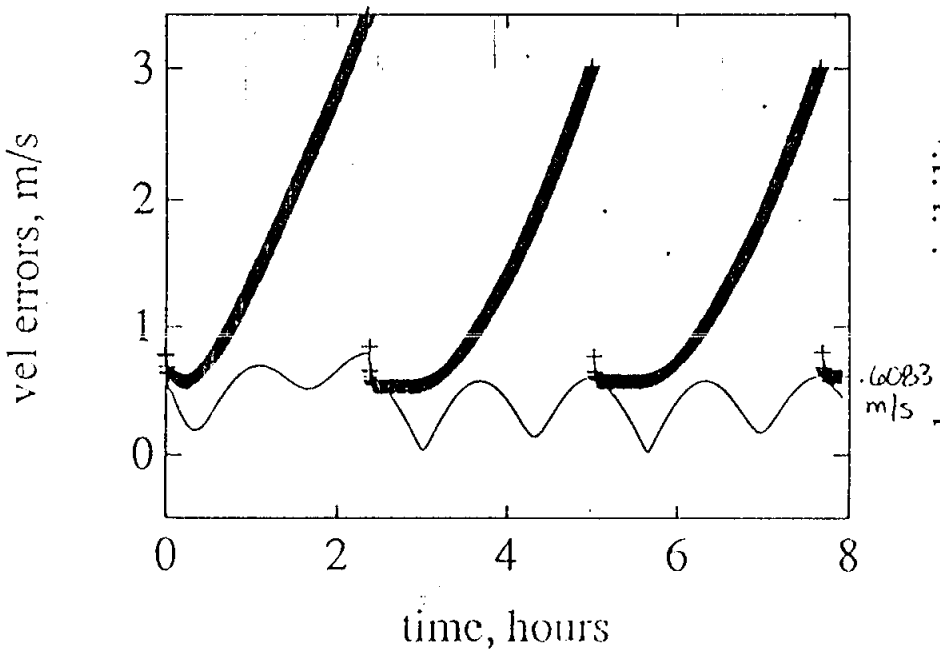
500 km Circular Equato Orbit, 3 revs, 3 beacons

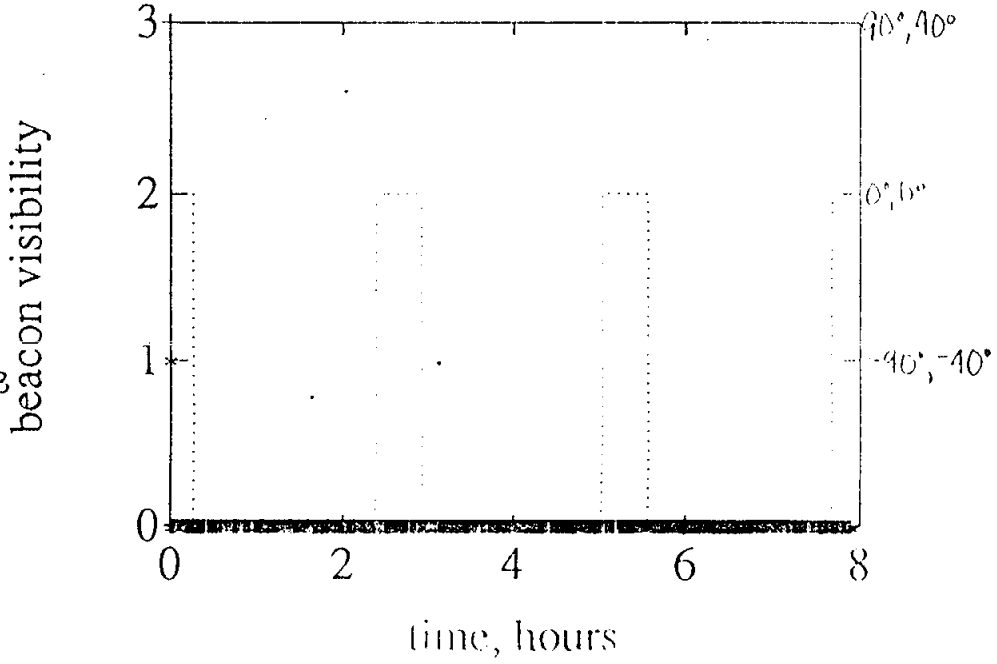
Init. Beacon Errors



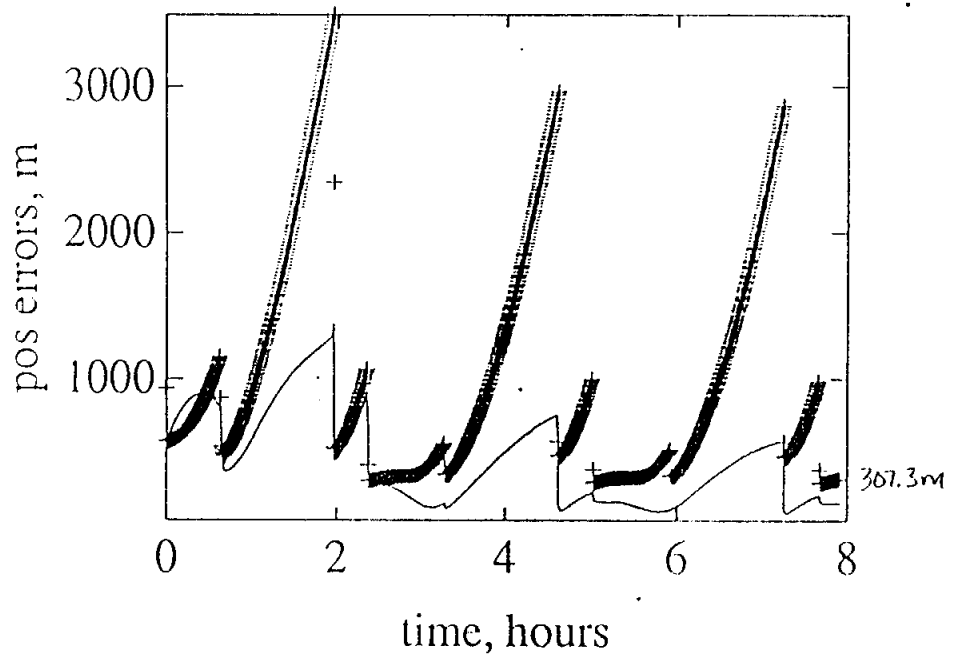


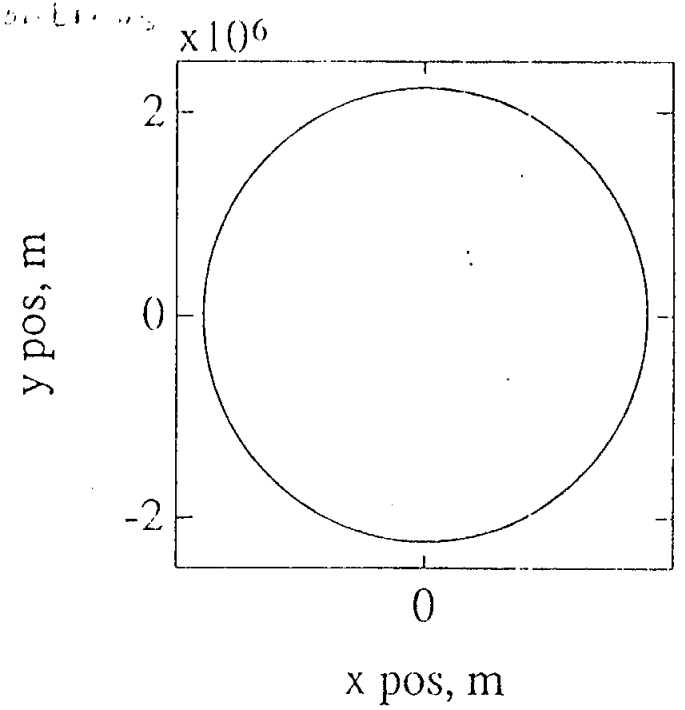


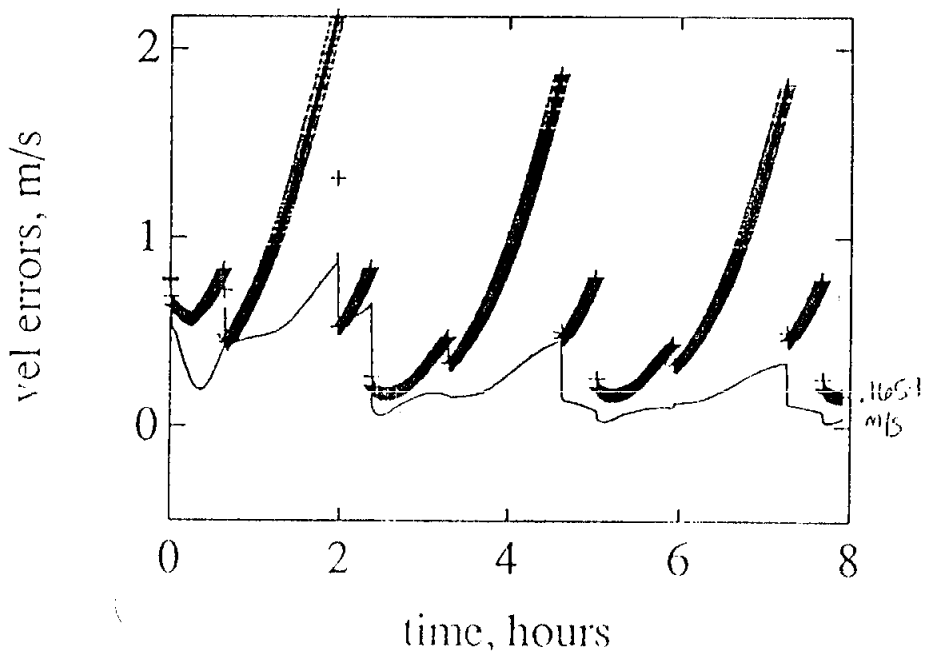


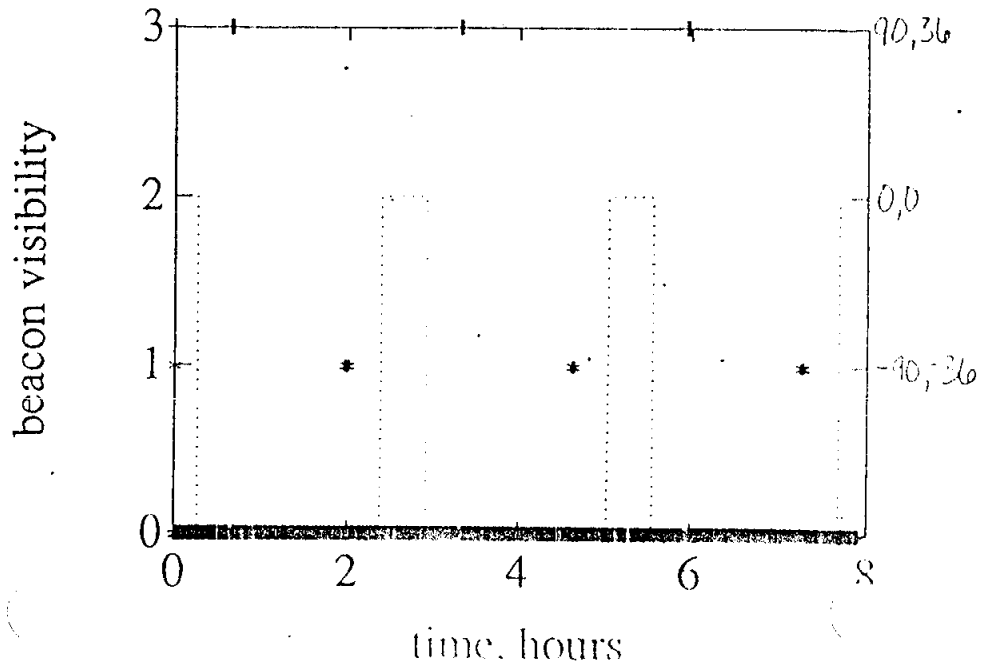


500 km Circular Equat. orbit - 3 revs, 3 beacons



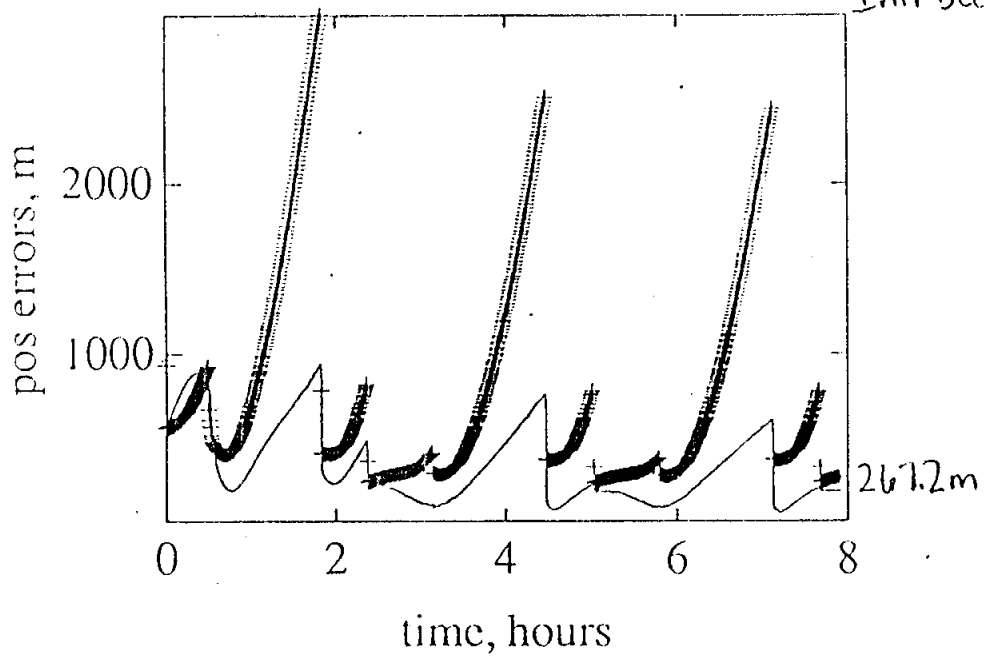


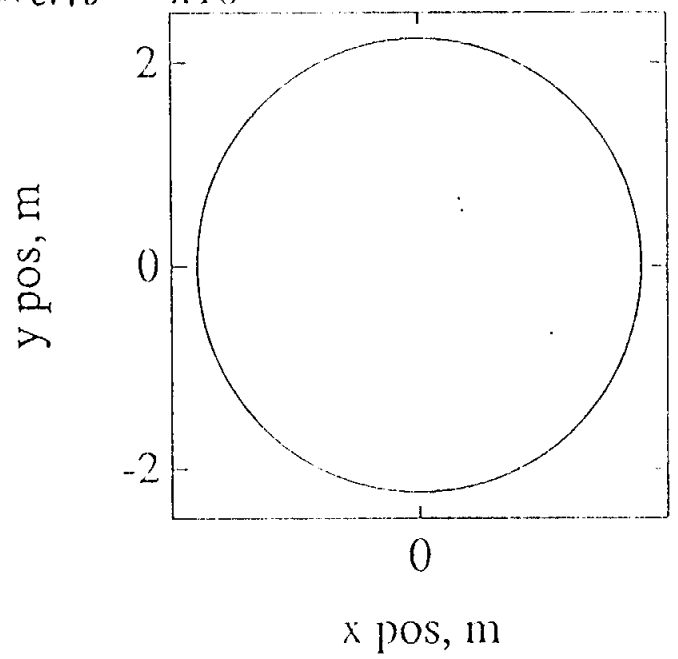


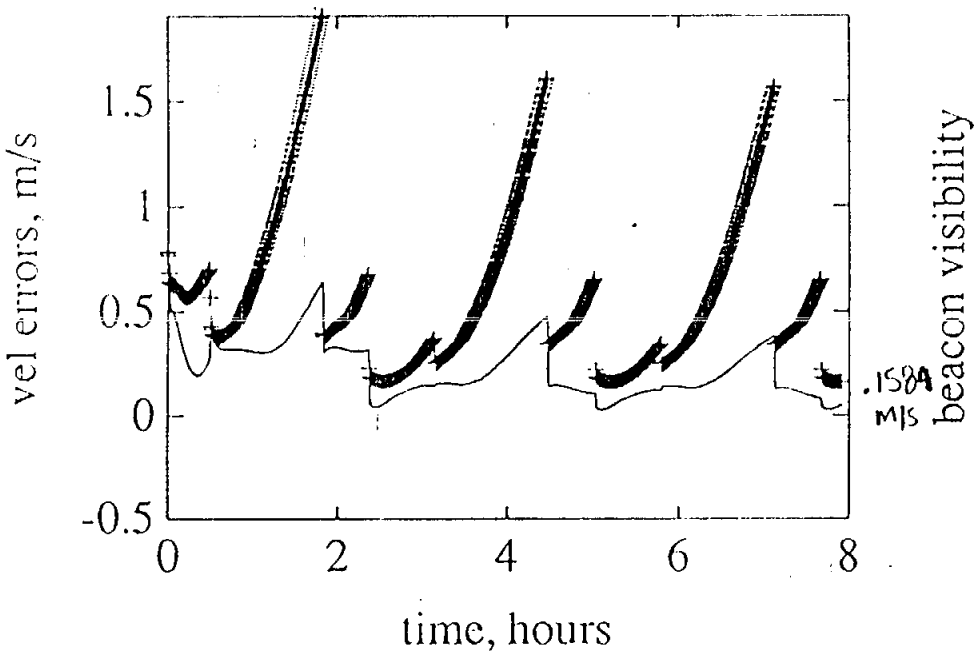


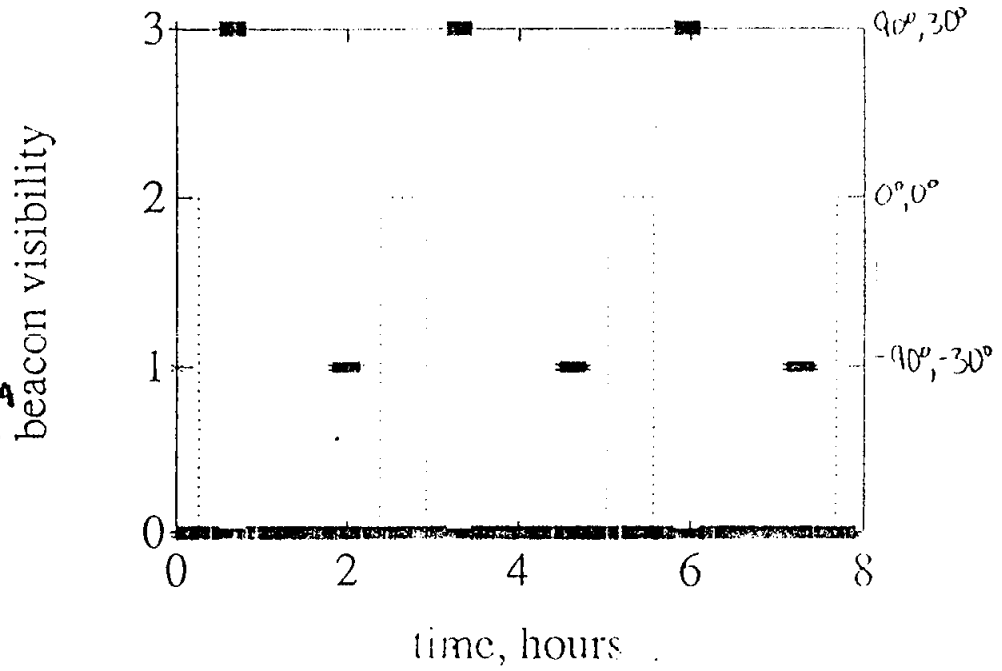
500 Km Circular Equatorial Orbit - 3 revs, 3 beacons

Init Beacon Errors x10^6



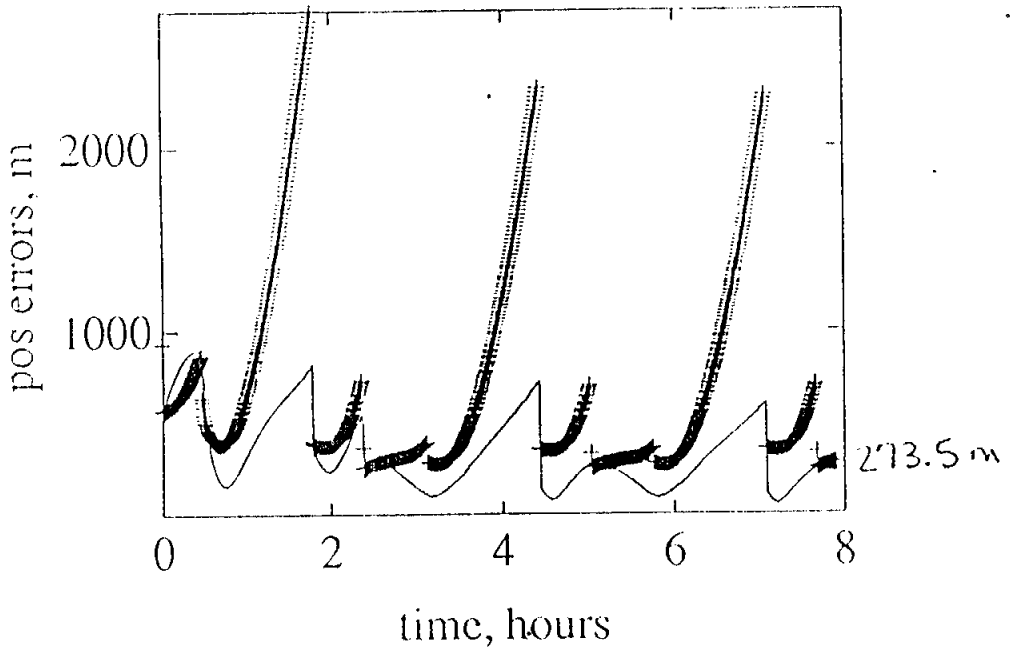


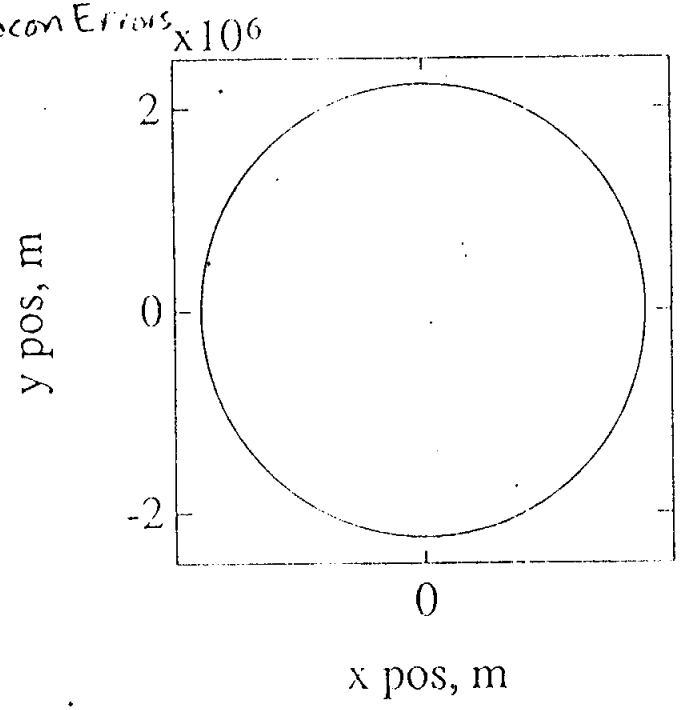


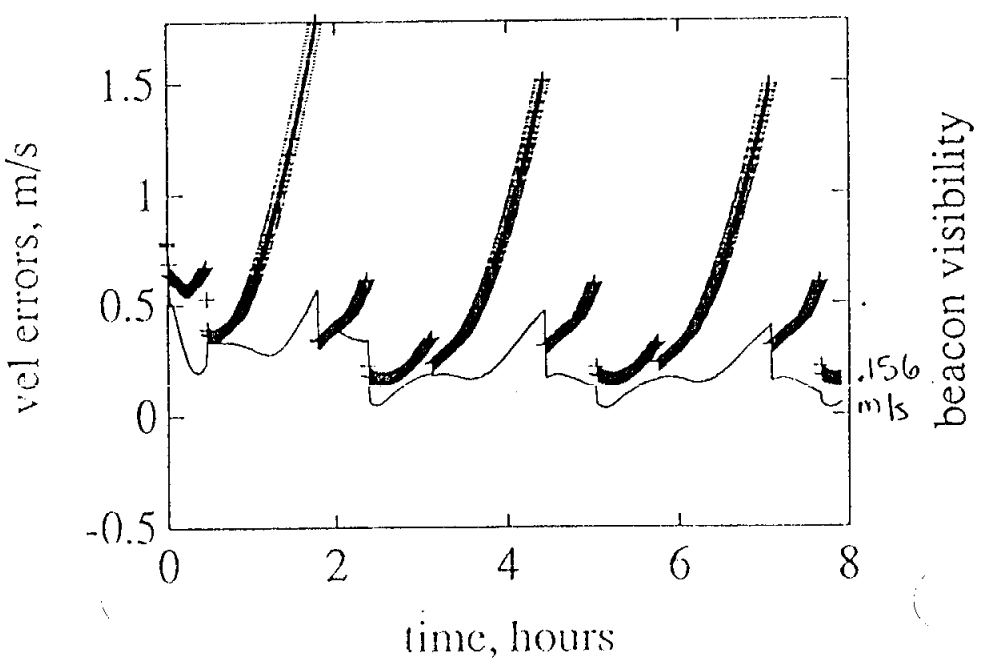


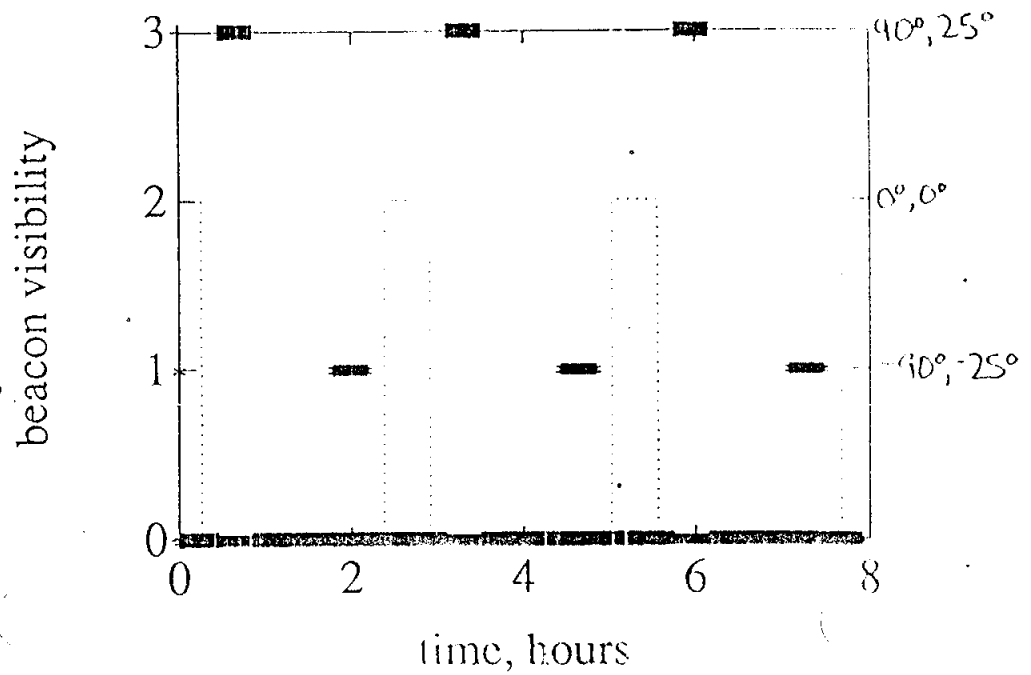
500 Km Circular E[illegible]rial Orbit - 3 reus, 3 be[illegible]

Init. Beacon Errors



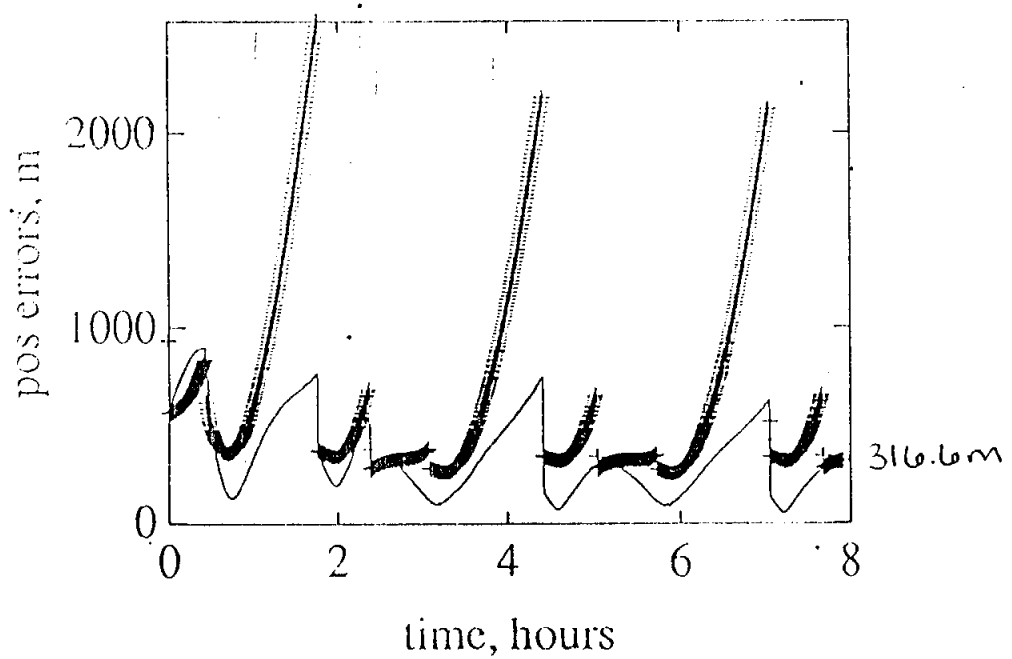


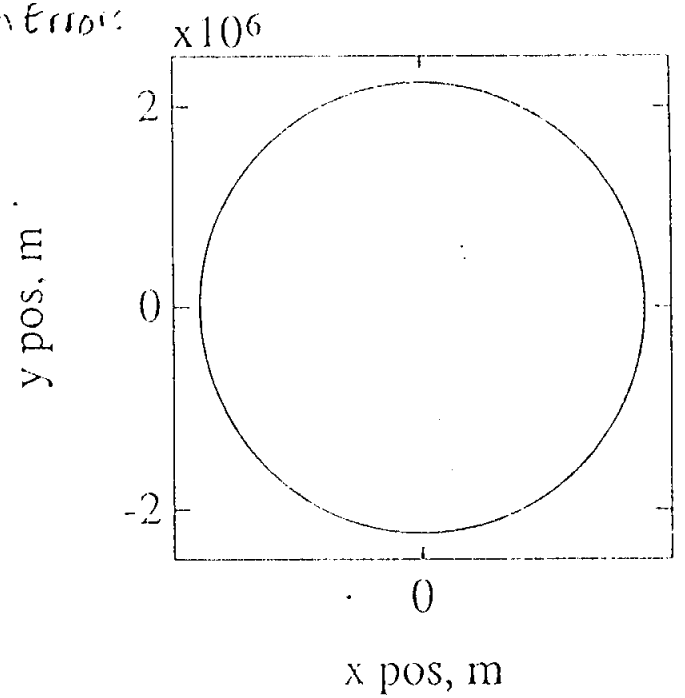


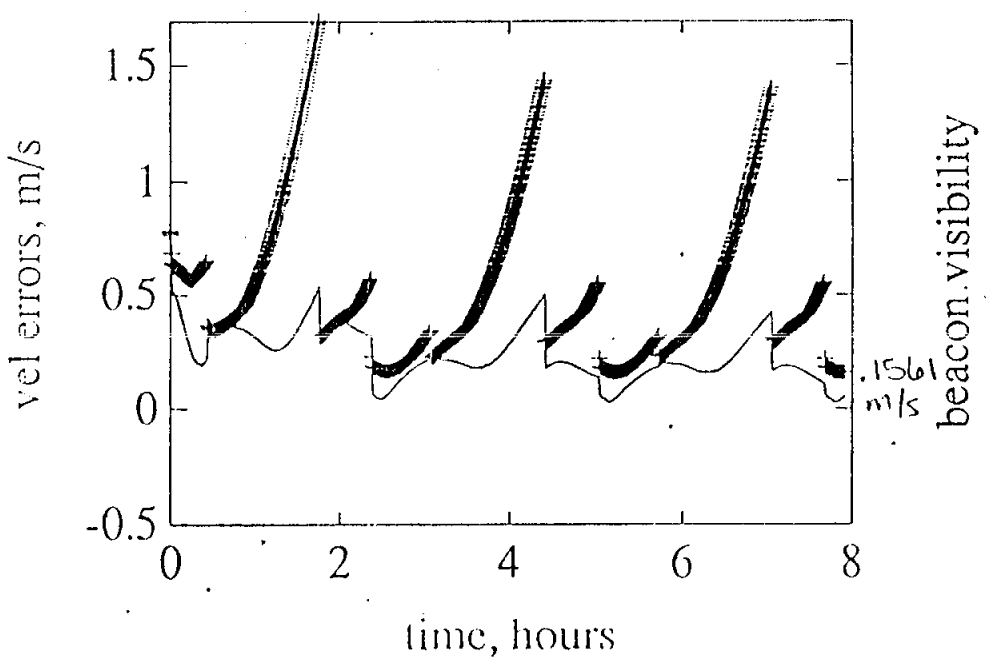


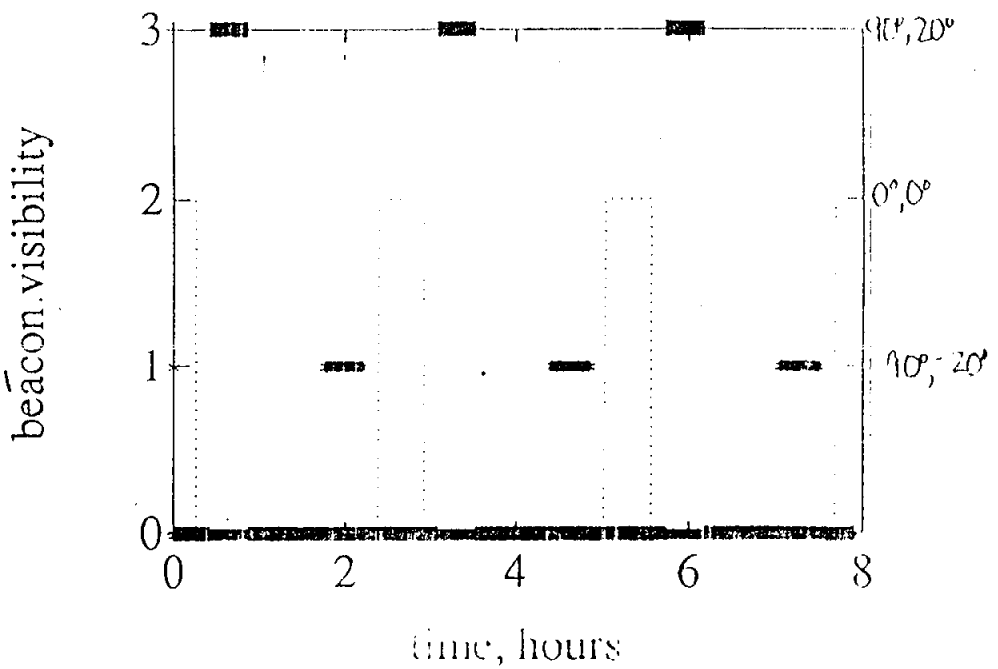
500 Km Circular Equa. Orbit - 3 revs, 3 beacons

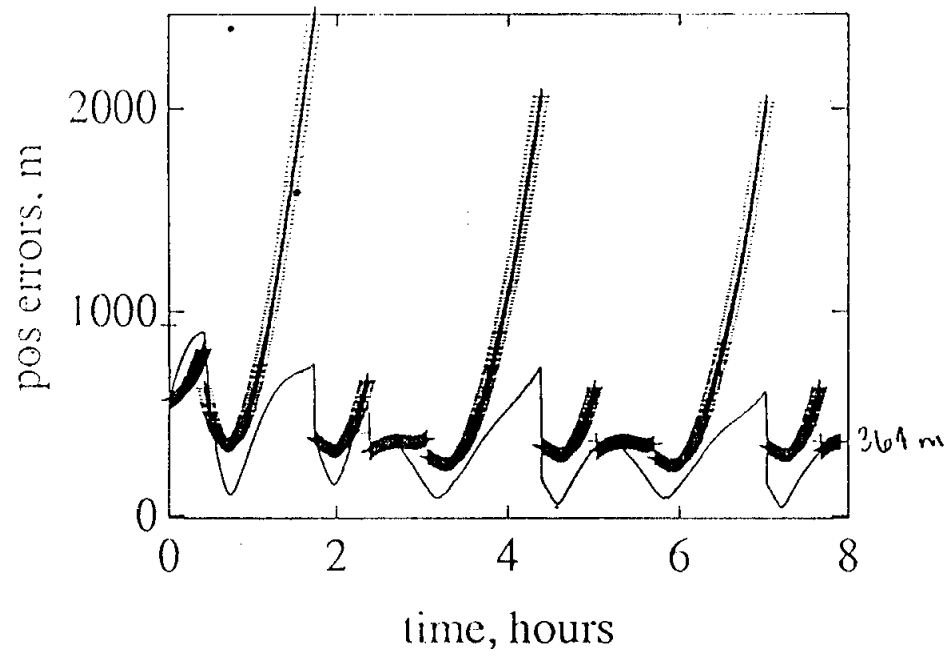
Init Beacon Errors










pos errors, m
2000
1000
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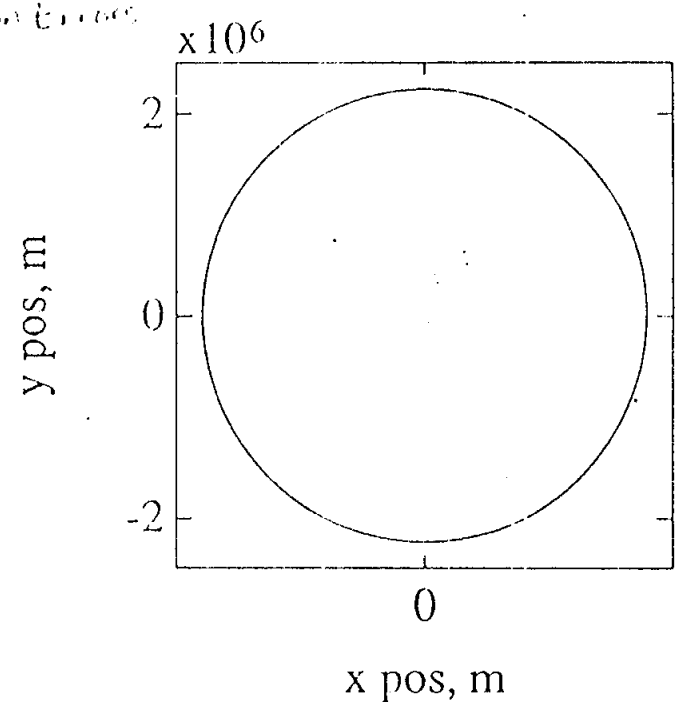
0
2
4
6
8
time, hours
369 m


x10^6
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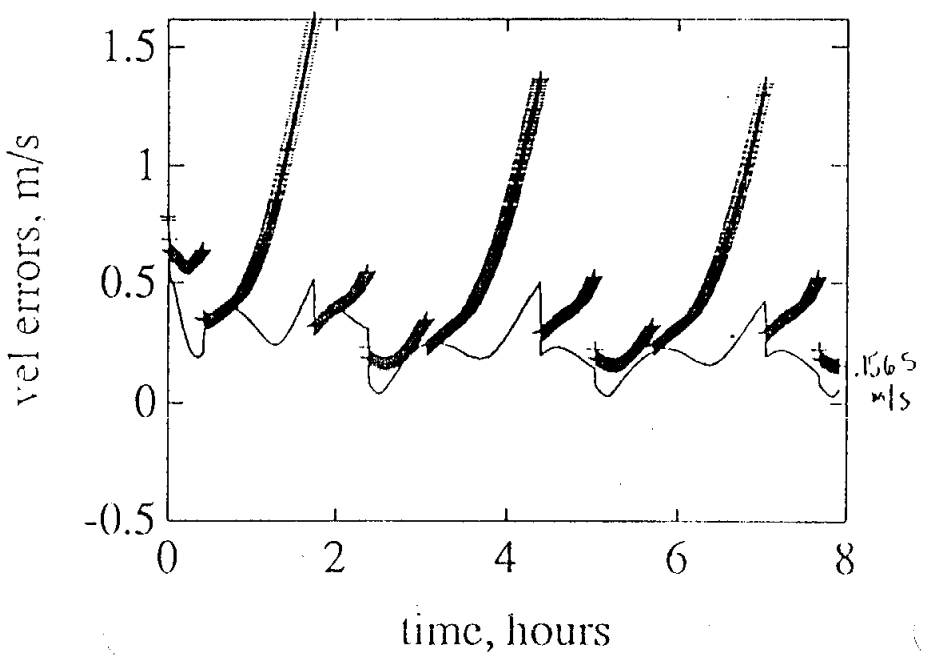
-2
y pos, m
0
x pos, m


vel errors, m/s
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0.5
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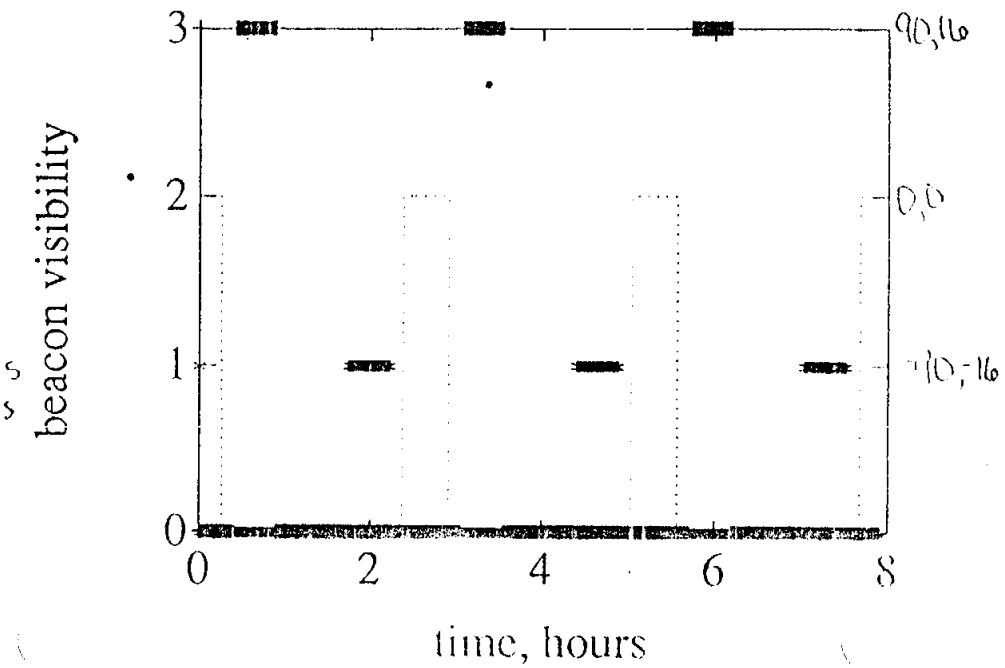
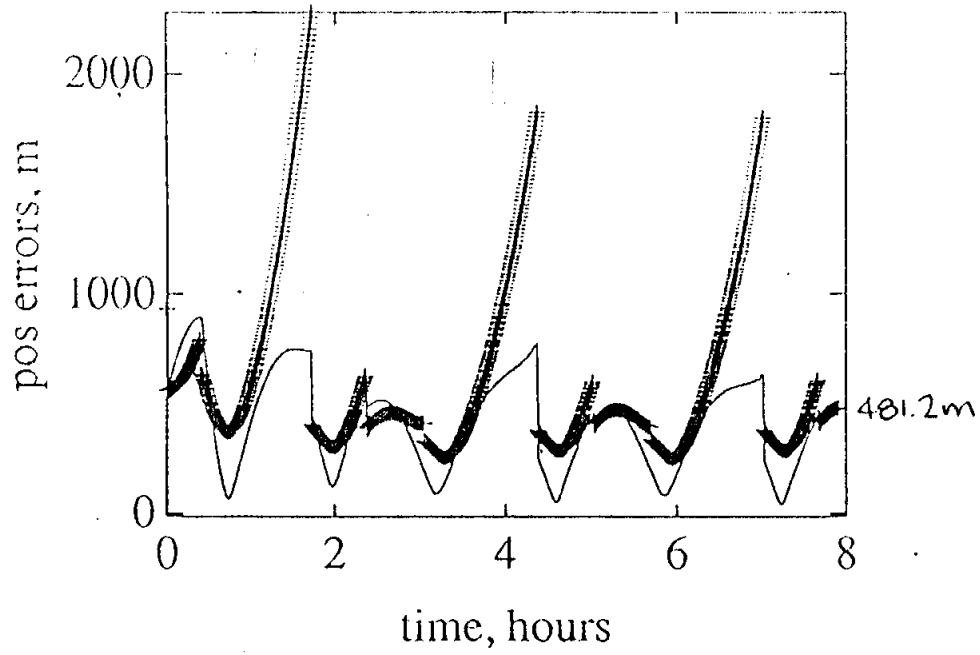
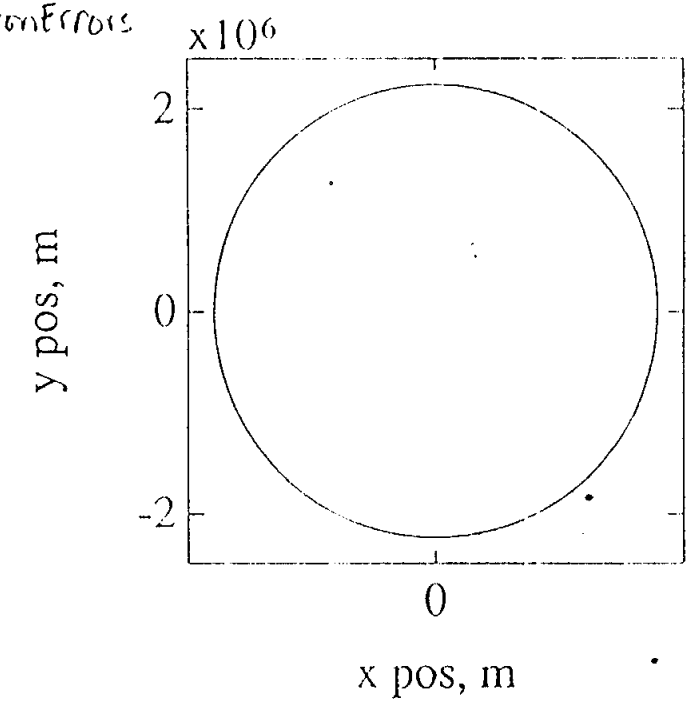
-0.5
0
2
4
6
8
time, hours
.1565 m/s

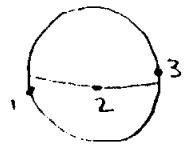

beacon visibility
3
2
1
0
0
2
4
6
8
time, hours
90,16
0,0
-10,-16

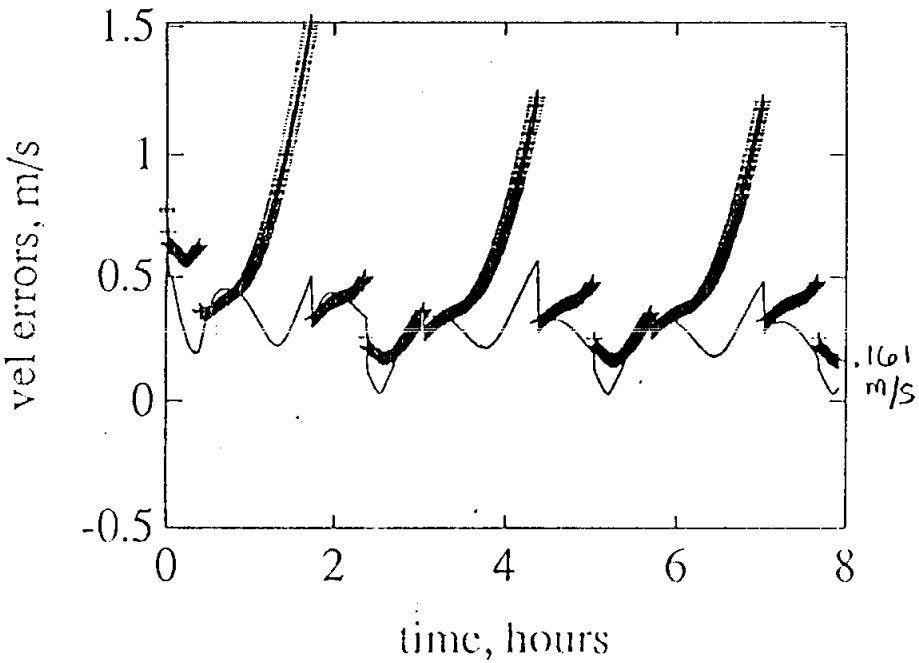
500 km Circular Equ 1 Orbit - 3 revs, 3 beaco

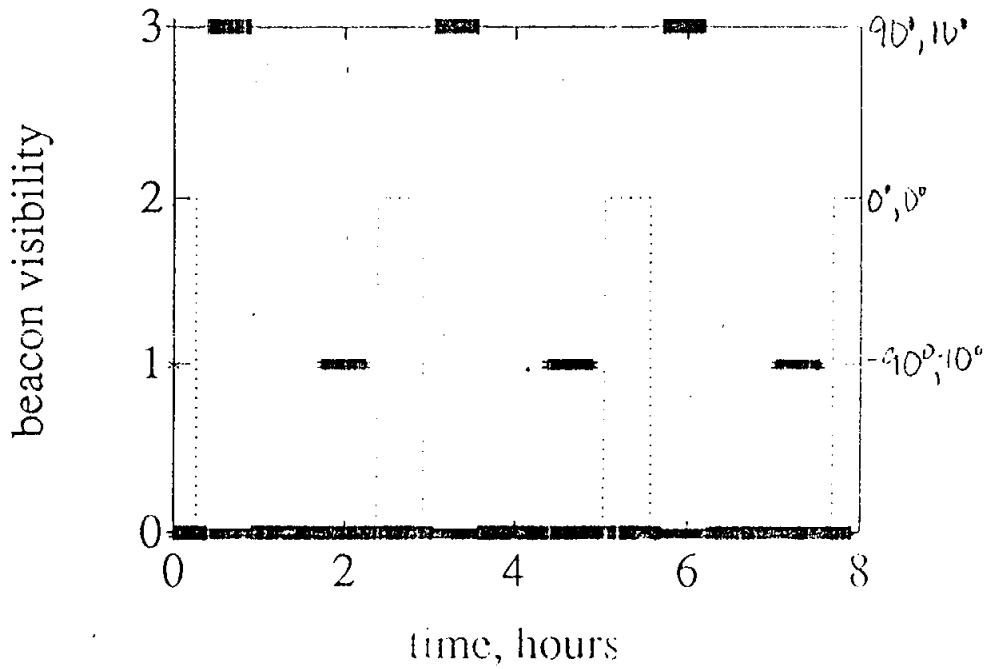
Init. Beacon Errors





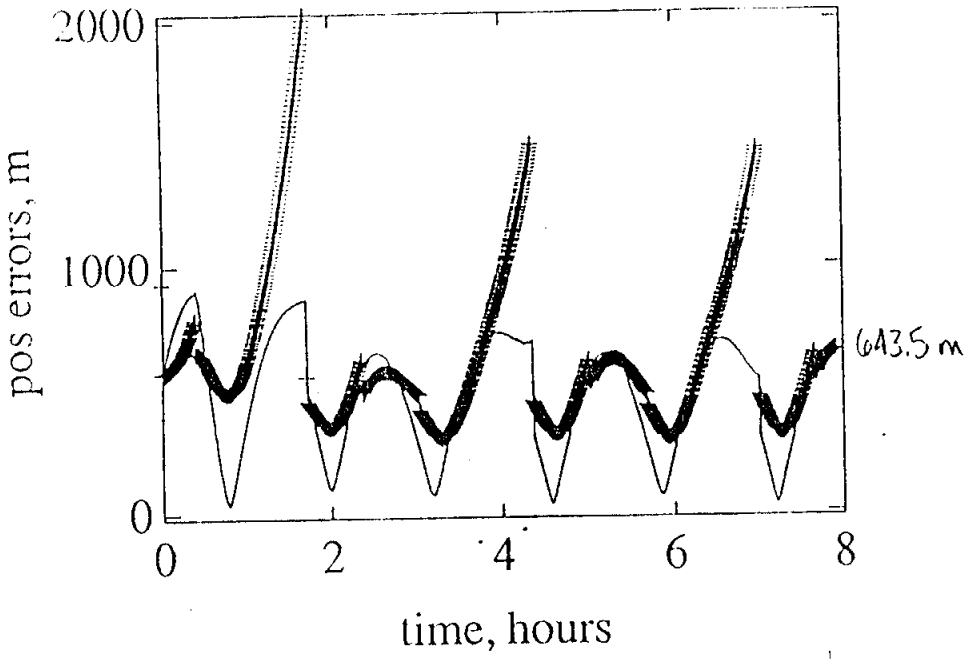


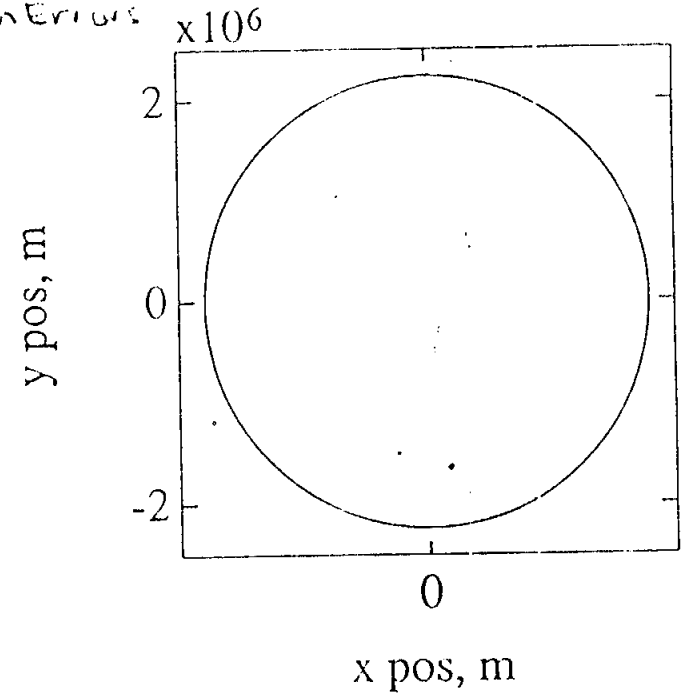


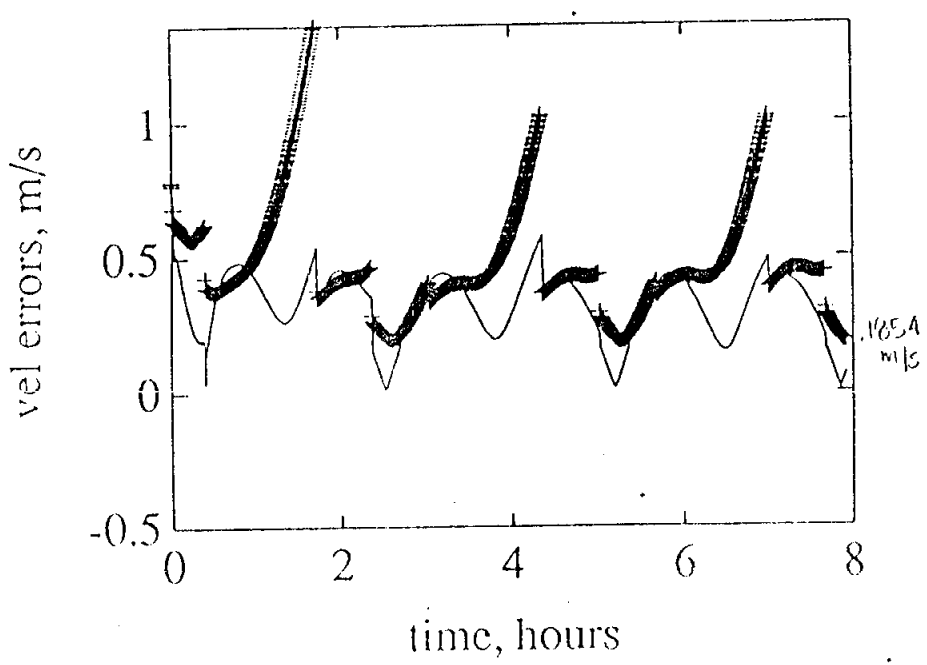


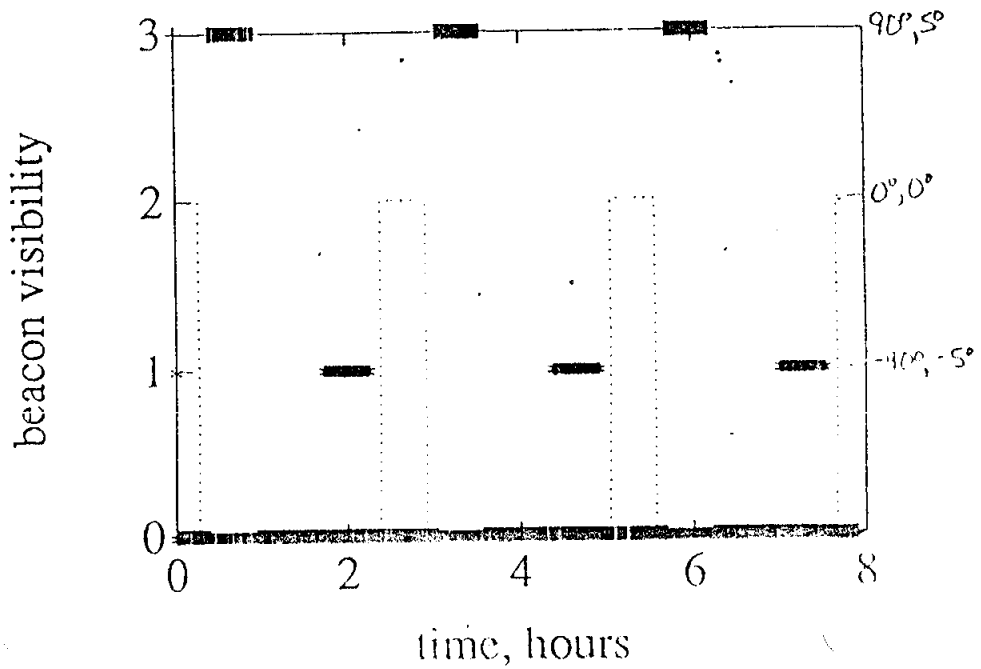
500 km Circular Equat Orbit - 2 revs, 3 beac

Ini Beacon Errors



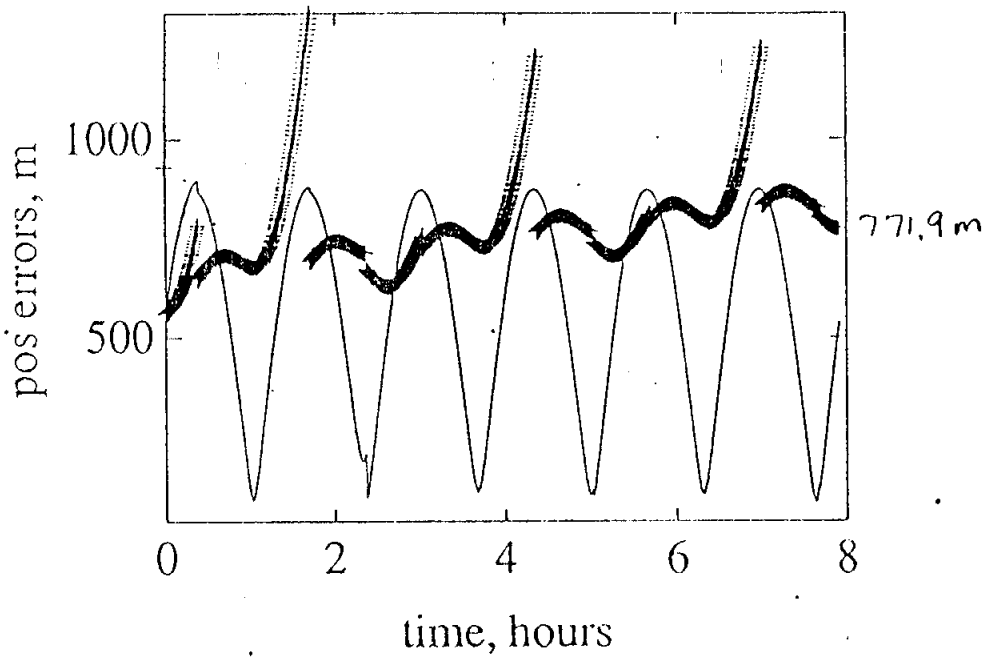


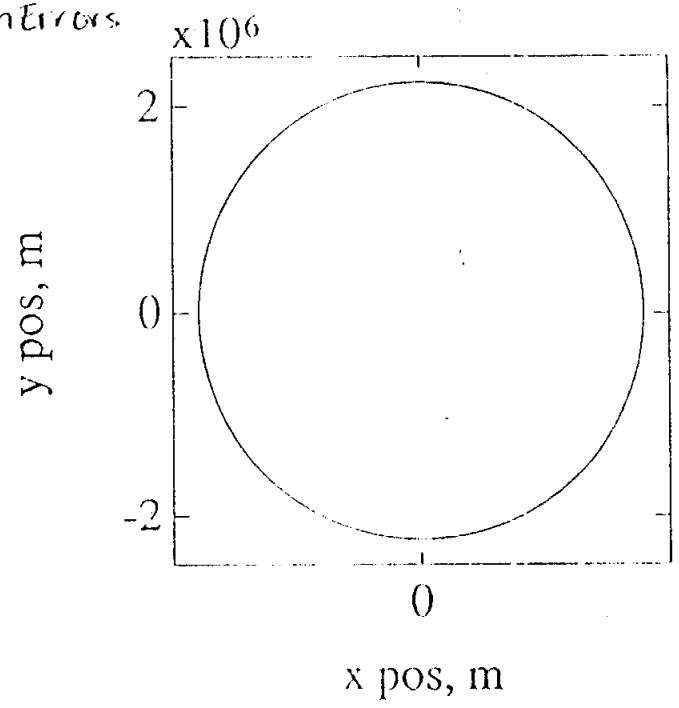


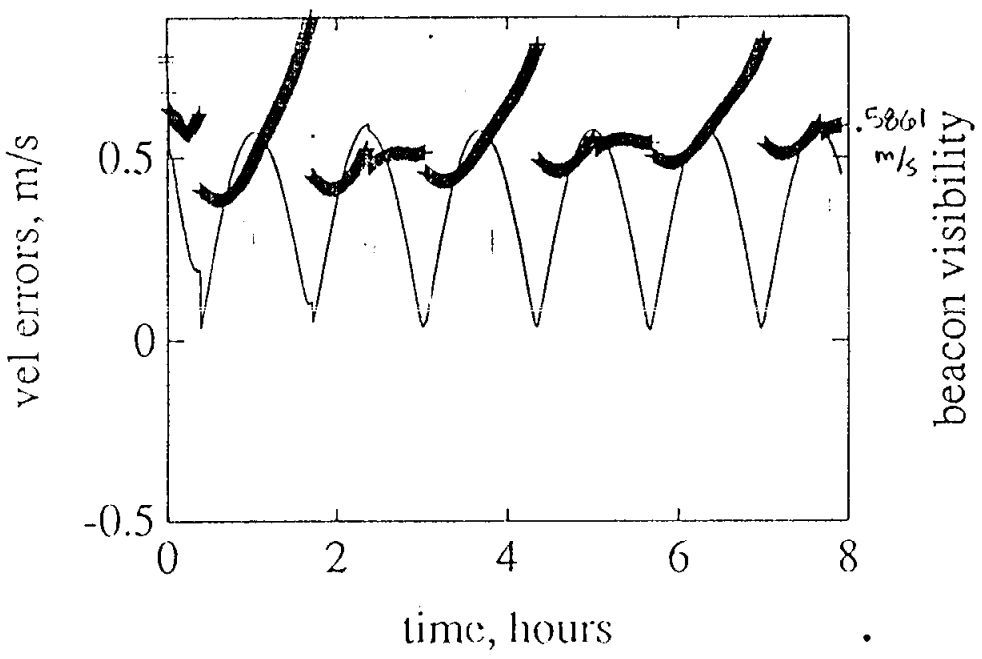


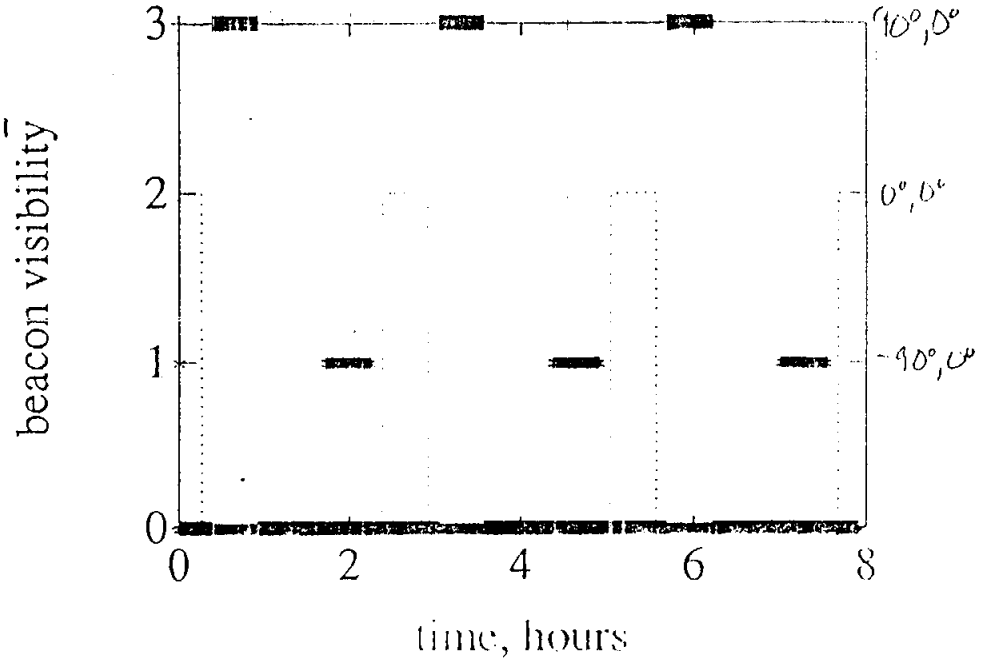
500 km Circular Orbit - 3 revs, 3 bea

Init. Beacon Errors

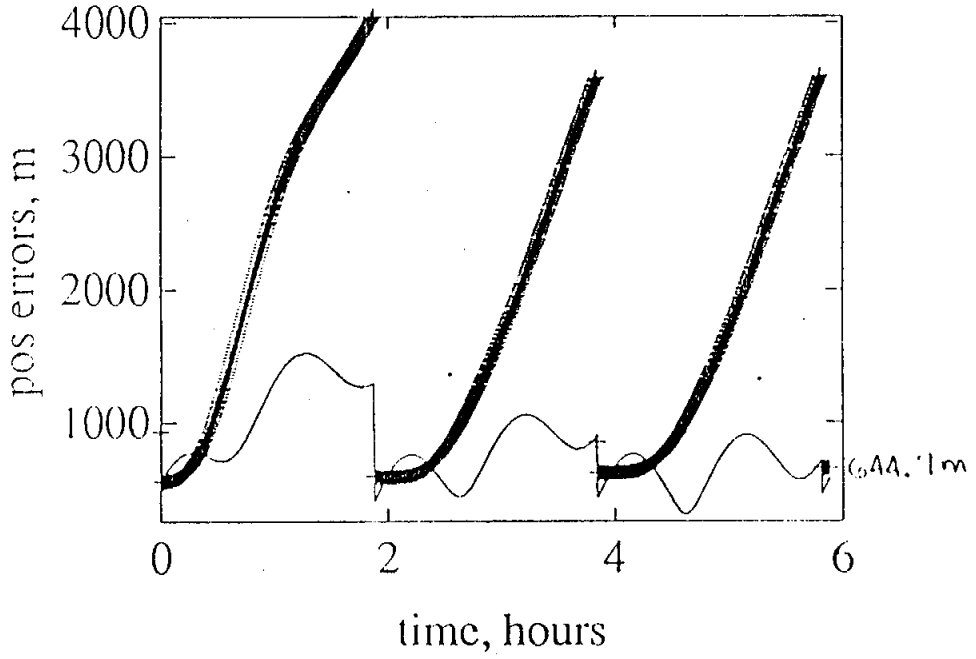


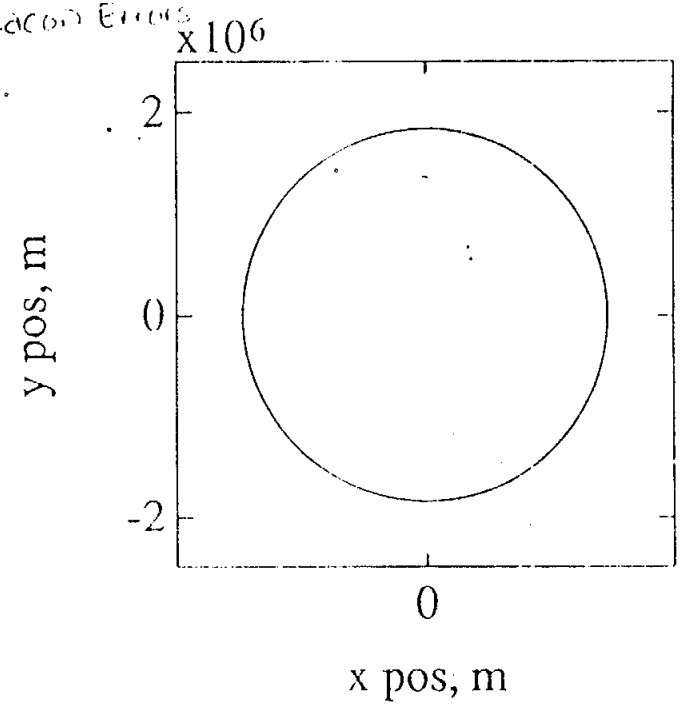


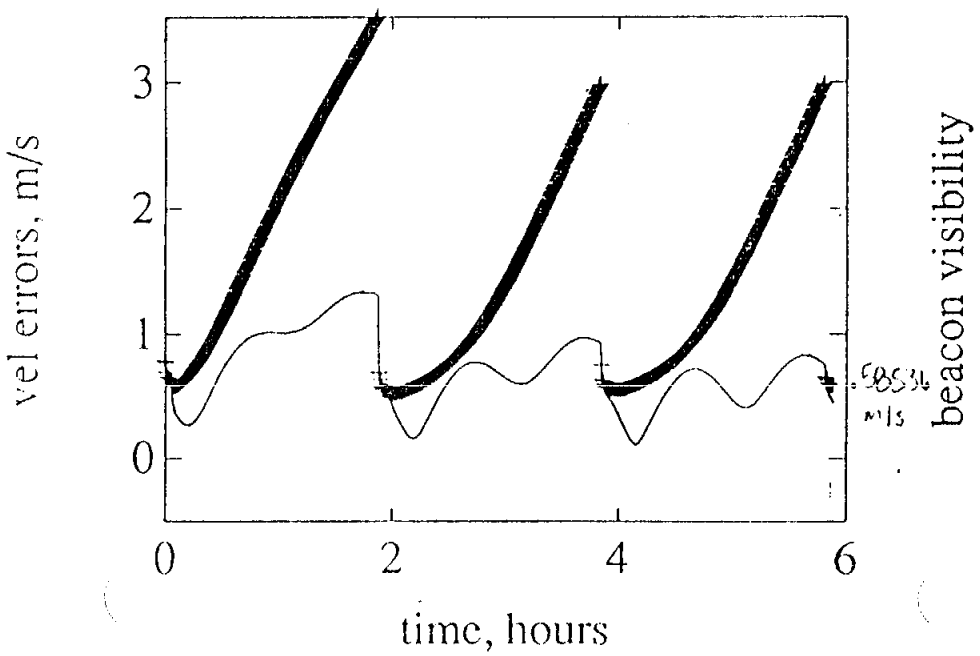


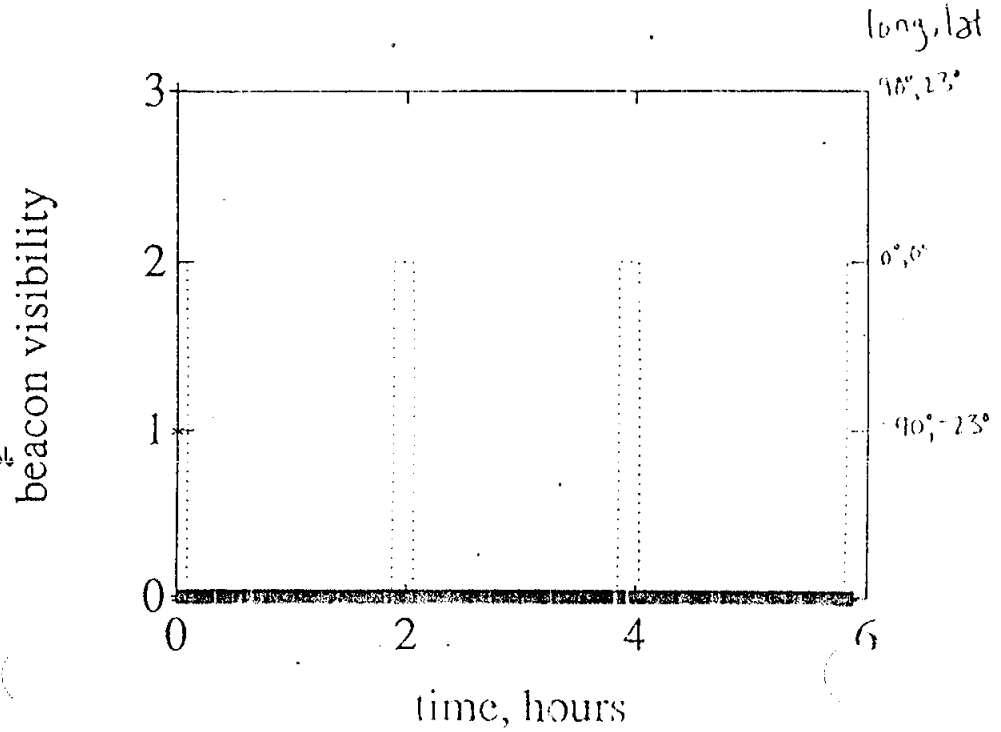


100 km Circular LEO Initial Errors, 3 beacons



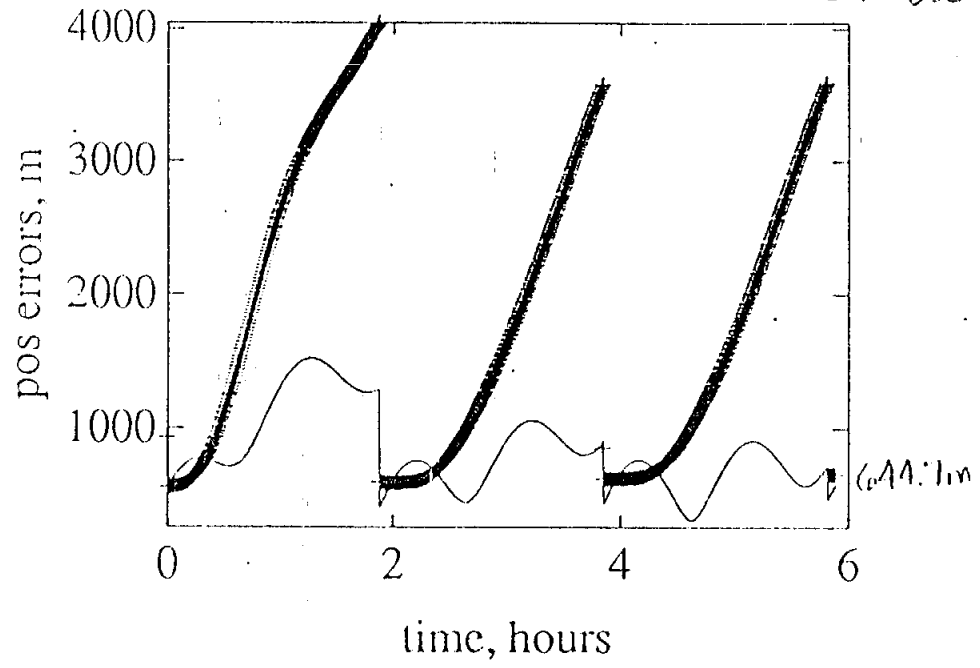


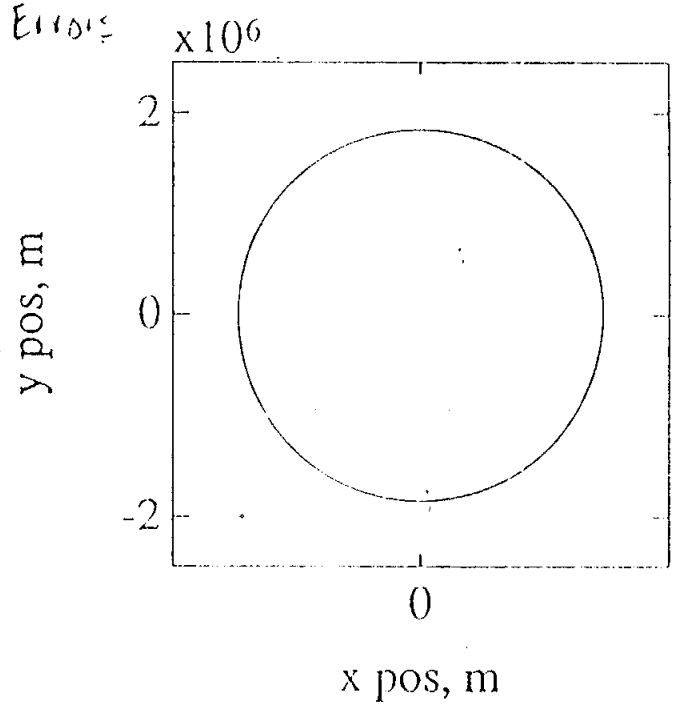


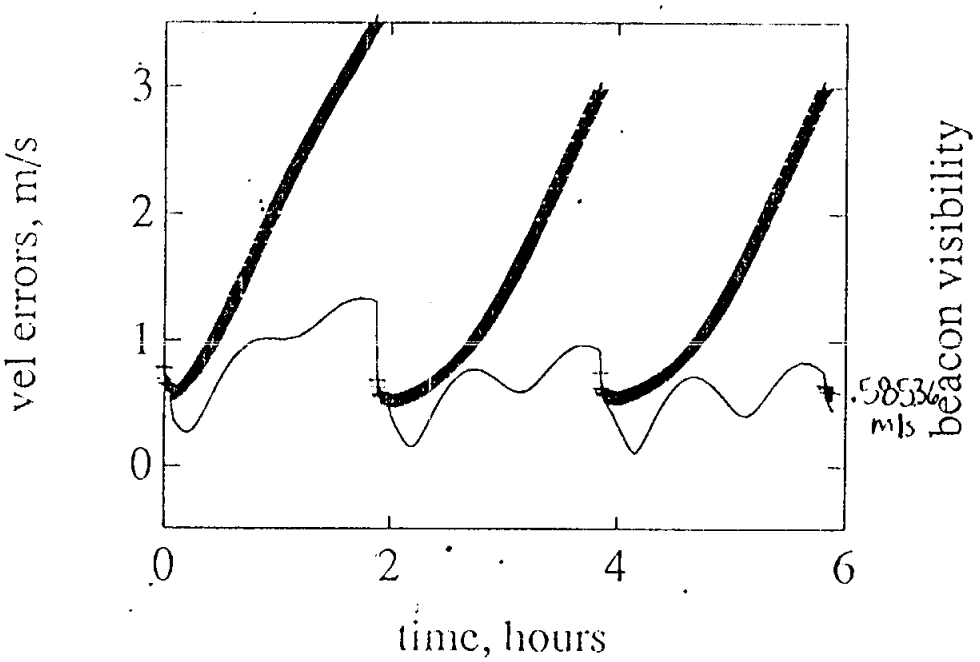


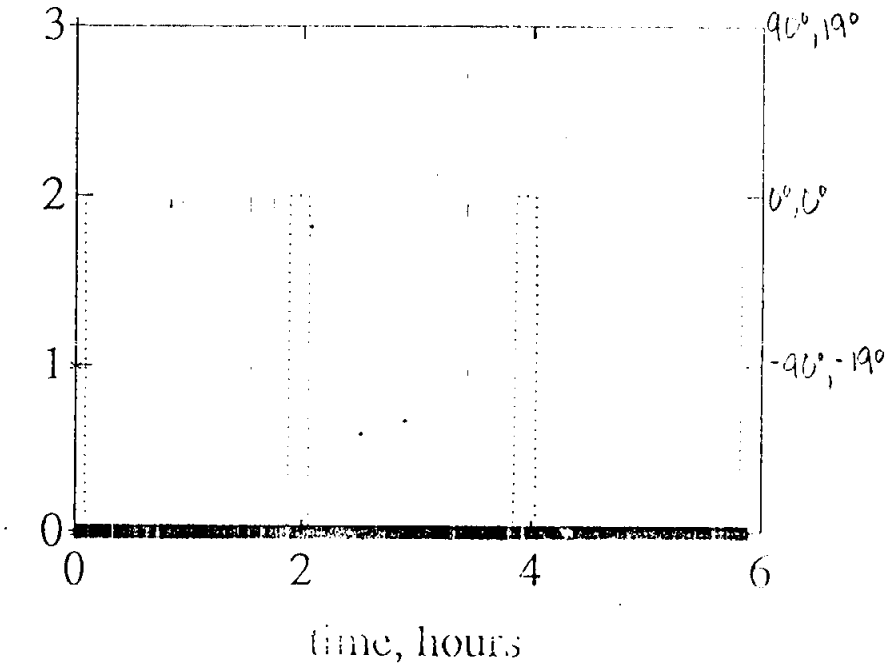
100 km Circular Equato Orbit - 3 revs, 3 beacons

Init Beacon Errors

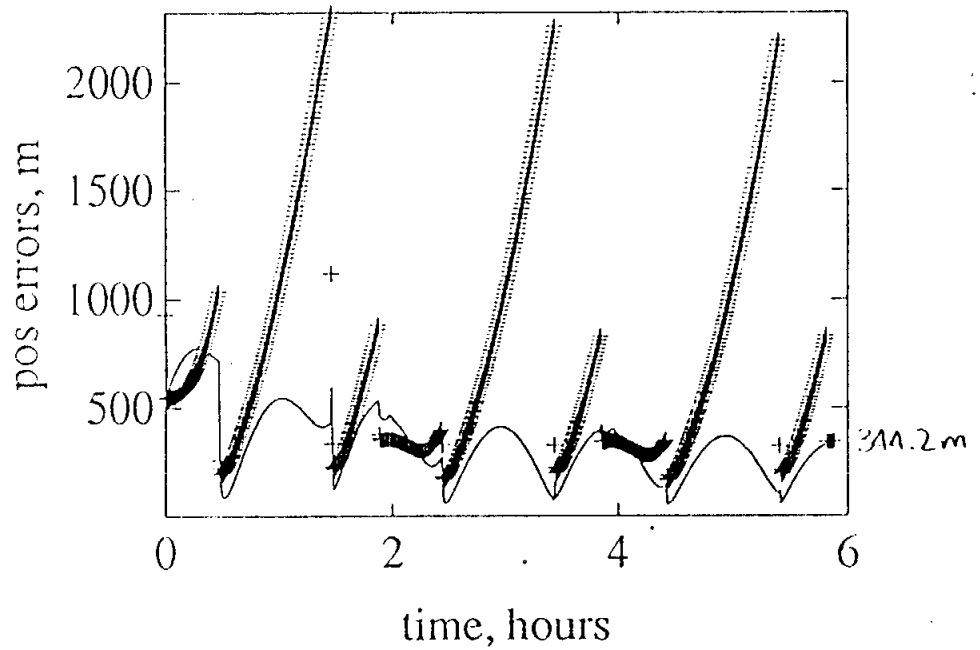


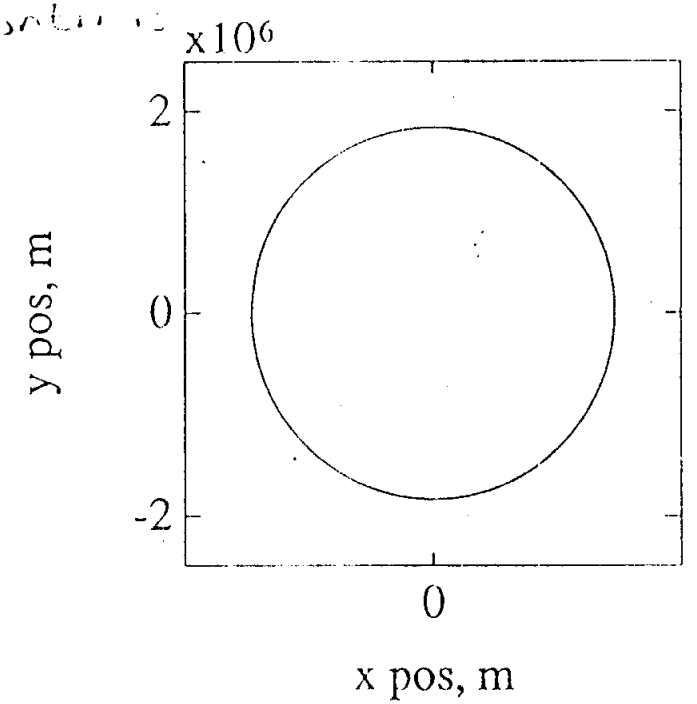


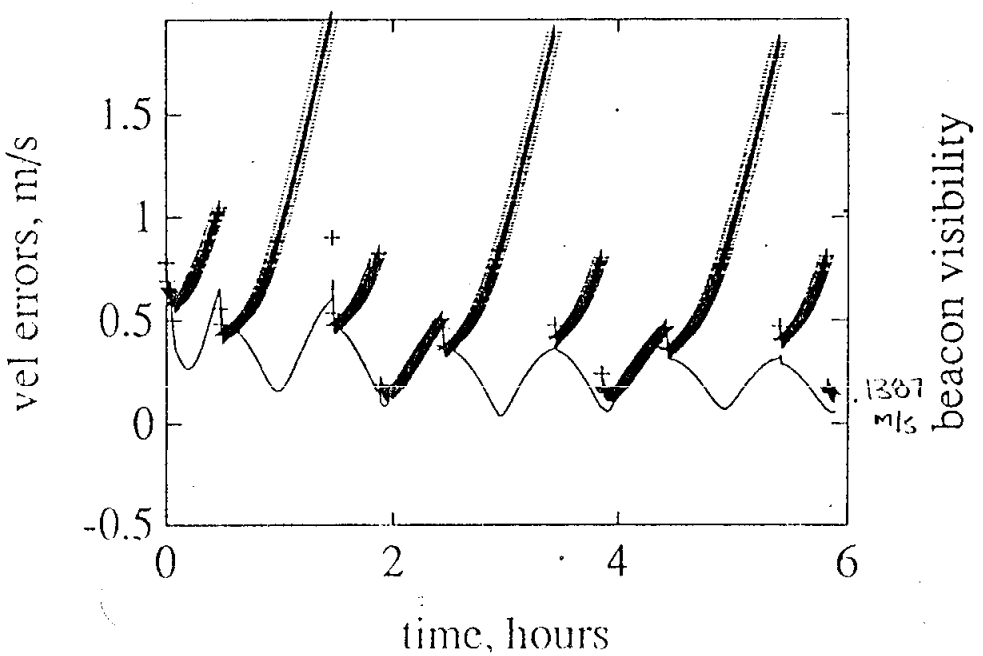


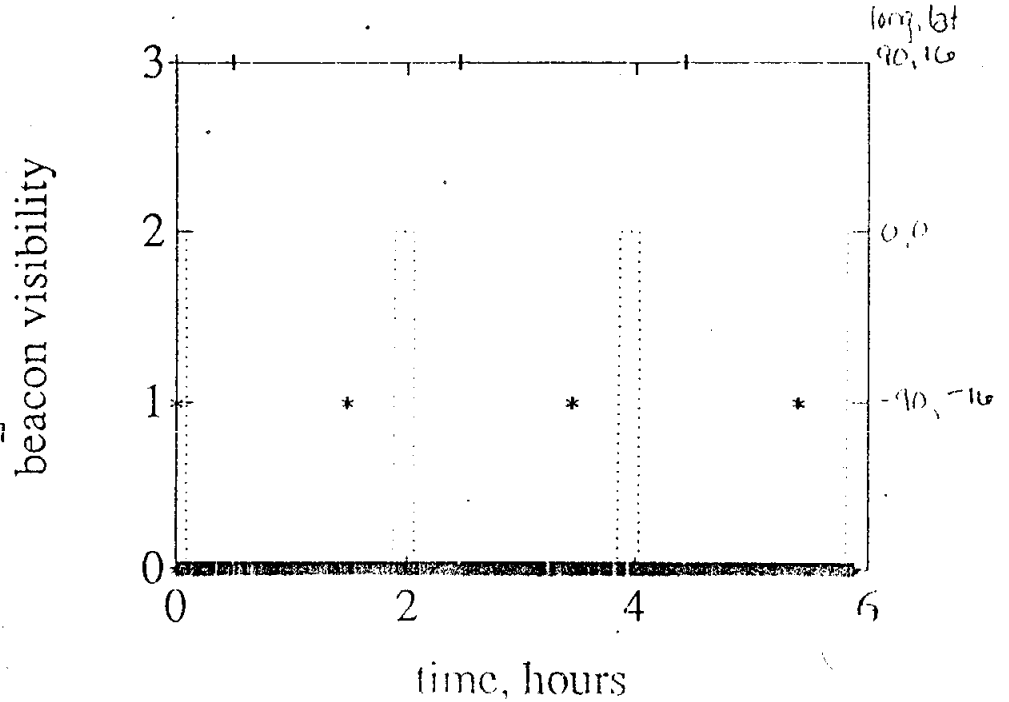


100 km Circular Eq[illegible]rial Orbit - 3 revs, 3 be[illegible]s

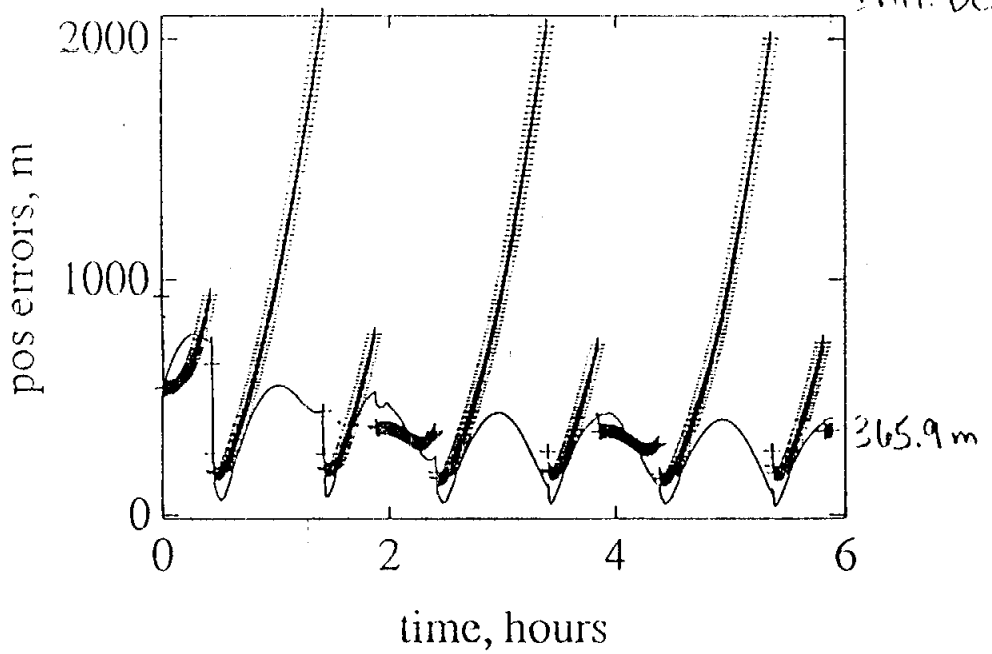


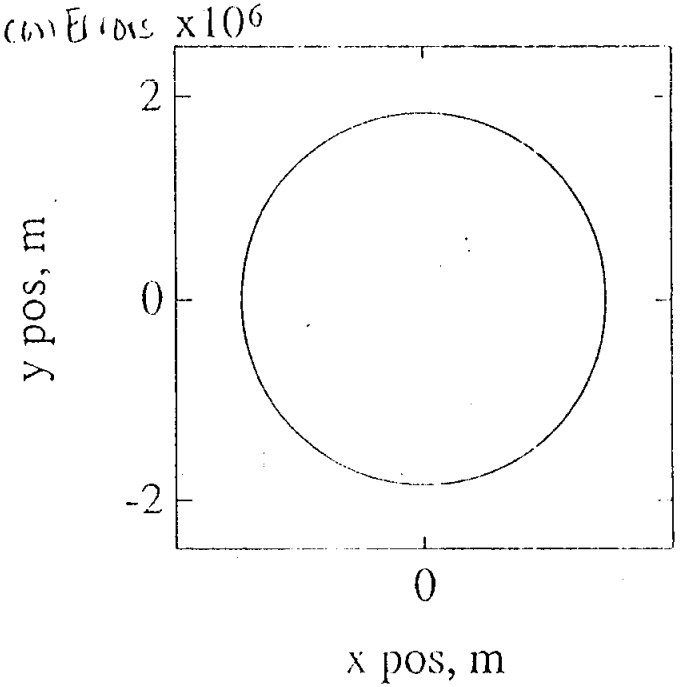


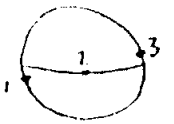


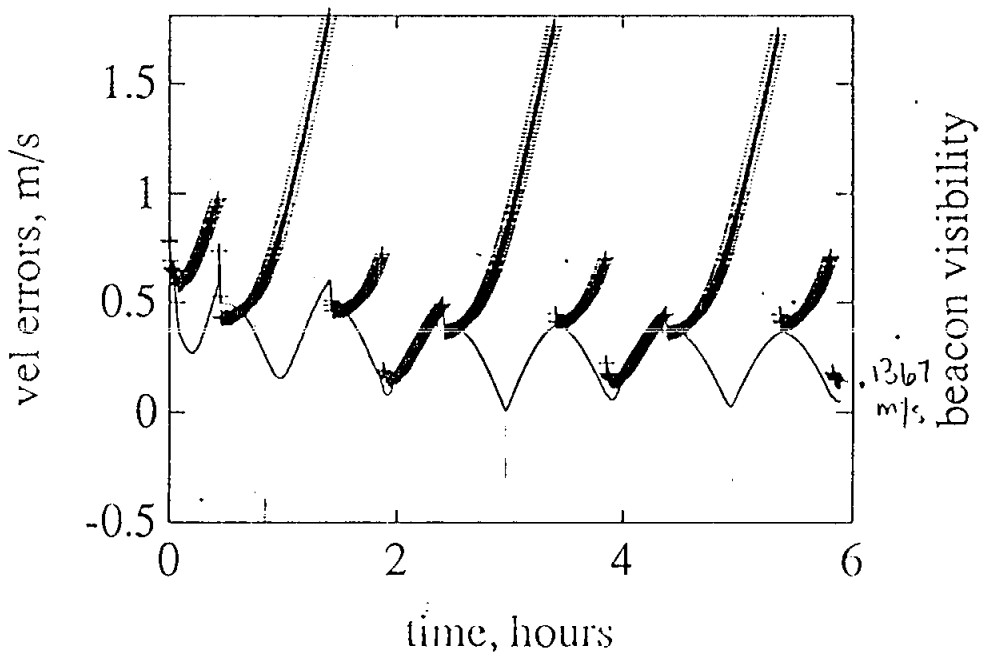


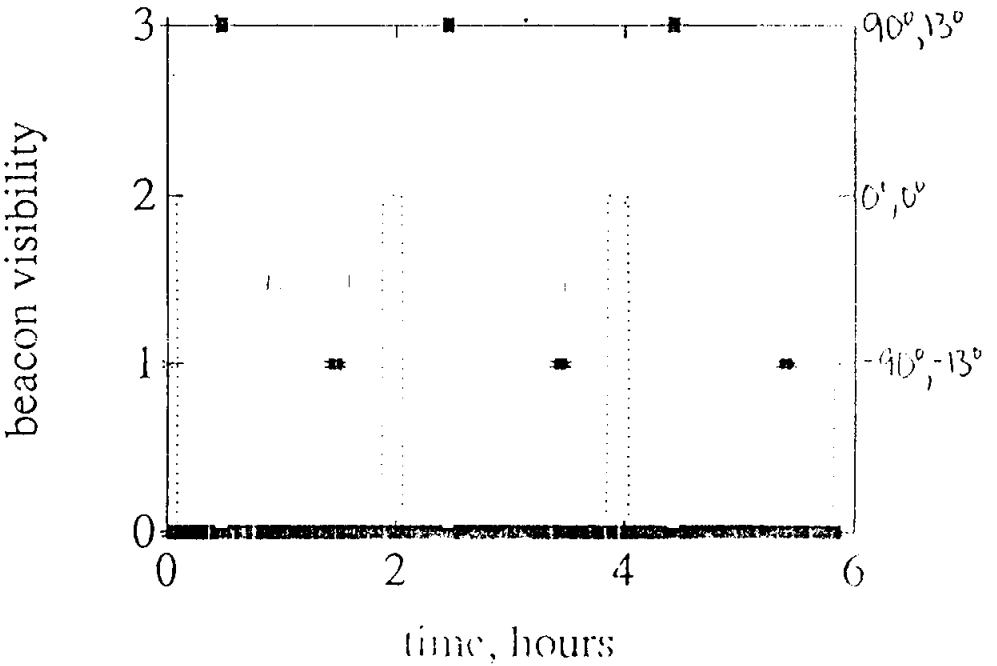
100 Km Circular Equator orbit - 3 revs, 3 beacons





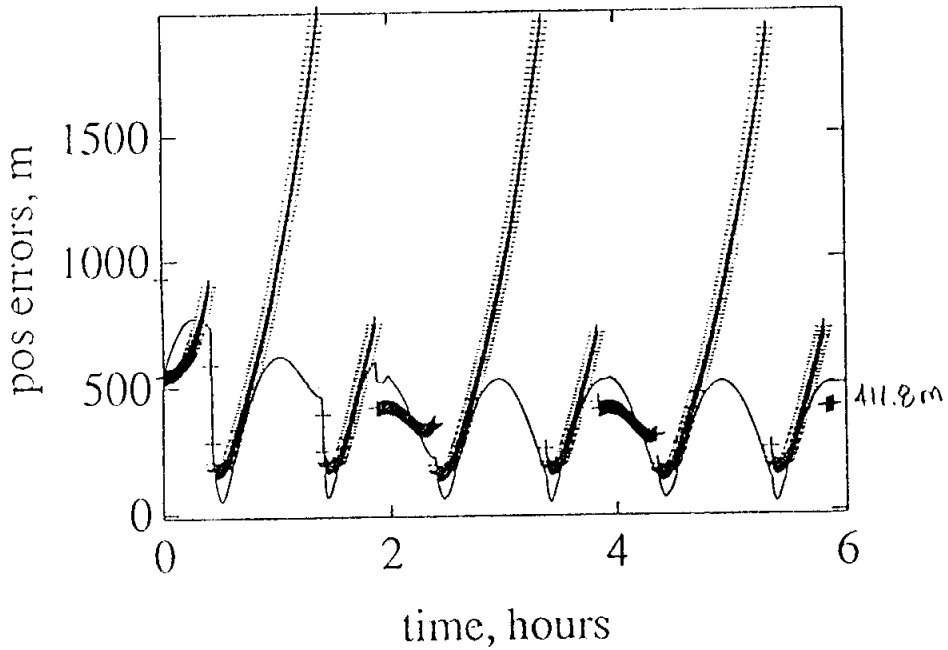


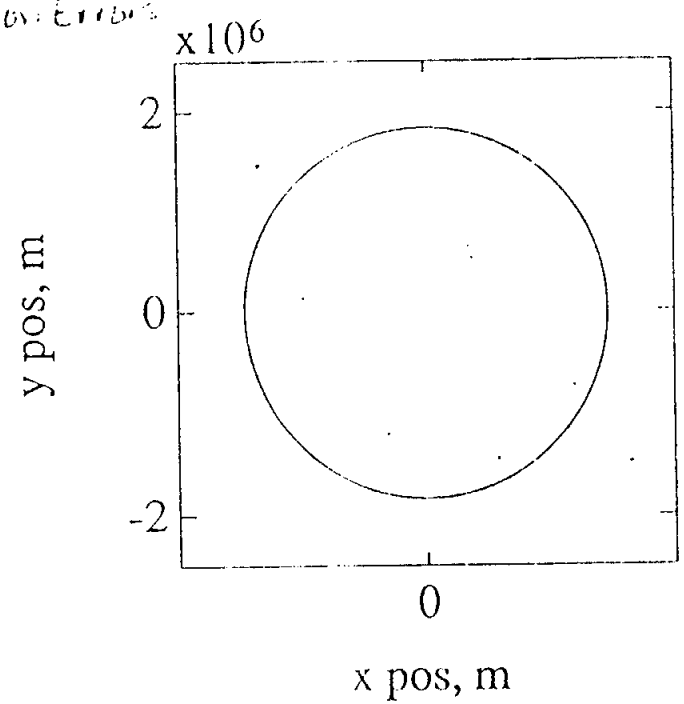


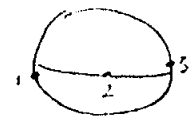


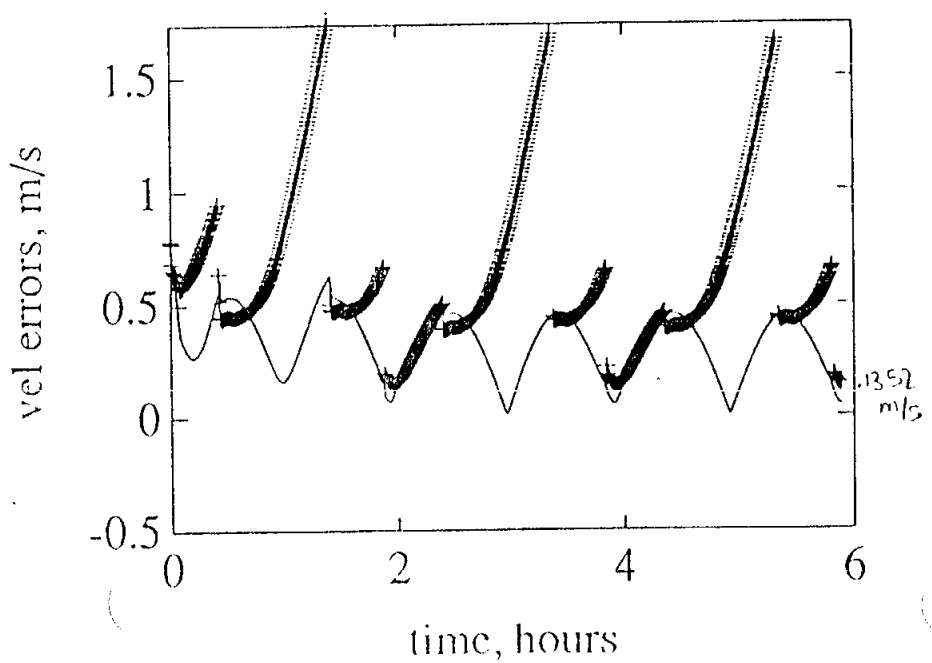
100 km Circular Equator bit - 3 revs, 3 beacons

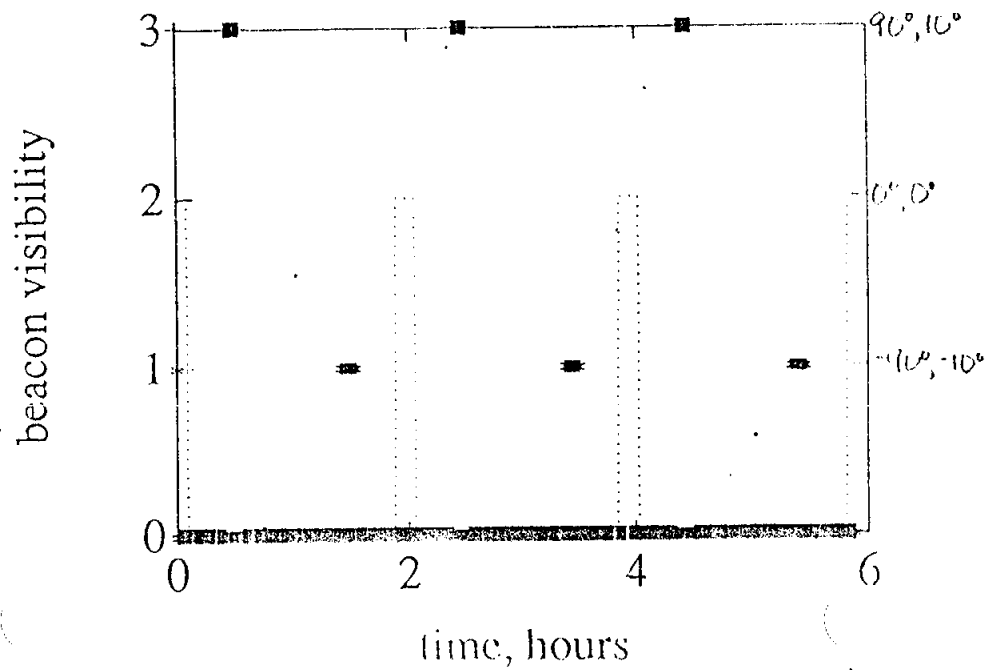
Init Beacon Errors





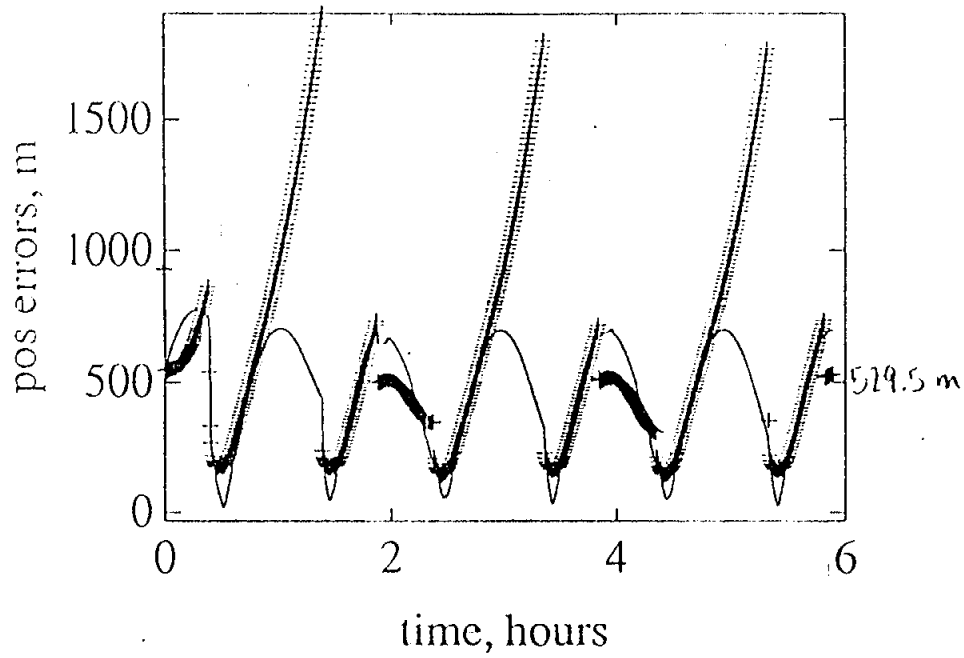


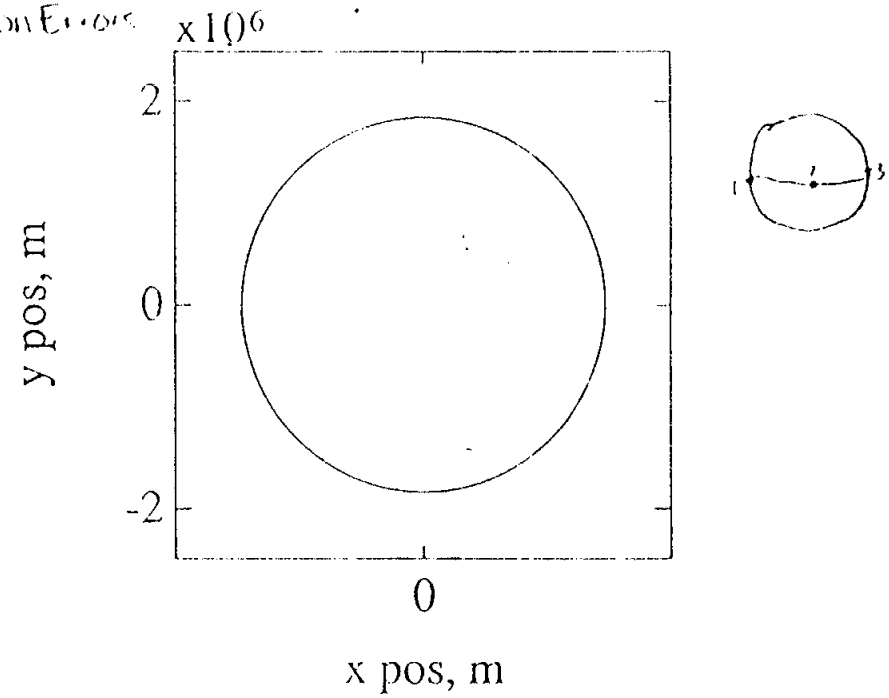


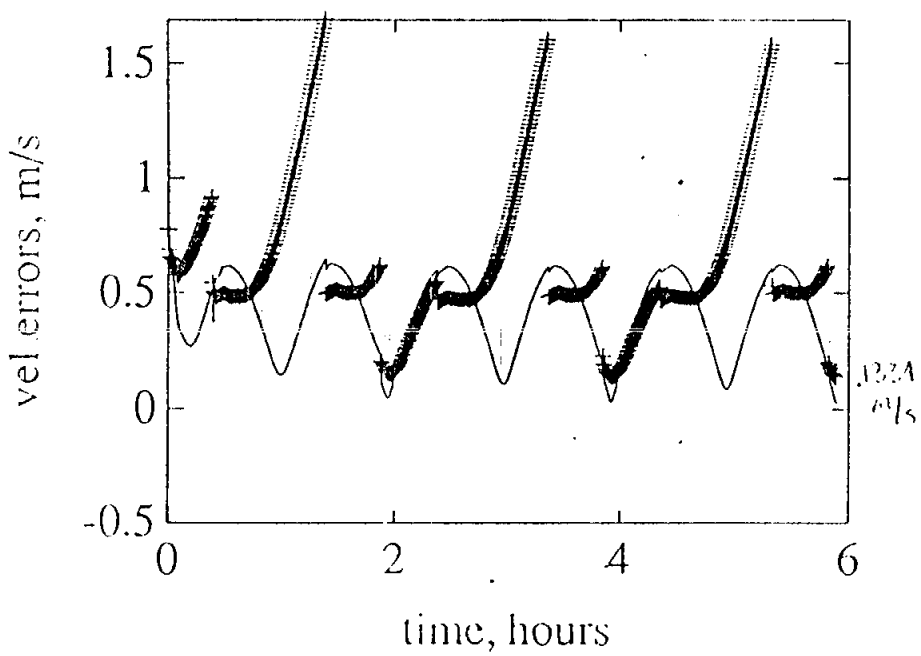


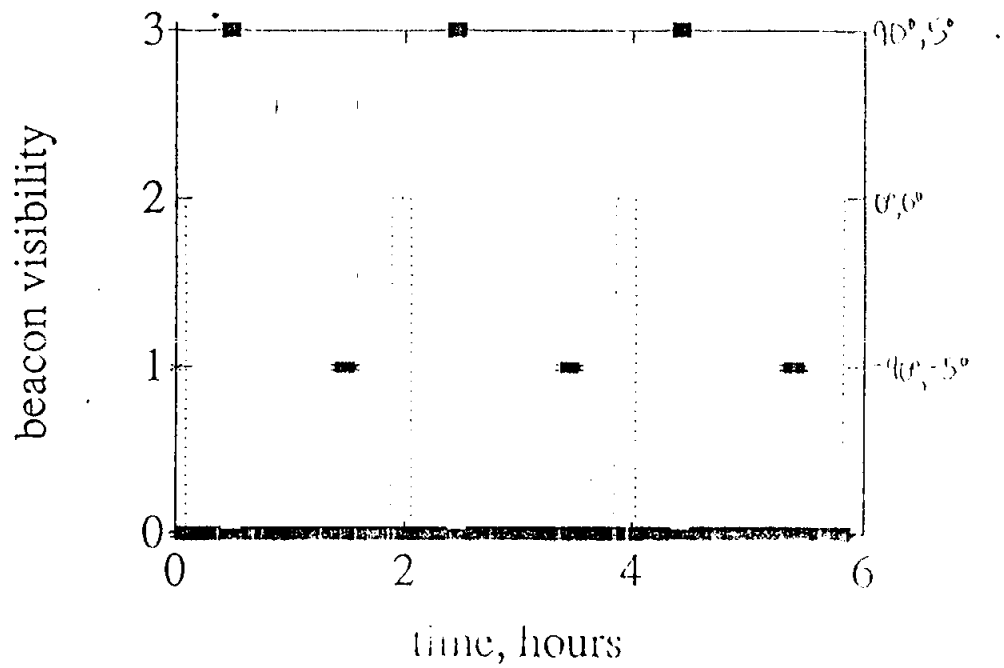
100 Km Circular Equa. .1 Cr bcl - 3 ITEUS, 3 beacc

Init. Beacon Errors

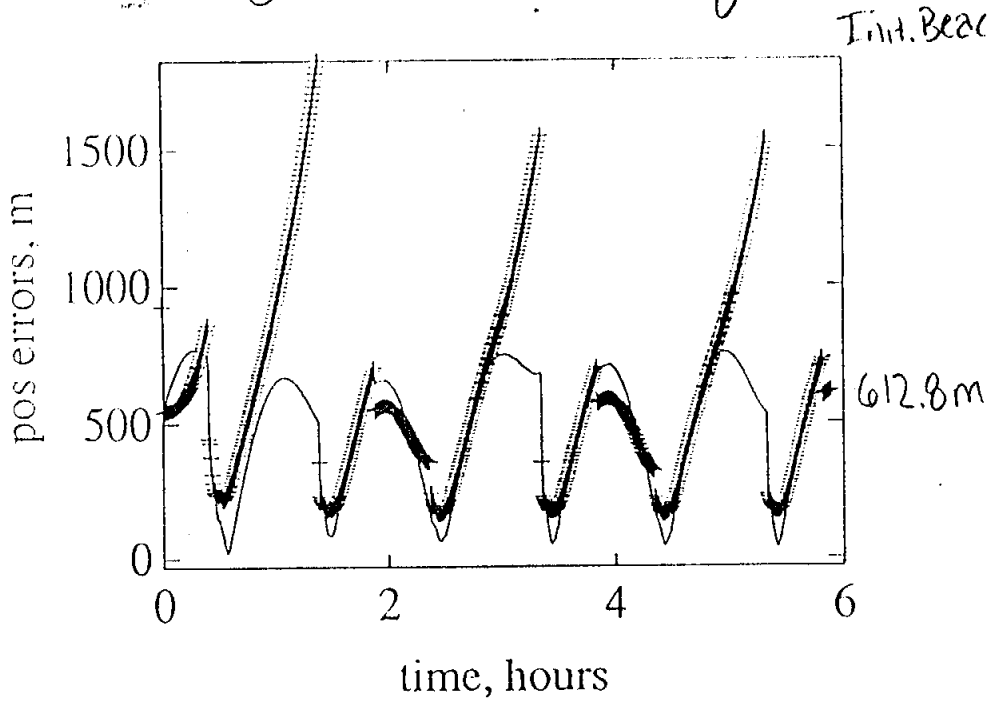


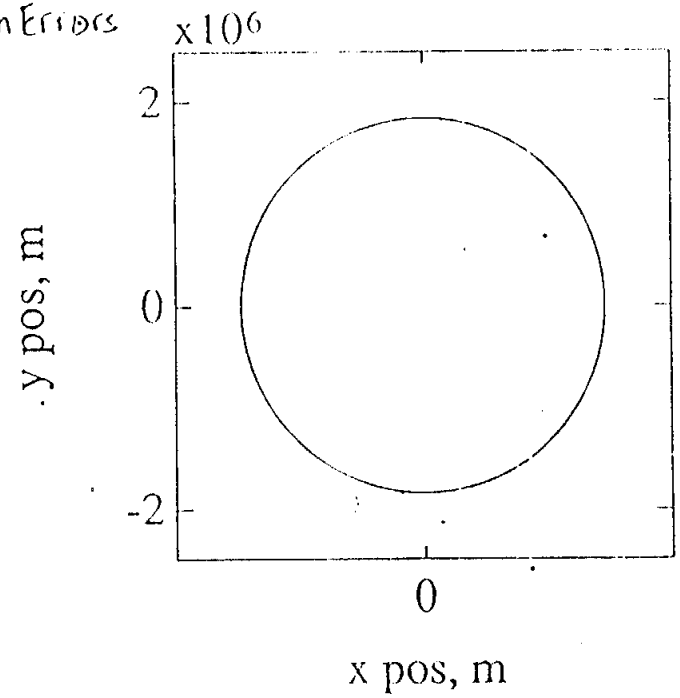


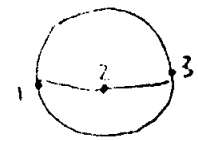


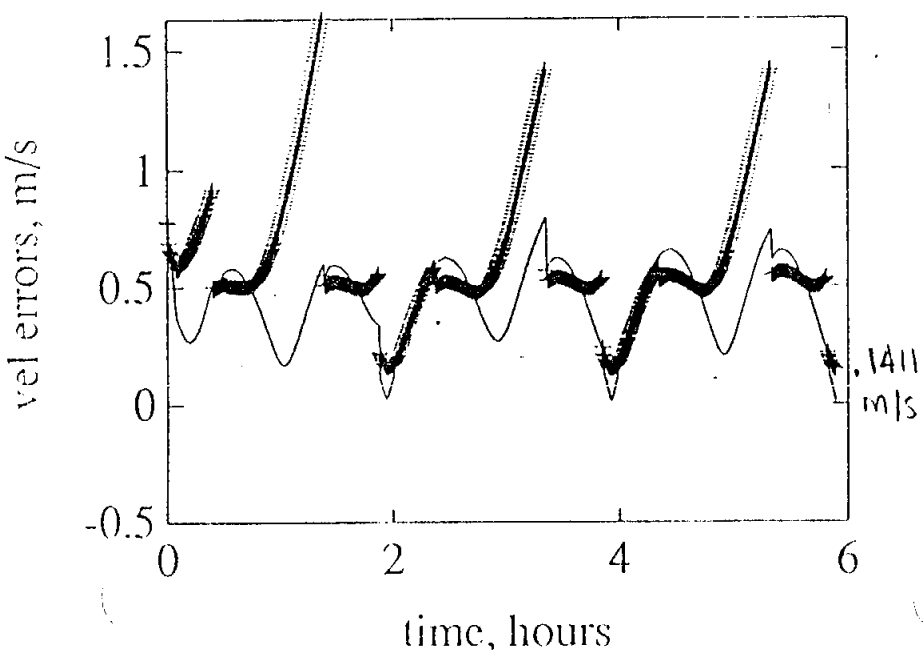


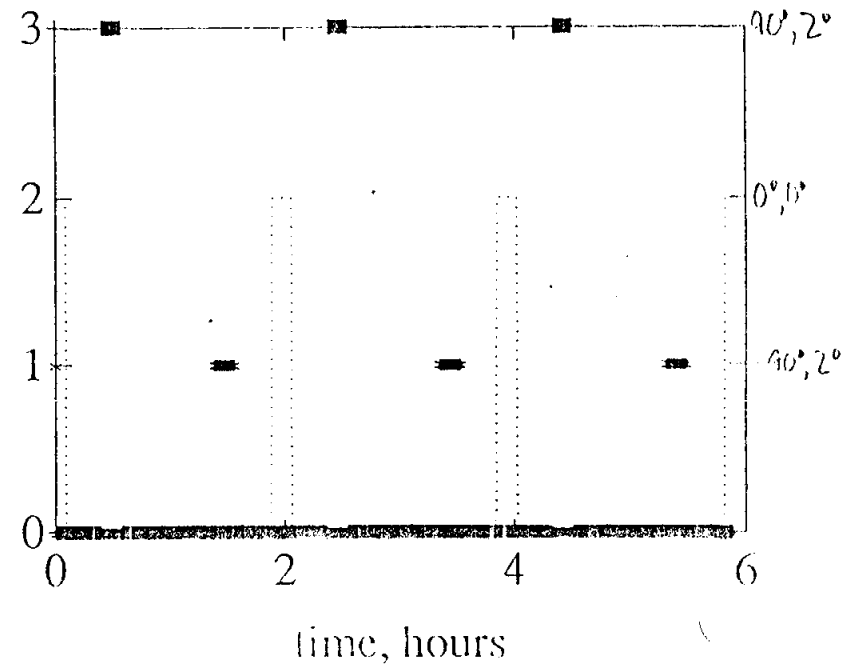
100 km Circular Equatorial Orbit - 3 revs, 3 beacons





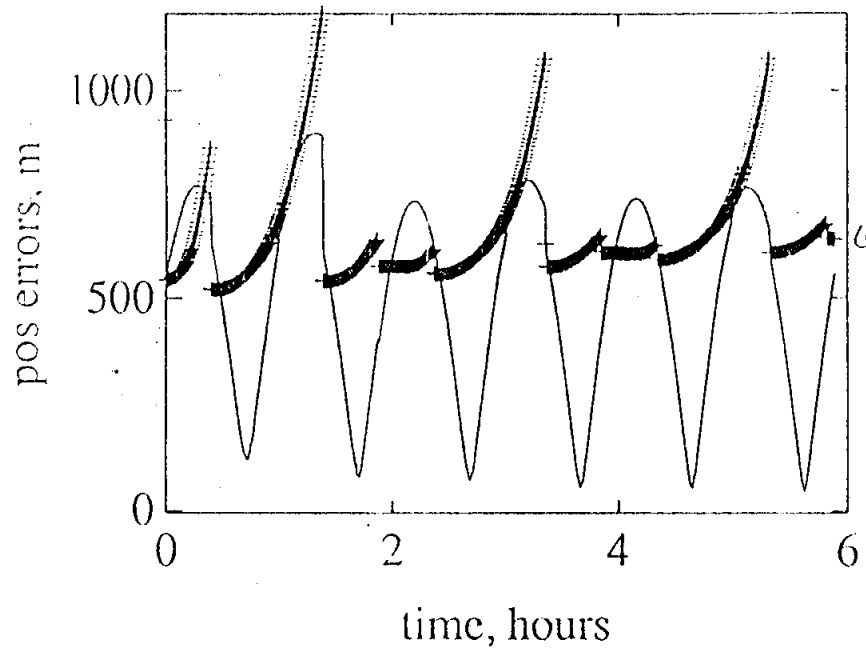


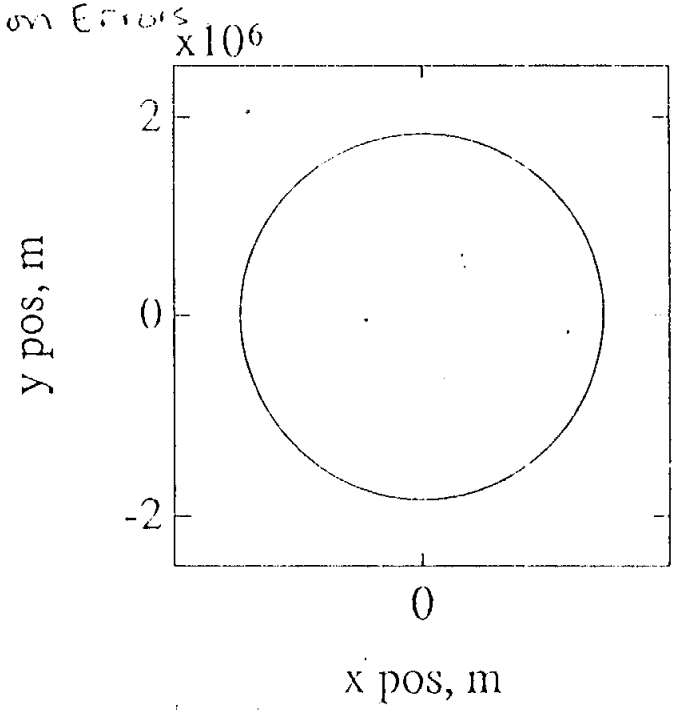


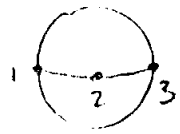


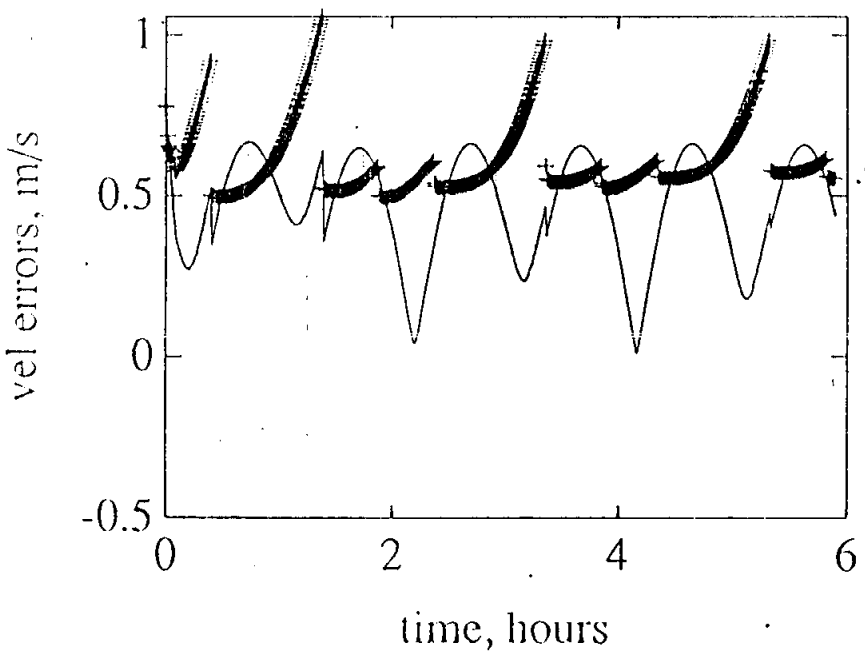
160 km Circular Lgo N Solid Errors, 3 beacons

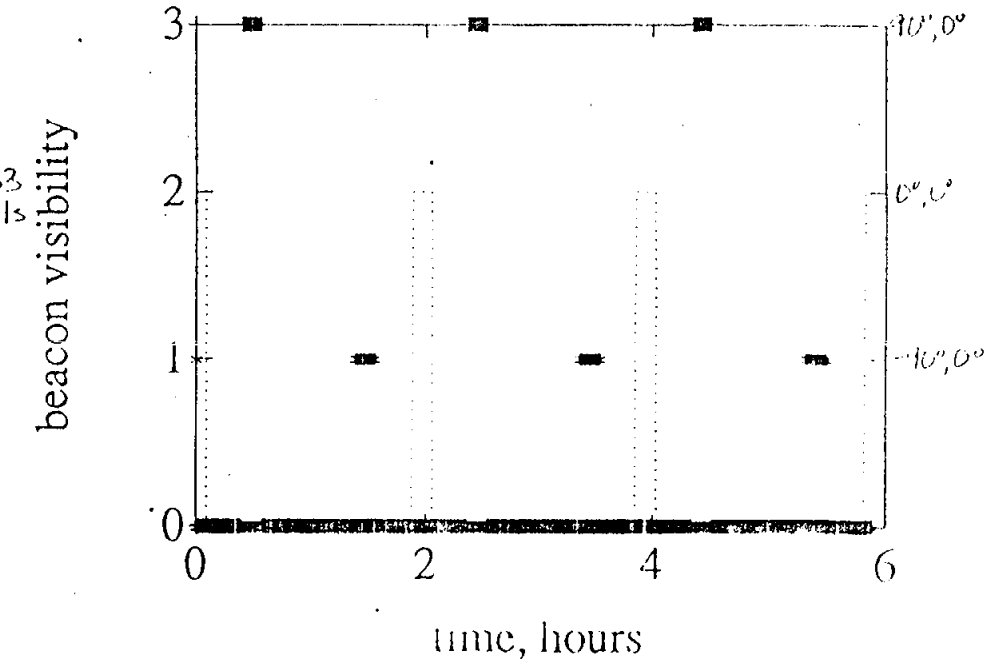
Int. Beacon Errors



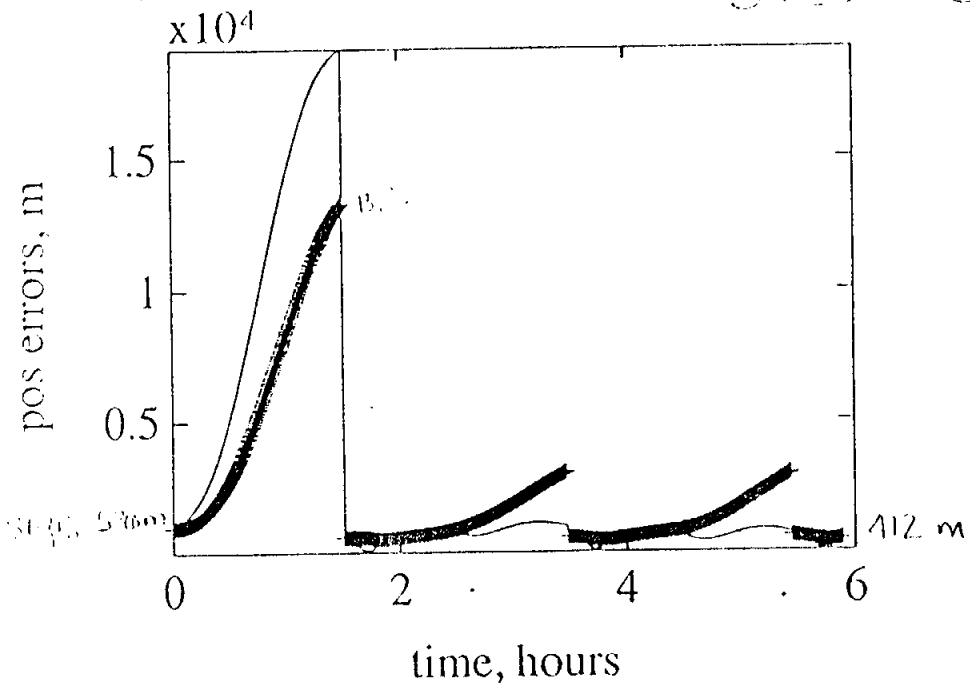


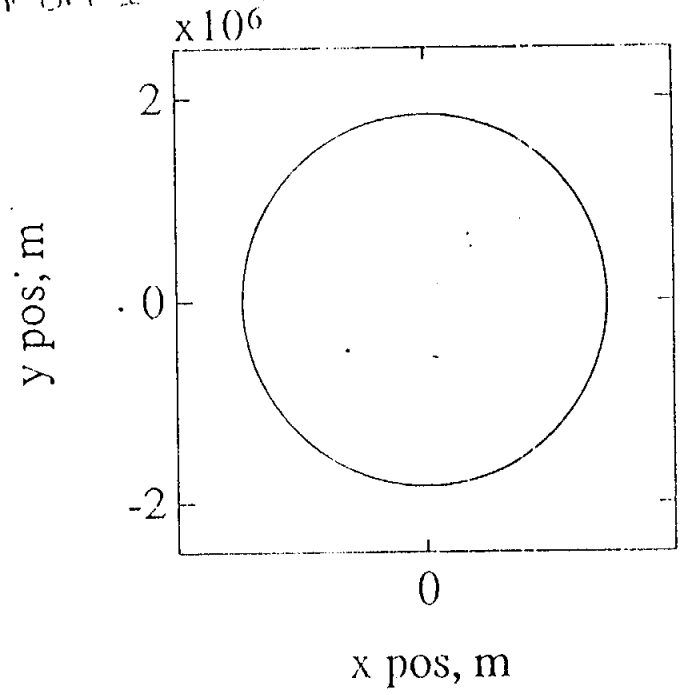




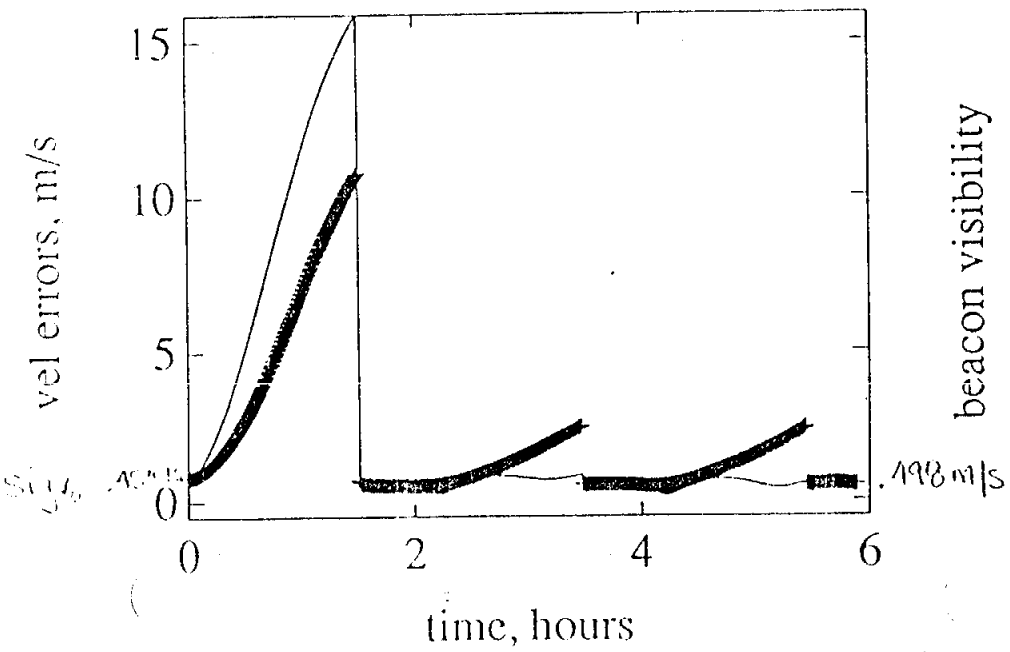


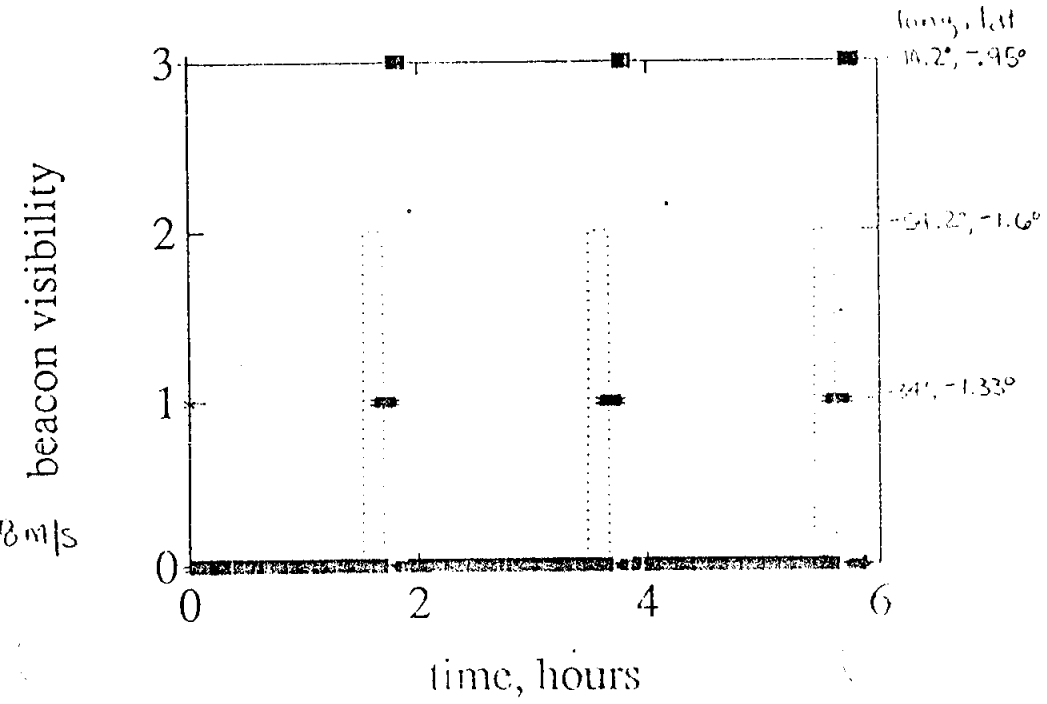
750 km Circular Orbit, [illegible] 18° – Beacon Overlap 16°
3 meter – Beacon Init Errors



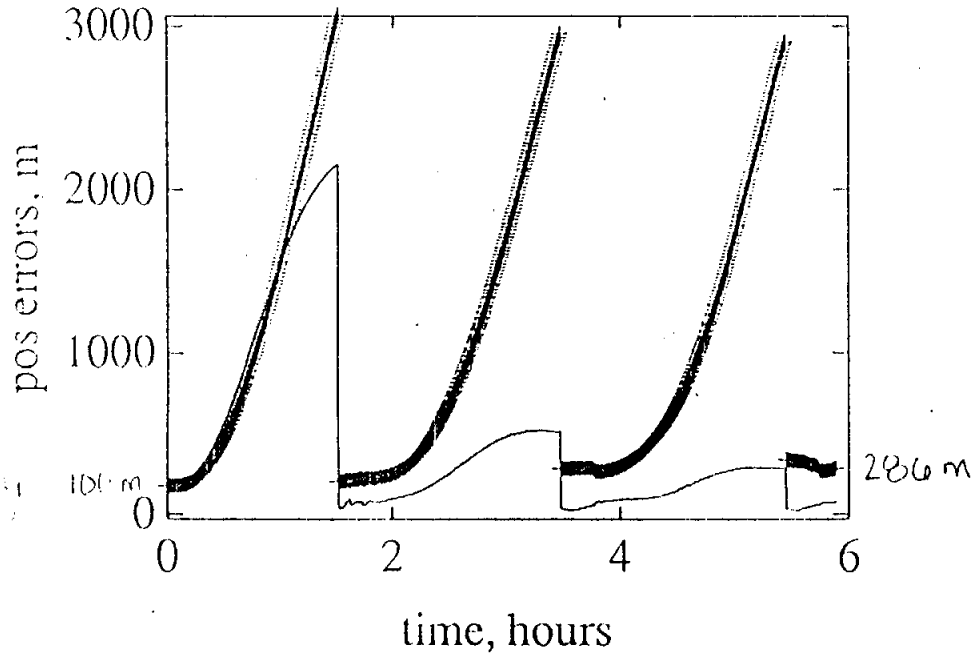


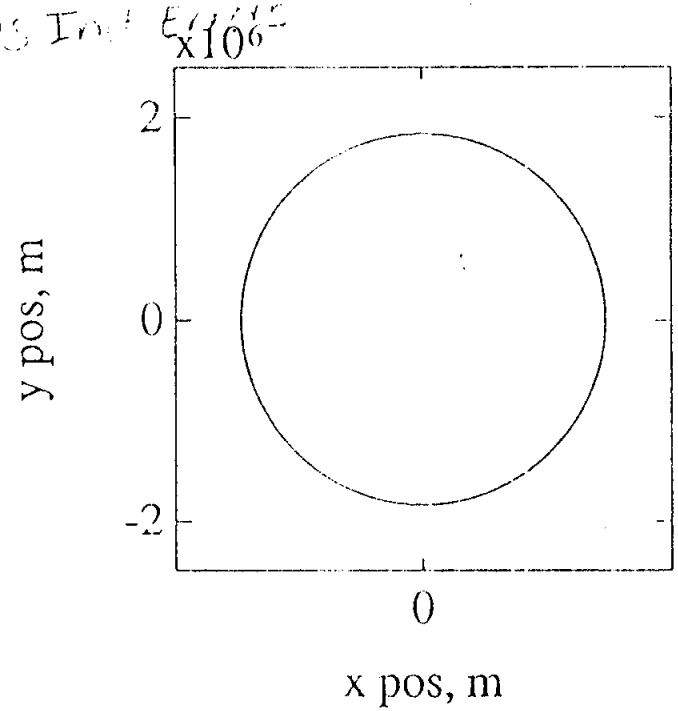
(1", 1", 10 offset from groundtrack)



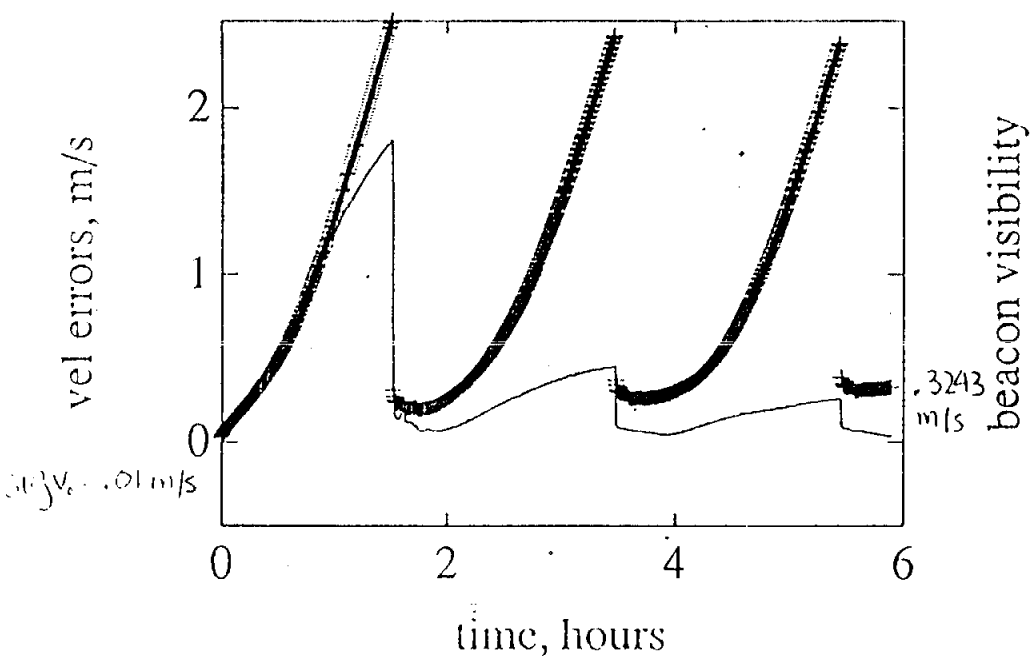


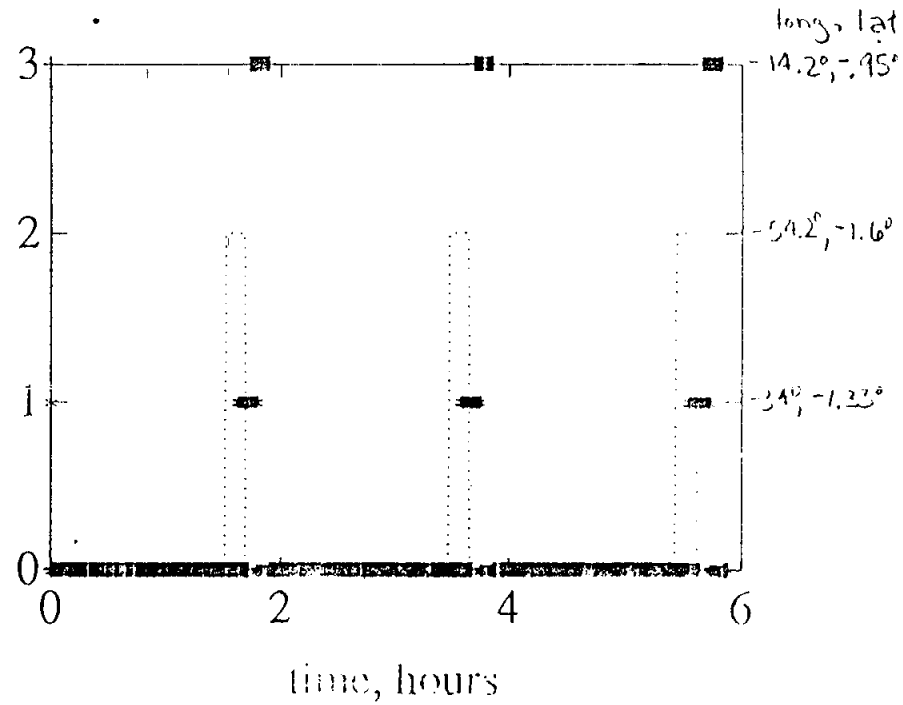
100 Km Circular Orbit, i = 1.8° Beacons Overlay
Free - INS Int Errors



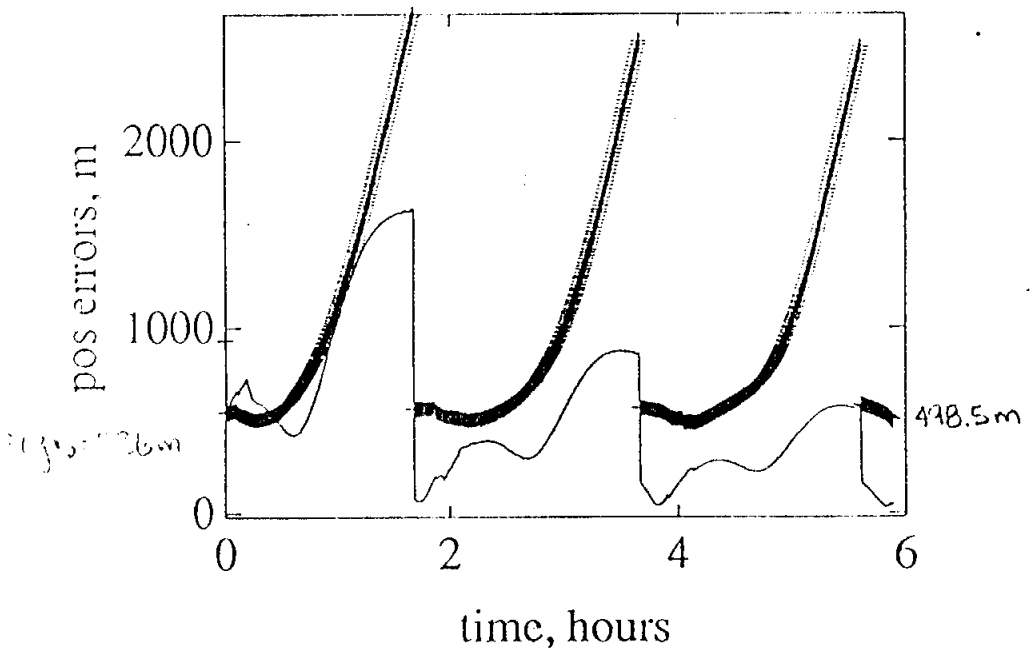


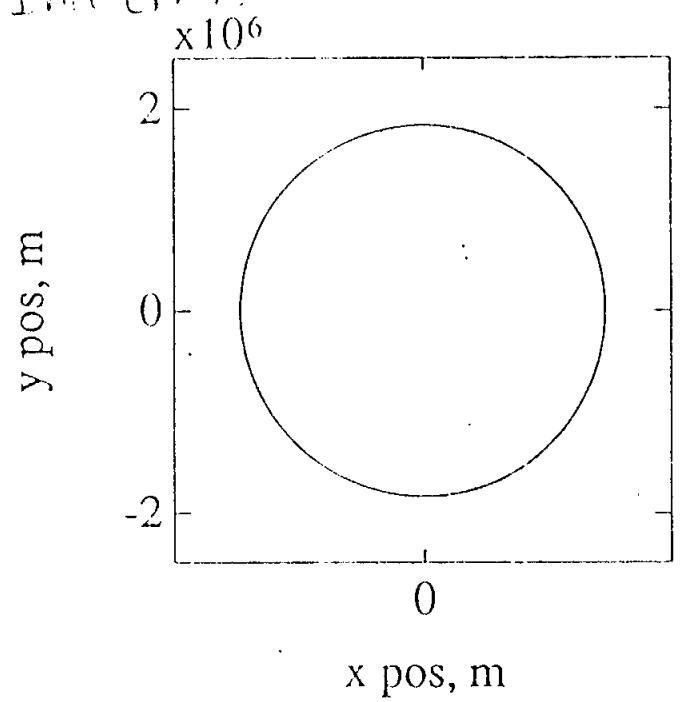
(#2, #3 4° offset from Groundtrack



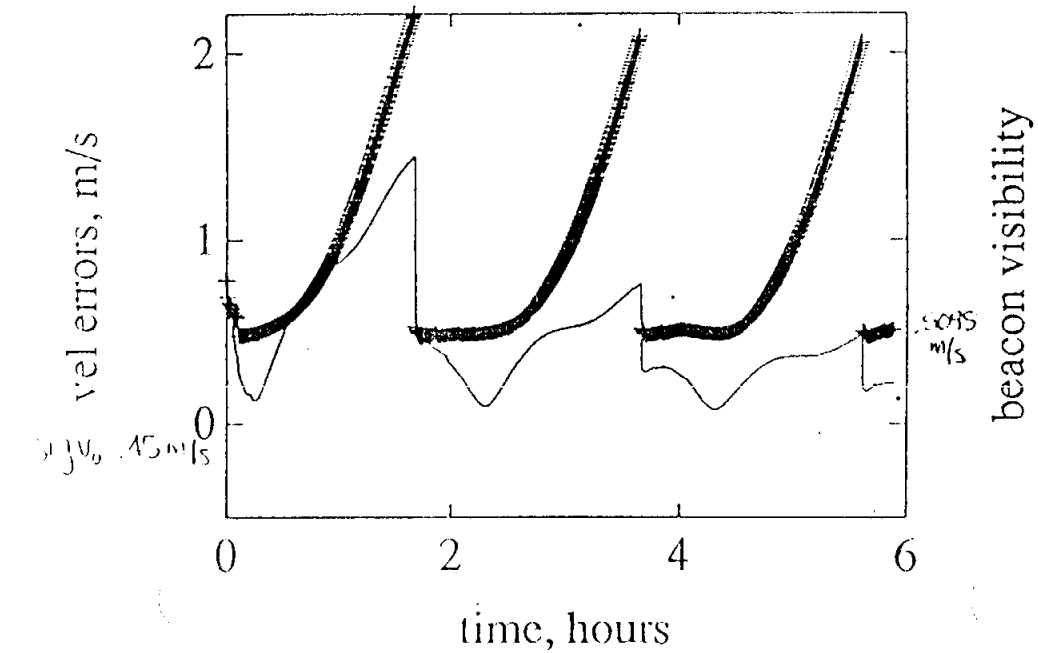


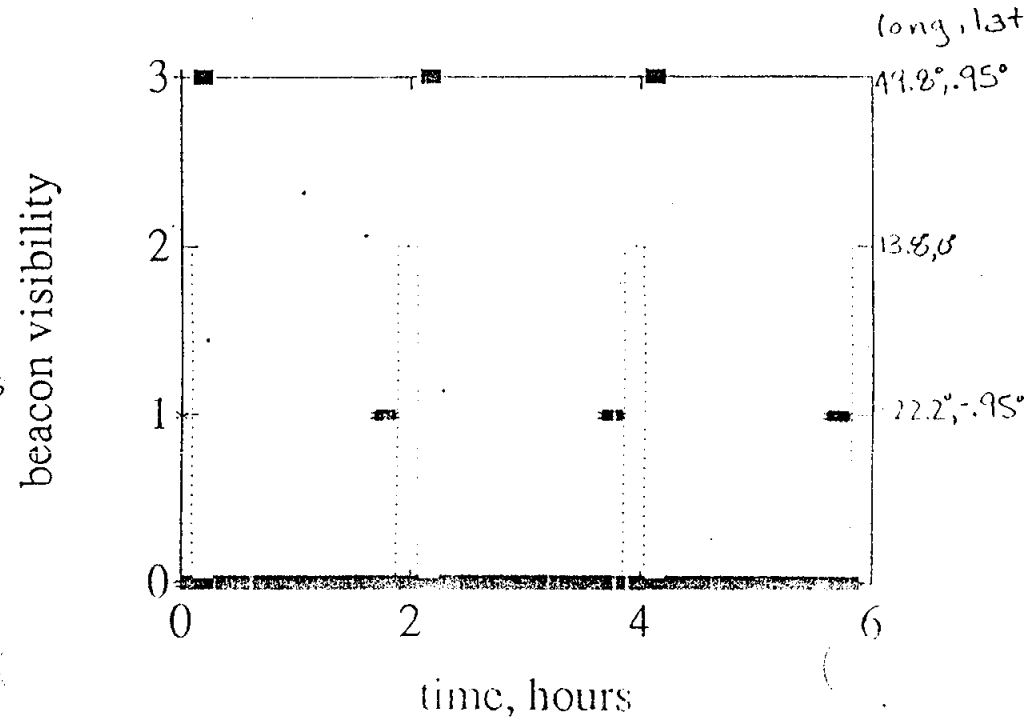
100 km Circular Orbit, 3°, 1 Sec - Beacons are 3 Secs - Beacon Inst Error





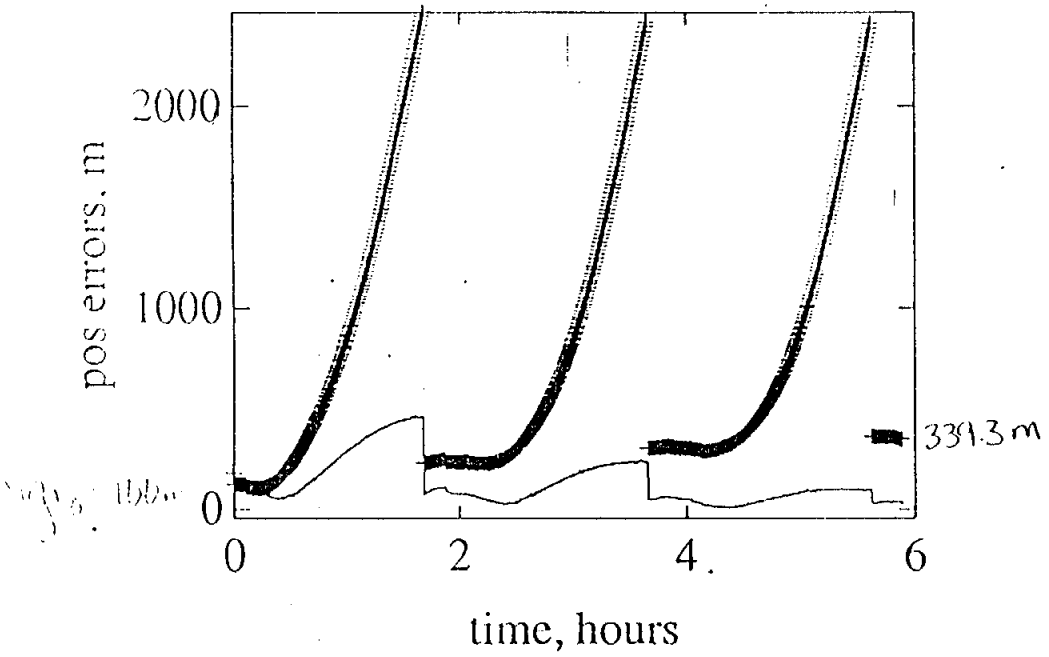
(#1, #3 90° offset from groundtrack

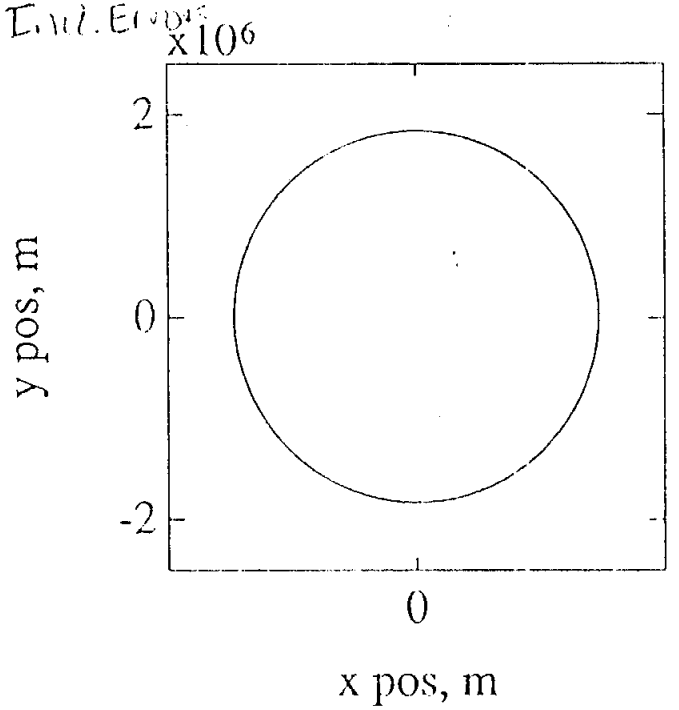




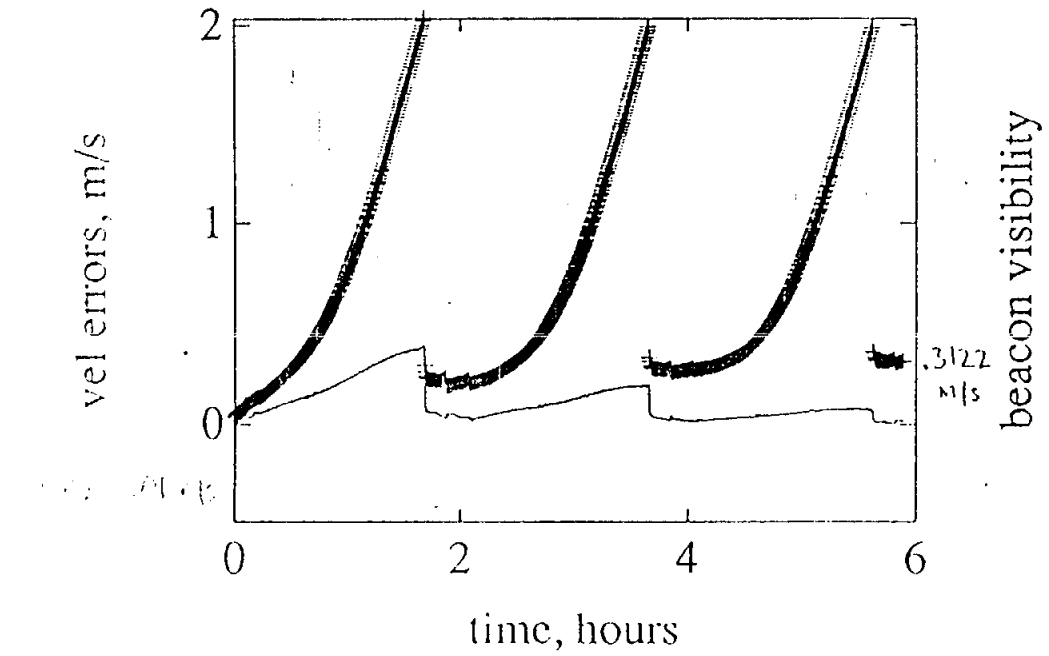
...10 Km Circular Orbit, Ω = 13.8 3° - Beacons are 32° Apart

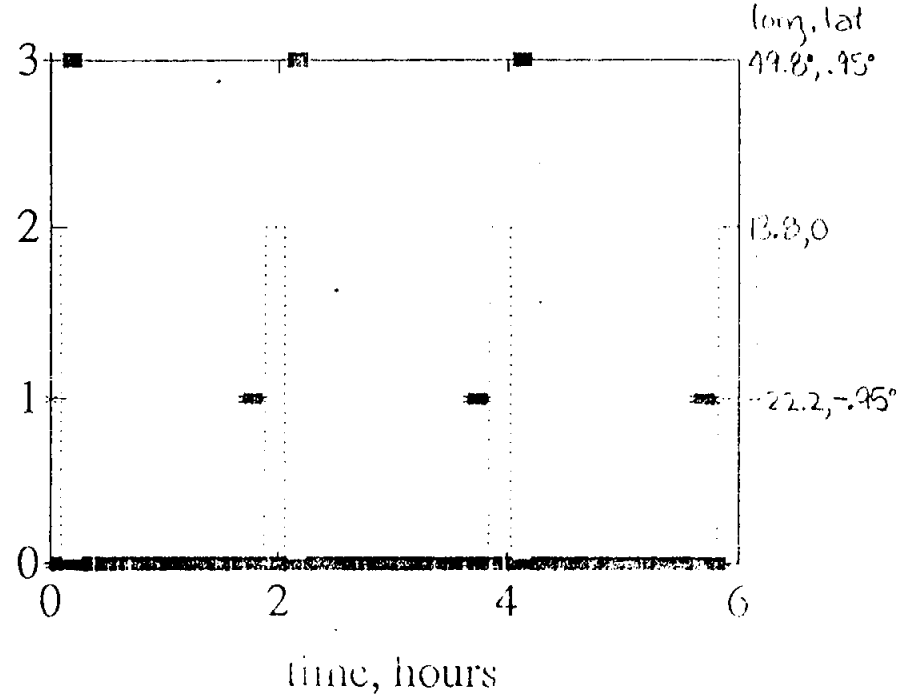
3 revs - DNS Init. Errors





(#1, #3 4° offset from groundtrack)





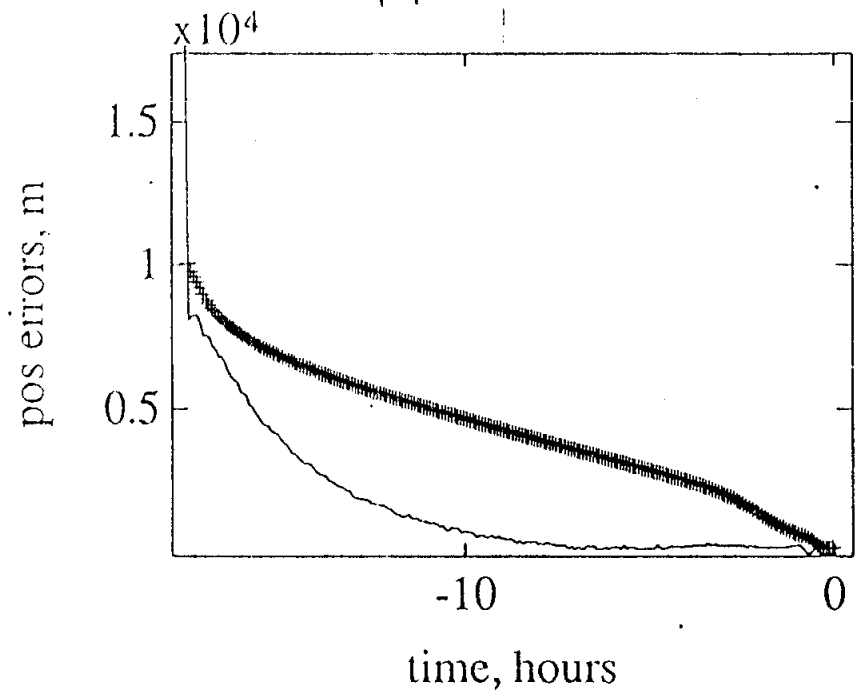
SET #2

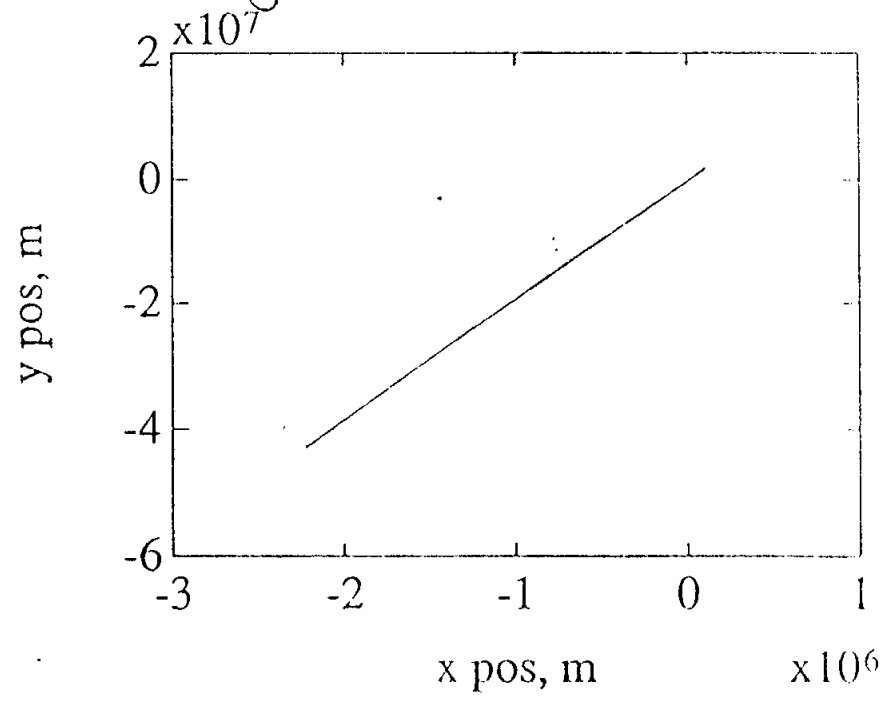
The following plots are for the circular parking orbit problem. They include, in order:

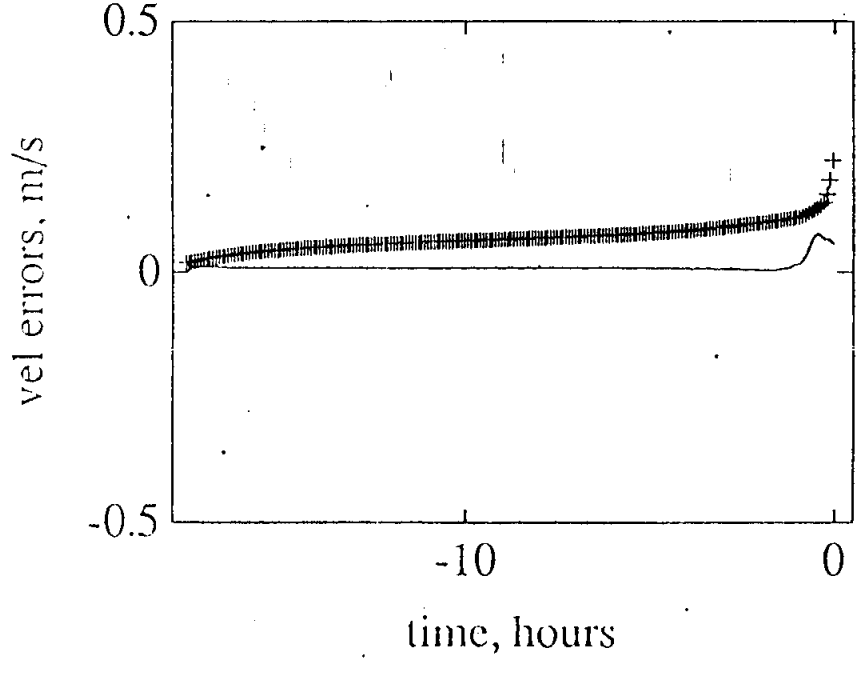
(1) the navigation plots for the nominal parking orbit with
   a. beacons 32 deg apart and DSN initial errors
   b. beacons 32 deg apart and beacon initial errors
   c. beacons overlapping 16 deg and DSN initial errors
   d. beacons overlapping 16 deg and beacon initial errors

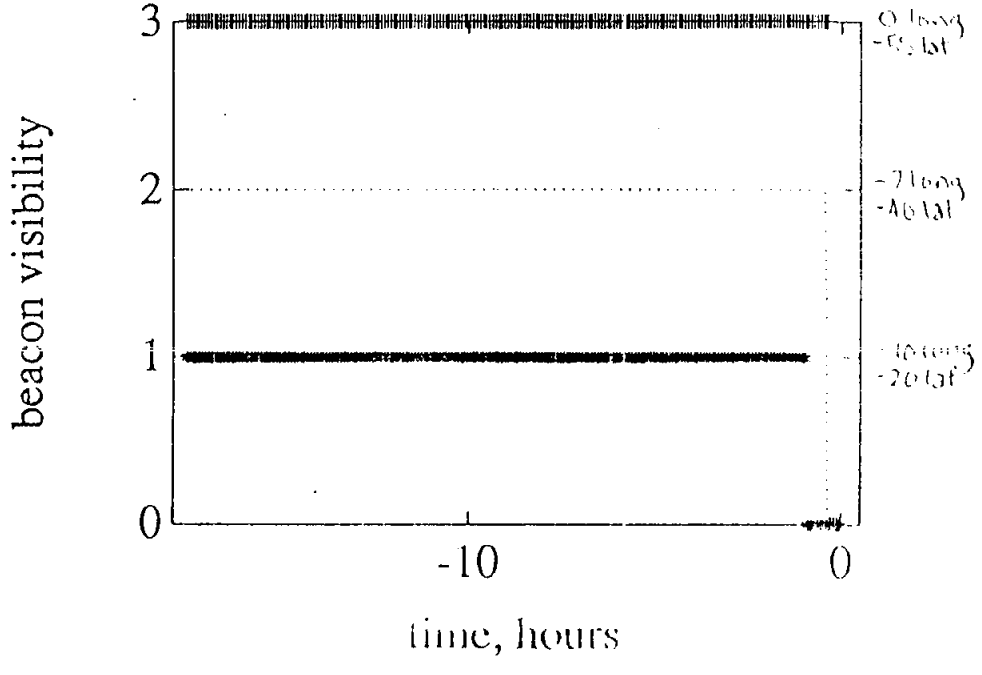
(2) for 100 km, 500 km equatorial orbits:
   a. all navigation plots used to generate the visibility and error vs latitude graphs

Approach V[illegible] of [illegible], 8[illegible] long 3 beacons [illegible]

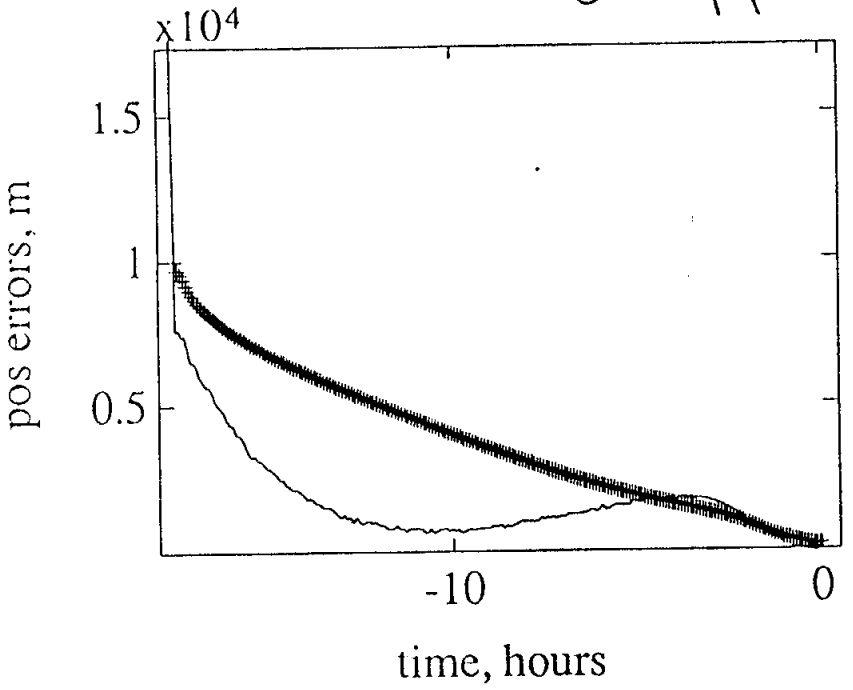


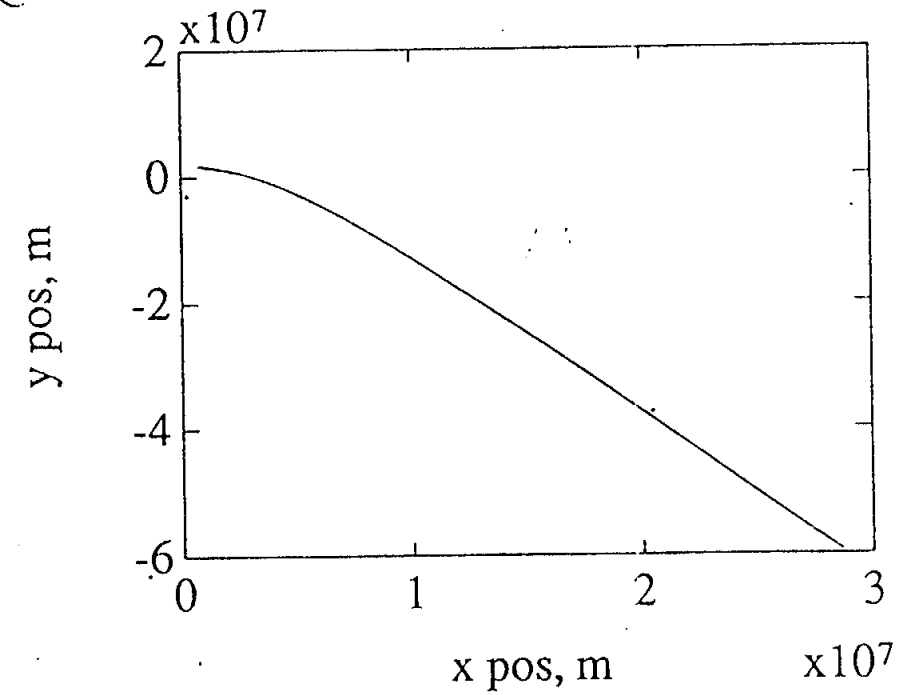


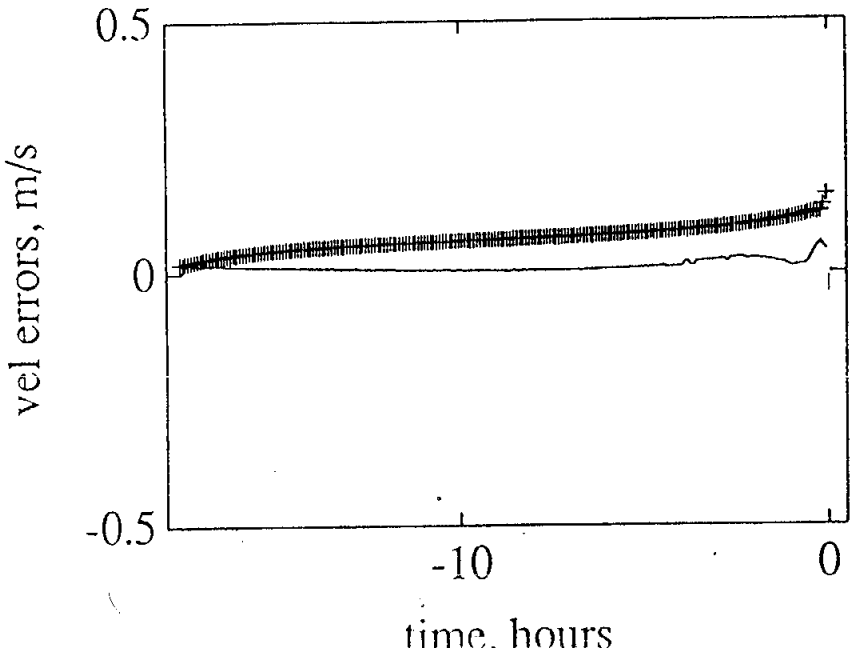


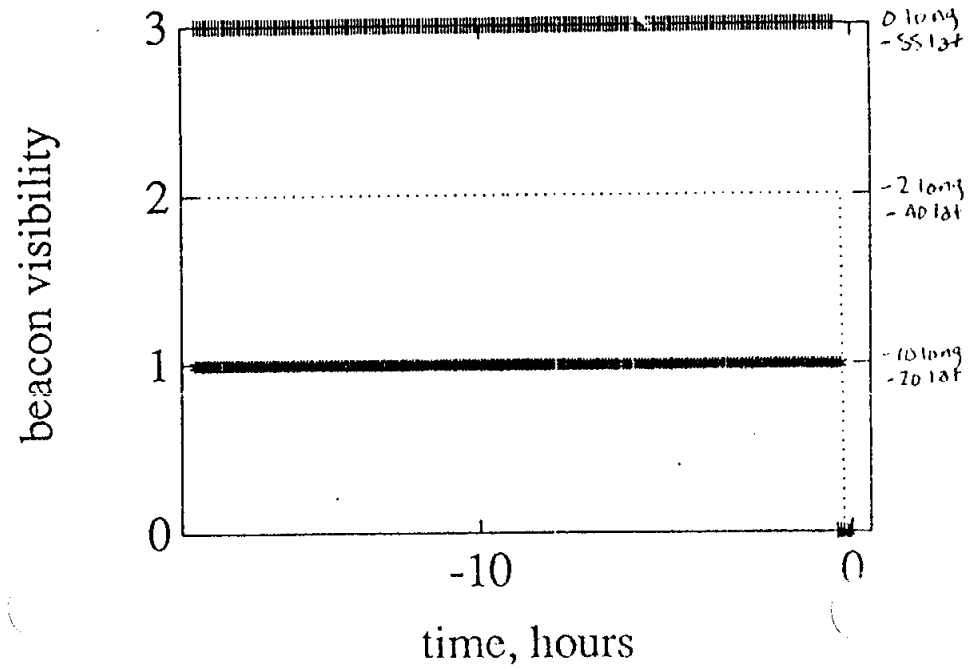


Nominal Approac.. - 3 Beacons (Try 2)

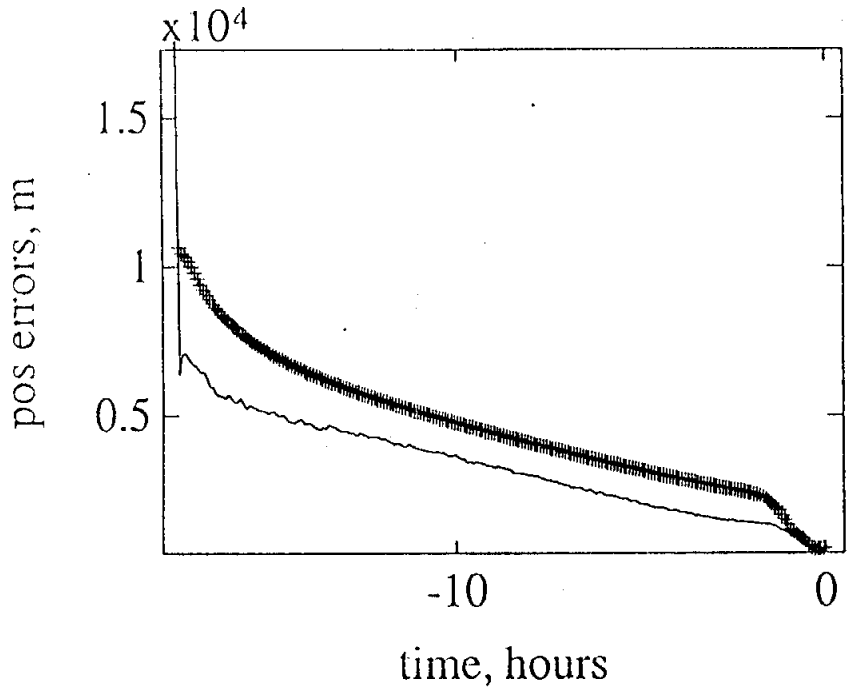


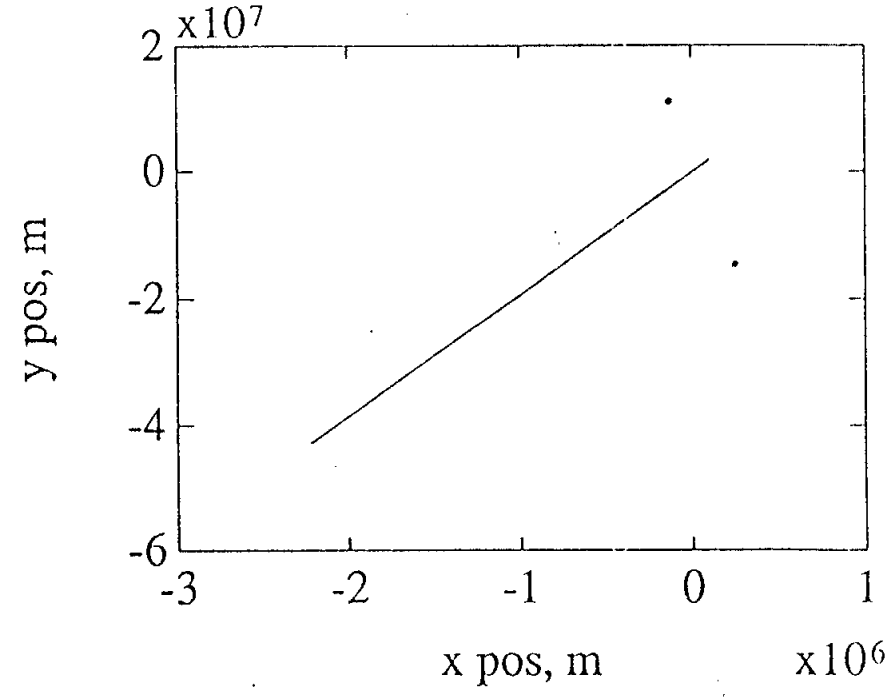


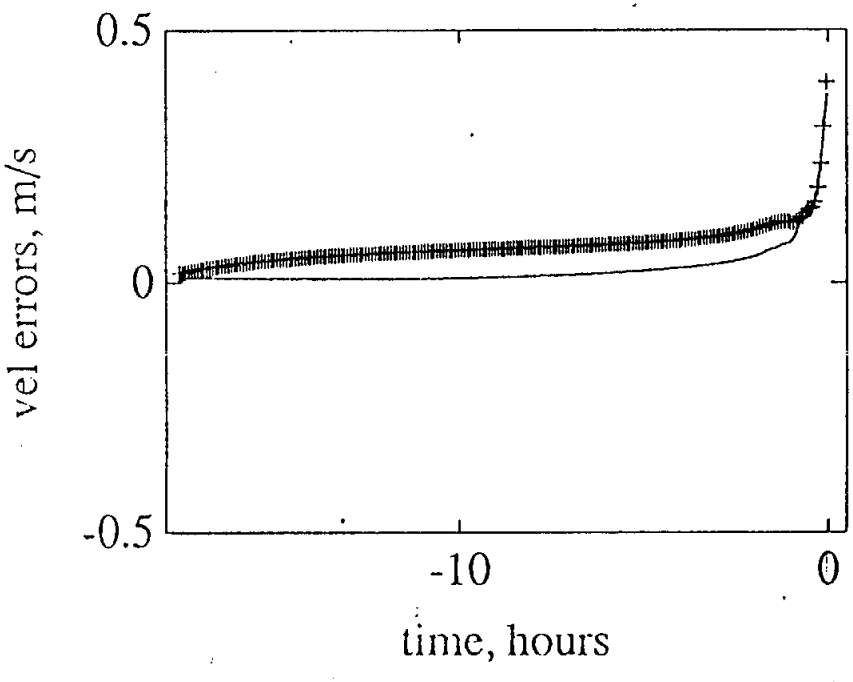


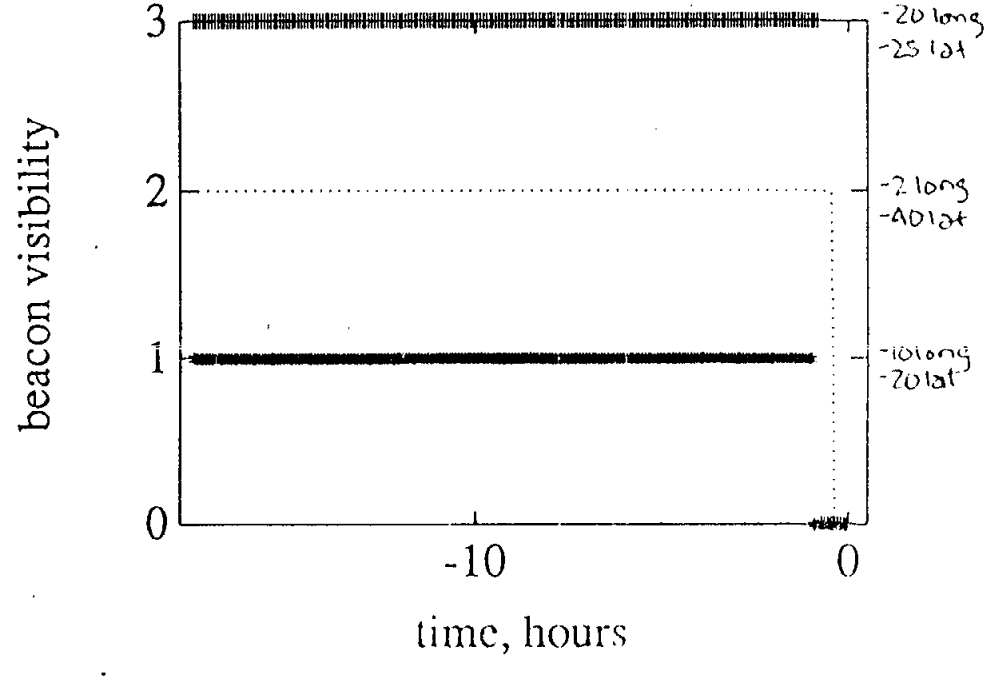


Approach Vint at, ASI 1° long - 3 Beacons (1r

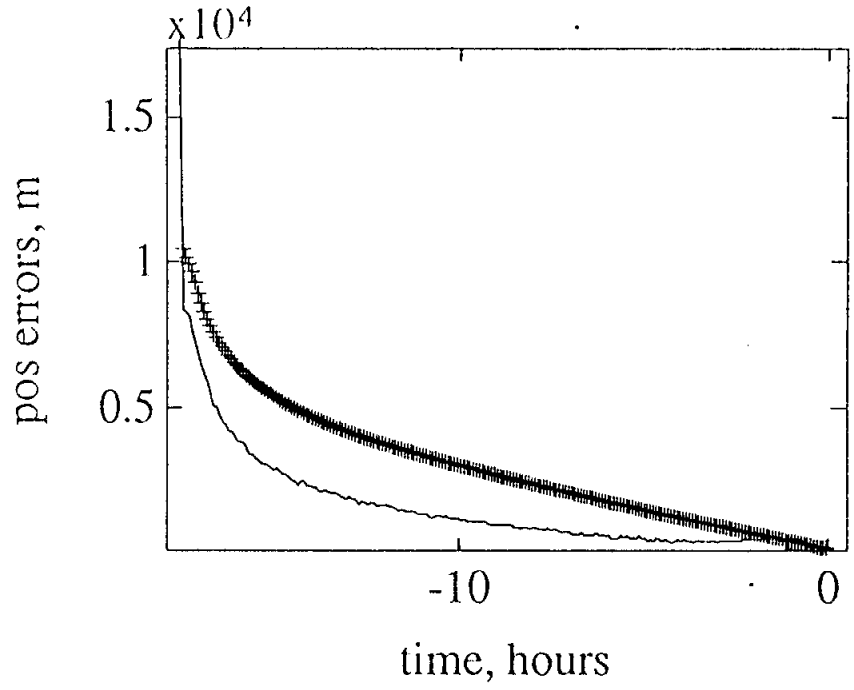


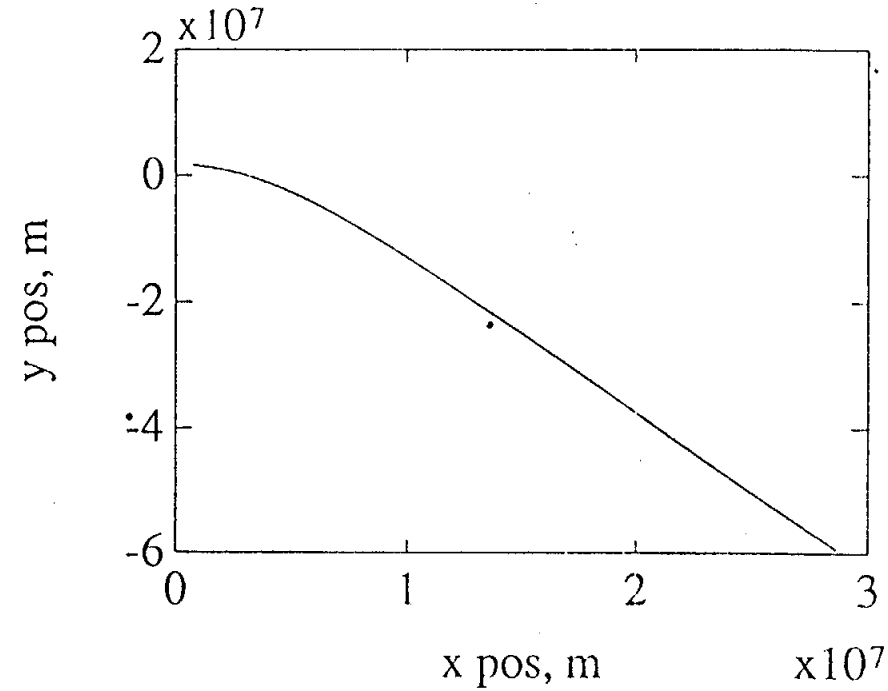


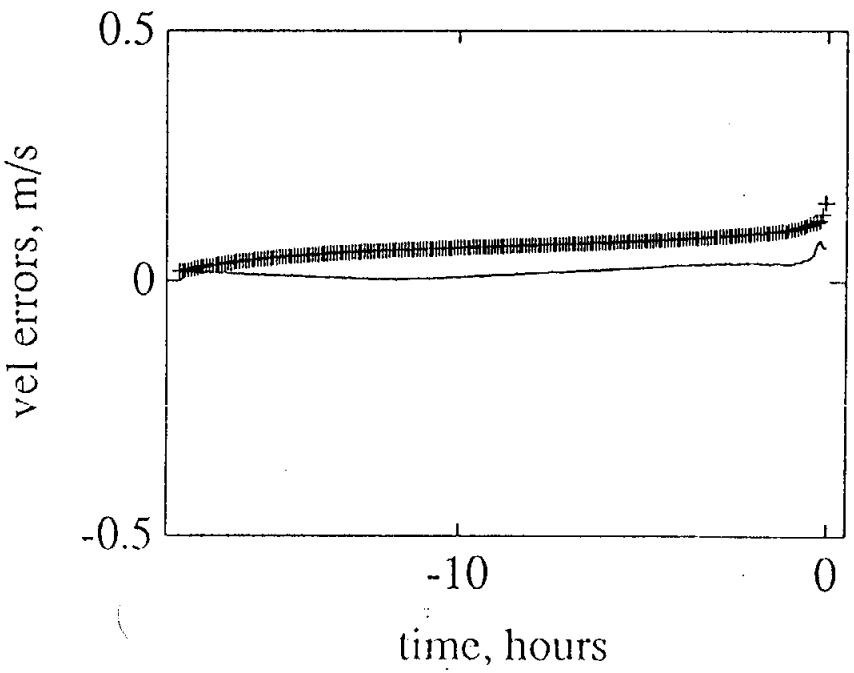


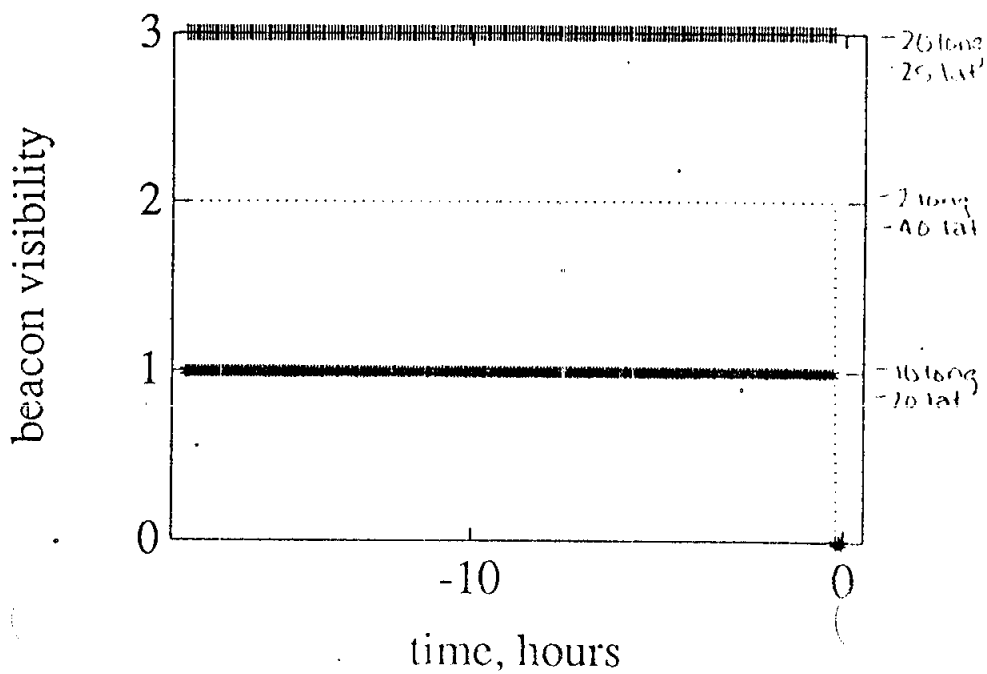


Nominal Approach 3 Beacons (Try 1)



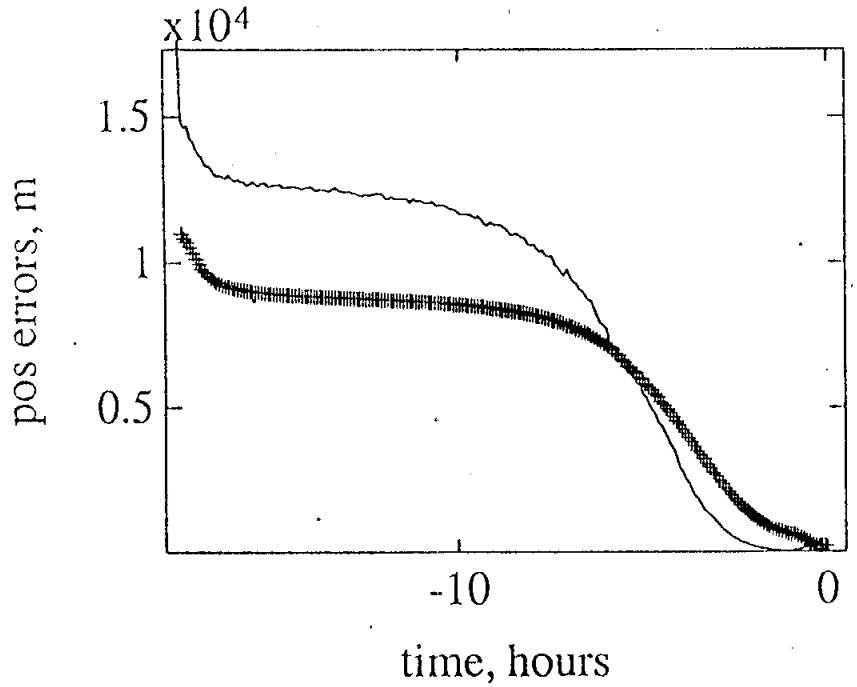


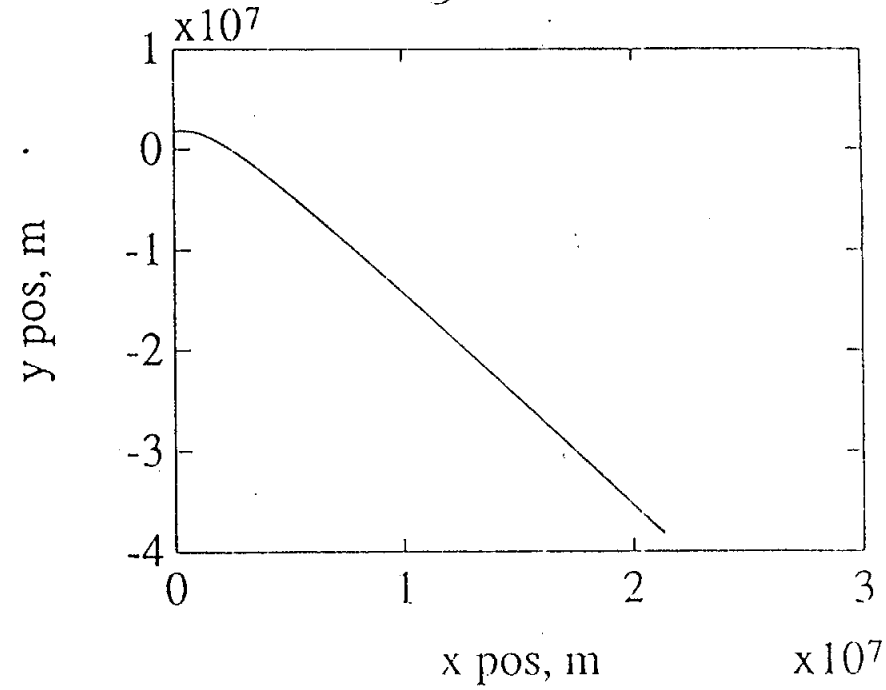


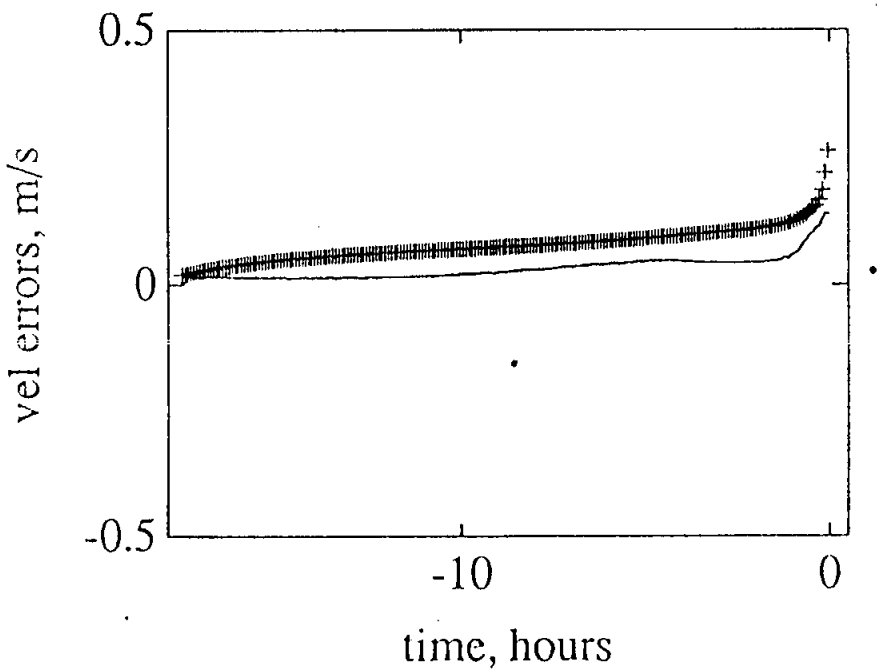


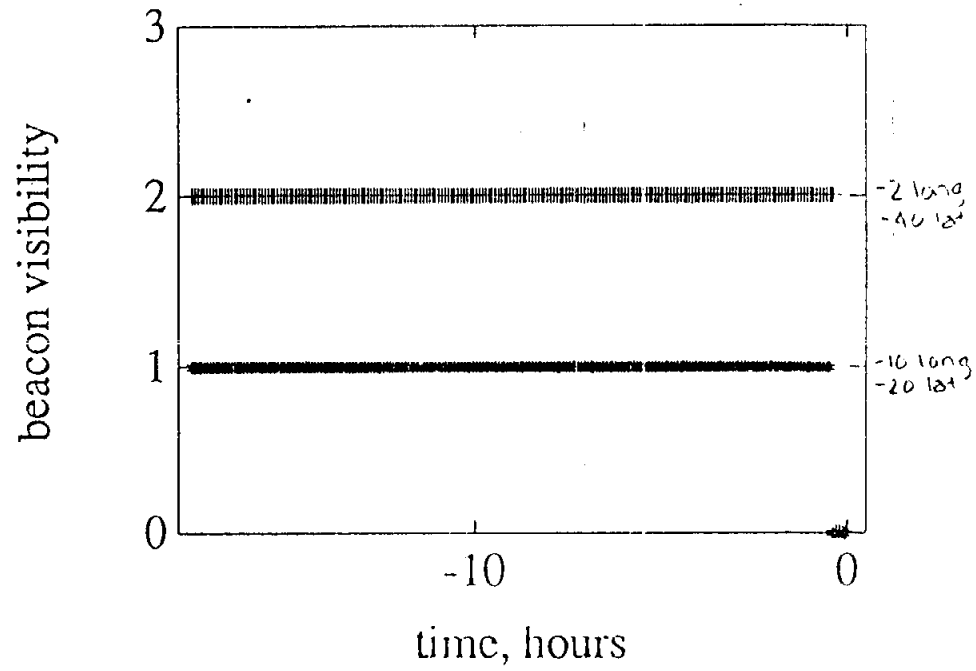
Approach Vinf

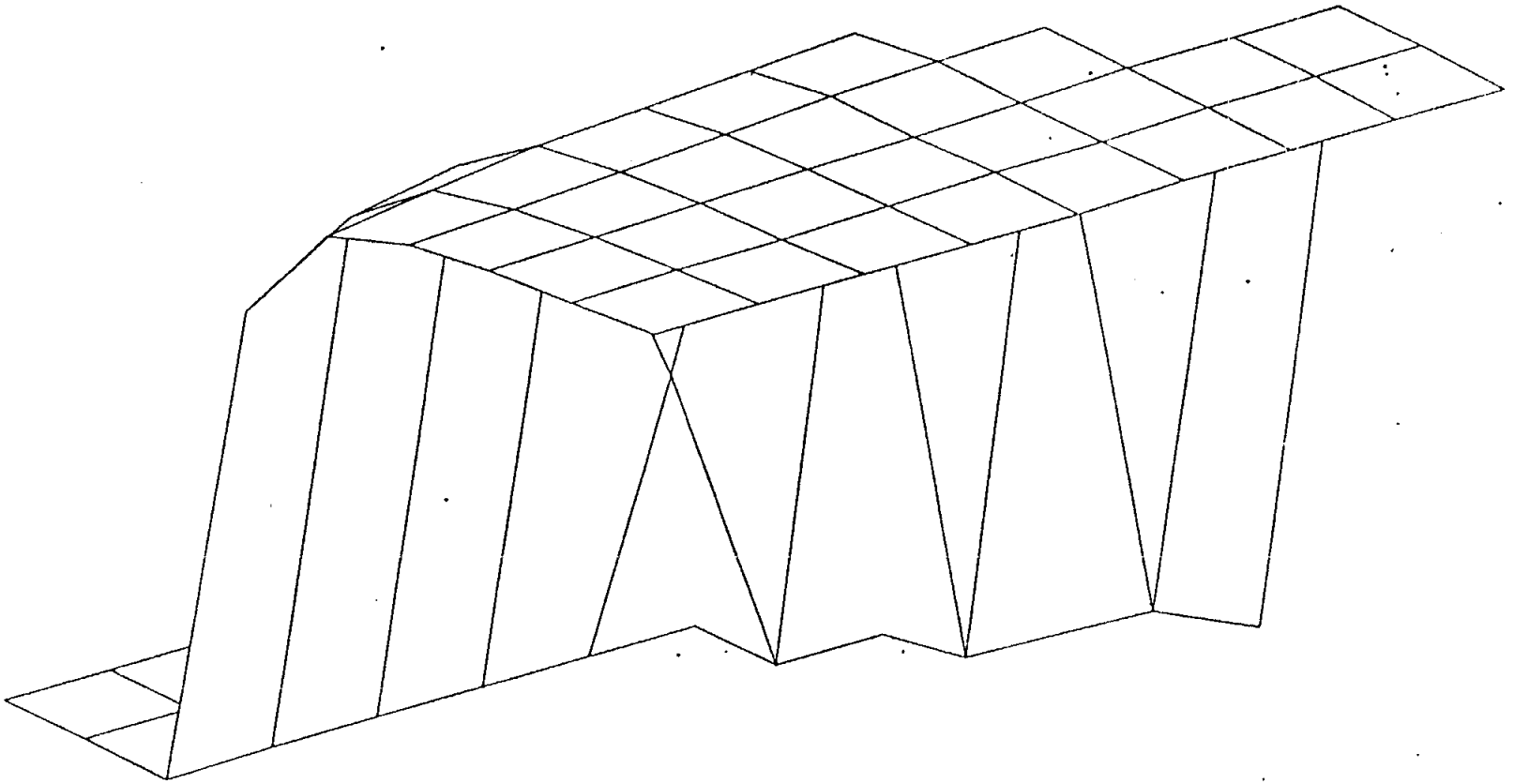
45° lat. 115° long (i=64°)

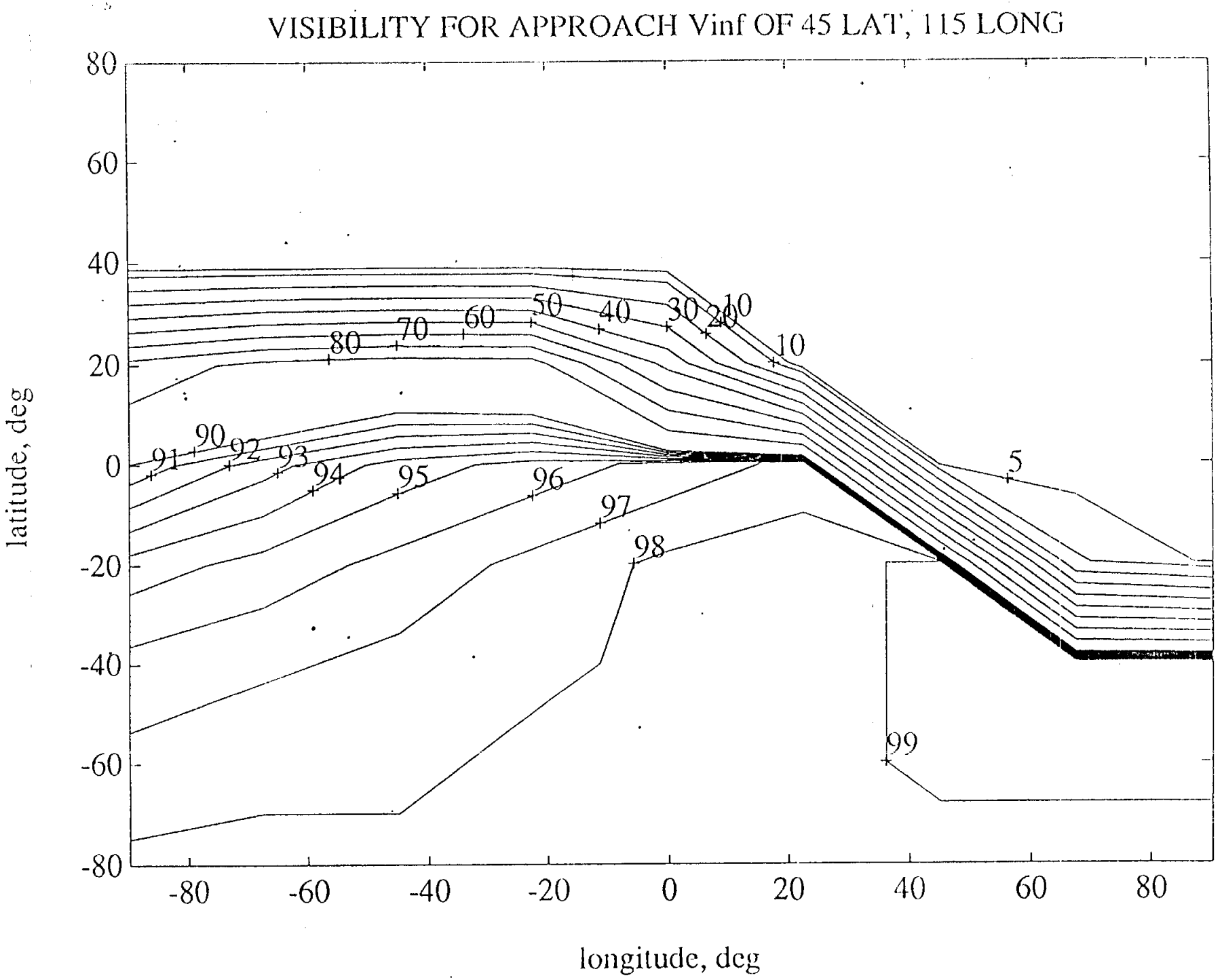




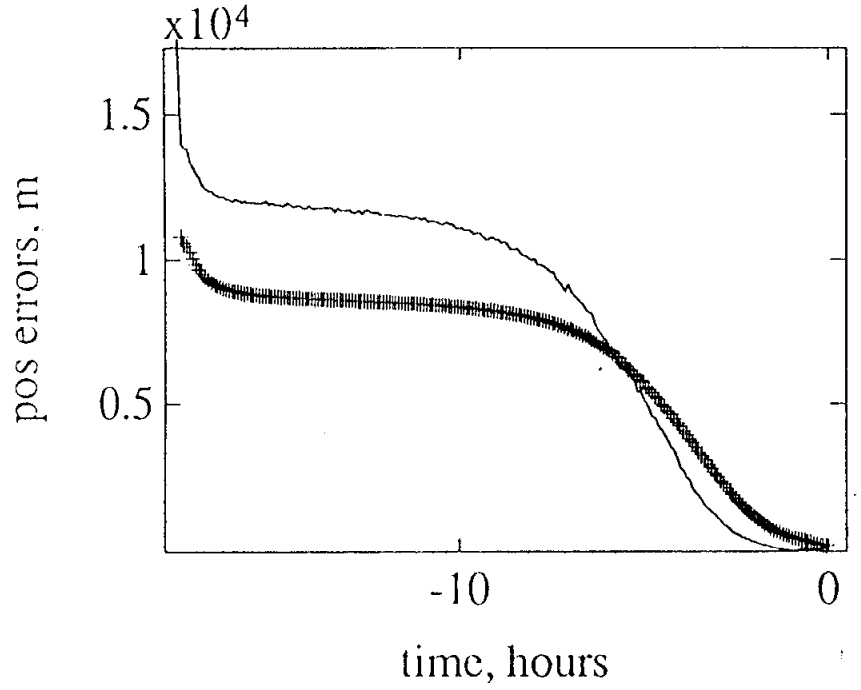


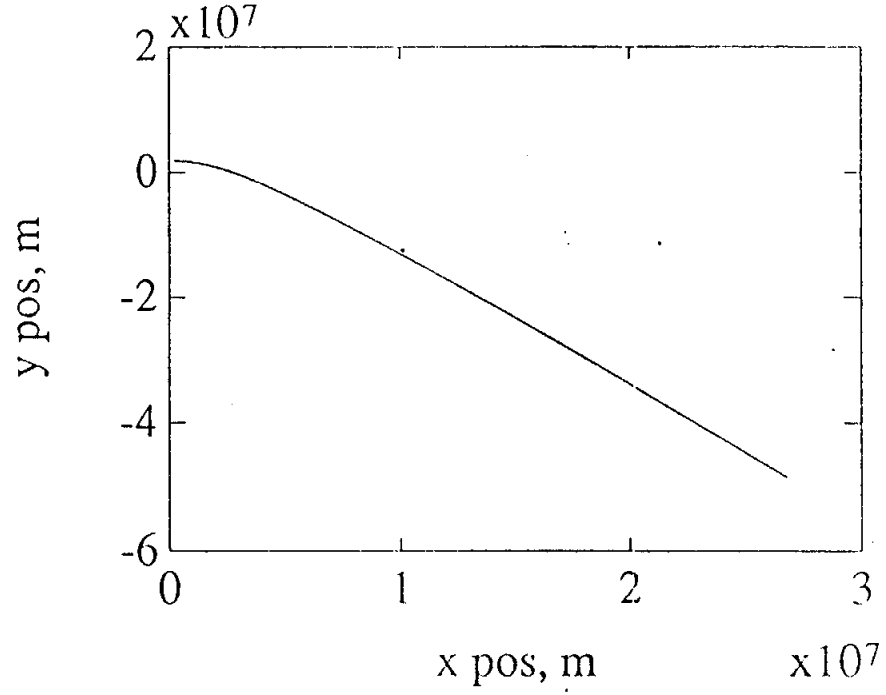


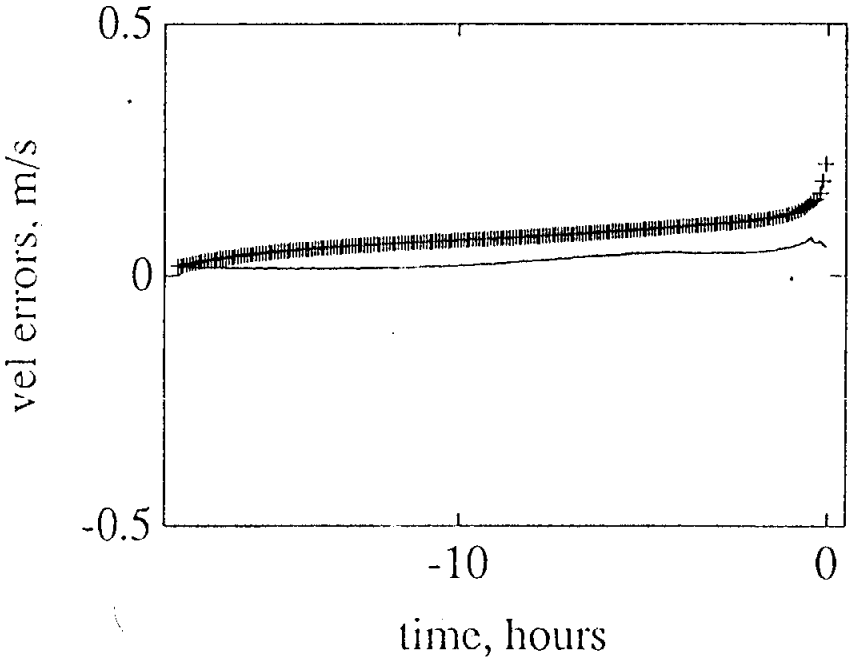


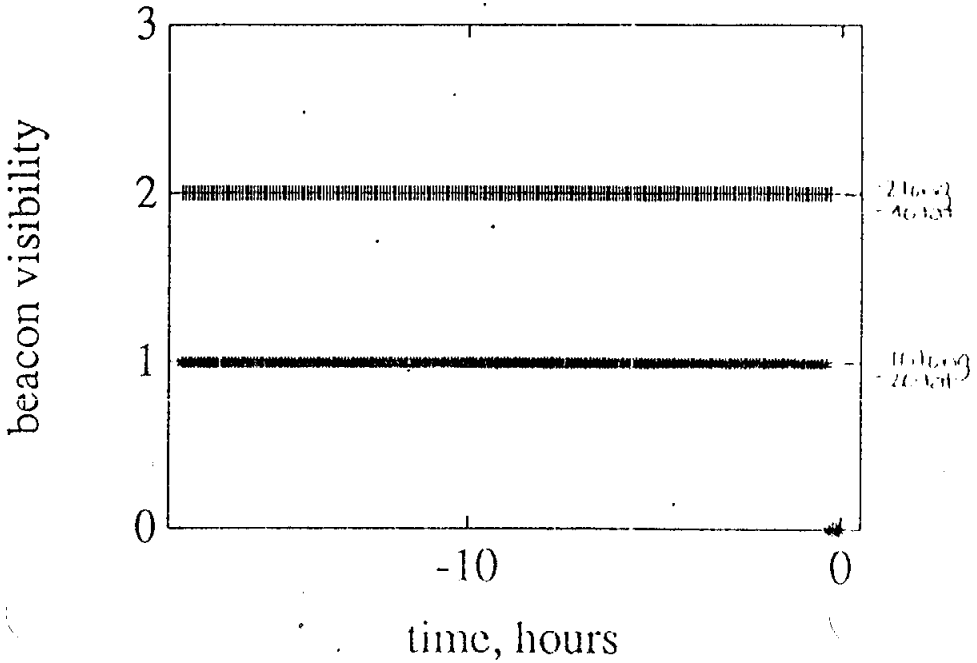

VISIBILITY FOR APPROACH Vinf OF 45 LAT, 115 LONG
latitude, deg
longitude, deg
80
60
40
20
0
-20
-40
-60
-80
-80
-60
-40
-20
0
20
40
60
80
10
20
30
40
50
60
70
80
90
91
92
93
94
95
96
97
98
99
5

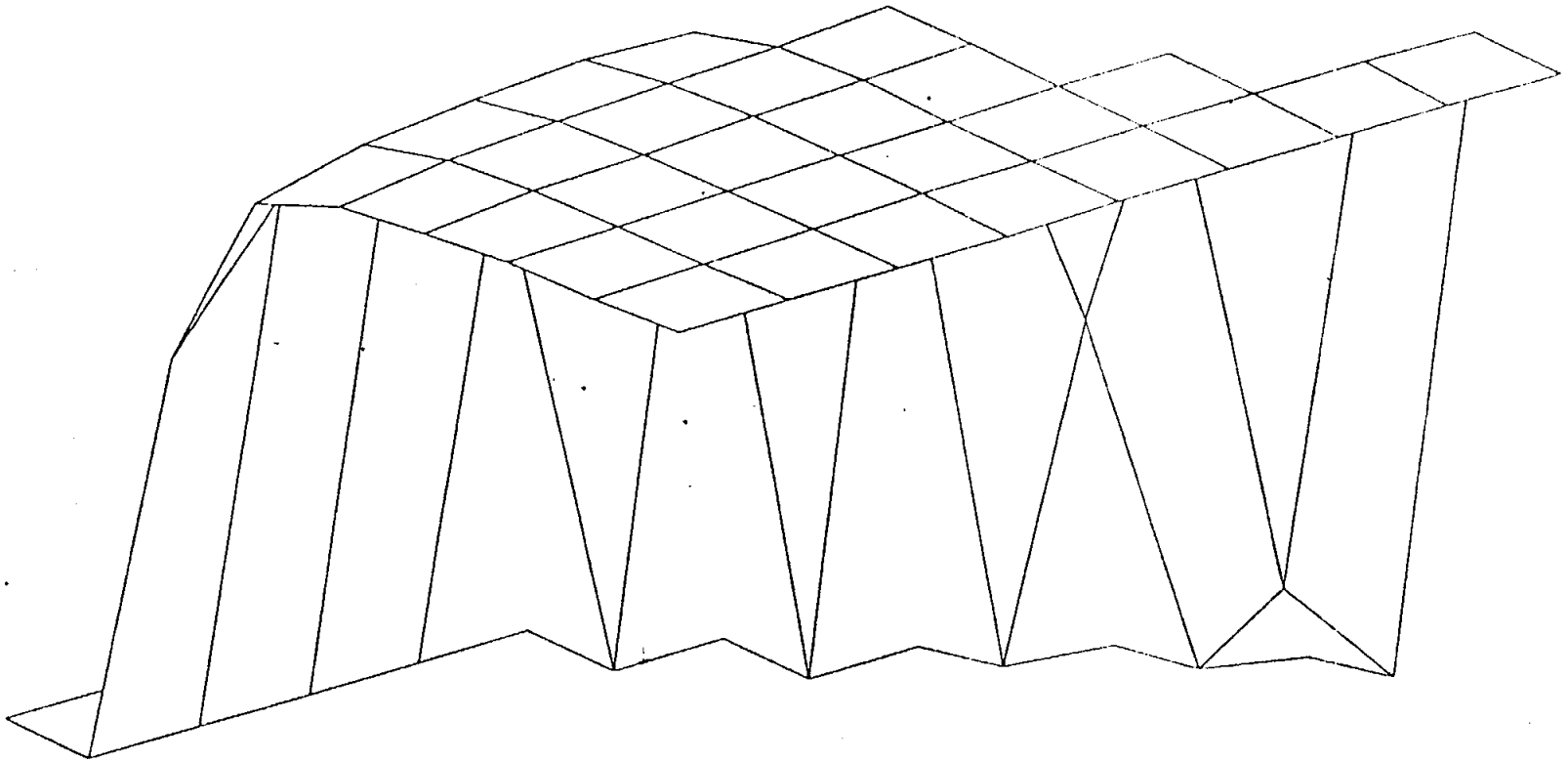
Approach Vinf °lat, 115° long (i = 49.4°)

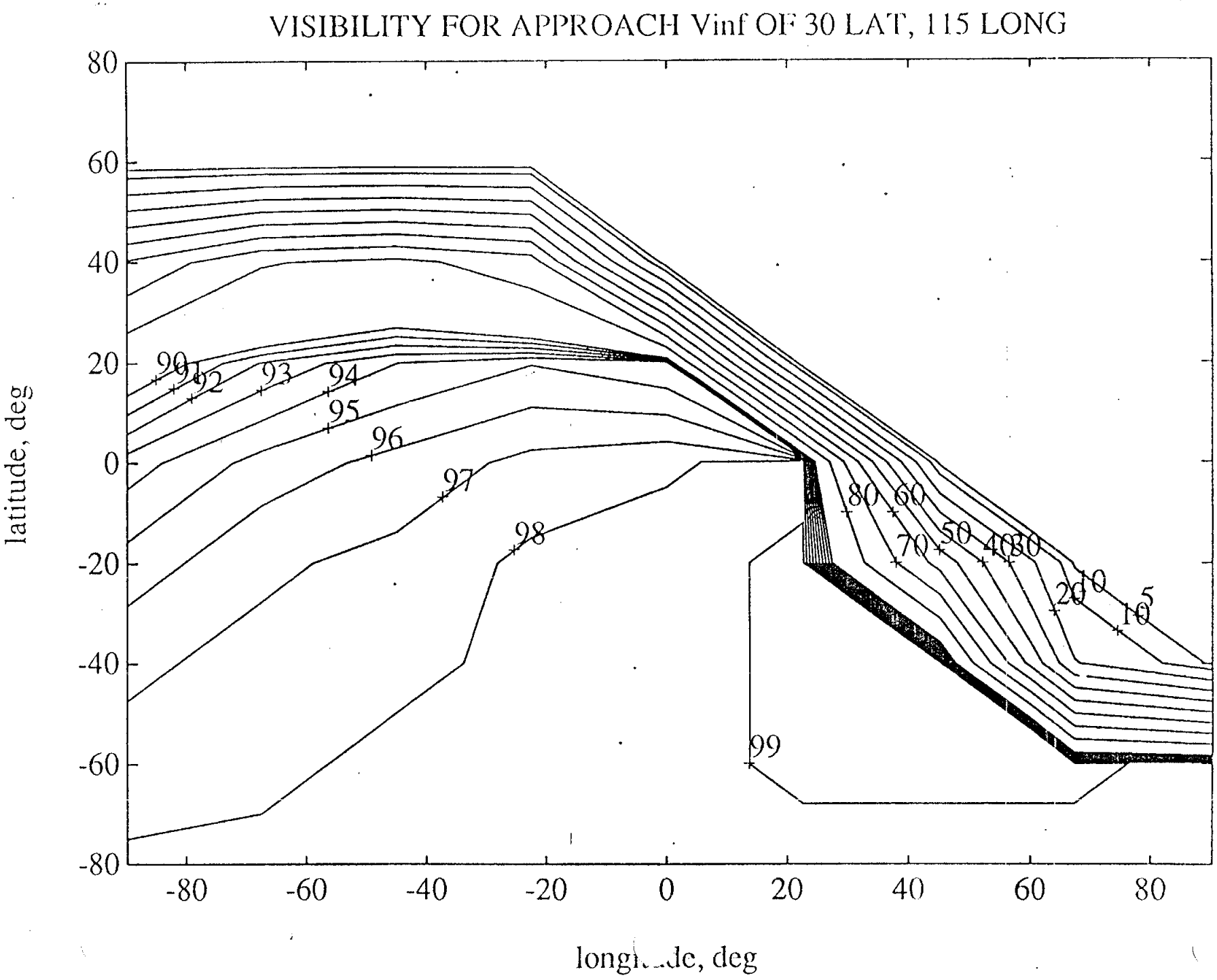




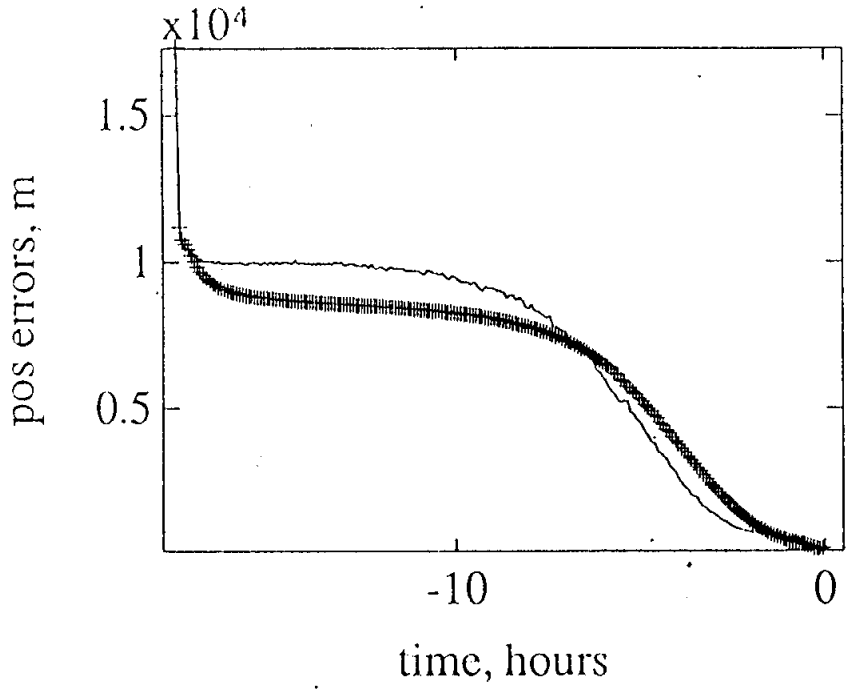


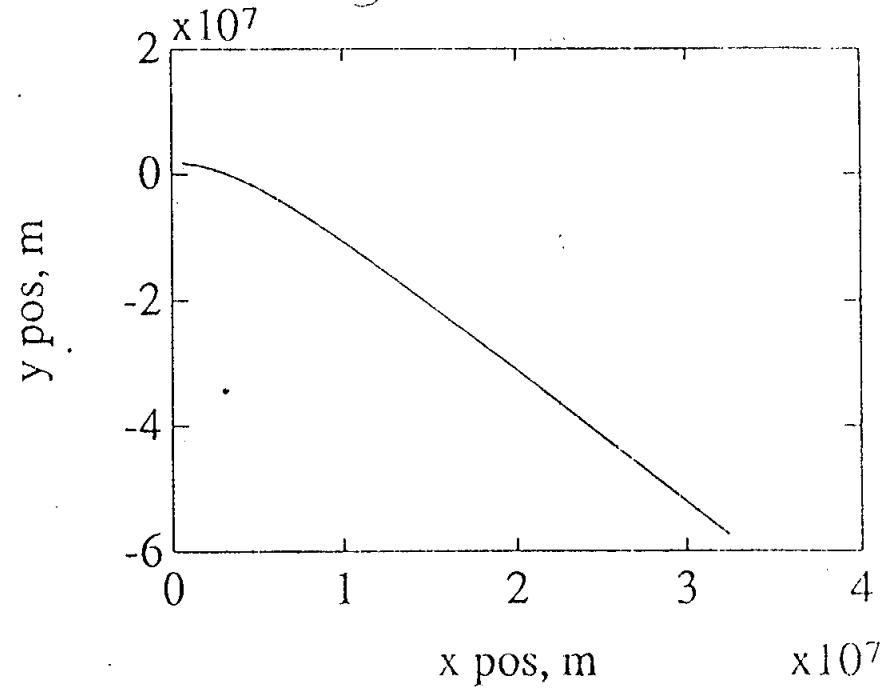


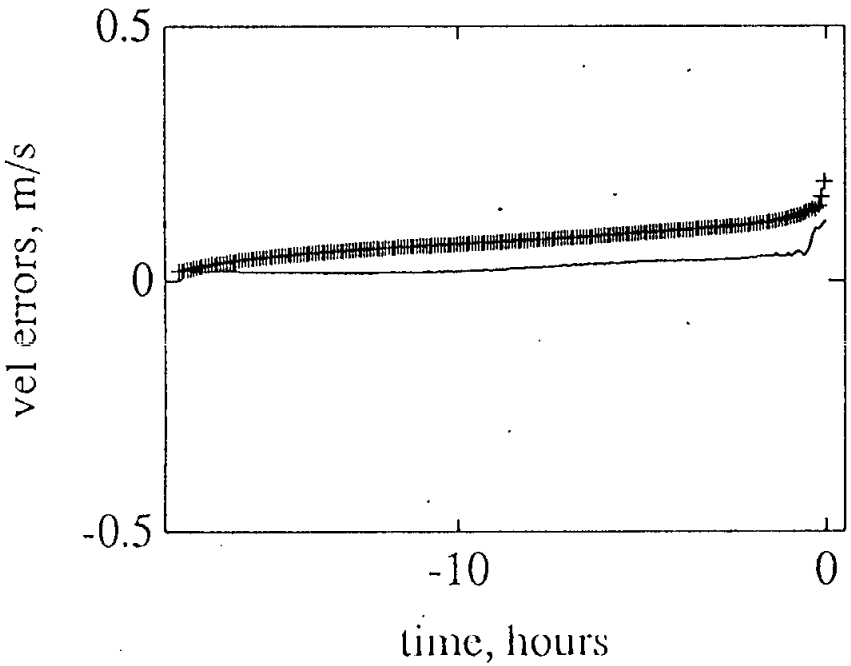


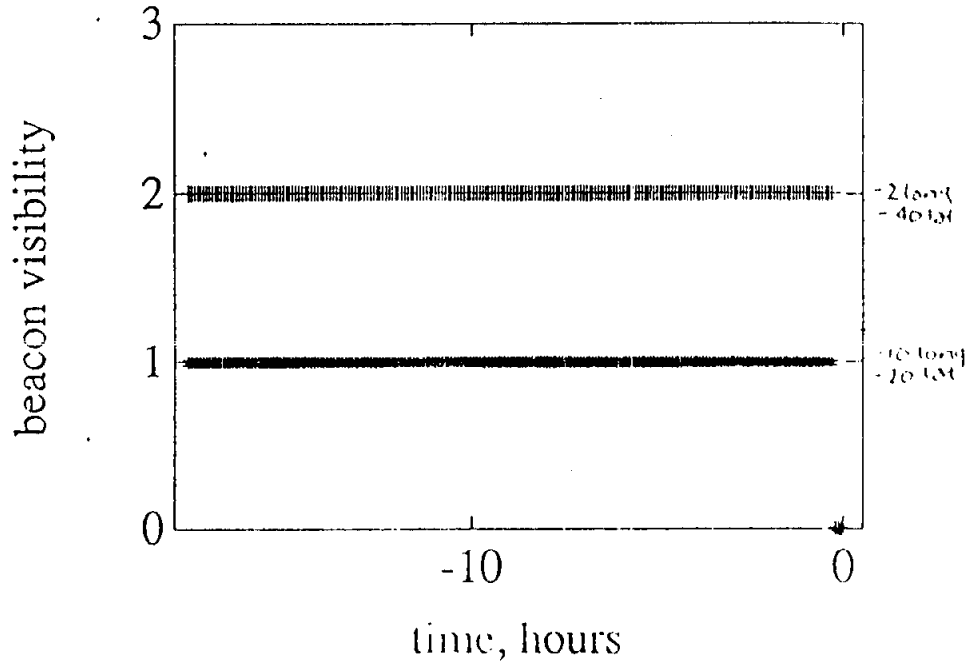

VISIBILITY FOR APPROACH Vinf OF 30 LAT, 115 LONG
latitude, deg
longitude, deg
80
60
40
20
0
-20
-40
-60
-80
-80
-60
-40
-20
0
20
40
60
80
90
91
92
93
94
95
96
97
98
99
80
60
70
50
40
30
20
10
10
5

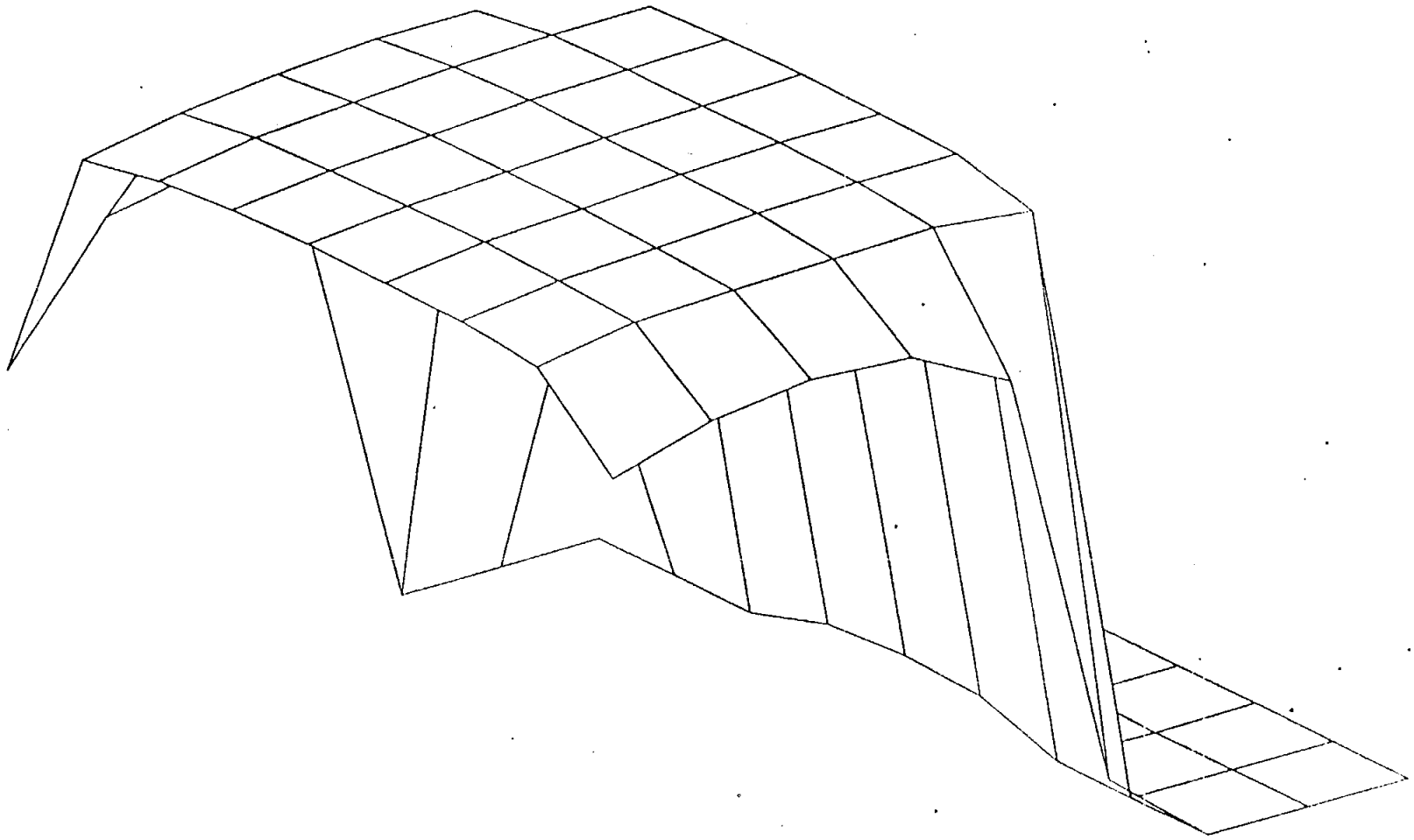
Approach Vid lat, 115° long (c-2°)

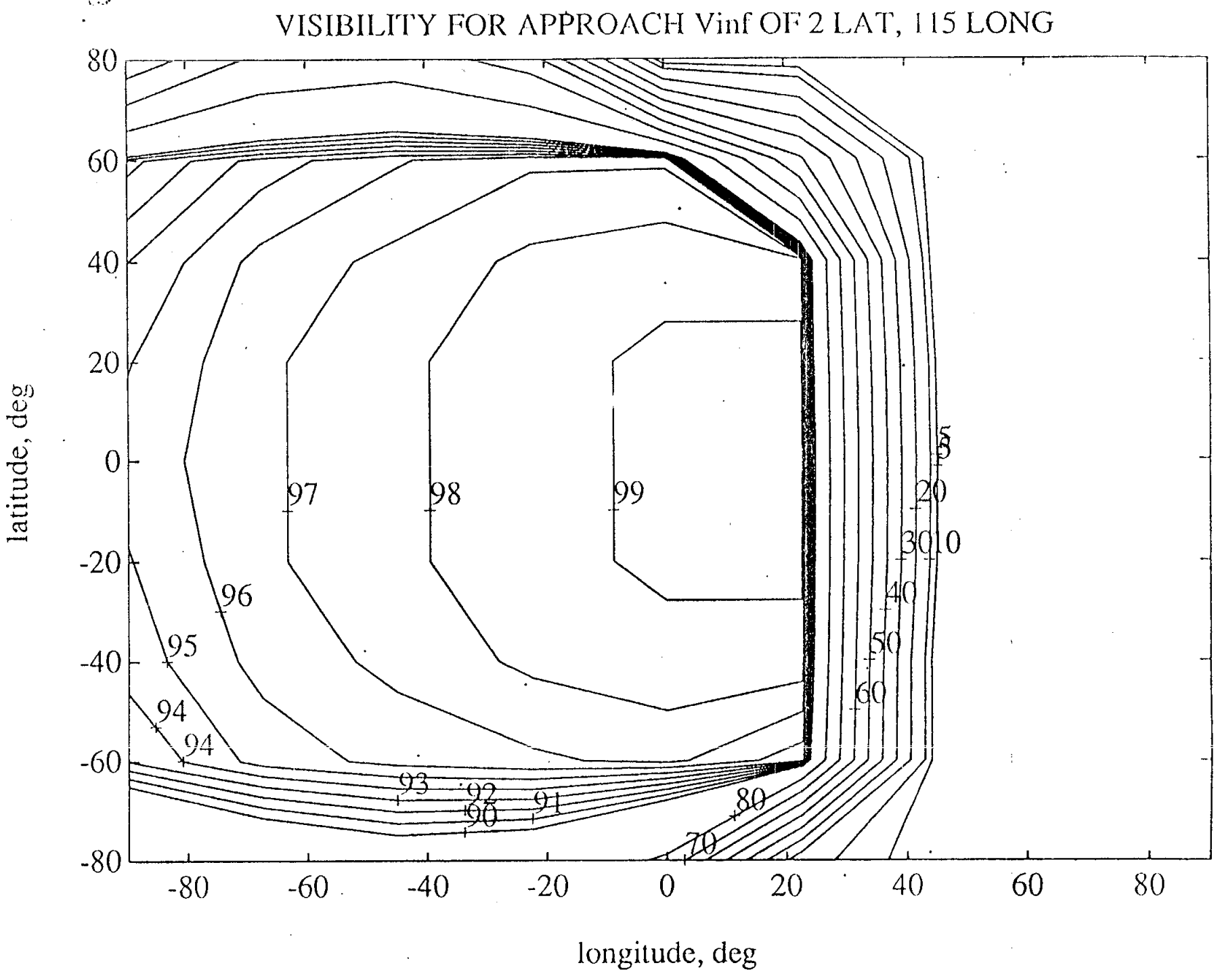




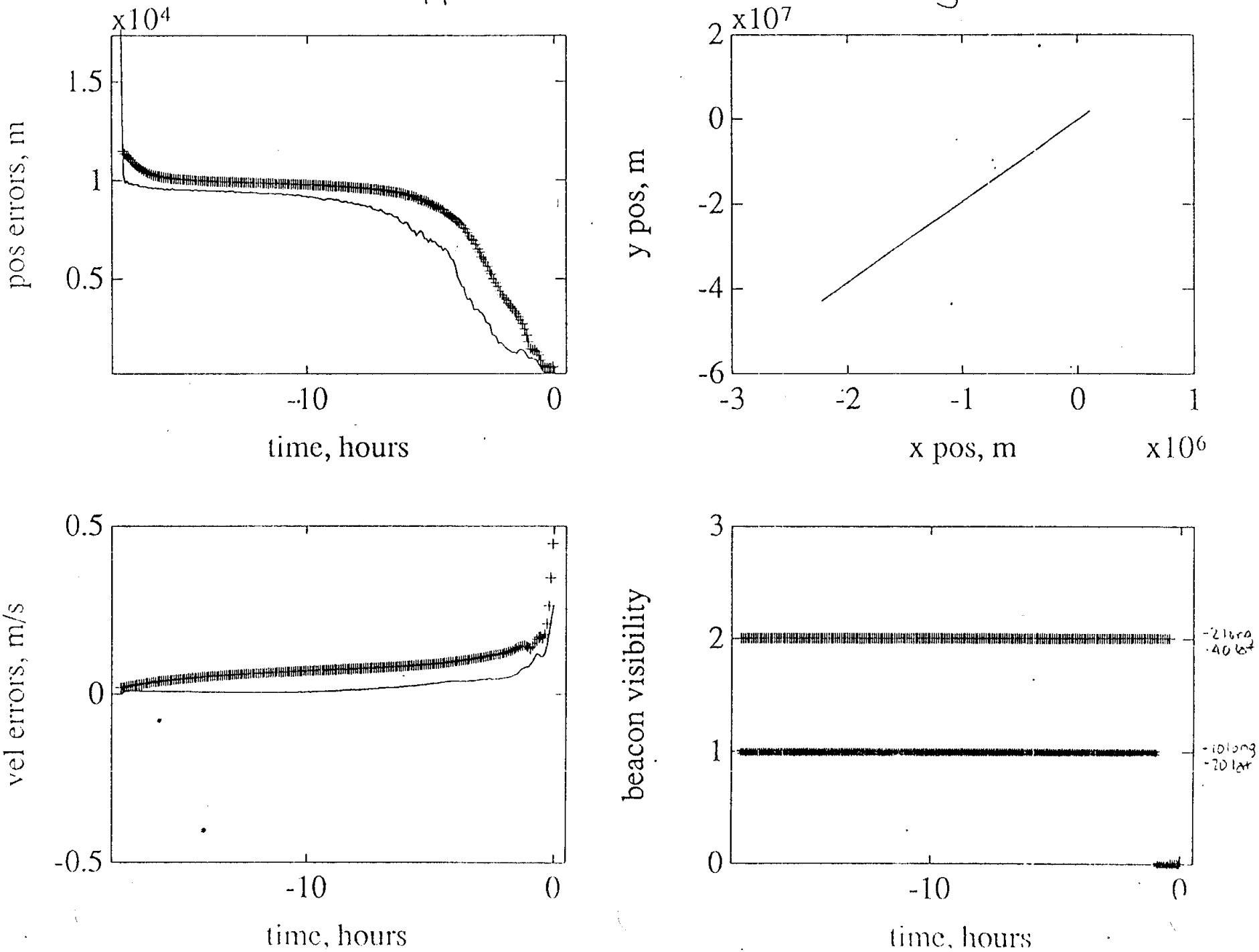


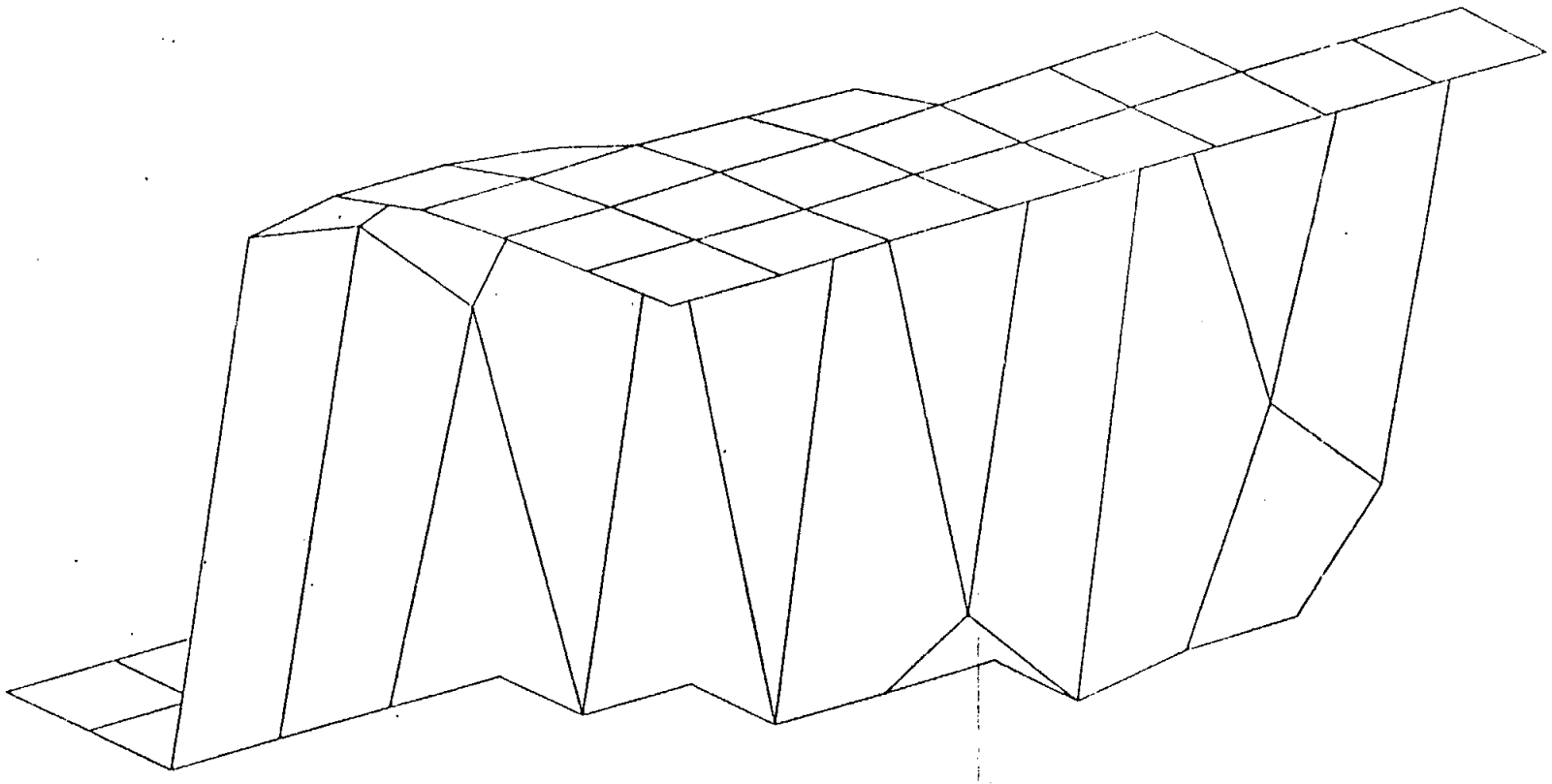


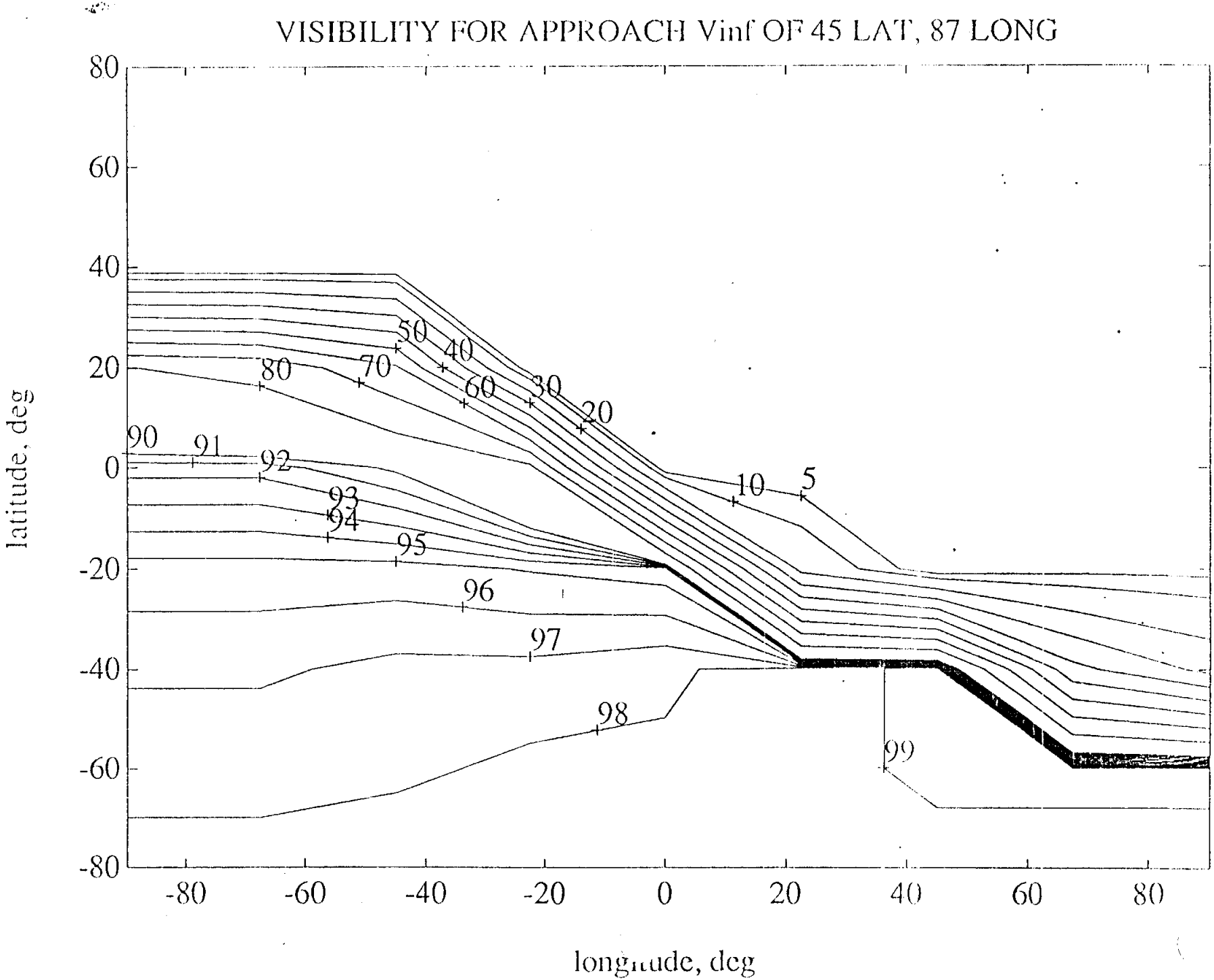



VISIBILITY FOR APPROACH Vinf OF 2 LAT, 115 LONG
latitude, deg
longitude, deg
80
60
40
20
0
-20
-40
-60
-80
-80
-60
-40
-20
0
20
40
60
80
97
98
99
96
95
94
94
93
92
90
91
80
70
20
30
10
40
50
60

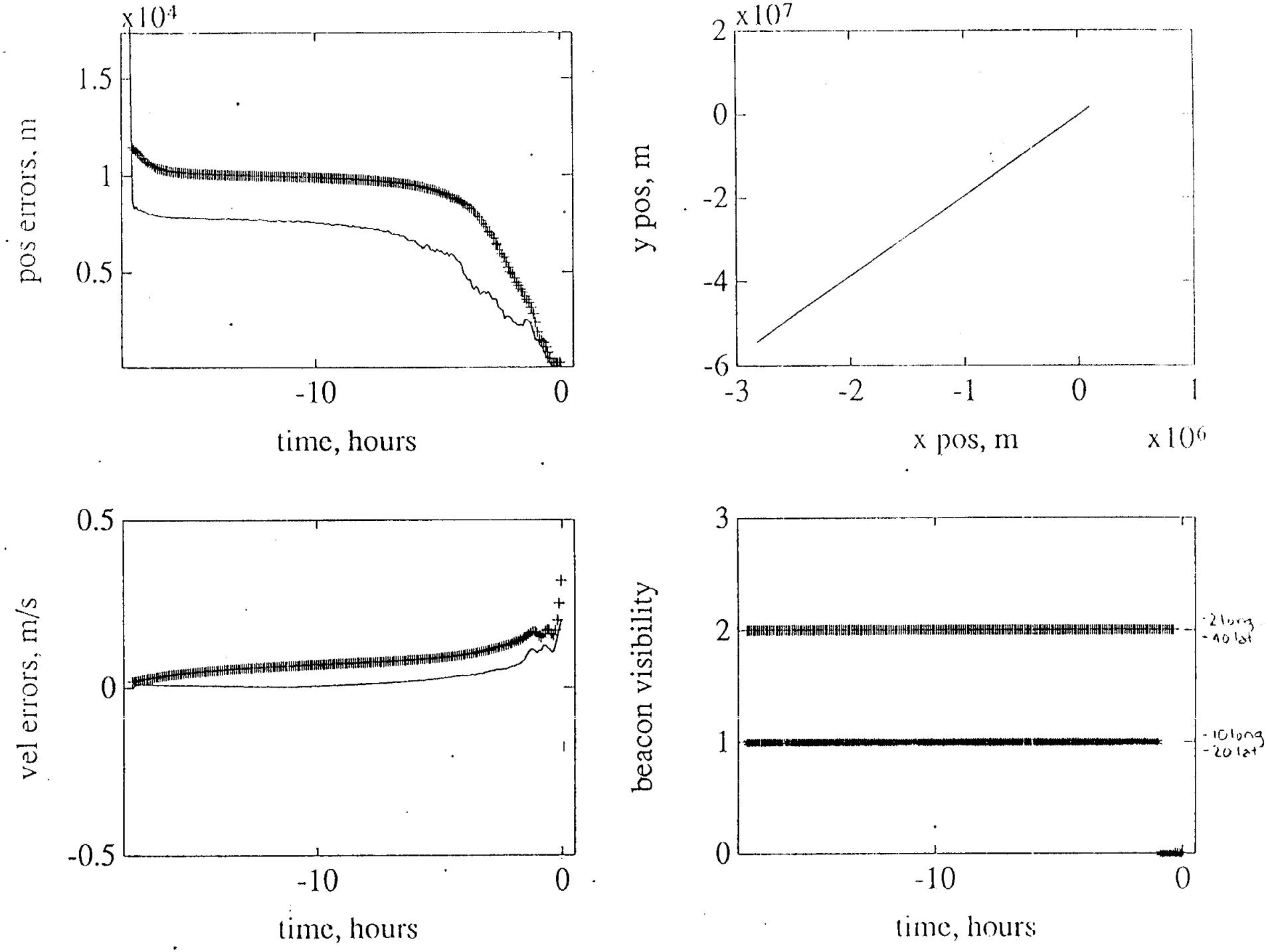
Approach vint. @ 45° lat, 87° long (i = 89.8°)

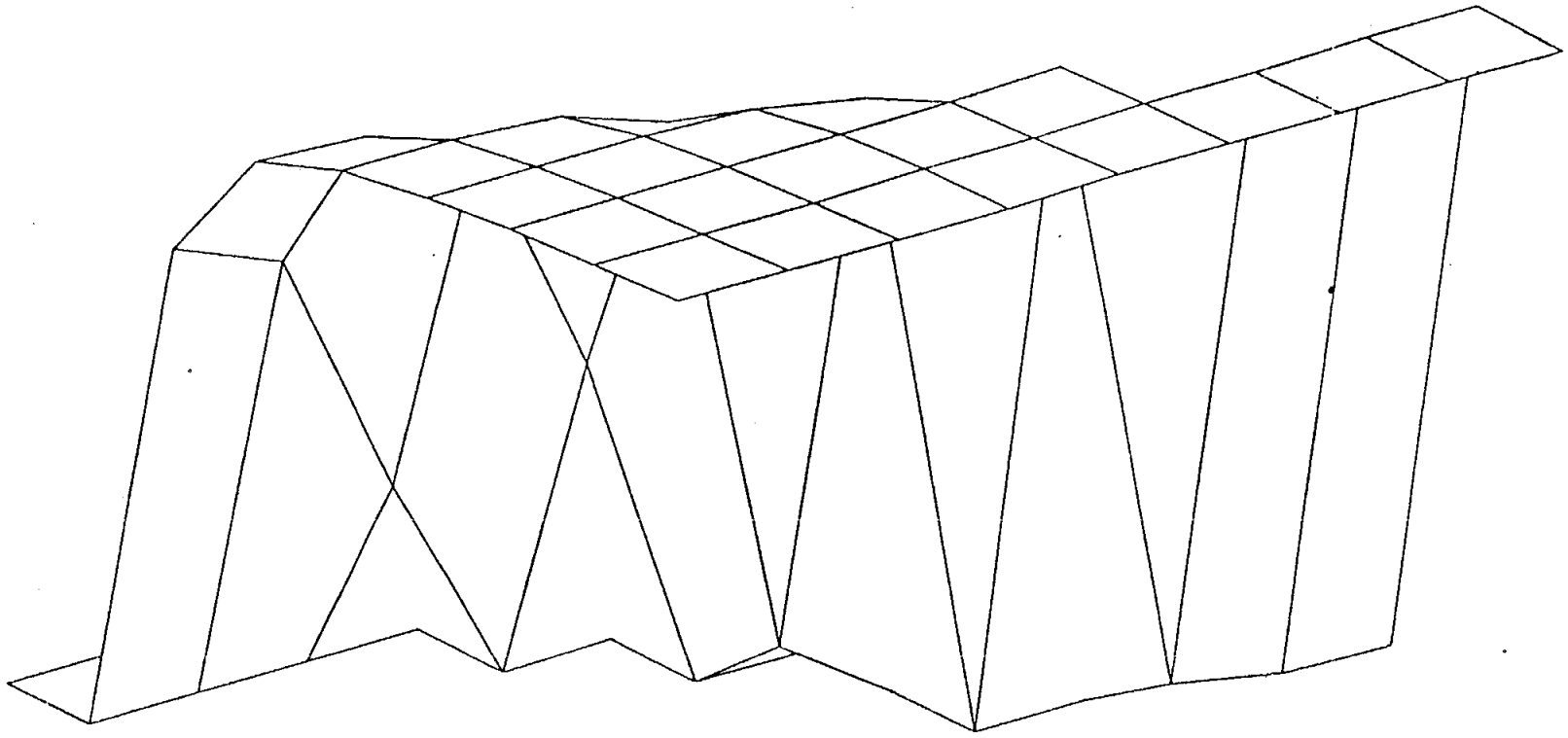


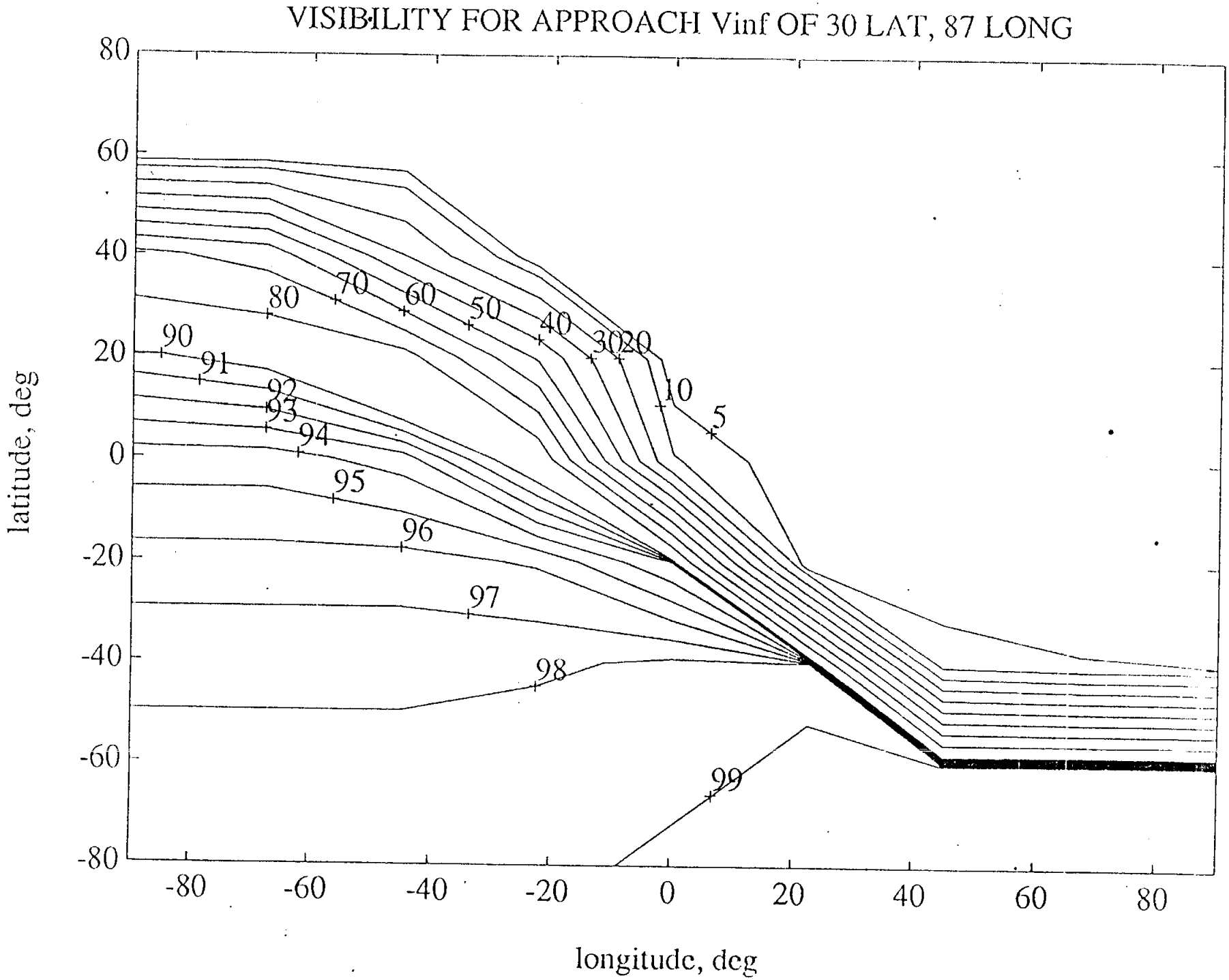



VISIBILITY FOR APPROACH Vinf OF 45 LAT, 87 LONG
latitude, deg
longitude, deg
80
60
40
20
0
-20
-40
-60
-80
-80
-60
-40
-20
0
20
40
60
80
90
91
92
93
94
95
96
97
98
99
80
70
50
40
60
30
20
10
5

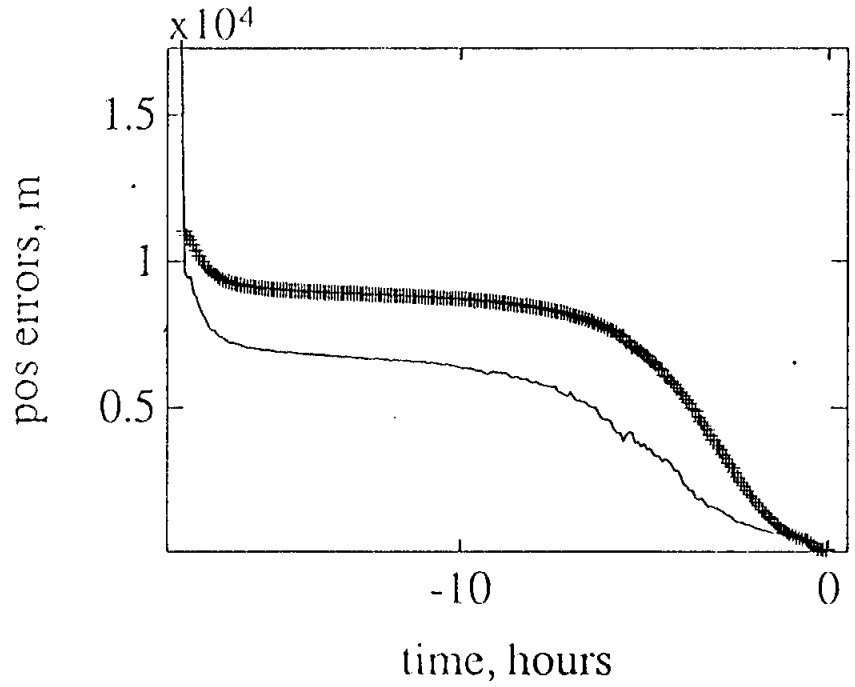
Approach ... : 30° lat, 81° long (i = 89.7°)

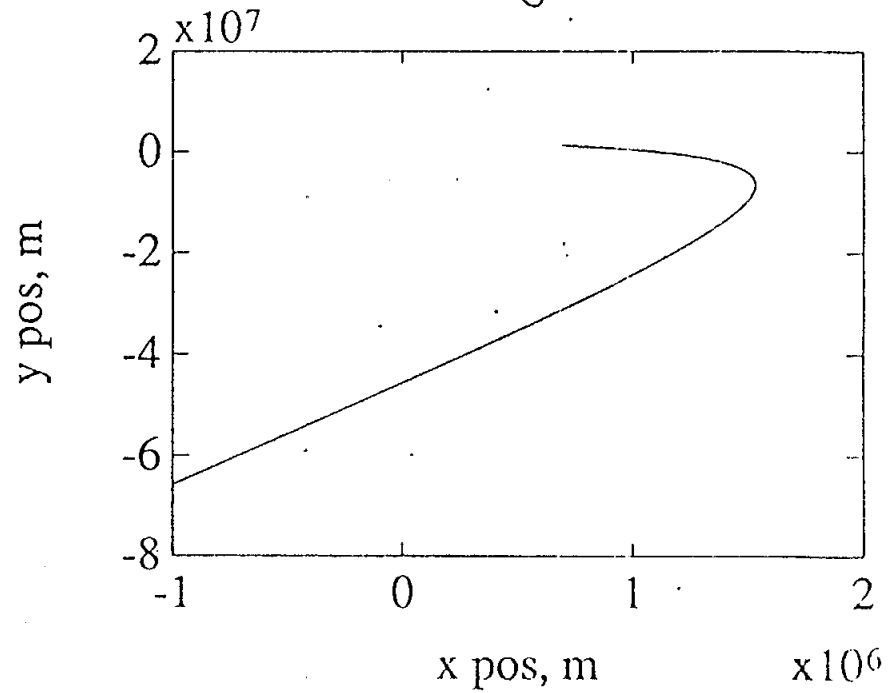


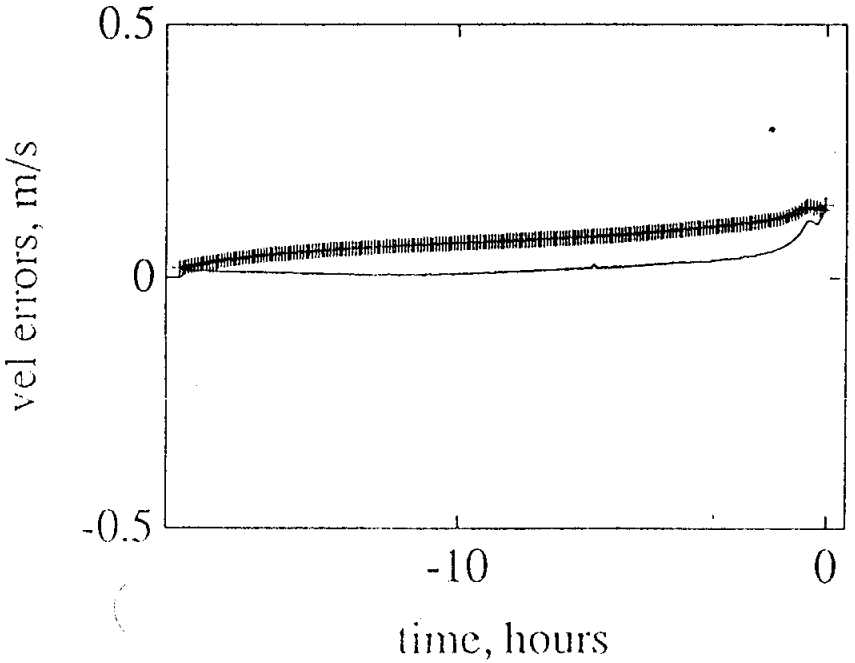


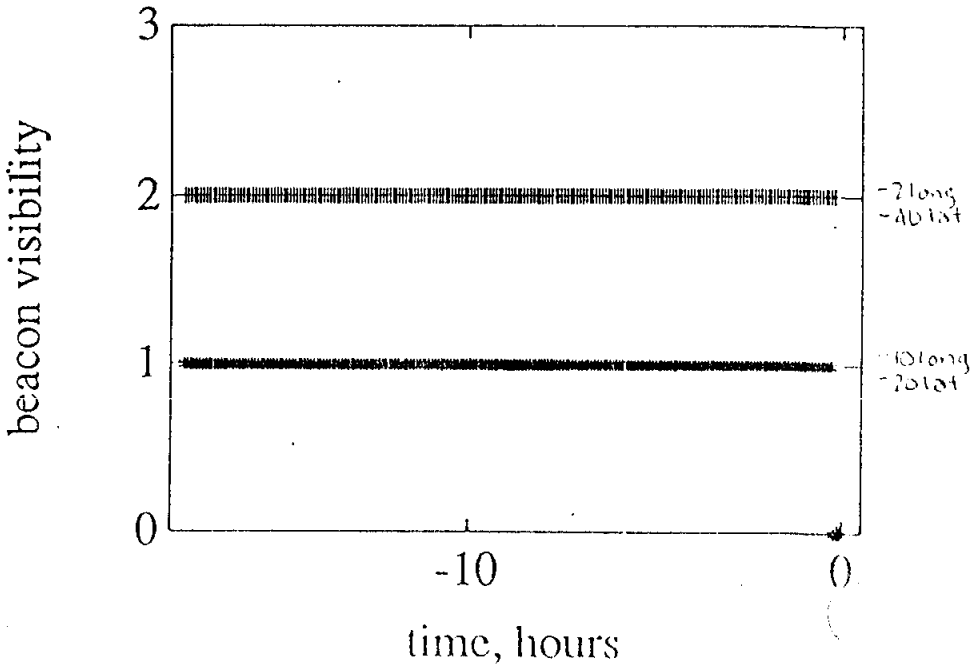

VISIBILITY FOR APPROACH Vinf OF 30 LAT, 87 LONG
latitude, deg
longitude, deg
80
60
40
20
0
-20
-40
-60
-80
-80
-60
-40
-20
0
20
40
60
80
90
91
92
93
94
95
96
97
98
99
80
70
60
50
40
30
20
10
5

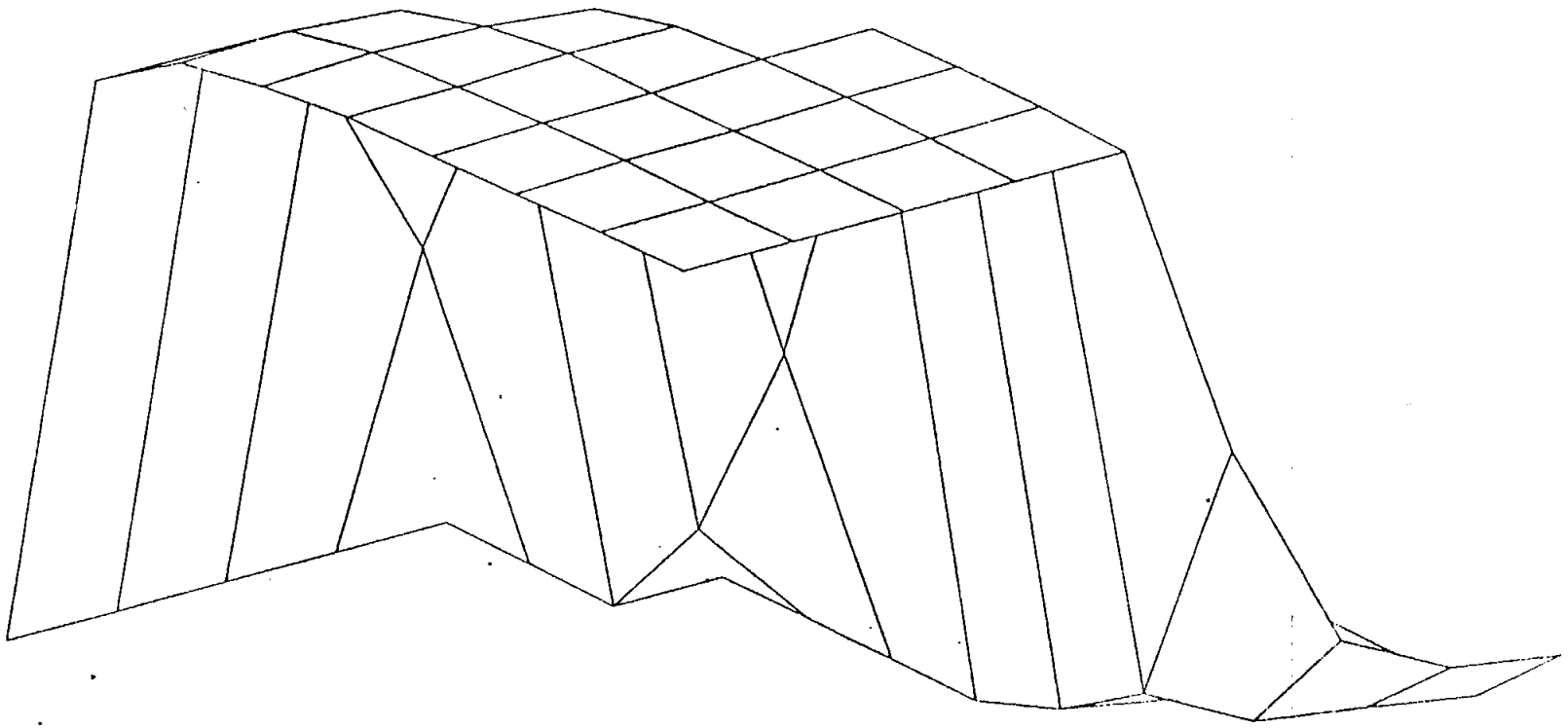
Approach of 2° lat, 87° long (i=61.8°)



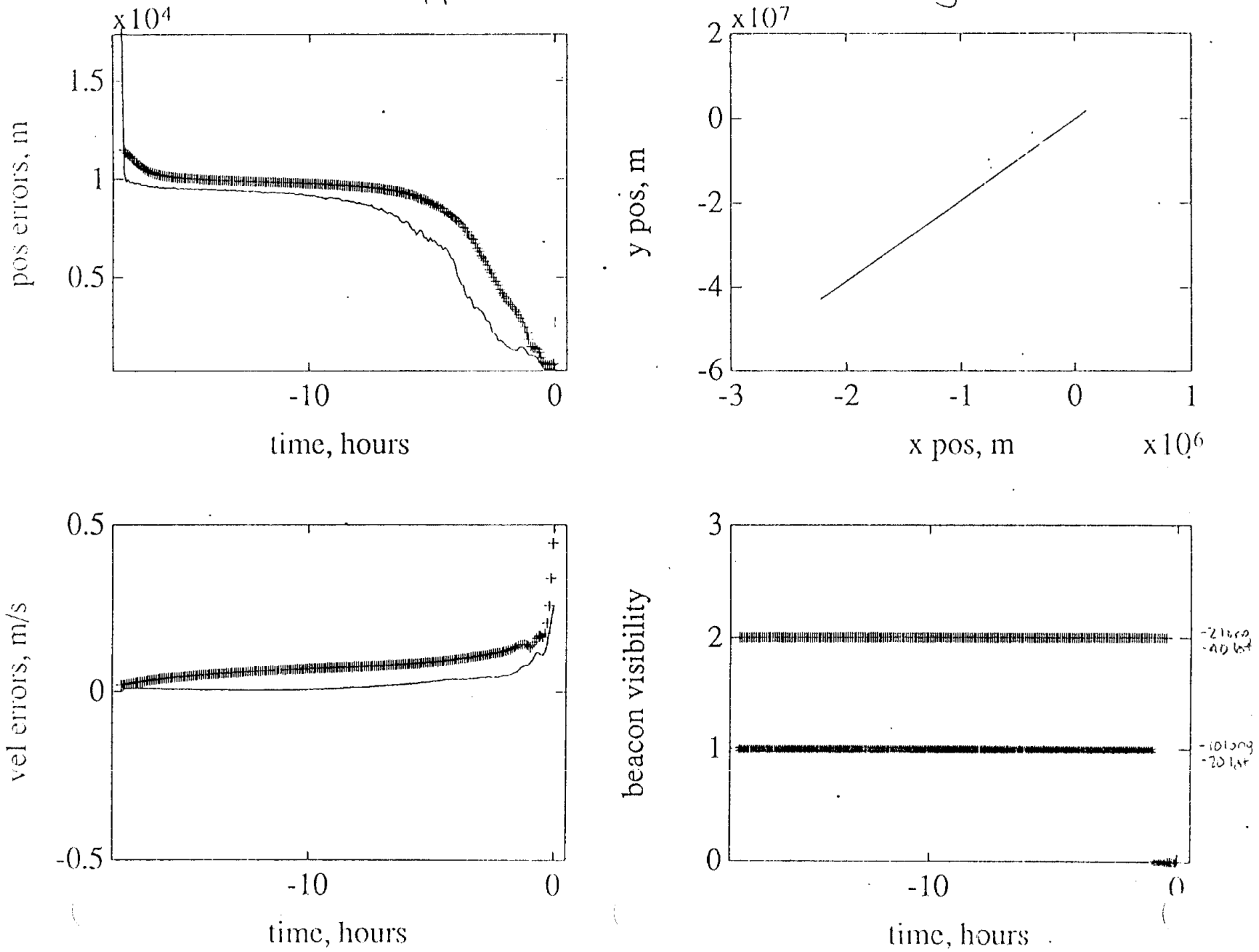


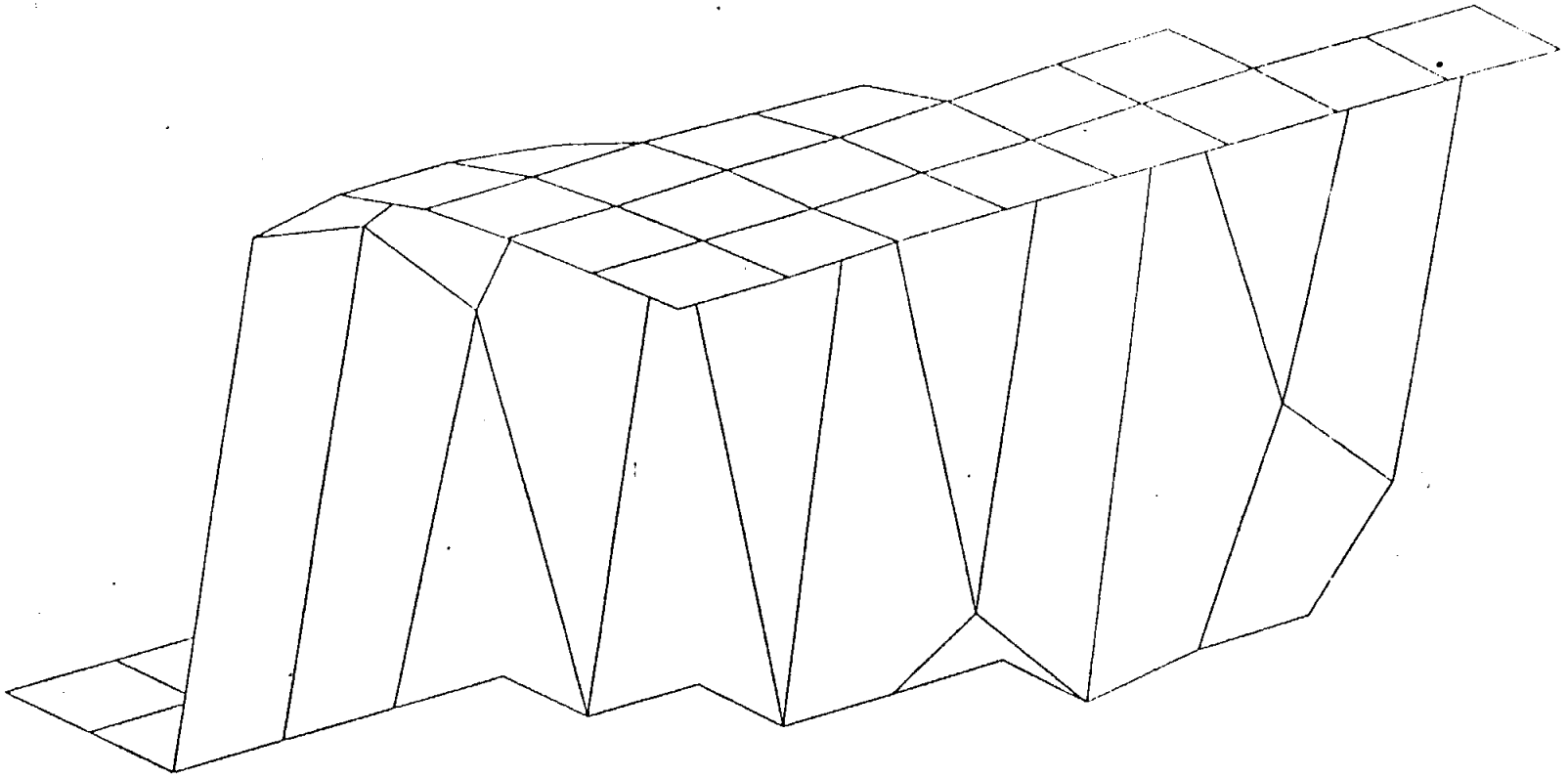


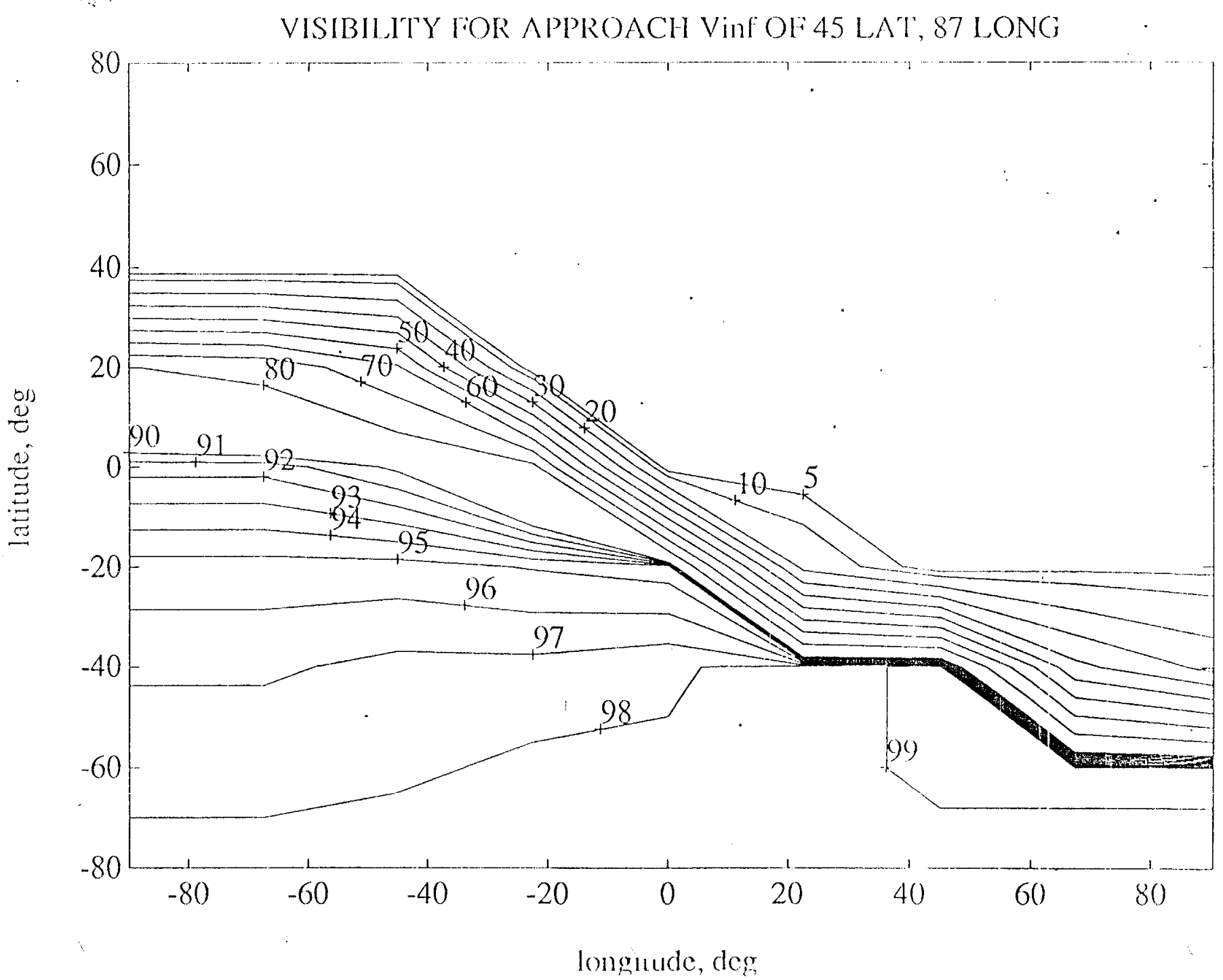




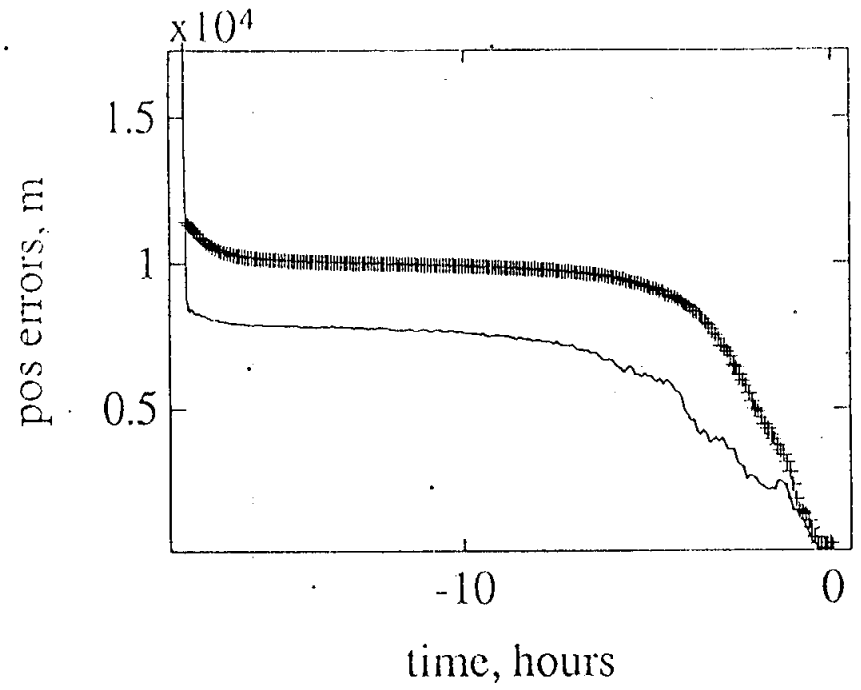
Approach with f 45° lat, 81° long (i=89.8°,

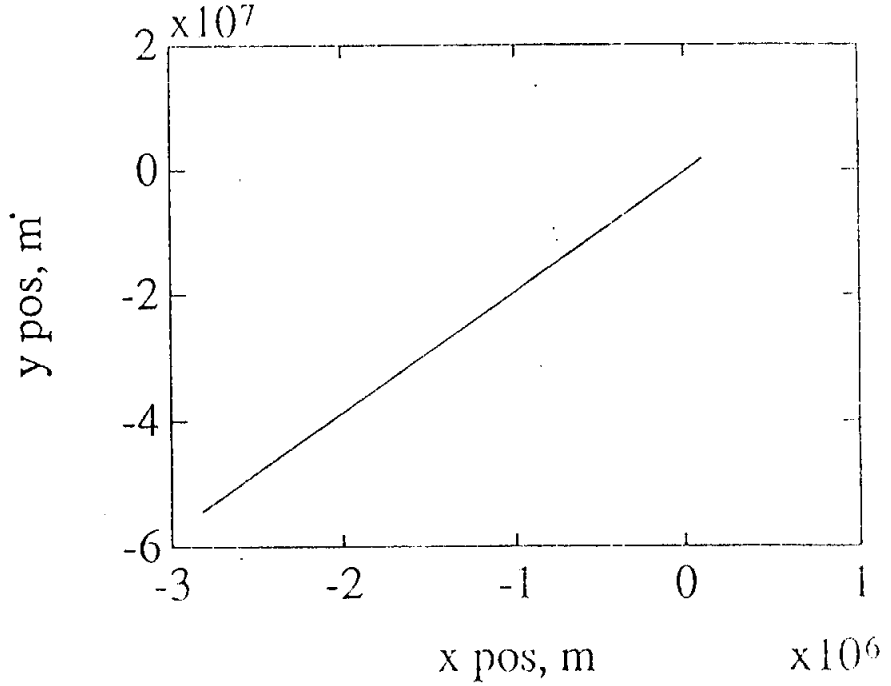


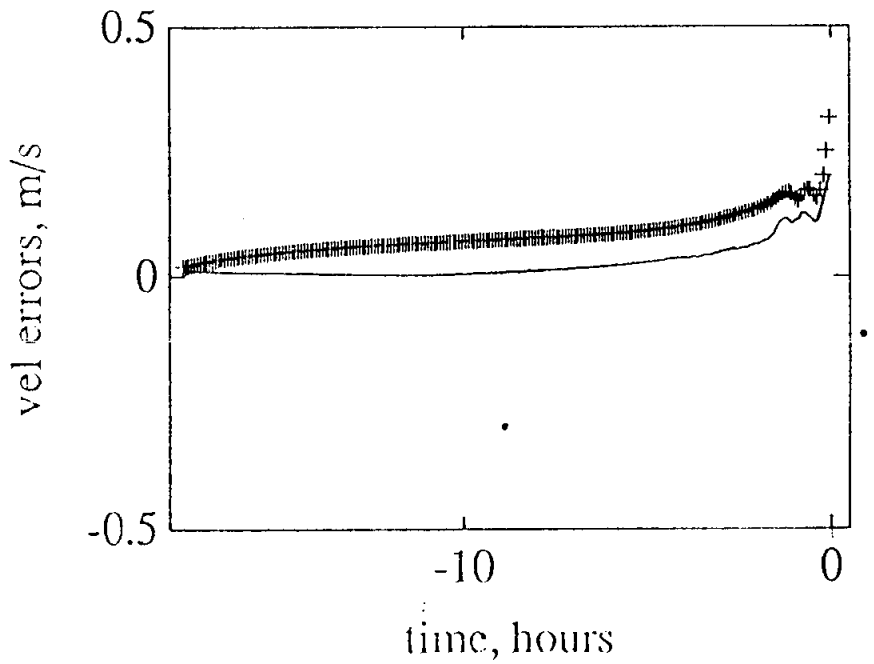


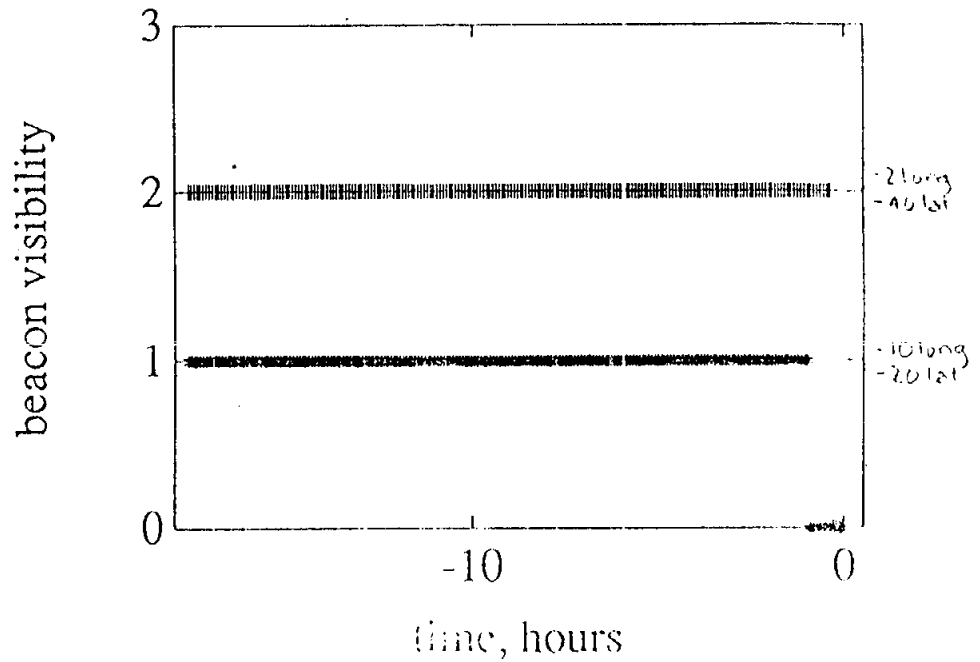

VISIBILITY FOR APPROACH Vinf OF 45 LAT, 87 LONG
latitude, deg
longitude, deg
80
60
40
20
0
-20
-40
-60
-80
-80
-60
-40
-20
0
20
40
60
80
90
91
92
93
94
95
96
97
98
99
80
70
50
40
60
30
20
10
5

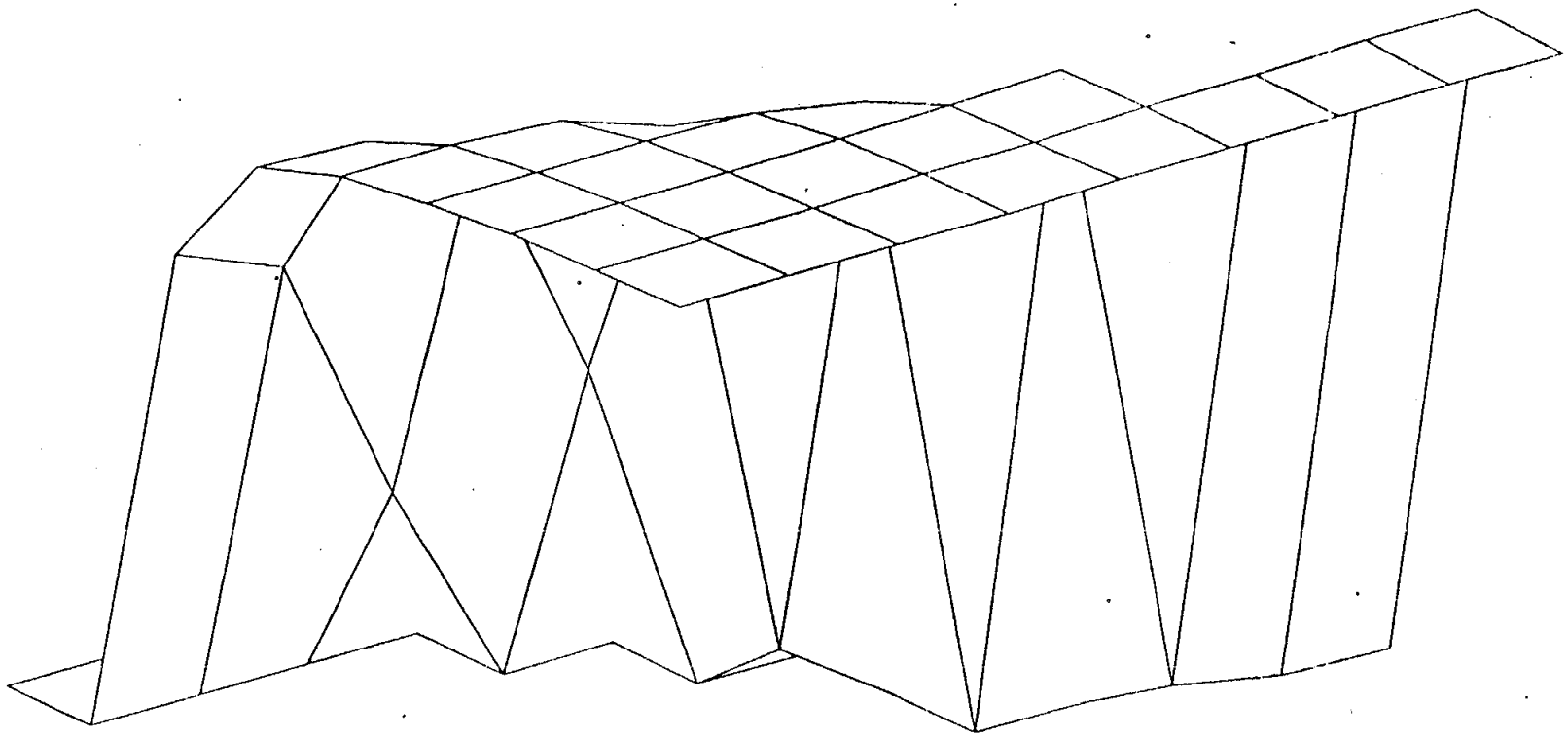
Approach ... : 30° lat, 81° long (i = 89.1°)

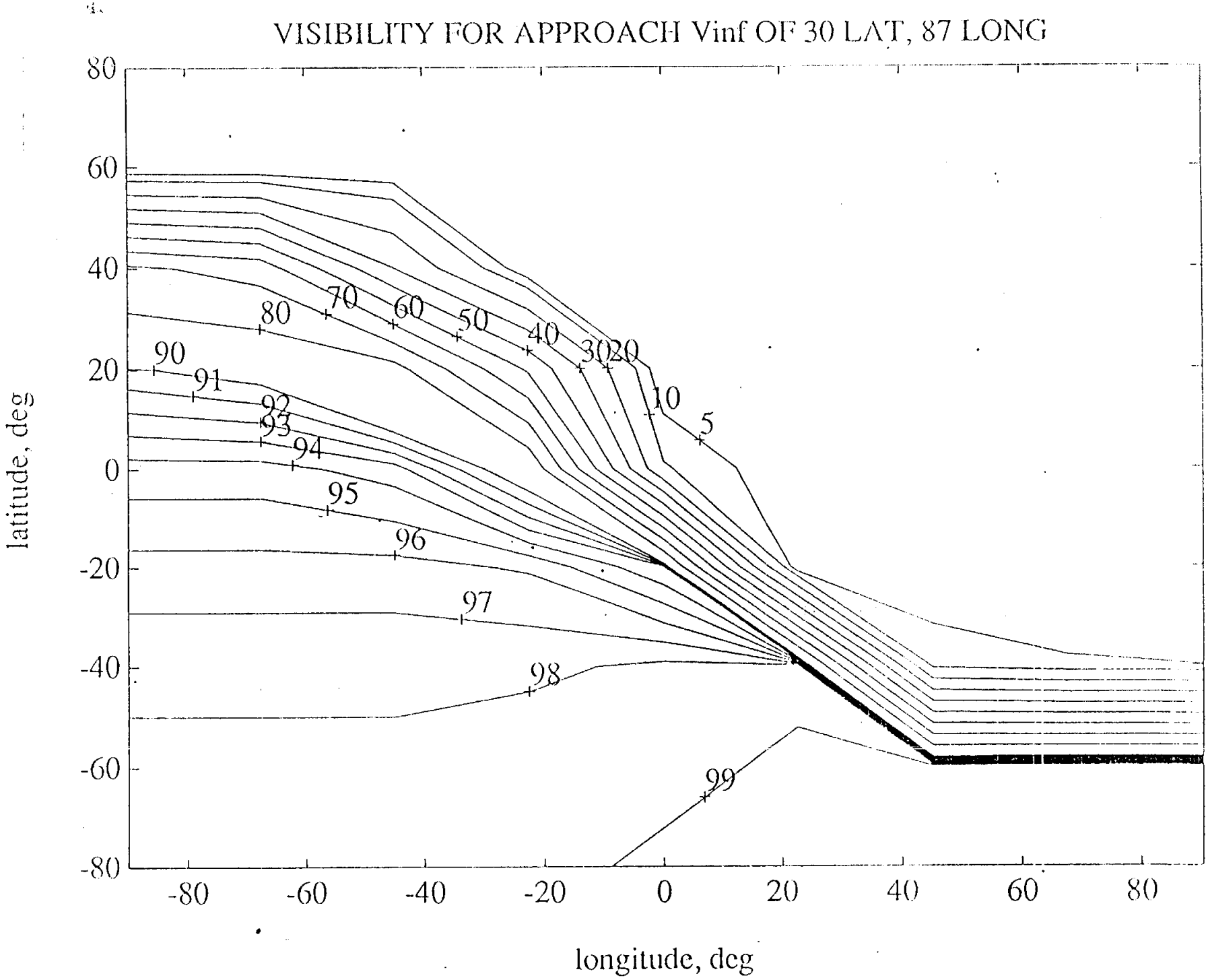




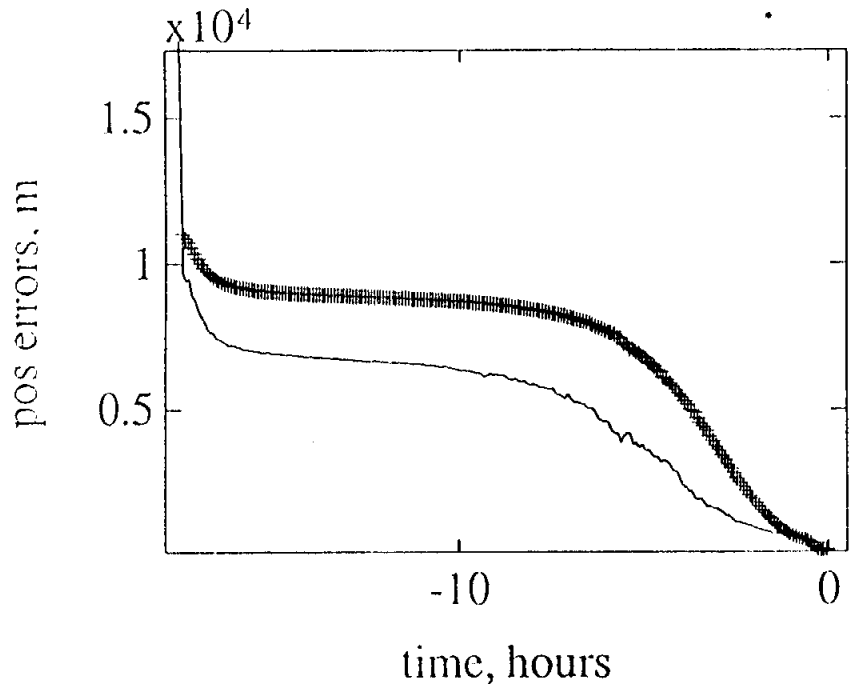


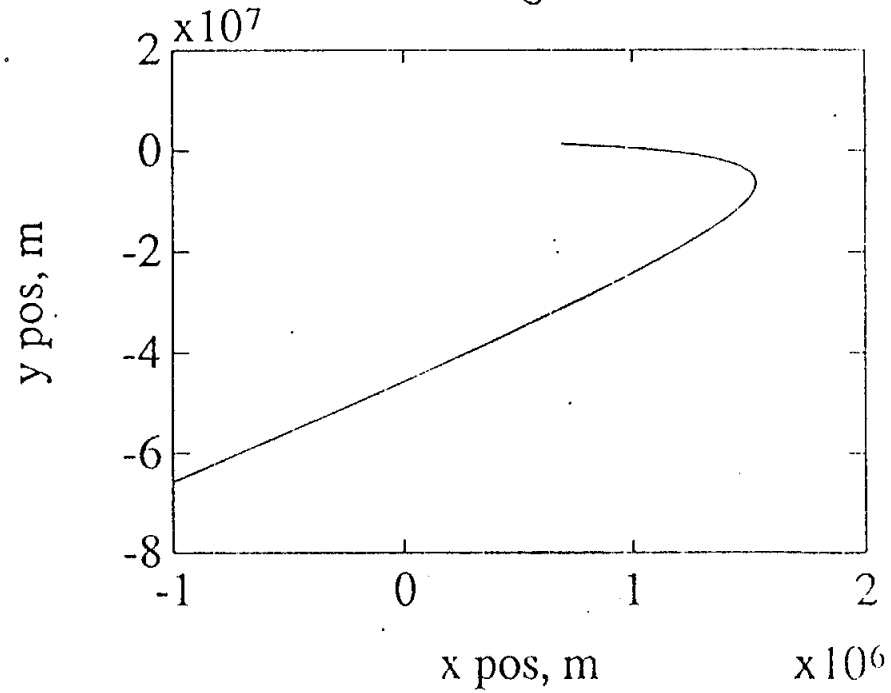


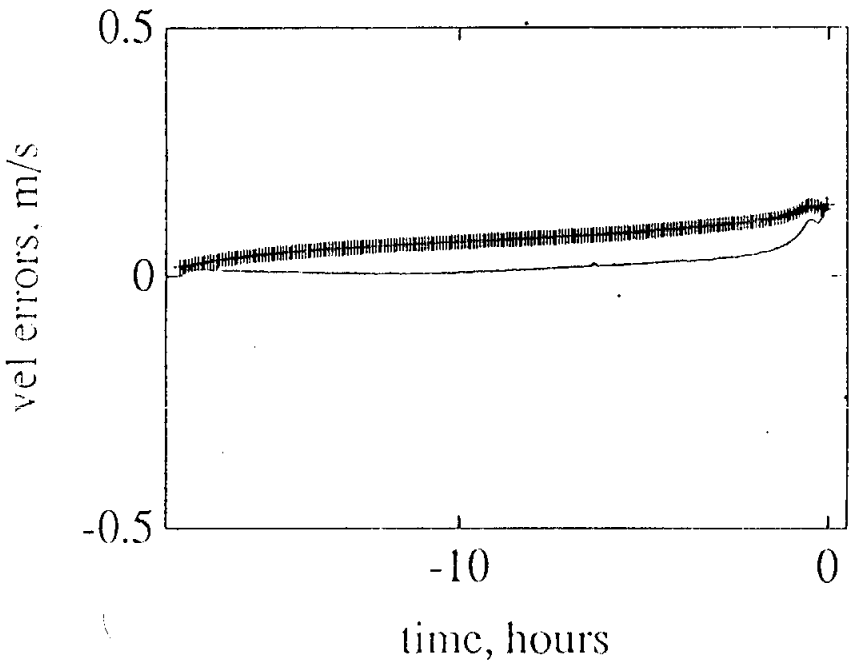


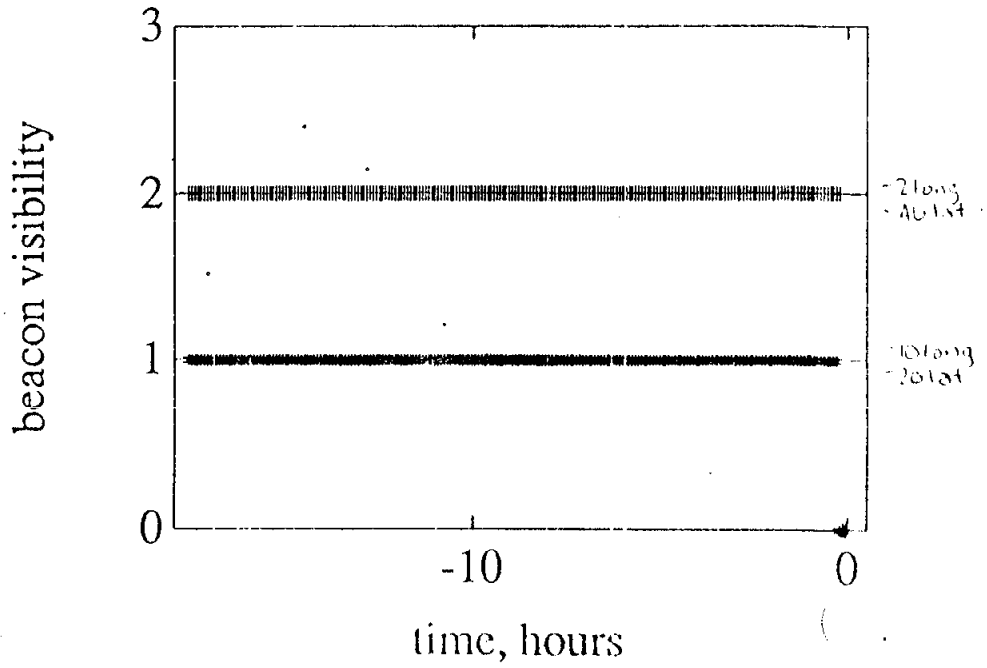

VISIBILITY FOR APPROACH Vinf OF 30 LAT, 87 LONG
latitude, deg
longitude, deg
80
60
40
20
0
-20
-40
-60
-80
-80
-60
-40
-20
0
20
40
60
80
90
91
92
93
94
95
96
97
98
99
80
70
60
50
40
30
20
10
5

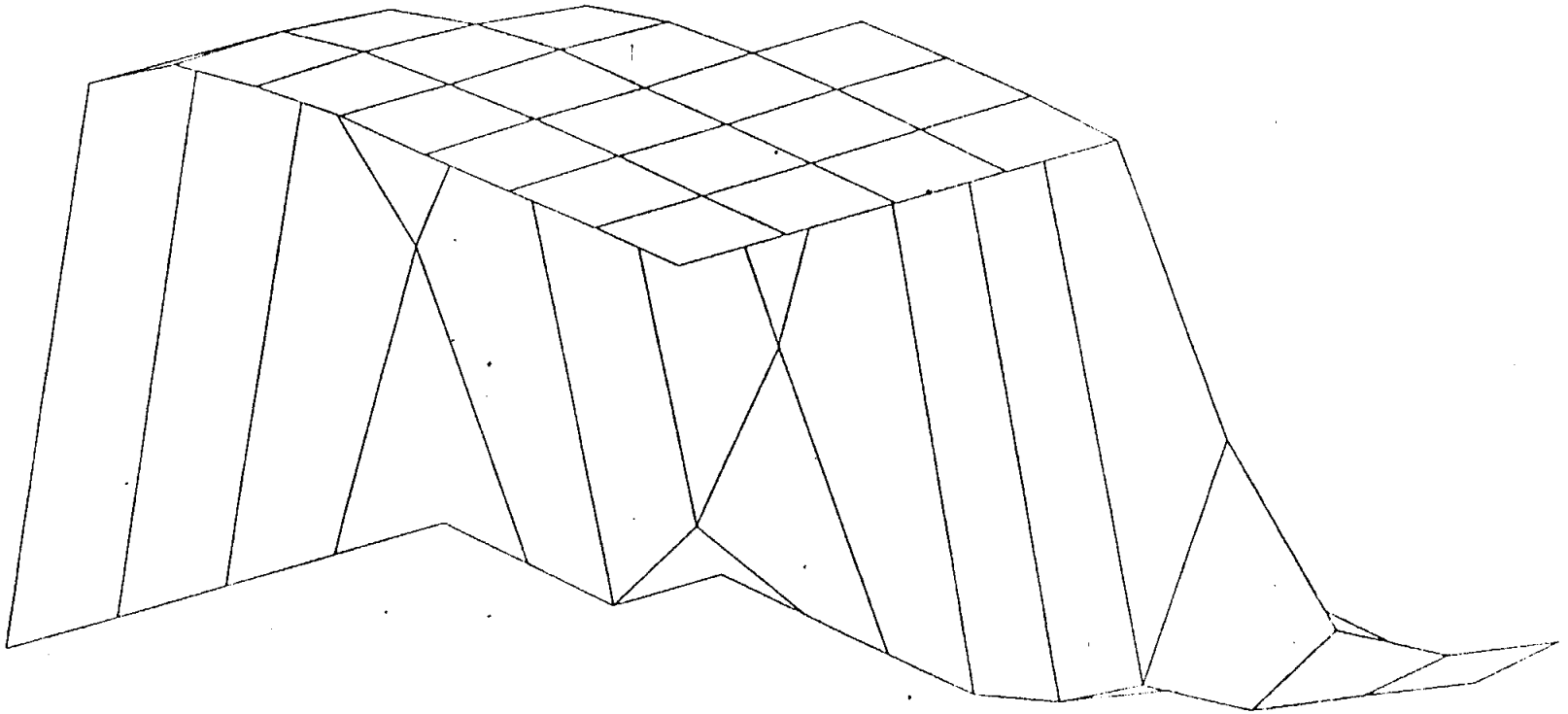
Approach " of 2° lat, 81° long (L=61.8°)

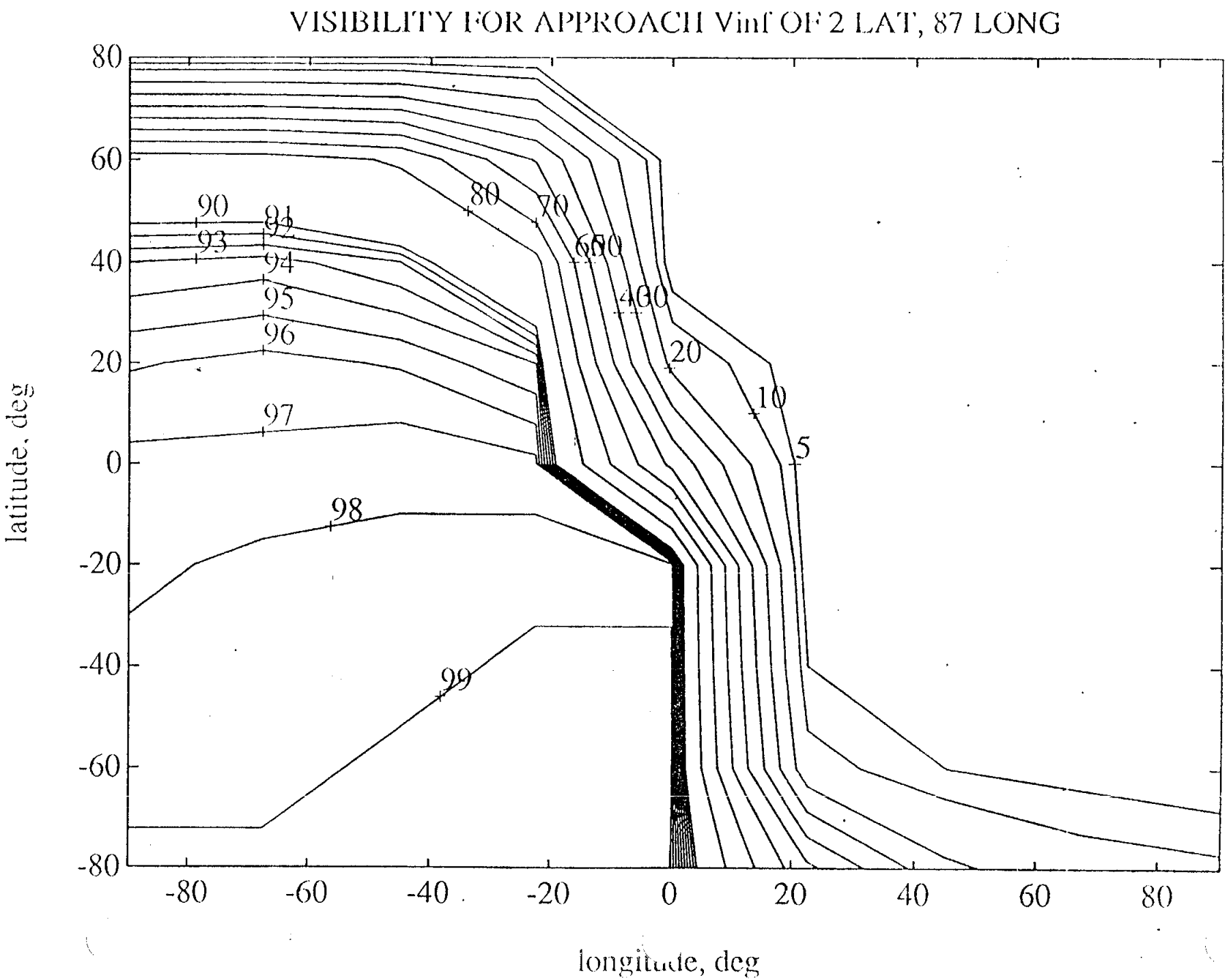


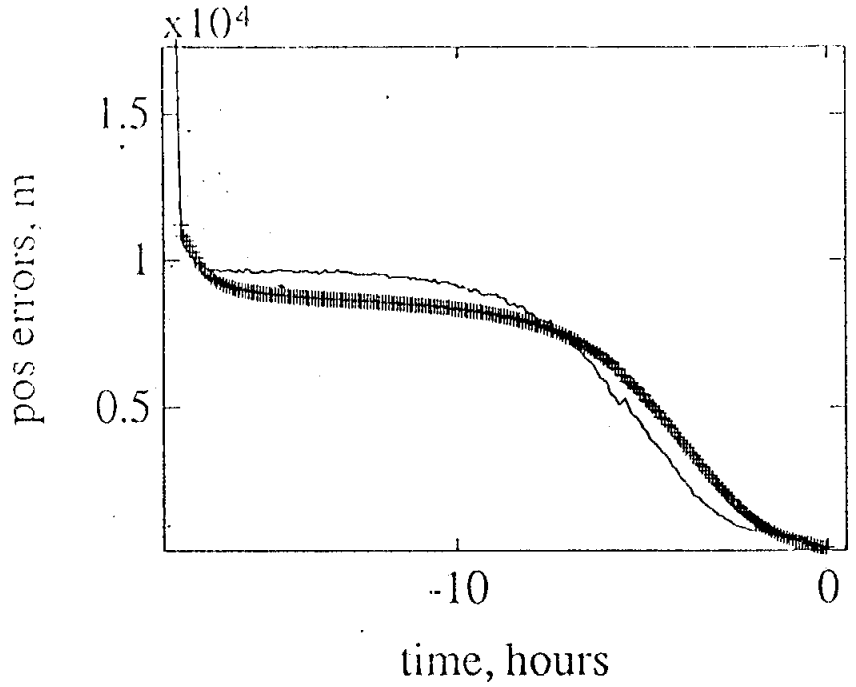


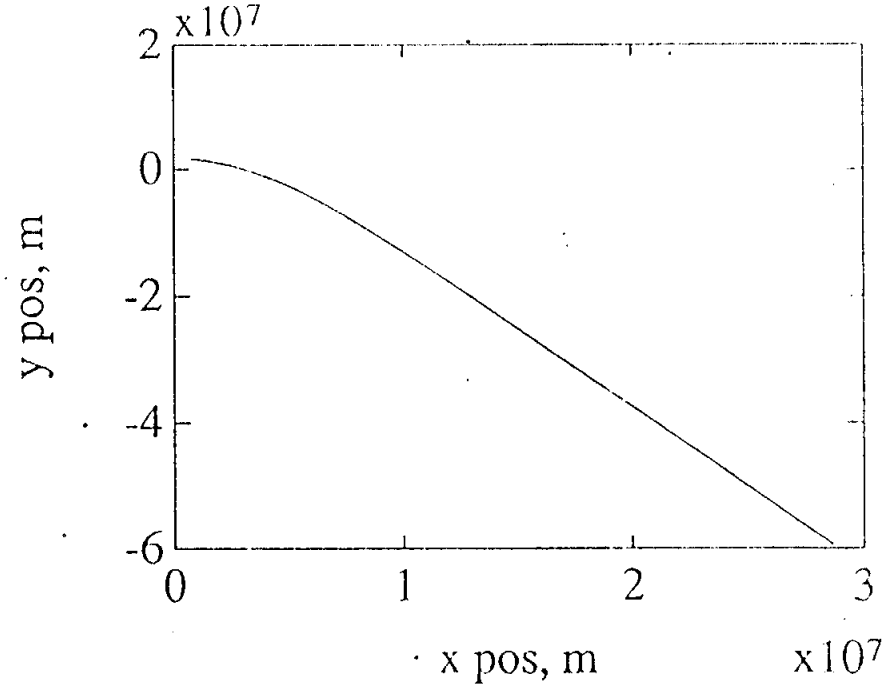


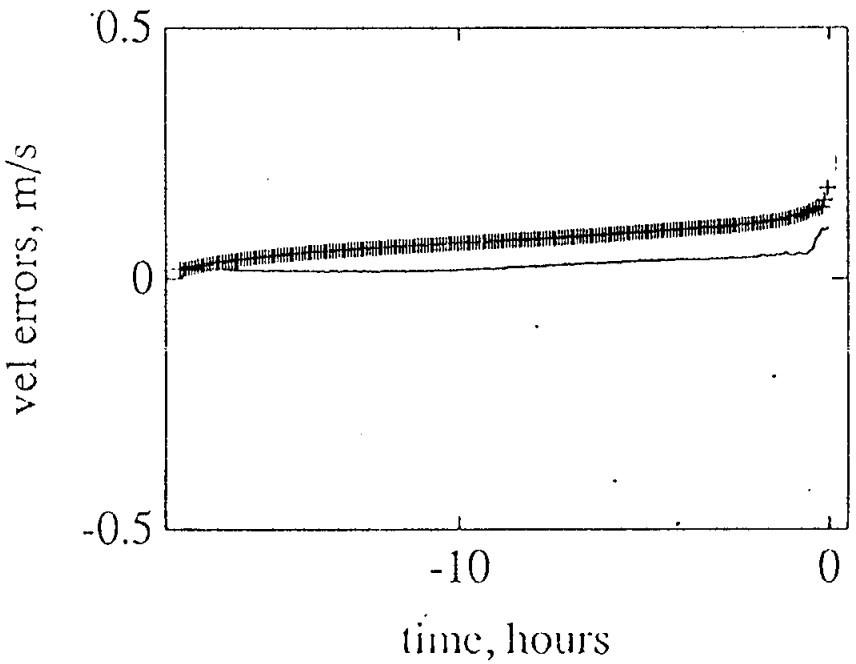


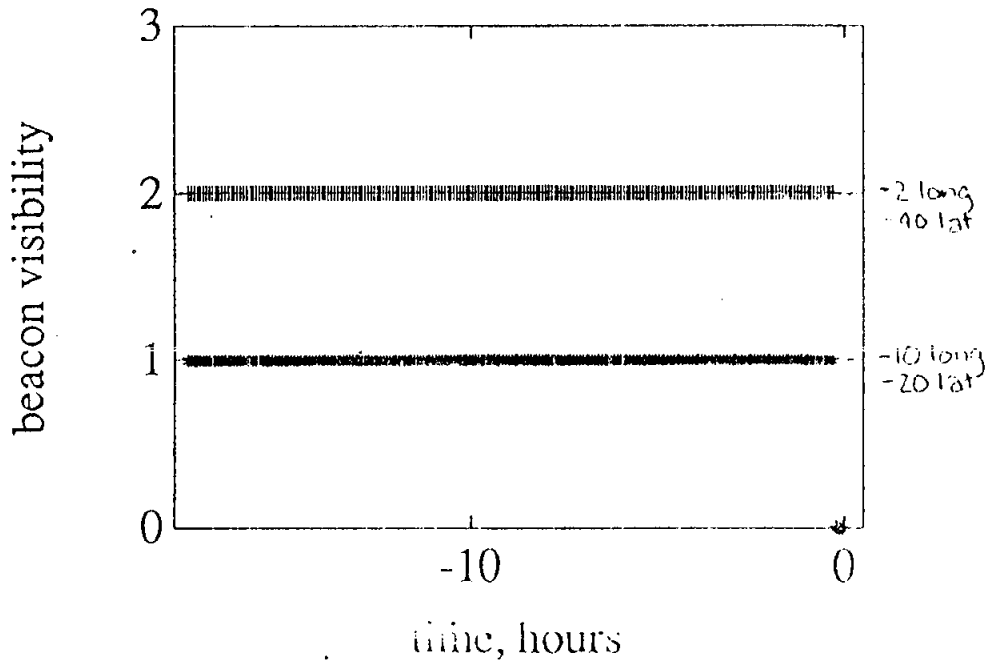



VISIBILITY FOR APPROACH Vinf OF 2 LAT, 87 LONG
latitude, deg
longitude, deg
80
60
40
20
0
-20
-40
-60
-80
-80
-60
-40
-20
0
20
40
60
80
90
91
92
93
94
95
96
97
98
99
80
70
60
50
40
20
10
5

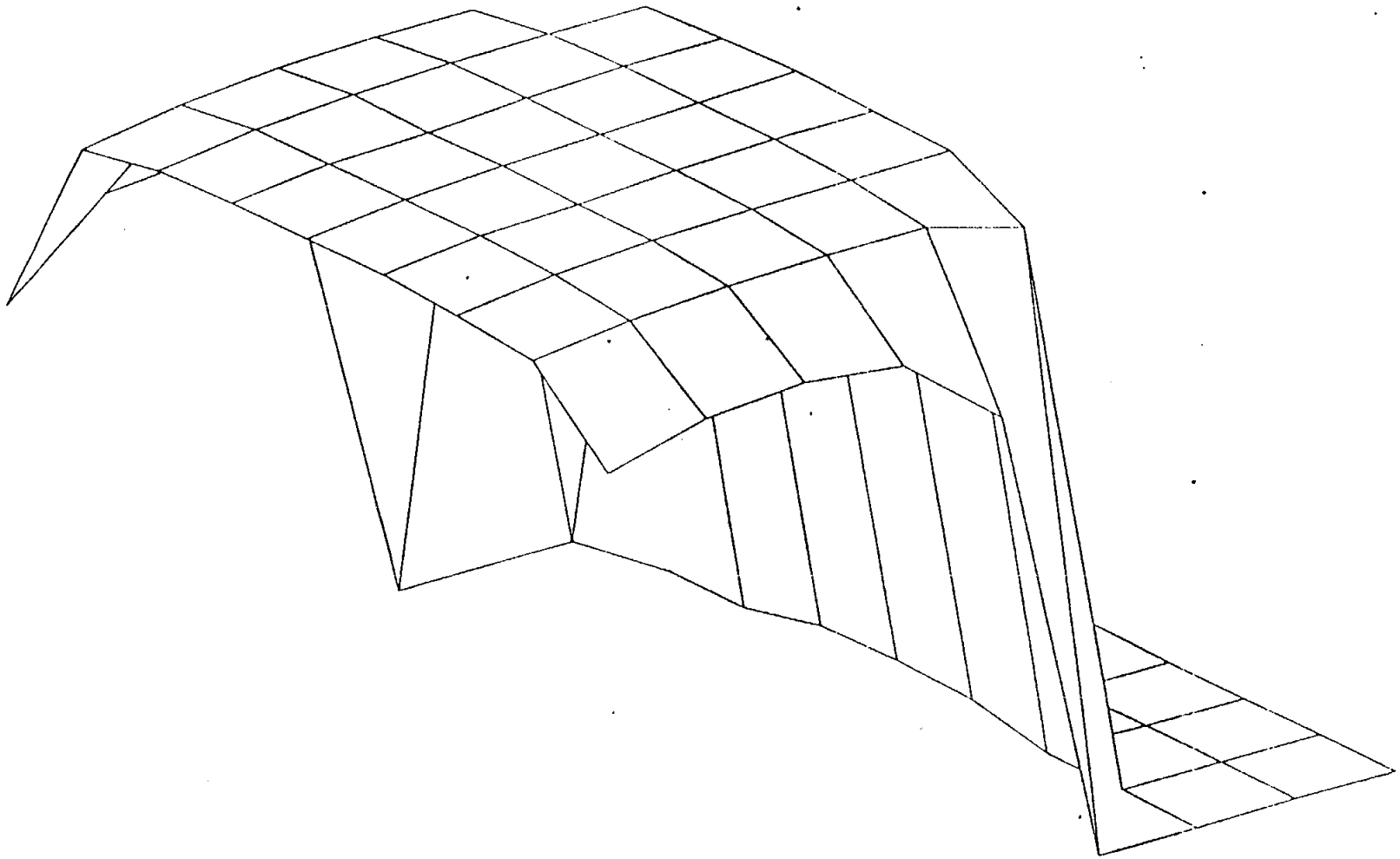

x10^4
1.5
1
0.5
pos errors, m
-10
0
time, hours


x10^7
2
0
-2
-4
-6
y pos, m
0
1
2
3
x pos, m
x10^7

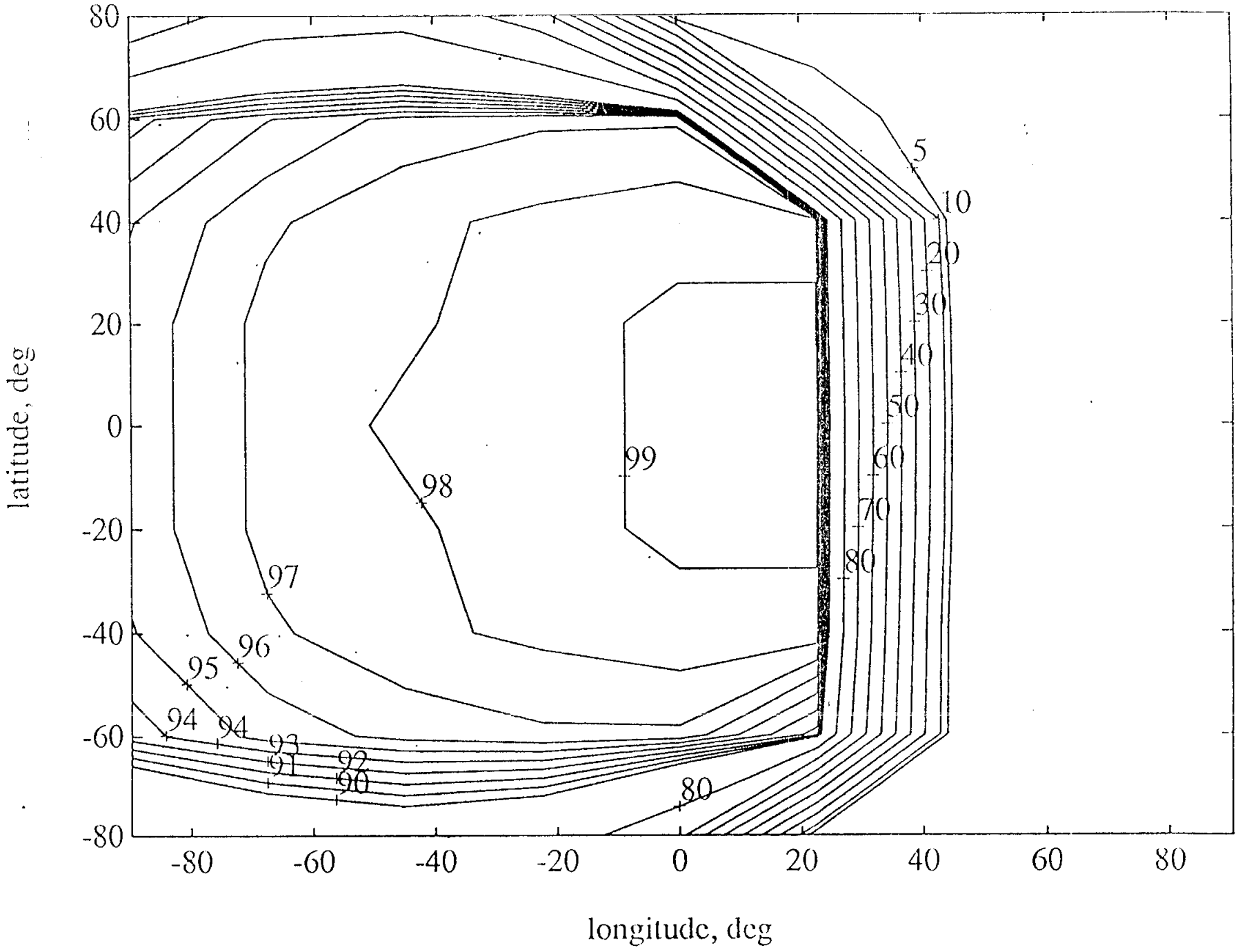

0.5
0
-0.5
vel errors, m/s
-10
0
time, hours


3
2
1
0
beacon visibility
-10
0
time, hours
-2 long
-90 lat
-10 long
-20 lat

Normal Use Mesh

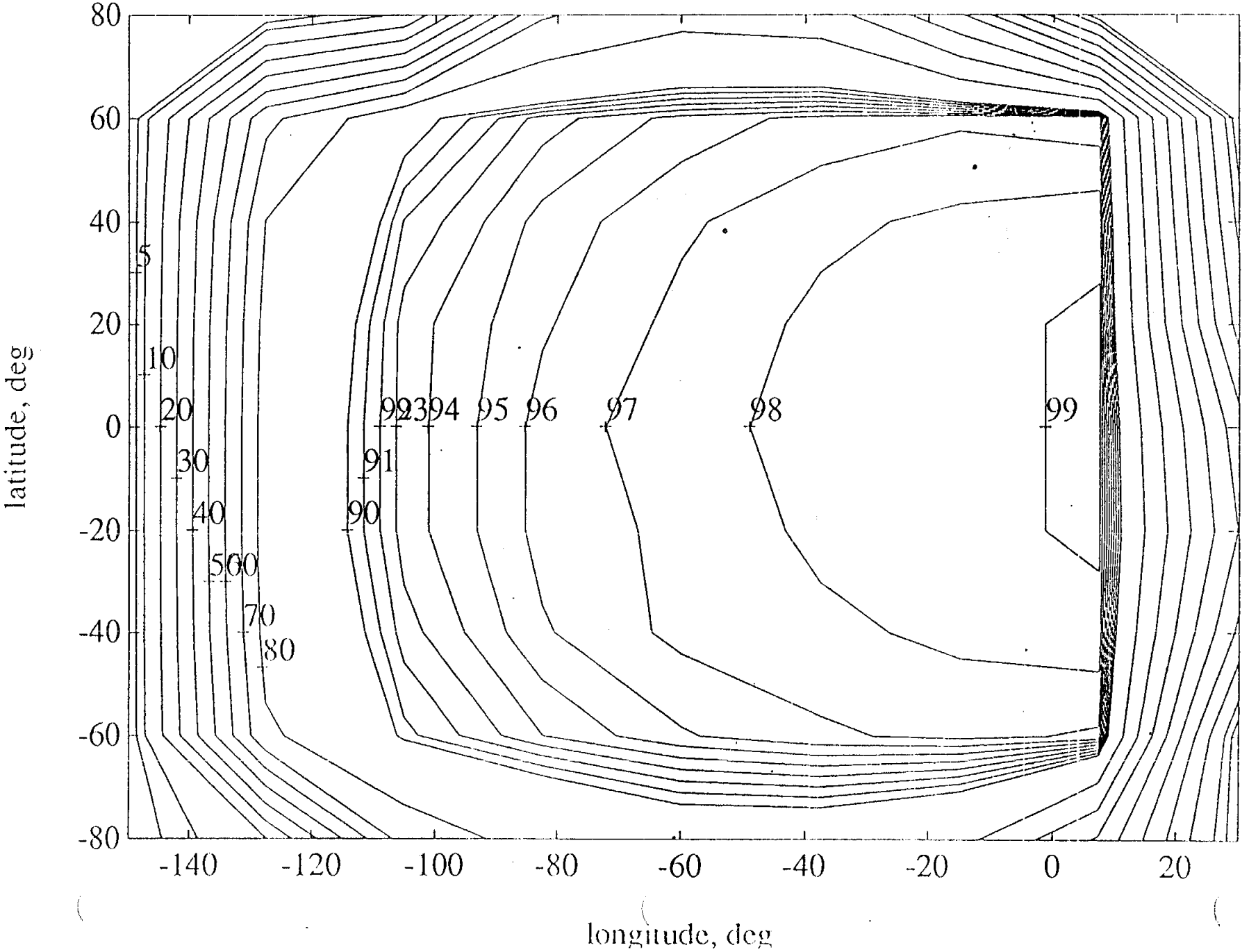


VISIBILITY FOR APPROACH Vinf OF 1.8 LAT, 111.3 LONG (Nominal)

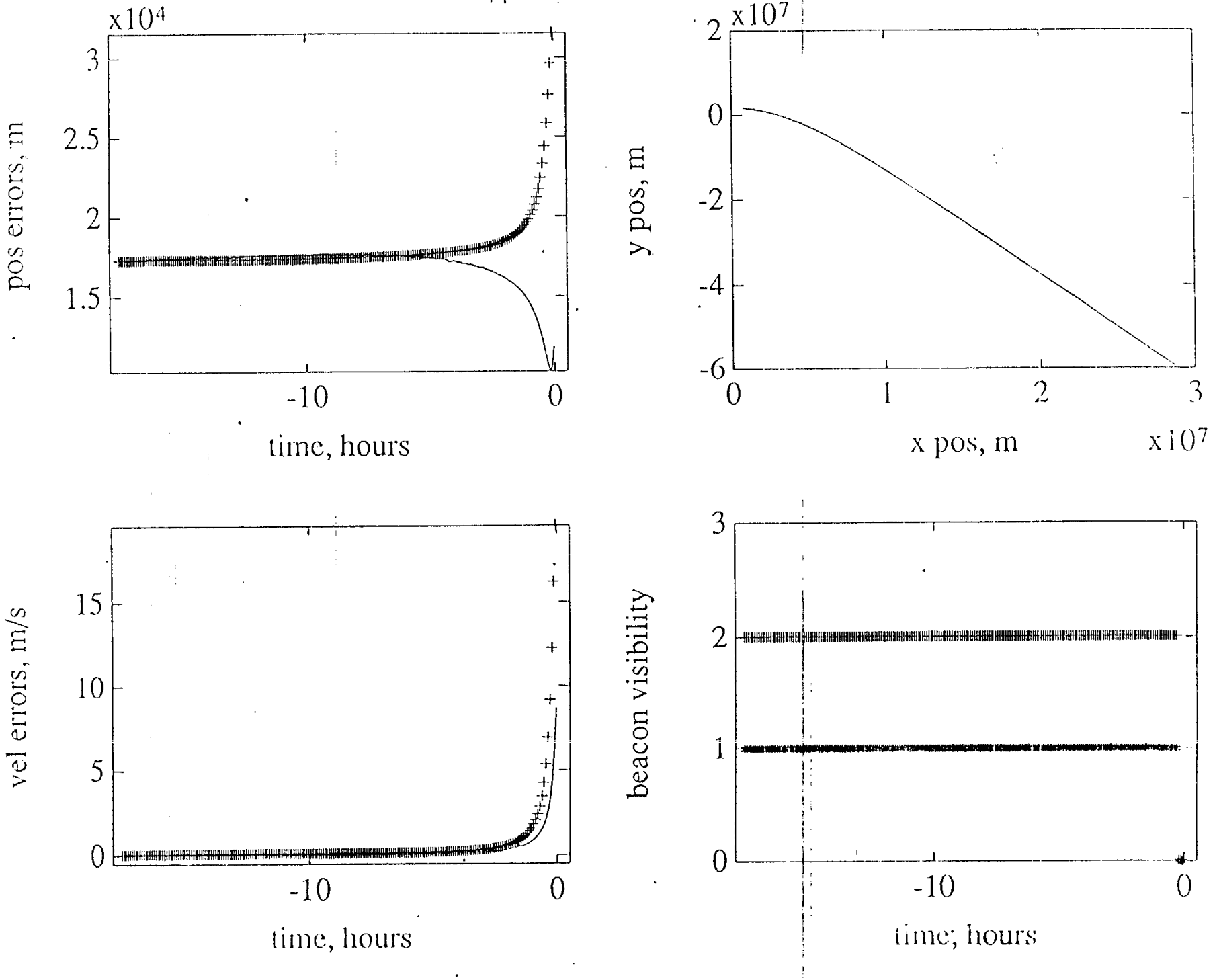


(Draft)

VISIBILITY FOR APPROACH Vinf OF 1.8 LAT, 111.3 LONG (Nominal)



Nominal Approach No measurements



SET #1

The following plots are for the hyperbolic approach problem.
They include, in order:

(1) the contour plots, 3-D mesh pictures, and navigation plots for the nominal and extreme cases mentioned (2 beacons each),

(2) the nominal and worst case navigation plots for 3 beacons, with beacon #3 at -20 long, -25 lat

(3) the nominal and worst case navigation plots for 3 beacons, with beacon #3 at 0 long, -55 lat

APPENDIX

LUNAR SURFACE BEACON NAVIGATION STUDY

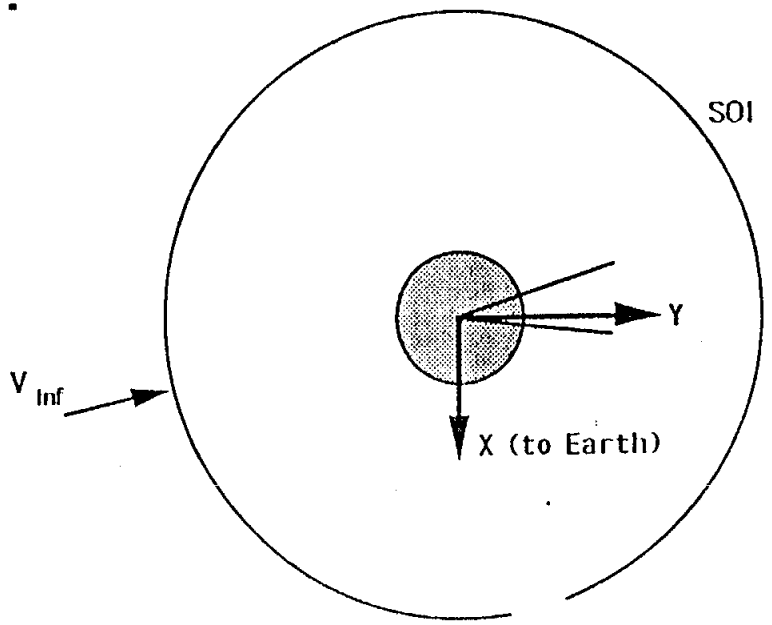
Summary Of Results

Michelle M. Munk
February 20, 1992

HYPERBOLIC LUNAR APPROACH

* GOAL: To assess the utility of Lunar surface beacons for navigation during hyperbolic approach to the Moon.
  - How many beacons are necessary?
  - Can they be placed such that they will be effective for all forseeable geometric extremes?

GEOMETRY:



HYPERBOLIC LUNAR APPROACH

ASSUMPTIONS:

- Landing Site at 1.7 deg latitude, 85.8 deg latitude

- Considered posigrade approaches only
- Nominal Vinf = 1.8 deg latitude, 111.3 deg longitude
  Yields parking orbit with i = 1.8 deg, asc node = 13.8 deg

- Allowed Vinf to vary:
  Latitude: 2 deg to 45 deg
  Longitude: 87 deg to 115 deg

  Yielded parking orbits with i = 2 deg to 89.8 deg
  asc node = 28.8 deg to 88.6 deg

- Considered 7 cases: Nominal Vinf, and combinations of 2 deg, 30 deg, and 45 deg latitude with 87 and 115 deg longitude

HYPERBOLIC LUNAR APPROACH

- The landing site cannot be "seen" during the approach since it is on the opposite side of the Moon. (Things are "visible" when they are at least 3 degrees above the horizon.)

- Determined that more than one beacon was necessary for good nav

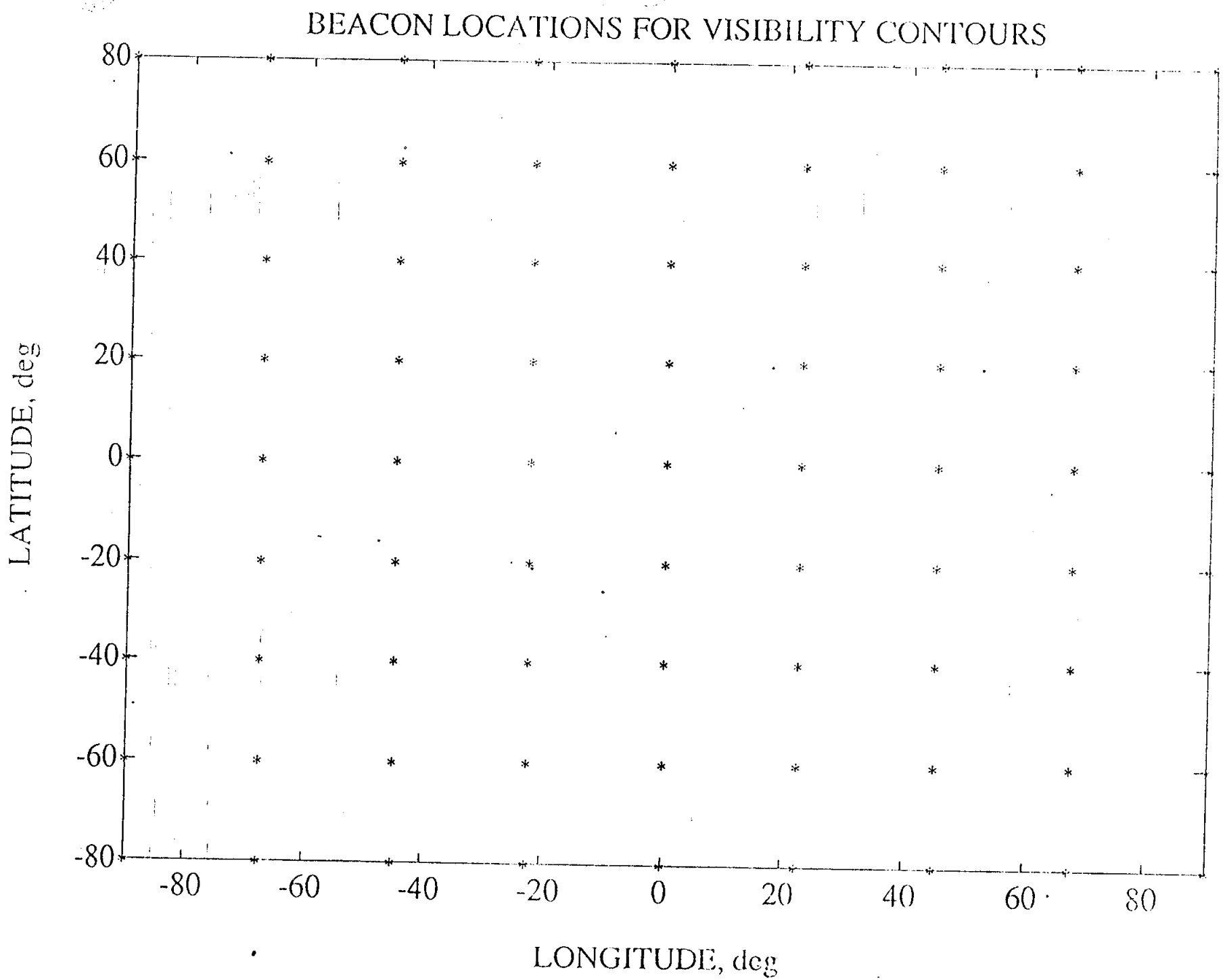
DIFFICULT to know where to put beacons for optimal navigation:

- Constructed a grid of 81 beacons on the front side of the Moon
- Computed the visibility of each point (% of time from SOI to perilune that it could be seen)
- CONTOUR PLOTS made for each of the 7 approach cases; when overlapped, show the best regions in which to place beacons

- Placed 2 beacons at -10 deg long, -20 deg lat, and -2 deg long, -40 deg lat
  - All cases show relatively good nav (results follow)

- Added third beacon at -20 deg long, -25 deg lat
  - Nominal and "worst" cases redone with some improvement (see following)

- Moved third beacon to 0 deg long, -55 deg lat
  - Nominal and "worst" cases redone with more improvement (see following)


BEACON LOCATIONS FOR VISIBILITY CONTOURS
LATITUDE, deg
LONGITUDE, deg
80
60
40
20
0
-20
-40
-60
-80
-80
-60
-40
-20
0
20
40
60
80

The following filter values are for 2 beacons at -10° long, -20° lat and -2° long, -40° lat. They were recorded at perilune and 1 hour prior.

Approach Vinf lat, long	Filter Position Error (km) -1 hour	per.	Filter Velocity Error (m/s) -1 hour	per.
1.8° lat, 111.3° long (nom)	.455	.165	.1217	.1798
2° lat, 87° long	.660	.119	.1243	.1445
30° lat, 87° long	2.394	.381	.1630	.3167
45° lat, 87° long	1.807	.536	.1395	.4458
2° lat, 115° long	.450	.184	.1220	.1941
30° lat, 115° long	.496	.194	.1214	.2209
45° lat, 115° long	.674	.252	.1218	.2604

The nominal and "worst" cases were run with 3 beacons, the third at -20° long, -25° lat. The results for the same times were:

1.8° lat, 111.3° long (nom)	.384	.155	.1010	.1560
45° lat, 87° long	1.277	.475	.1205	.3966

The final values were improved by putting the third beacon in a better location: 0° long, -55° lat. The results were:

1.8° lat, 111.3° long (nom)	.377	.142	.1060	.1542
45° lat, 87° long	.762	.235	.1084	.2226

APPROACH STUDY CONCLUSIONS

- At least two beacons must be available simultaneously for acceptable navigation.
- Beacons should be placed so that they are visible for a large (>90) percentage of the approach time.
  - Determination of the optimum placement is facilitated by contour plots of surface visibility
- Adding a third beacon to two "optimal" ones helps marginally
- The beacons generally need to be separated by some latitudinal distance to be effective

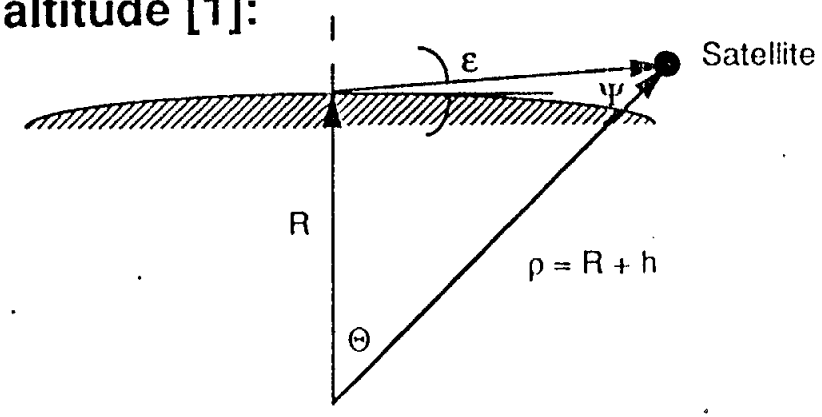
CIRCULAR LUNAR ORBIT

*GOAL: To assess the utility of Lunar surface beacons in the navigation of a vehicle in a circular orbit about the Moon.

- How many beacons must be visible simultaneously?
- Where should they be placed, based on the geometry (alt, i) of the parking orbit?

APPROACH:

- We can analytically compute the angle of visibility of a beacon from a given altitude [1]:



$$\Theta = -\varepsilon + \mathrm{acos}\left[\left(\frac{R}{(R+h)}\right)\sin(\varepsilon + pi/2)\right] \qquad (\varepsilon = 3\ \mathrm{deg})$$

[1] Material developed by David Hoffman, ET4, 17-JAN-1992.

CIRCULAR LUNAR ORBIT

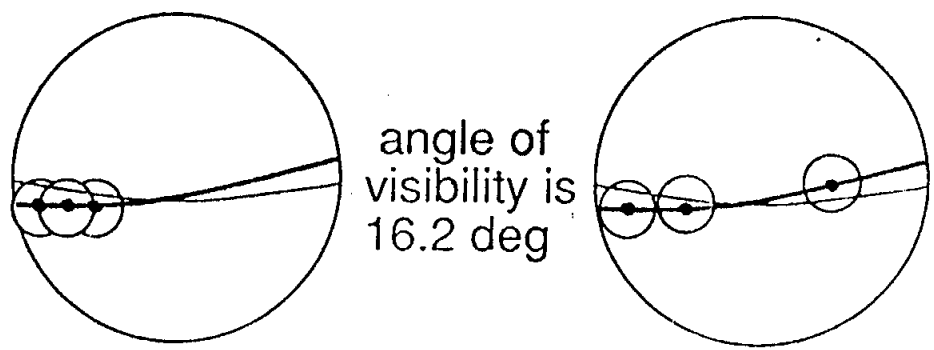
APPROACH (cont'd):

- For 100 km, 300 km, and 500 km altitudes, the visibility angles are 16.2, 28.6, and 36 deg, respectively. (Using the analytics on the previous page.)

RUNS AND RESULTS, PART 1:

- How many beacons must be visible simultaneously for acceptable nav?
- How much do the initial filter errors affect the results?

  - Used the nominal parking orbit: alt = 100 km, i = 1.8 deg, node = 13.8 deg
  - Tried 2 beacon placements:

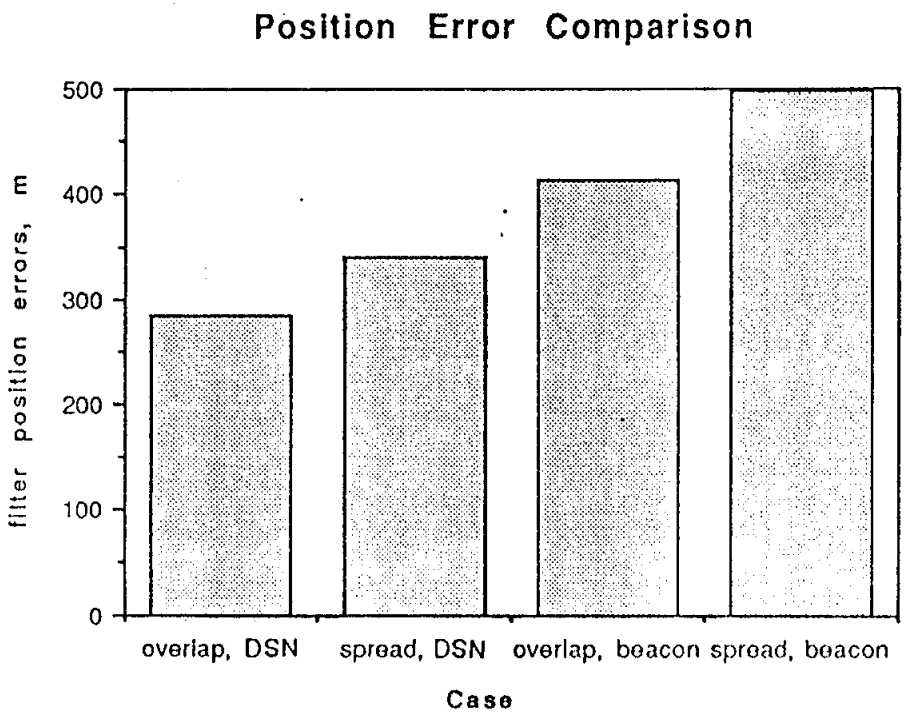


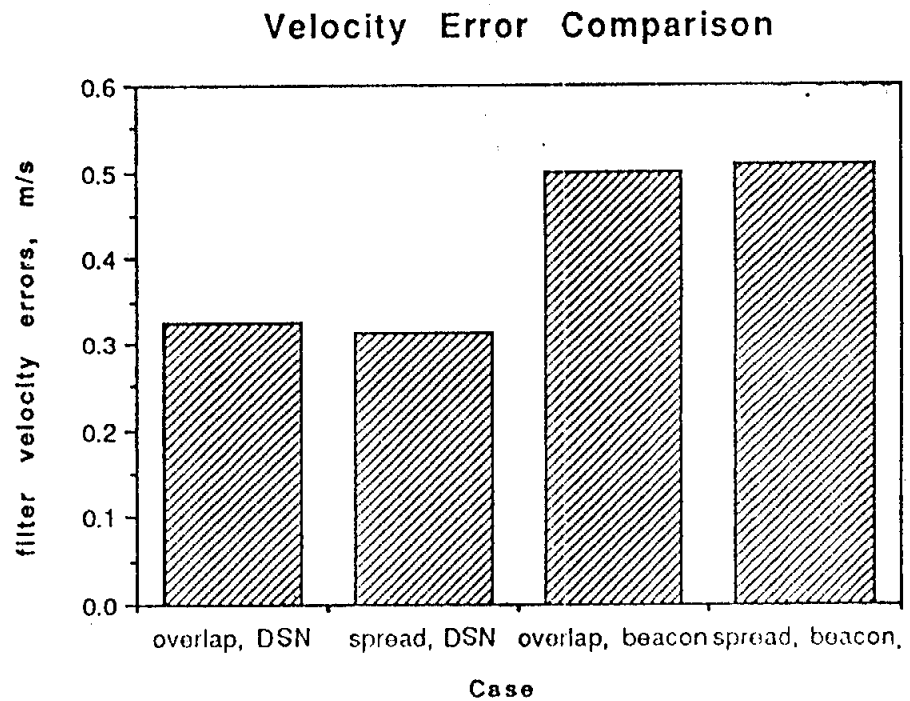
see two simultaneously see only one at a time

CIRCULAR LUNAR ORBIT

- Each case was initialized with either:
  (1) Deep Space Network (DSN) initial errors (100 m, .01 m/s), or
  (2) Worst beacon errors from approach cases (536 m, .45 m/s)

- RESULTS:

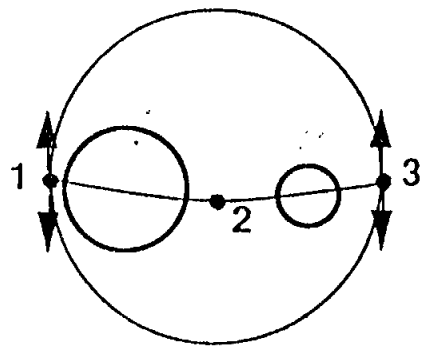




CIRCULAR LUNAR ORBIT

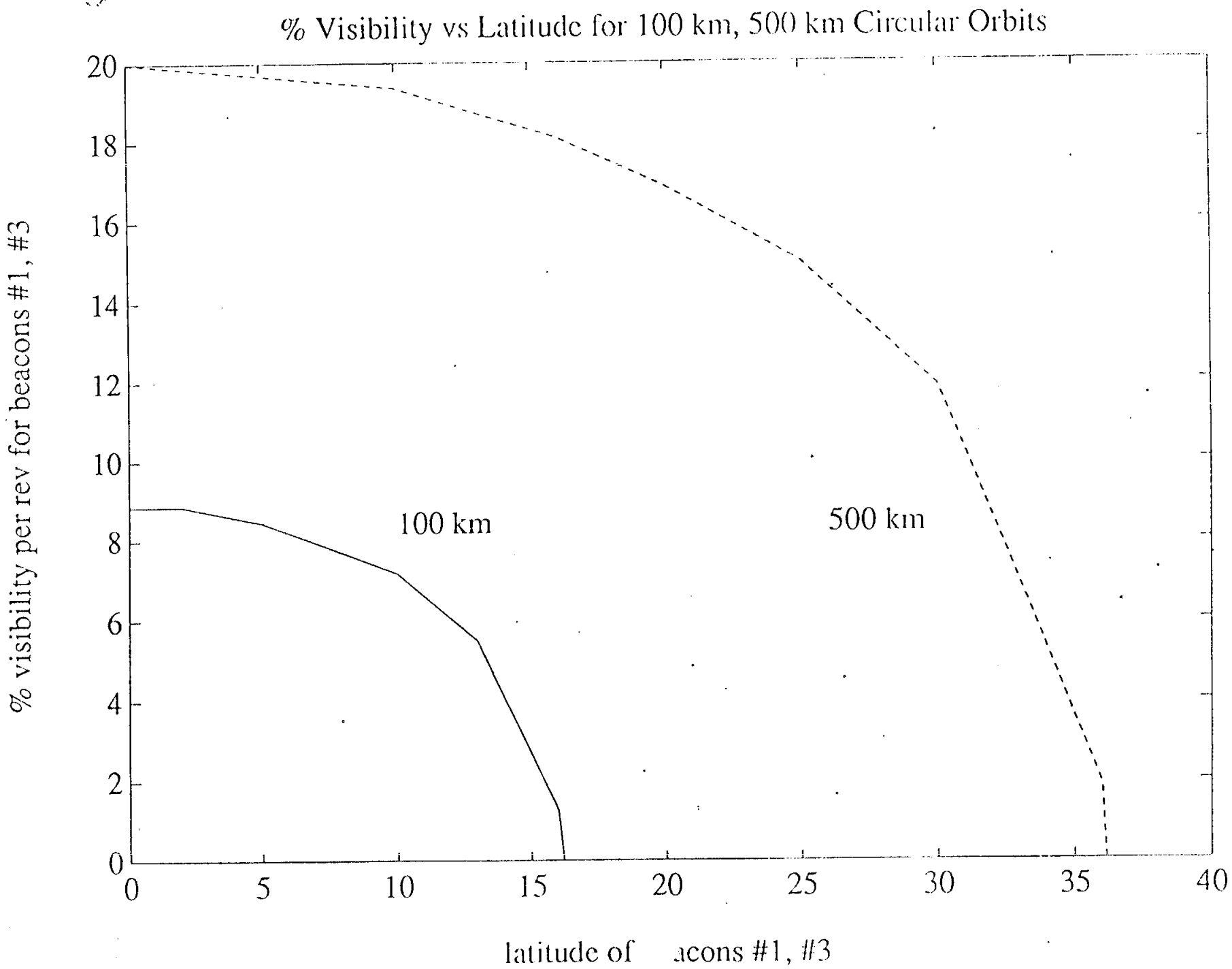
RUNS AND RESULTS, PART 2:

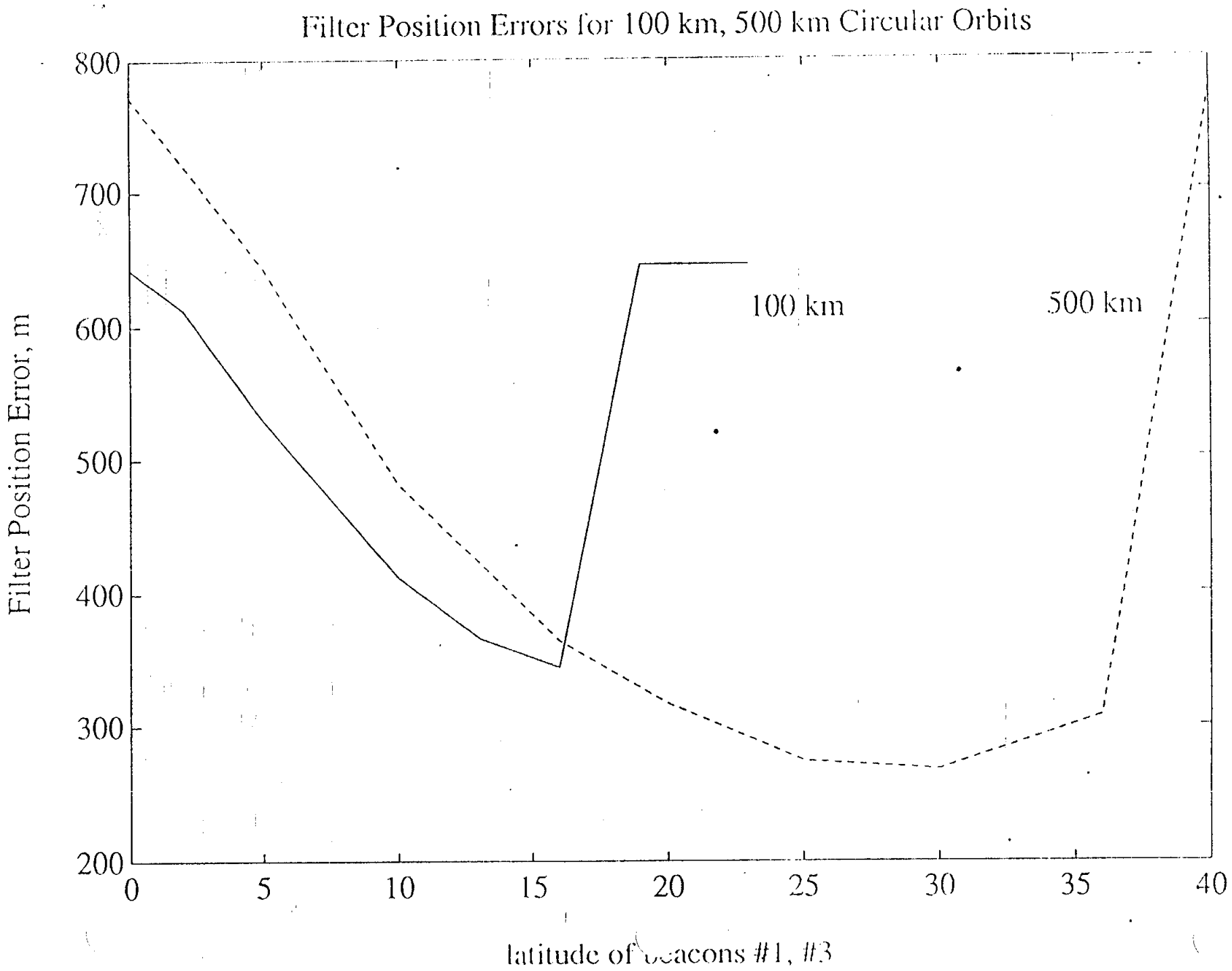
- Where should beacons be placed for a given orbit altitude and inclination?
  - Used 100 km and 500 km equatorial orbits with beacon initial errors
  - Spread beacons longitudinally (-90 deg, 0 deg, 90 deg) to receive measurements from as close to the dark side as possible.
  - Varied latitude of beacons at -90 deg and 90 deg (#1 and #3)
  - Expect errors to increase as latitude increases (beacon is visible for a shorter time)

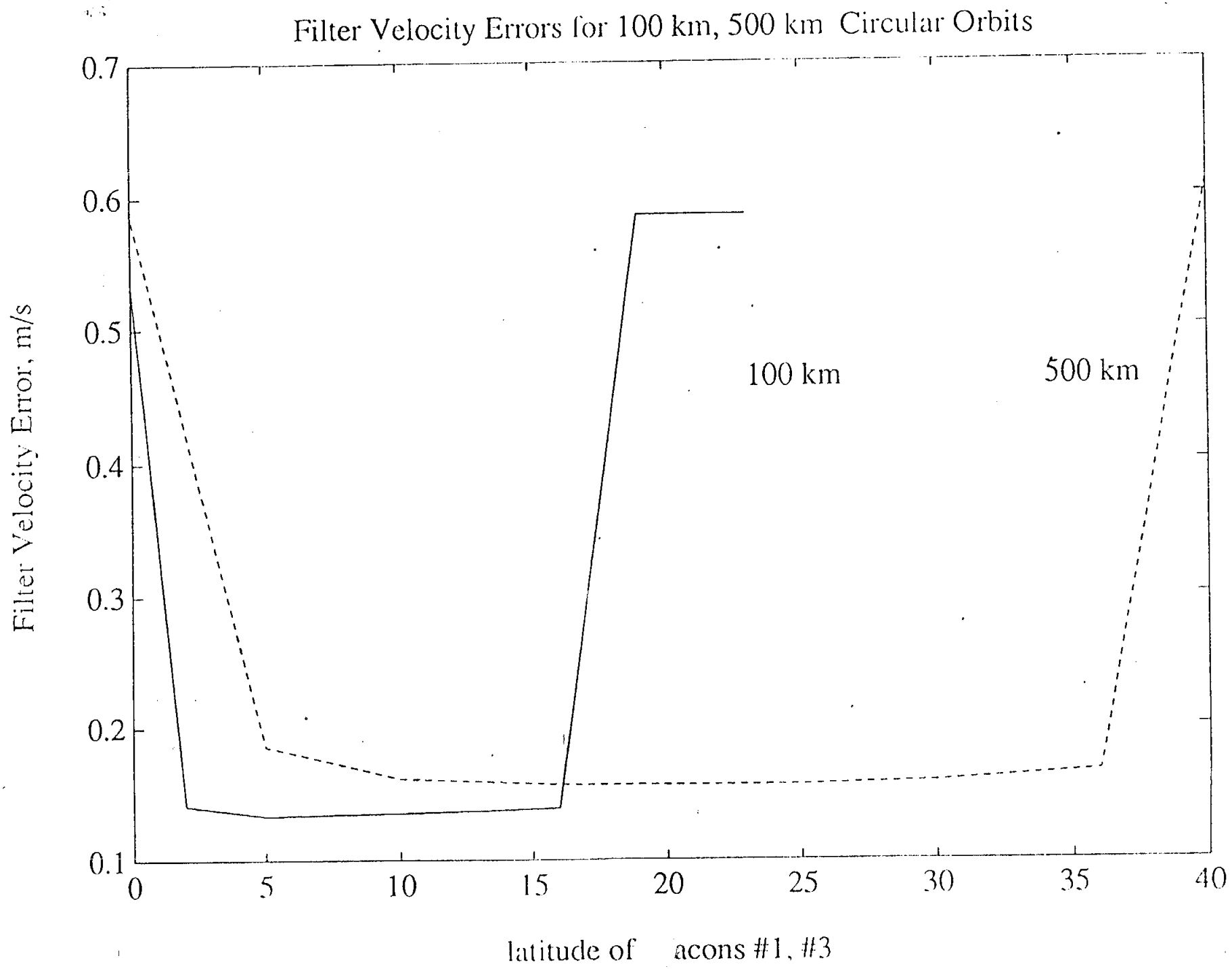


RESULTS:

- Errors actually decrease as visibility decreases because you are receiving measurements from further out-of-plane


% Visibility vs Latitude for 100 km, 500 km Circular Orbits
% visibility per rev for beacons #1, #3
latitude of acons #1, #3
100 km
500 km
0
2
4
6
8
10
12
14
16
18
20
0
5
10
15
20
25
30
35
40


Filter Position Errors for 100 km, 500 km Circular Orbits
Filter Position Error, m
800
700
600
500
400
300
200
0
5
10
15
20
25
30
35
40
100 km
500 km
latitude of beacons #1, #3


Filter Velocity Errors for 100 km, 500 km Circular Orbits
Filter Velocity Error, m/s
0.7
0.6
0.5
0.4
0.3
0.2
0.1
0
5
10
15
20
25
30
35
40
100 km
500 km
latitude of acons #1, #3

CIRCULAR ORBIT CONCLUSIONS

CONCLUSIONS FROM PART 1:

- There is no advantage to seeing more than one beacon at a time.
- Initial errors have little effect on results.

CONCLUSIONS FROM PART 2:

- Regardless of orbit inclination, beacons should be placed as far out of the orbital plane as possible, while still allowing minimal visibility.
- The orbital altitude will determine the distance out-of-plane to be used.

LOCATING A BACKSIDE BEACON

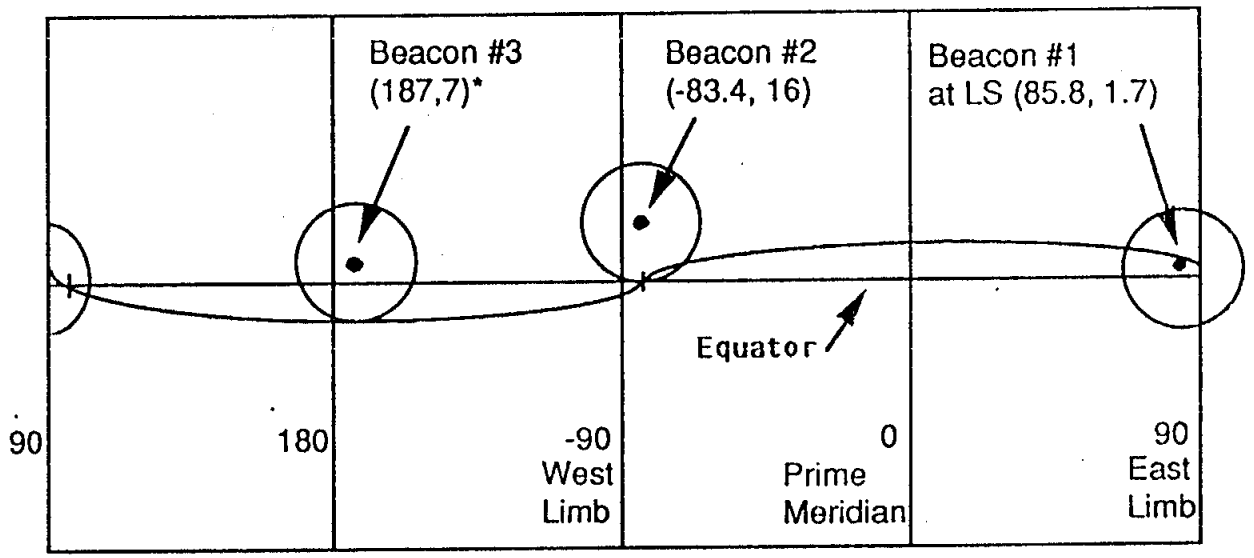
*GOAL: To determine if the position of a beacon on the back side of the moon can be determined by an orbiting vehicle, with the help of two beacons on the front side.

ASSUMPTIONS:

- Initial beacon errors for front side: 50 m radially, 50 m north, 50 m east
- Initial beacon errors for back side: 500 m radially, 1500 m north and east
- Cargo Mission, with 100 km circular parking orbit: i = 9 deg
  asc node = -83.4 deg
- Beacon #1 at Landing Site (85.8 deg long, 1.7 deg lat)
- From circular orbit conclusions, should put beacons as far out of the orbital plane as possible, still allowing visibility (see following figure)

LOCATING A BACKSIDE BEACON

PROBLEM GEOMETRY



This geometry was used for a first run of the problem. For the second run, Beacon #1 was moved from the LS to further out of the orbit plane. The results with both DSN and Beacon initial errors follow.

LOCATING A BACKSIDE BEACON

RESULTS THUS FAR:

Case	B3 sigr	B3 pos	Vehicle Pos. Errors min	Vehicle Pos. Errors end of run	Vehicle Vel. Errors min	Vehicle Vel. Errors end of run
B3 at LS						
DNS	233	255	163	838	.224	.8375
Beacon	420	372	171	854	.265	.9074
B3 out of plane						
DNS	259	188	166	1100	.271	1.084
Beacon	363	215	223	1104	.344	1.109

(Errors in meters and m/s.)

CONCLUSIONS

- Study results represent a quick look and are preliminary.
- Stable high altitude lunar orbits exist (for both 5000 km and 10,000 km), i.e., lifetimes exceed 5 years without maneuvers.
- 10,000 km orbits provide larger coverage areas for communication and navigation support
- Stable orbits exist for inclinations of 0, 30, 150, and 180 degrees at both altitudes.
- Orbits are unstable at inclinations of 60, 90, and 120 degrees at both altitudes.
- 10,000 km orbits are more unstable than 5,000 km orbits.
- Unstable orbits can be maintained with periodic corrections to the orbit eccentricity.
- 10,000 km unstable orbits require more frequent burns resulting in More ΔV over mission lifetime.

HABITAT LANDER DEORBIT TO LANDING NAVIGATION STUDY

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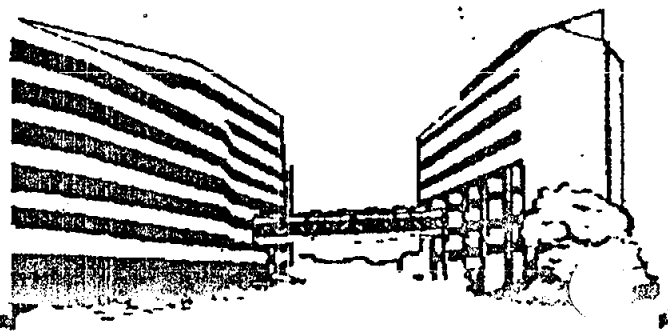
MARCH 17, 1992

ESB-92-149
LMI-92-013

THE
CHARLES STARK DRAPER
LABORATORY, INC.



STUDY OVERVIEW

- OBJECTIVE
  - — Define Habitat Lander navigation sensor and initial condition requirements for landing
    - IMU
    - IMU alignment (star tracker)
    - Radars (altitude, Doppler)
    - Initial state vector accuracy
    - Possible Navsat or uprange Navsite navaid(s)

- APPROACH
  - — Linear covariance analysis
    - Statistical measure of navigation performance on reference trajectory
    - Does not evaluate G&N interaction, particularly the timeliness of nav updates

- Trajectory obtained from JSC (Condon and Bryant)
  - — Equatorial approach to landing site on the equator
  - — Hohmann transfer (180 deg) down to 18 km altitude
  - — Powered descent starts a perilune

INTRODUCTORY COMMENTS

- This is preliminary data; models and assumptions must be reviewed carefully

- Key questions
  - (1) Do we need Doppler (ground speed) radar?
  - (2) When do we need altimeter update (will single-position radar pointing be sufficient)?
  - (3) Is good terrain information necessary (or what is our sensitivity to altitude uncertainty under the ground track and at landing)?
  - (4) What navigation performance is possible?

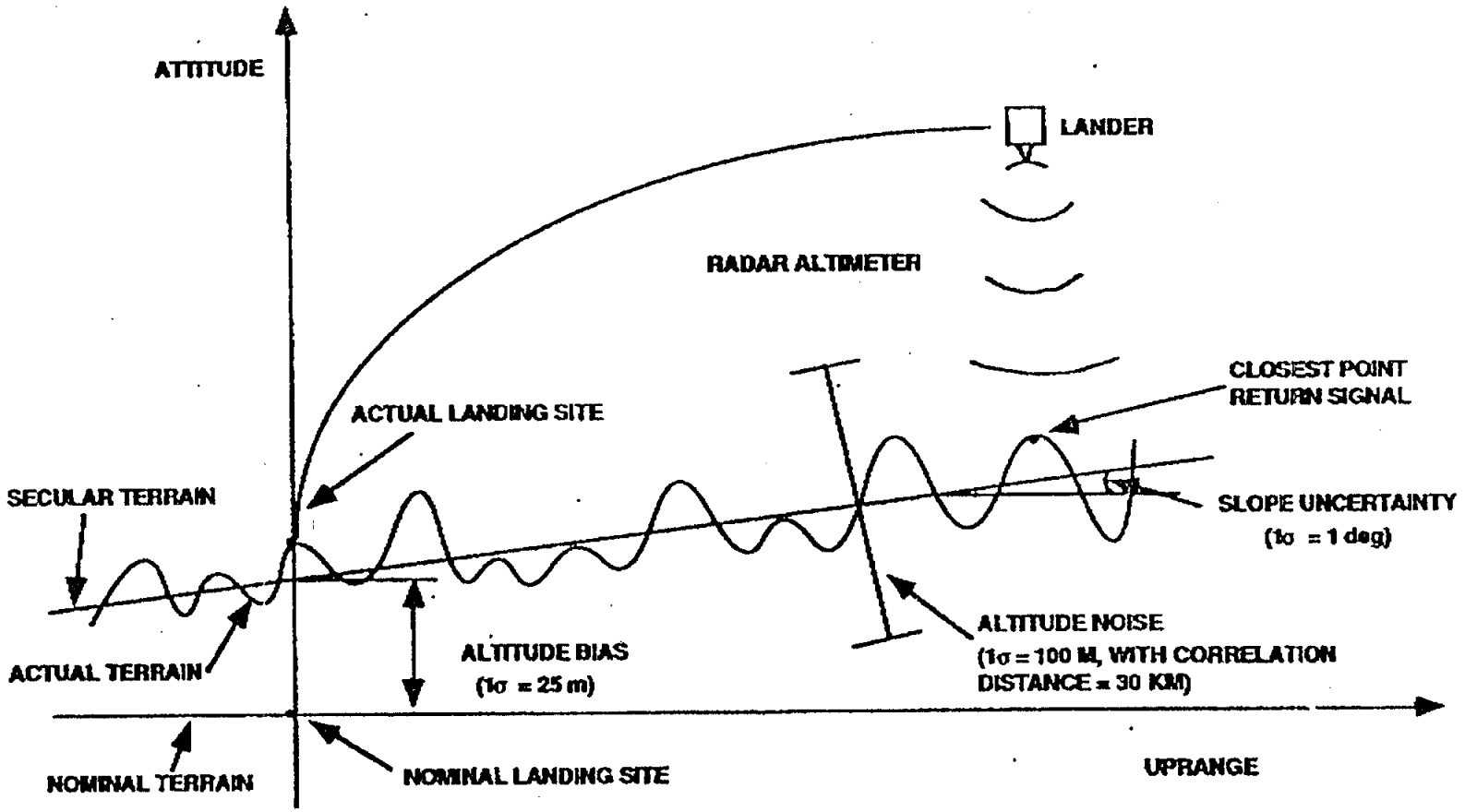
BASELINE SCENARIO ASSUMPTION

- Habitat in 100 km equatorial orbit
- Essentially perfect (< 100 m) DSN state vector update 1 rev prior to deorbit
- Error growth during 1 rev coast prior to deorbit (due to gravity)
  - — Model gravity uncertainty as process noise
  - — Baseline "sized" for 2 km 3σ downtrack error growth/rev
  - — Crosstrack growth approximately 40% of downrange
  - — LVLH process noise values used:
    $3.18\ 10^{-7}$, $3.18\ 10^{-7}$, $1.51\ 10^{-5}$ m^2/s^3 (DR, VER, CT)
  - — Growth: 667, 136, 260 m (DR, VER, CT) (1σ)
- Deorbit burn (ΔV = 18.6 m/s)
  - — IMU alignment accuracy = 10 arcsec
  - — Small burn has only minor impact on nav

BASELINE SCENARIO ASSUMPTION (cont'd)

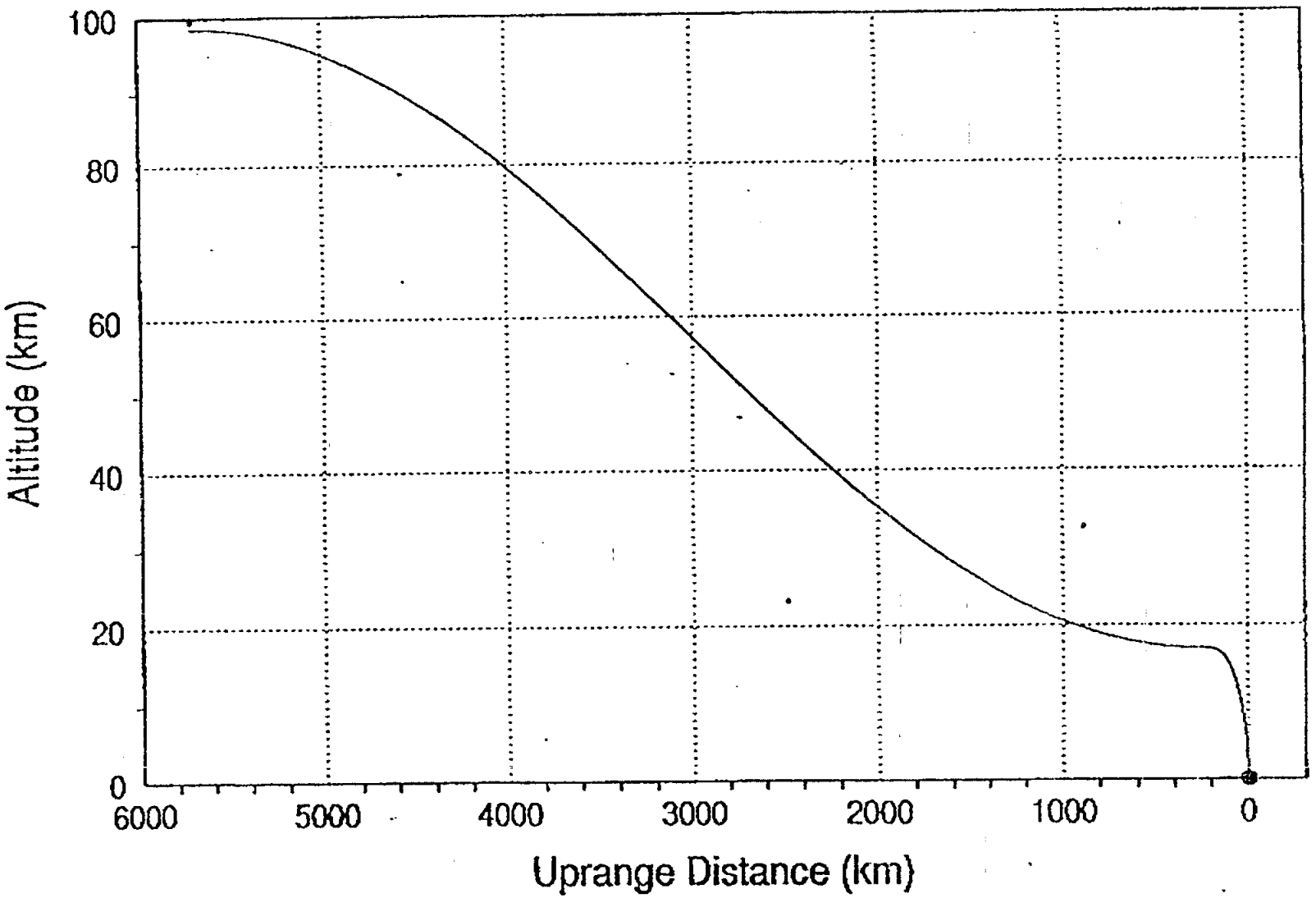
- Landing area terrain model (see next chart)
  - — Local terrain (within 5 km radius) assumed not well known
  - — Site altitude uncertainty = 25 m (may be optimistic)
  - — Terrain slope uncertainty assumed = 1 deg (equivalent to ≈~~90~~ 18 m/km)
  - — Approach terrain "noise" = 100 m (Markov process with correlation distance = 30 km)
  - — Operation of radar altimeter over "bumpy" terrain or crater near/at the landing site needs more thought (add spacecraft altitude dependence to combined terrain/radar model)

FINAL APPROACH TERRAIN MODELING



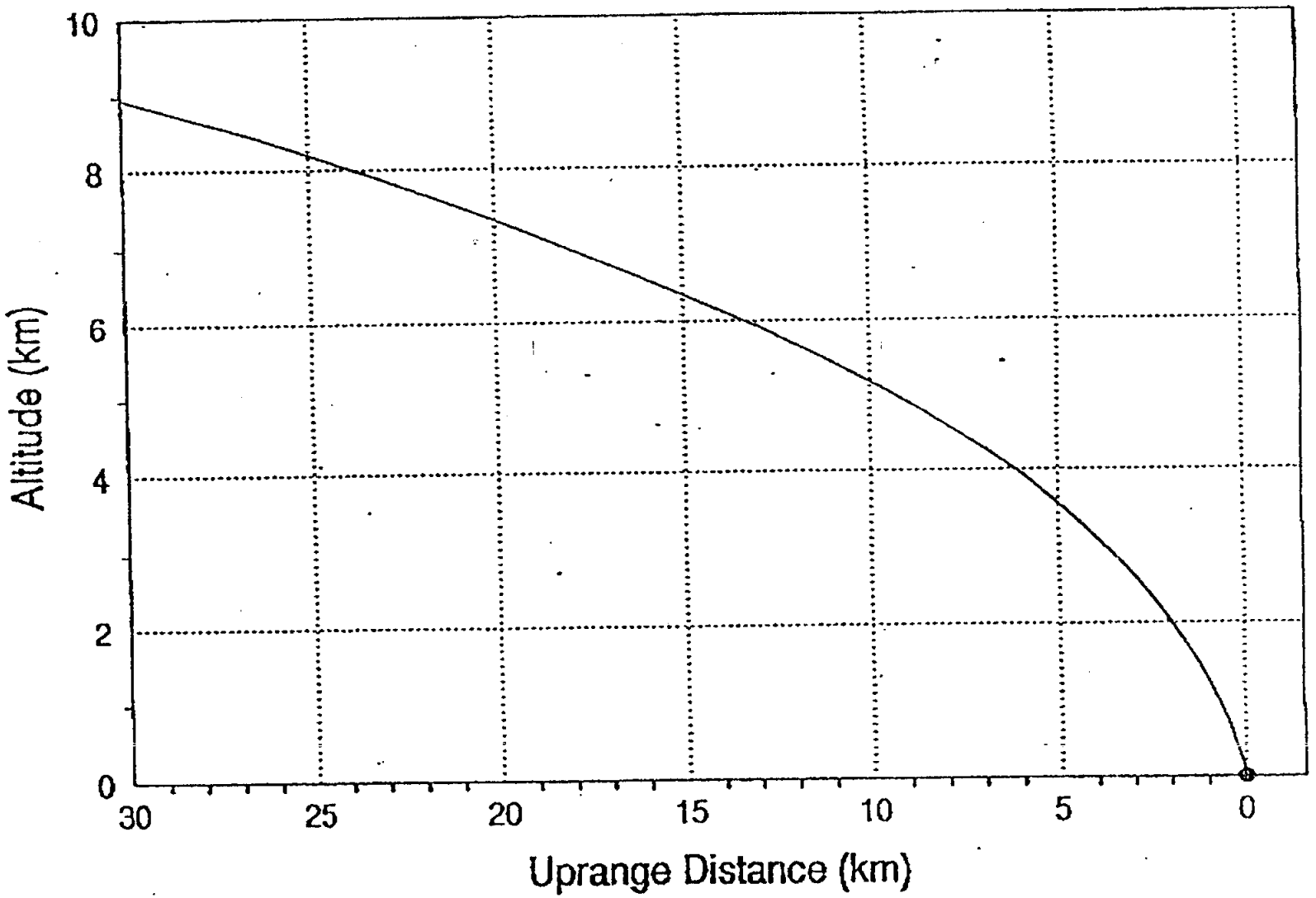
LUNAR DEORBIT TO LANDING TRAJECTORY

FULL TRAJECTORY PROFILE



LUNAR DEORBIT TO LANDING TRAJECTORY

FINAL RELATIVE APPROACH



BASELINE INSTRUMENT ACCURACY ASSUMPTIONS

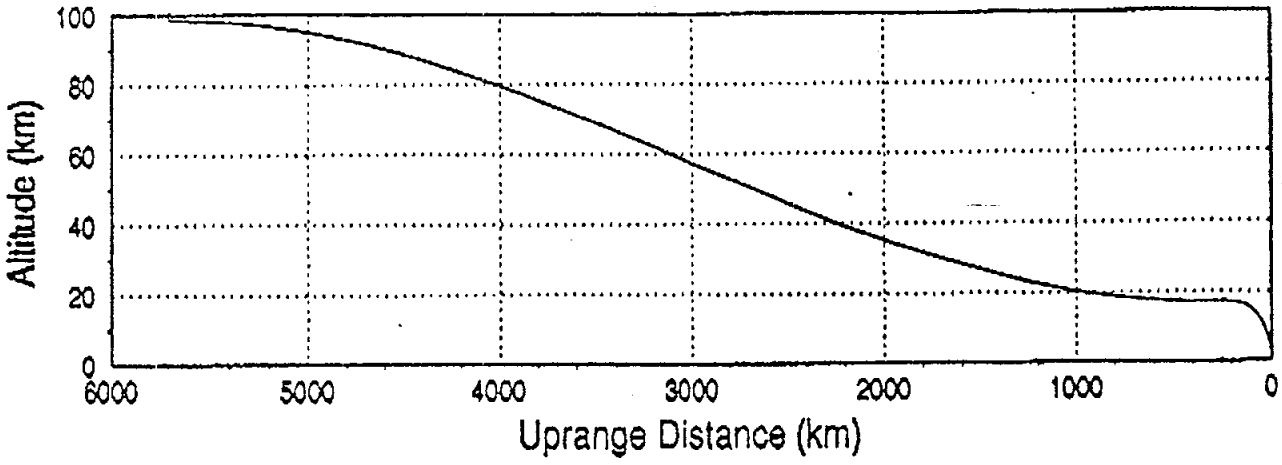
• **IMU Accuracy (1σ)**	
— Initial misalignment at ignition	20 arcsec (tracker driven)
— Misalignment drift rate*	0.015 deg/hr
— Accelerometer bias	50 μg's
— Accelerometer scale factor	0 ppm (to be added)
• **Radar altimeter accuracy** (1σ)	
— Measurement bias	0 m (to be added)
— Measurement noise	1.3 + 0.05 H m (H < 300 m) 9.0 + 0.05 H m (H > 300 m)
• **Doppler radar accuracy** (1σ)	
— Antenna misalignment	0.67 deg
— Measurement bias (τ = 30000 s)	0.30 m/s
— Measurement noise	0.30 + 0.02V m/s (V < 200 m/s) 0.30 + 0.03V m/s (V > 200 m/s)

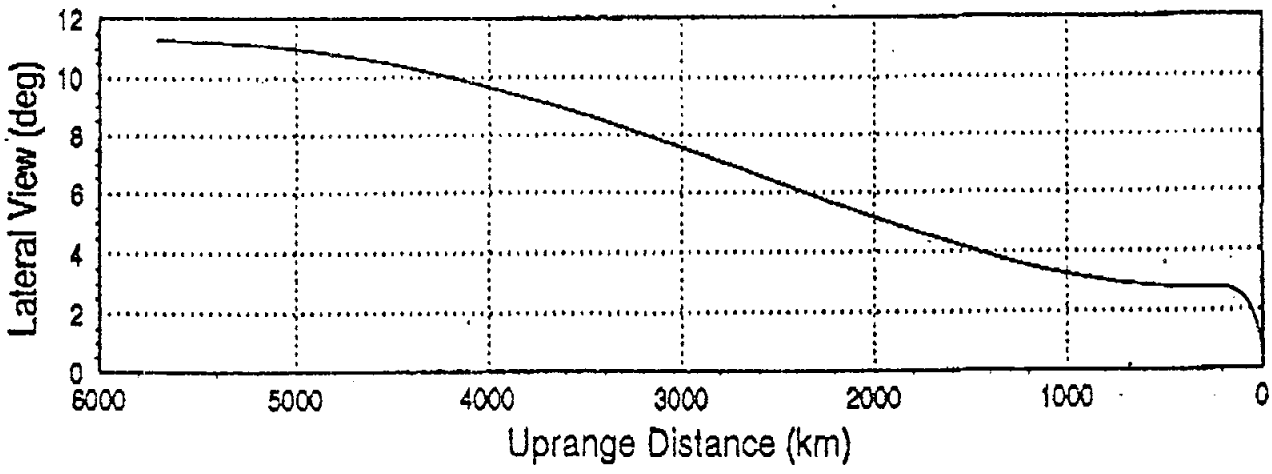
*Consider state (not estimated)
**Acquisition altitude = 10 km
radar altimeter measurement every 1 s (artificially slow)
Doppler radar measurements every 2 s (artificially slow)

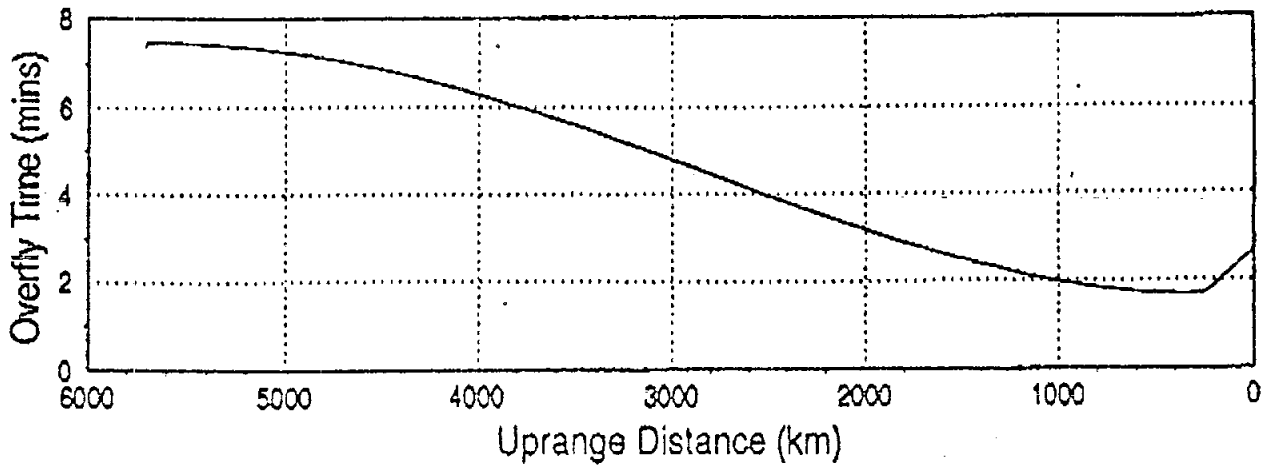
POTENTIAL ADDITIONAL NAVAIDS

- Navsat
  - — 5000 km circular orbit
  - — Located over landing site during powered descent
- Navsite
  - — Located uprange (flyover just before PDI)
  - — 10 deg elevation angle measurement requirement
  - — Yields about 5 measurements at 20 s intervals (see next chart)
- Each provides 2-way range measurements
  - — 3 m bias (due to electronic delays)
  - — 10 m/1000 km measurement accuracy (low power)

LUNAR DEORBIT TO LANDING TRAJECTORY SURFACE BEACON VISIBILITY ISSUES







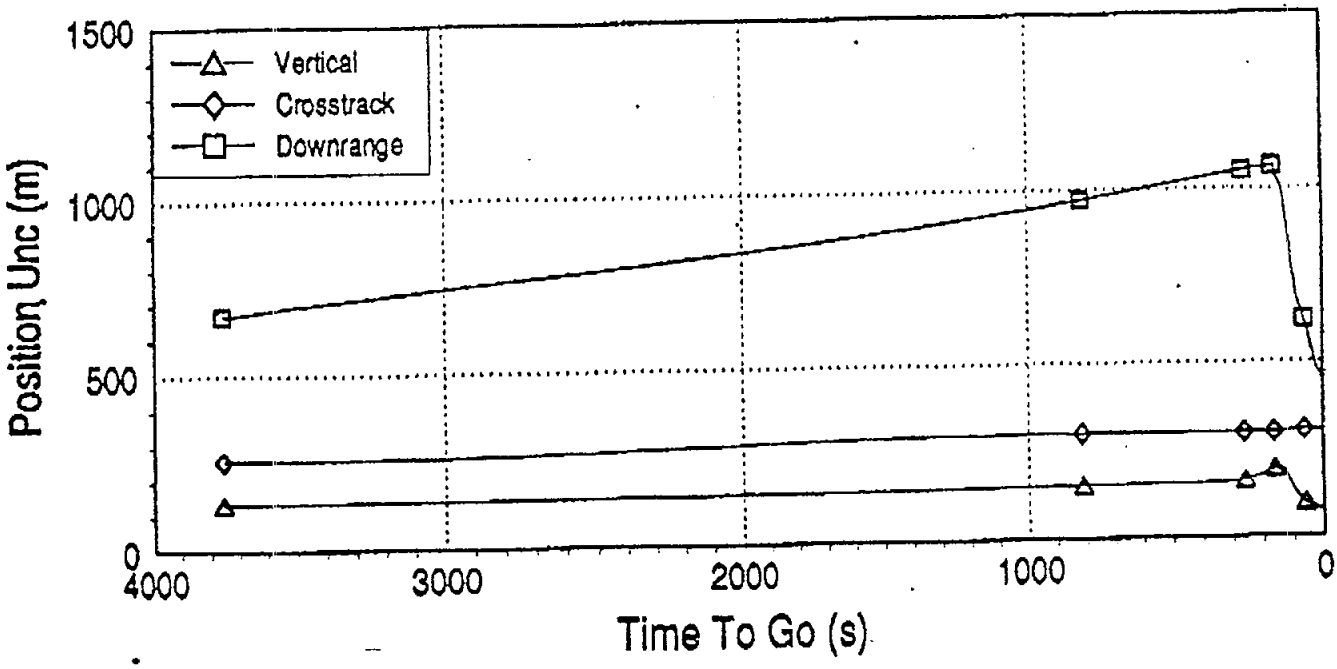
SITE RELATIVE STATE ERRORS AT LANDING (1σ)

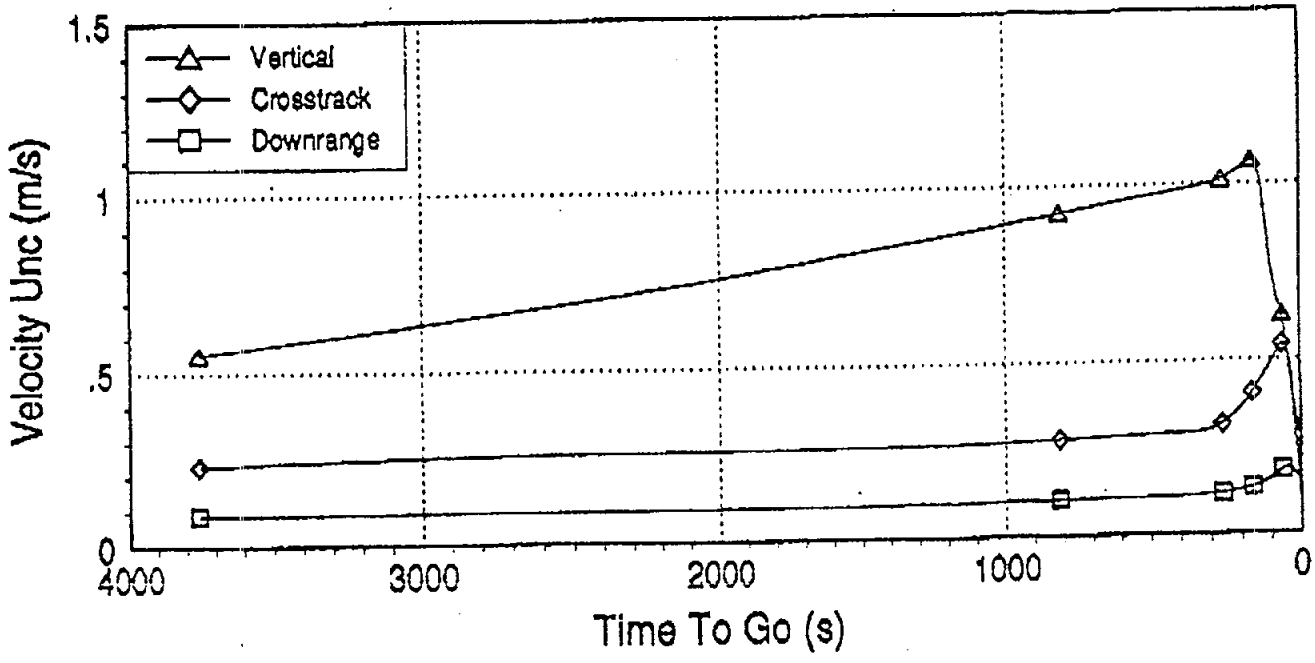
	Position Errors (m)			Velocity Errors (m/s)		
	DR	CT	ALT	DR	CT	ALT
(1) Baseline	450	307	0.60	0.07	0.28	0.08
(2) Isolate gravity effects						
(a) Perfect IMU	127	301	0.60	0.07	0.22	0.08
(b) Perfect IMU and no Doppler radar	130	301	0.60	0.08	0.30	0.08
(3) Isolate IMU effects						
(a) Perfect gravity	45	50	0.60	0.15	0.27	0.08
(b) Perfect gravity and no Doppler radar	50	90	0.60	0.17	0.50	0.08
(4) Effects of radar instruments						
(a) No Doppler radar	450	314	0.61	0.18	0.58	0.08
(b) No Doppler radar, no altimeter	1060	314	355	0.24	0.58	1.21
(5) IMU alignment sensitivity						
(a) 40 arcsec	491	308	0.60	0.16	0.29	0.08
(b) 10 arcsec	438	306	0.60	0.16	0.28	0.08
(c) Realign at PDI	260	303	0.60	0.16	0.25	0.08
(6) Initial condition sensitivity						
(a) 2 × initial state errors	529	605	0.60	0.17	0.30	0.08
(b) 1/2 × initial state errors	351	159	0.60	0.16	0.27	0.08
(c) 1/4 × initial state errors	232	91	0.60	0.15	0.27	0.08

SITE RELATIVE STATE ERRORS AT LANDING (1σ)

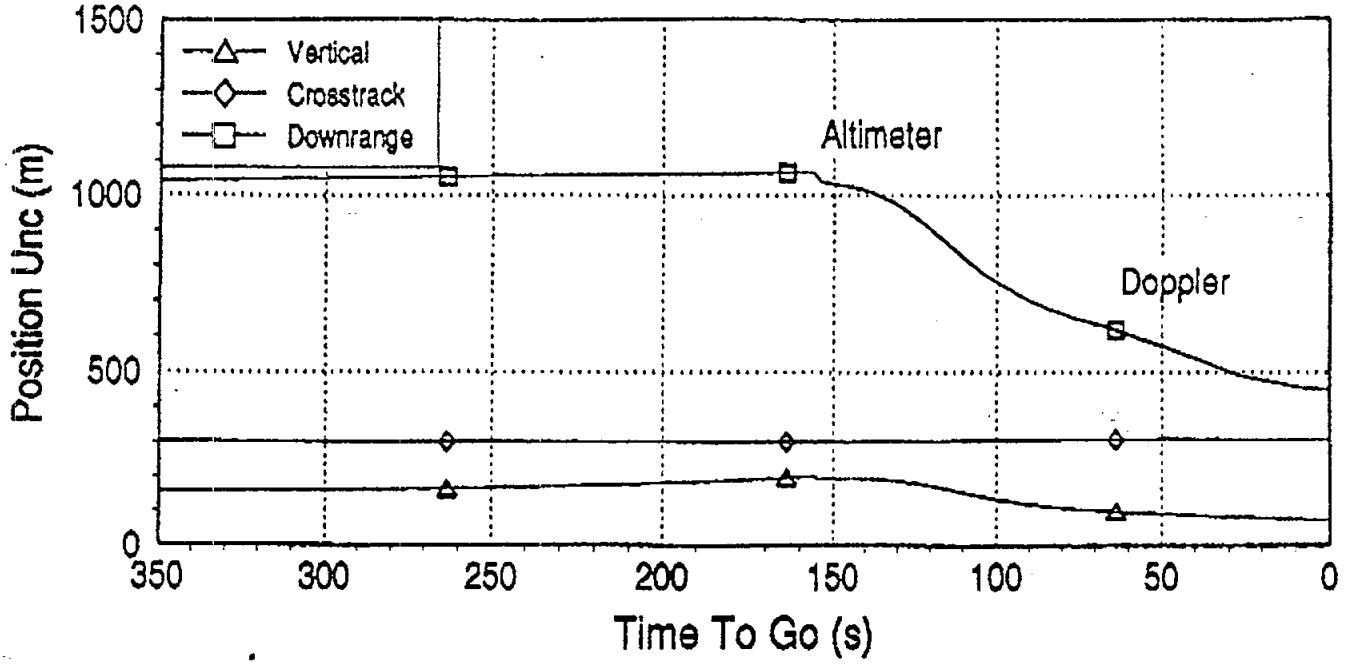
	Position Errors (m)			Velocity Errors (m/s)		
	DR	CT	ALT	DR	CT	ALT
(1) Baseline	450	307	0.60	0.07	0.28	0.08
(7) Loss of radar at 100 m						
(a) Lose altimeter (no Doppler)	525	314	12.56	0.21	0.58	0.46
(b) Lose altimeter	485	308	9.36	0.19	0.38	0.32
(8) Loss of radar at 33 m						
(a) Lose altimeter (no Doppler)	480	314	5.02	0.20	0.58	0.29
(b) Lose altimeter	466	307	3.92	0.17	0.31	0.23
(9) Additional navaids						
(a) Navsat over landing site	98	307	0.53	0.16	0.28	0.06
(b) Uprange Navsite (pre-PDI)	39	307	0.60	0.15	0.28	0.08

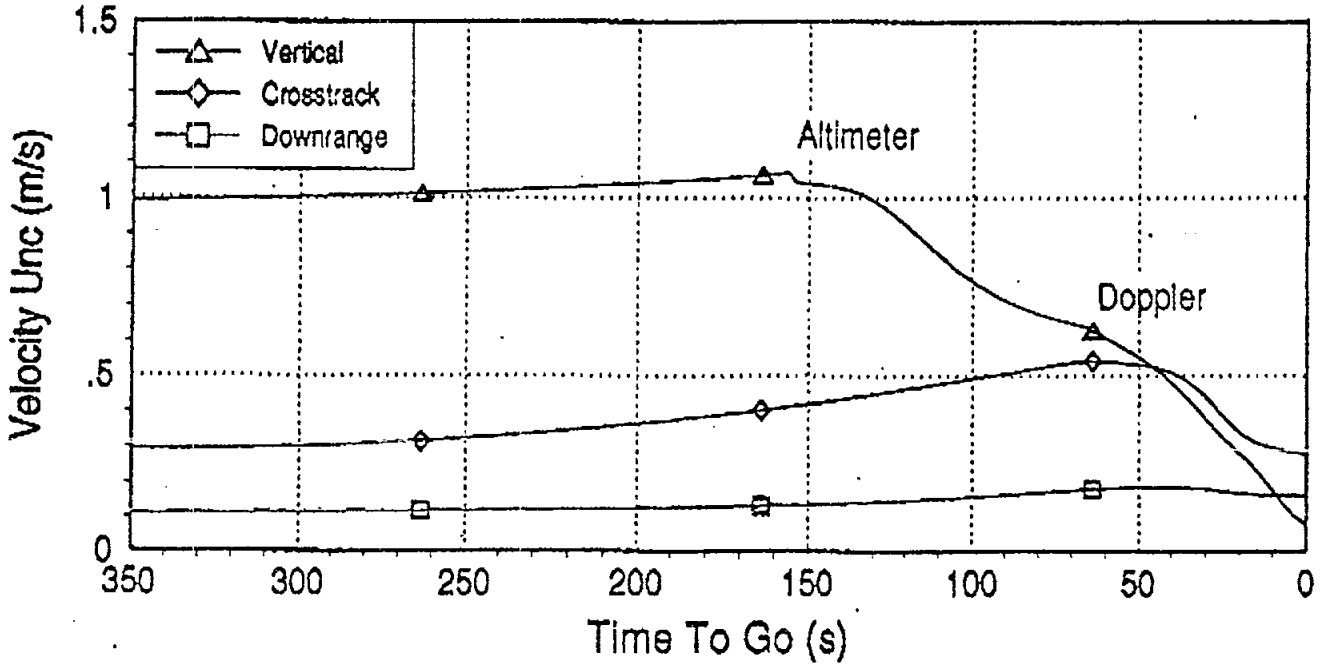
LUNAR DEORBIT TO LANDING TRAJECTORY LVLH STATE UNCERTAINTY HISTORY





LUNAR DEORBIT TO LANDING TRAJECTORY LVLH STATE UNCERTAINTY HISTORY FROM PDI





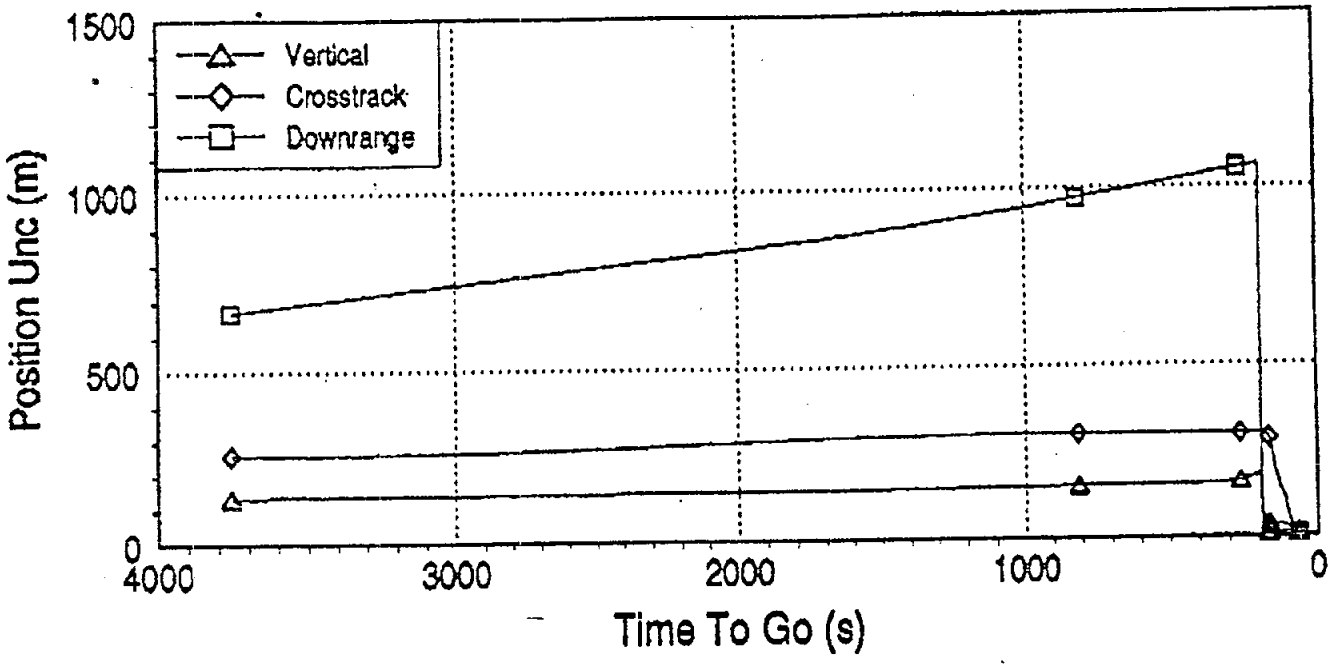
CONCLUSIONS

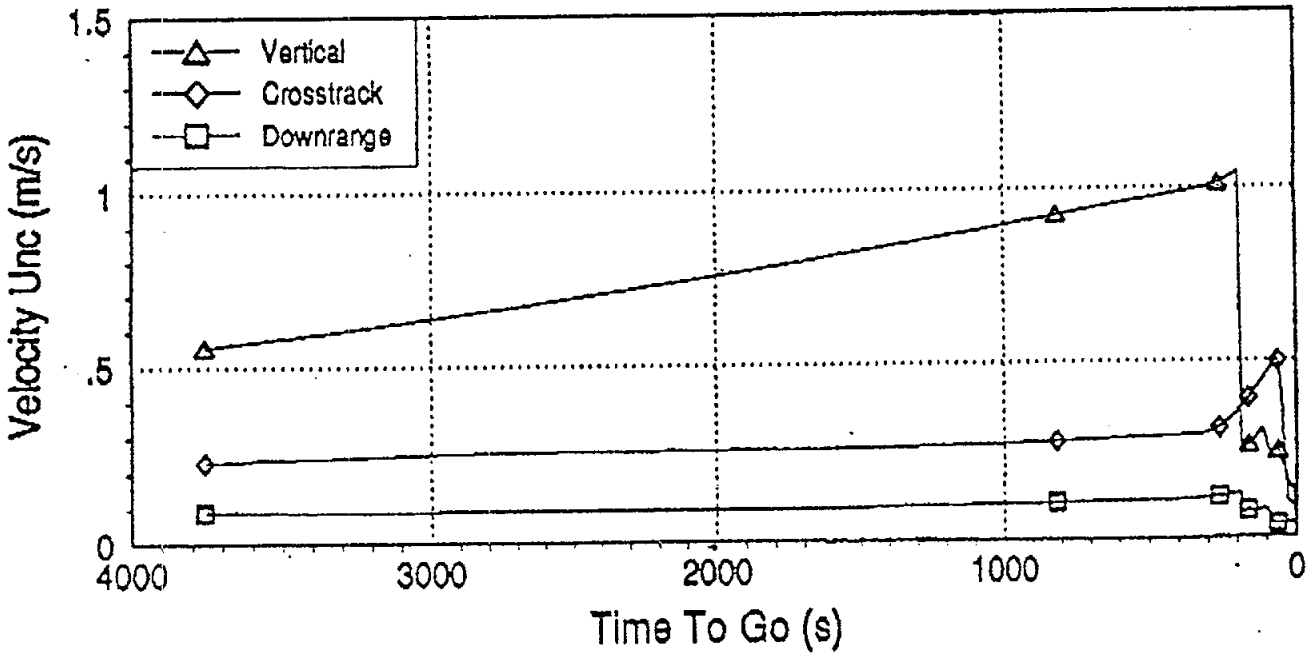
- Performance is primarily drive by gravity and terrain uncertainties
- Altimeter is essential for terminal altitude estimation
- Altimeter also provides terminal vertical velocity
- Stability of IMU inertial reference relatively unimportant because powered descent burn is only 5 min long and velocity errors have little time to integrate into position errors
- Doppler radar primary utility is in estimating horizontal velocity (surface relative) at touchdown. Whether it is needed depends on the horizontal velocity accuracy requirement (tbd) to prevent tipover
- We suggest swivel wheels to relax the requirement ☺

FUTURE WORK

- Review/improve models and assumptions
  - — Particularly gravity and terrain/altimeter models
- Impact of navigation updates on guidance needs evaluation
  - — How 'late' can updates such as radar altimeter be initiated
  - — Can JSC help?
- Include effects of RCS firings on nav during coast
- Consider constraint on radar acquisition based on vehicle attitude (fixed mount radar)
- Model Doppler radar with more fidelity including 3 non-orthogonal beams and correlated measurements

LUNAR DEORBIT TO LANDING TRAJECTORY
LVLH STATE UNCERTAINTY HISTORY





PRELIMINARY

LMI-92-014

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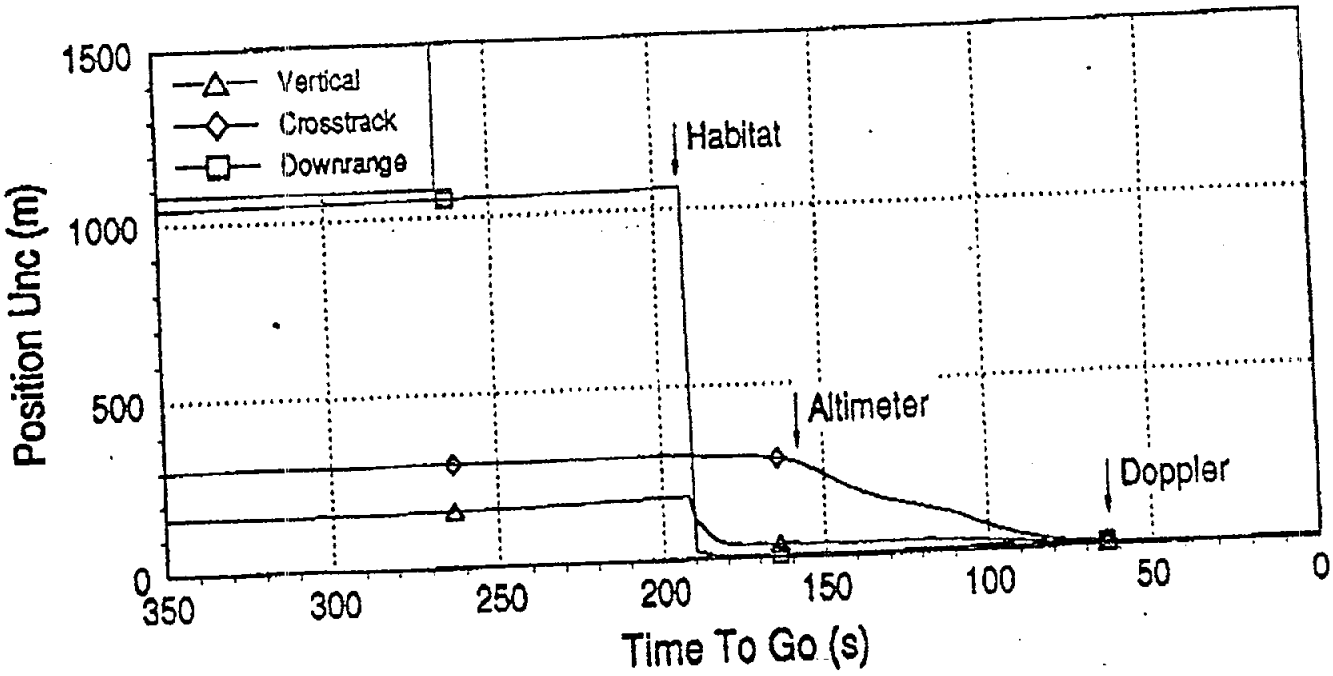
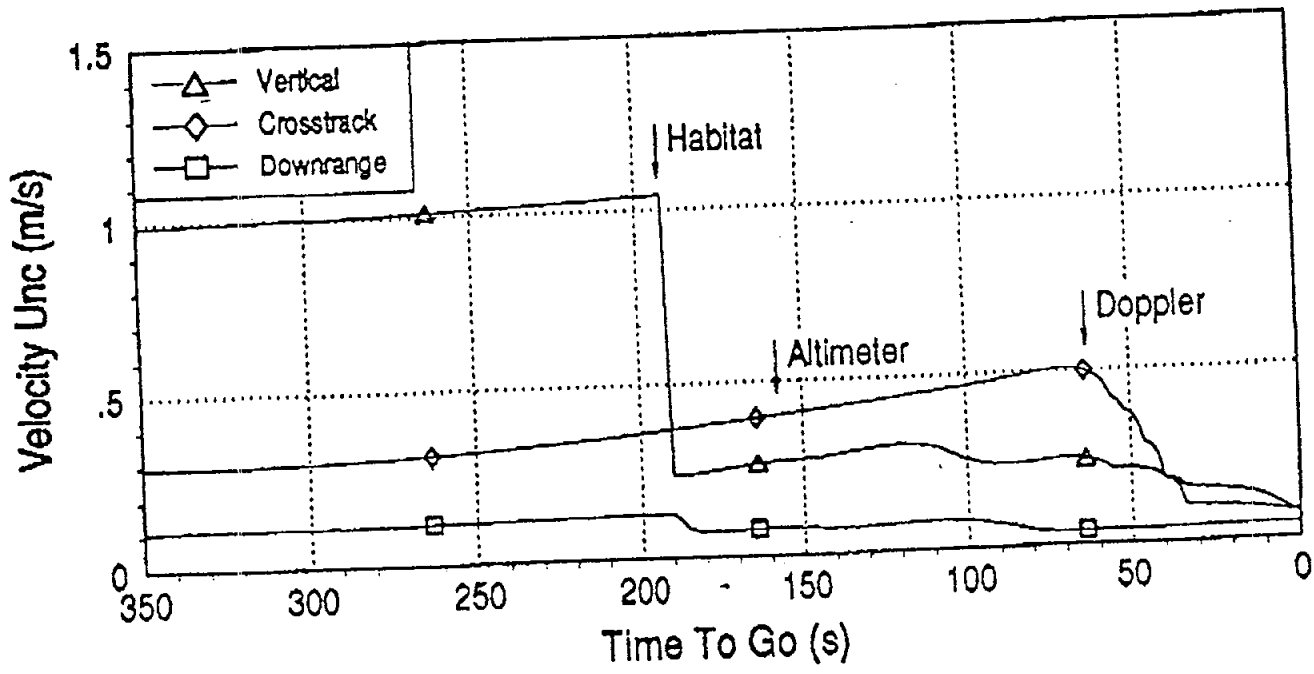
LUNAR DEORBIT TO LANDING TRAJECTORY LVLH STATE UNCERTAINTY HISTORY FROM PDI





PRELIMINARY

MANNED LANDER DEORBIT TO LANDING NAVIGATION STUDY

PRELIMINARY

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MARCH 17, 1992

LMI-92-015

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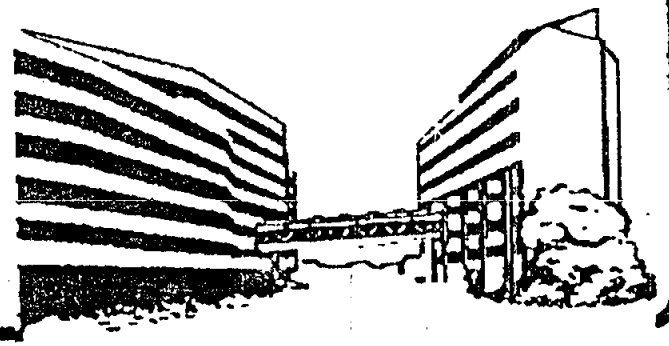

U.S. Gov't MAR 17 '92 CALTECH CELL-EBB 617 EEB 6874

HABITAT-RELATIVE STATE ERRORS AT LANDING (1σ)

	Position Errors (m)			Velocity Errors (m/s)		
	DR	CT	ALT	DR	CT	ALT
(1) Baseline (Habitat 1 km North)	5	4	0.57	0.04	0.07	0.07
(2) Effects of Radar Instruments						
(a) No Doppler radar	5	4	0.58	0.04	0.08	0.08
(b) No Doppler radar, no altimeter	9	5	99	0.05	0.12	0.16
(3) a) Habitat 1 km downrange	3	307	0.46	0.02	0.28	0.03
b) Habitat 1 km downrange, no Doppler	3	313	0.46	0.02	0.56	0.03
(4) IMU Alignment Sensitivity						
(a) 40 arcsec	5	4	0.57	0.04	0.07	0.07
(b) 10 arcsec	5	4	0.57	0.04	0.08	0.07
(c) Realign at PDI	5	4	0.57	0.04	0.08	0.07

HABITAT AND MANNED LANDERS

DEORBIT TO LANDING

NAVIGATION STUDY UPDATES

T. BRAND

S. SHEPPERD

MARCH 19, 1992

U.S. Gov't

HABITAT LANDER STUDY

- SOME CONCERN WAS EXPRESSED OVER THE 12 M ALTITUDE UNCERTAINTY FOR THE NO DOPPLER RADAR, LOSE ALTIMETER AT 100 M CASE
  - THE TERRAIN MODEL WAS SUSPECTED AS THE PROBLEM
  - SEVERAL EXPLORATION RUNS WERE MADE (SEE NEXT CHART)
  - TERRAIN WAS ONLY PART OF THE PROBLEM
  - EVEN WITH NO TERRAIN ERRORS, AT 100 M:
    - ALTITUDE UNCERTAINTY = 4 M
    - ALTITUDE RATE UNCERTAINTY = 0.24 M/S
    - LAST ALTIMETER MEASUREMENT NOISE = 1.3 + 100(0.05) = 6.3 M
    - NOISE COULD BE BEATEN DOWN WITH MORE MEASUREMENTS
    - NO ALTIMETER FOR 26 S SHOULD PRODUCE ALTITUDE UNCERTAINTY $\approx 4 + 26(0.24) \approx 10$ M

SITE-RELATIVE STATE ERRORS AT LANDING (1σ)

	POSITION ERRORS (m)			VELOCITY ERRORS (m/s)		
	DR	CT	ALT	DR	CT	ALT
1). BASELINE	450	307	0.60	0.07	0.28	0.08
2). NO DOPPLER, LOSE ALTIMETER AT 100 M	525	314	12.56	0.21	0.58	0.46
3). CASE (2) VARIATIONS						
a). TERRAIN: NO ERRORS	449	314	7.32	0.19	0.58	0.24
b). TERRAIN: 25 M ALTITUDE BIAS	451	314	7.35	0.19	0.58	0.24
c). TERRAIN: 1 DEG SLOPE UNCERTAINTY	456	314	8.58	0.20	0.58	0.30
d). TERRAIN: 100 M MARKOV NOISE MODEL	522	314	12.44	0.20	0.58	0.45

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MANNED LANDER STUDY

- SOME CONCERN WAS EXPRESSED OVER THE 99 M ALTITUDE UNCERTAINTY FOR THE NO ALTIMETER, NO DOPPLER RADAR CASE
  - THE TERRAIN MODEL WAS SUSPECTED AS THE PROBLEM
  - SEVERAL EXPLORATION RUNS WERE MADE (SEE NEXT CHART)
  - THE MARKOV PROCESS TERRAIN NOISE IS MAIN CULPRIT
  - HOWEVER, THE BIAS ALSO CONTRIBUTES, AND THE SLOPE SHOULD NOT BE IGNORED. ANY CROSS SLOPE WILL NOT BE ESTIMATED!
  - THE REAL MESSAGE IS THAT THE HABITAT <u>CANNOT</u> SUBSTITUTE FOR THE ALTIMETER IF IT IS 1 KM AWAY FROM THE LANDING SITE.
- SOME INTEREST WAS EXPRESSED IN A "NO DOPPLER RADAR" PLOT
  - THE ATTACHED PLOTS INDICATE THAT THE HABITAT IS PRIMARILY RESPONSIBLE FOR ESTIMATING CROSSTRACK VELOCITY, <u>NOT</u> THE

HABITAT-RELATIVE STATE ERRORS AT LANDING (1σ).

	POSITION ERRORS (m)			VELOCITY ERRORS (m/s)		
	DR	CT	ALT	DR	CT	ALT
1). BASELINE.	5	4	0.57	0.04	0.07	0.07
2). NO RADAR (NO ALTIMETER, NO DOPPLER)	9	5	99.50	0.05	0.12	0.16
3). NO RADAR VARIATIONS						
a). TERRAIN : NO ERRORS	9	5	9.30	0.05	0.12	0.16
b). TERRAIN : 25 m ALTITUDE BIAS	9	5	26.67	0.05	0.12	0.16
c). TERRAIN : 1 DEG SLOPE UNCERTAINTY	9	5	9.30	0.05	0.12	0.16
d). TERRAIN : 100 m MARKOV NOISE MODEL	9	5	96.31	0.05	0.12	0.16

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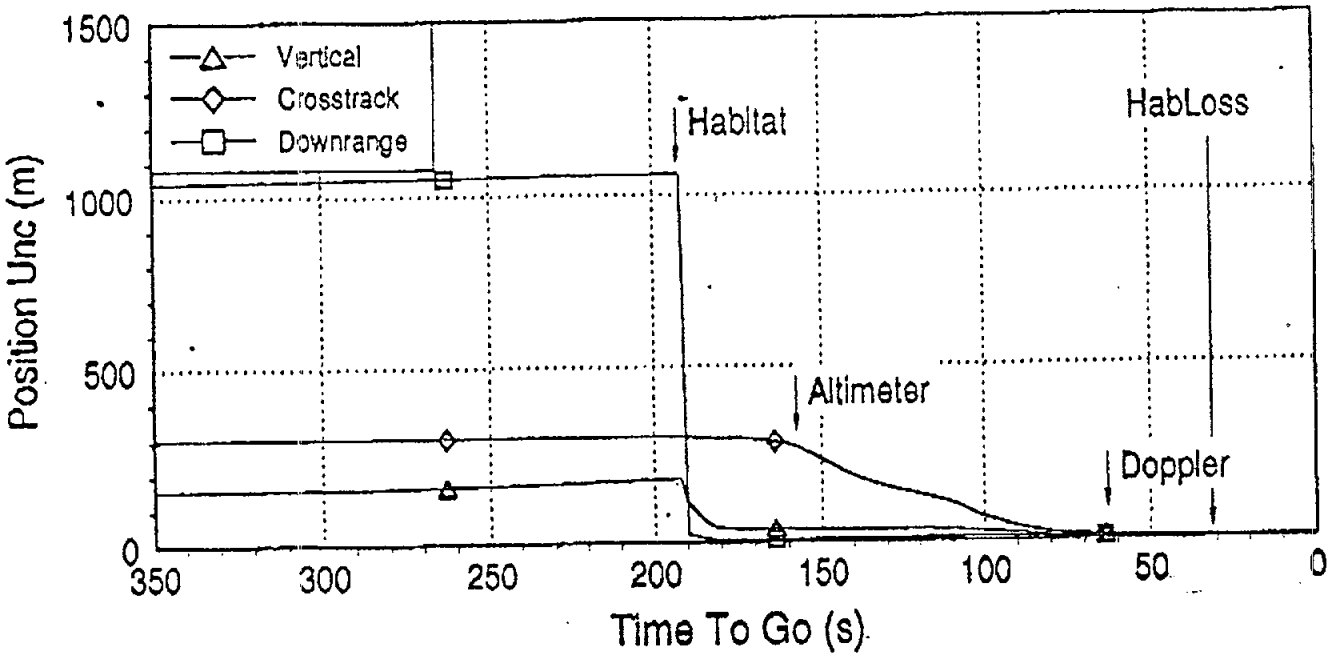
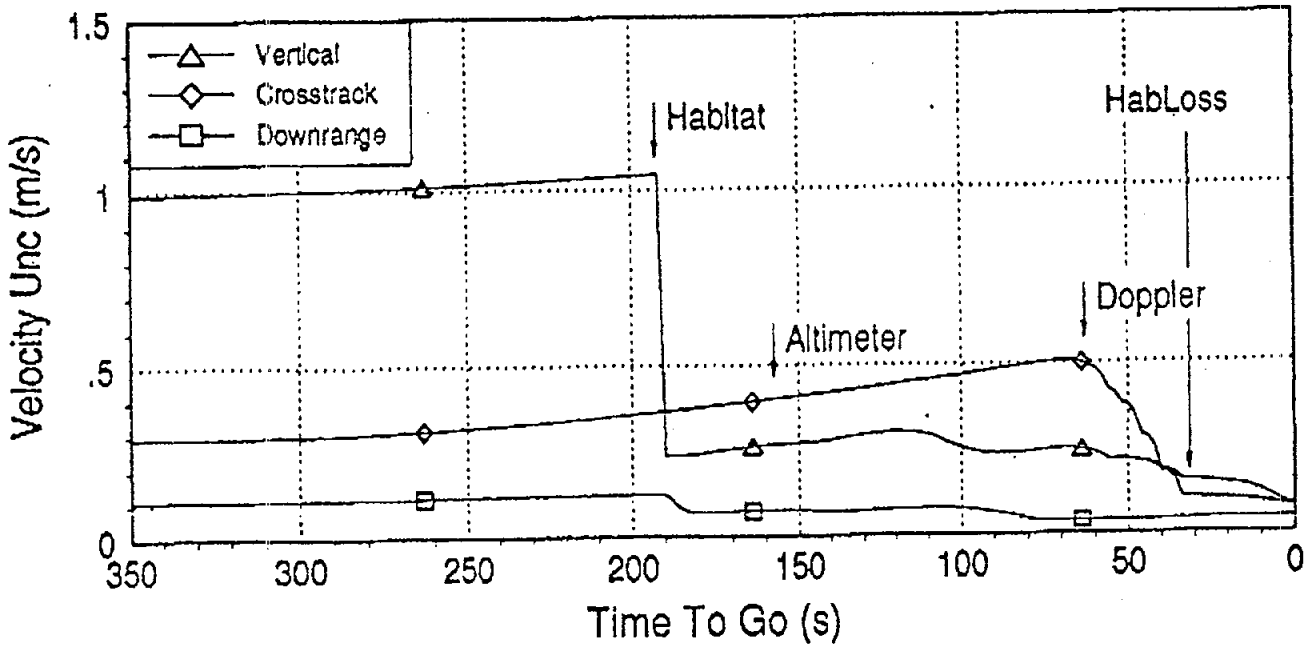
Lunar Deorbit to Landing Trajectory

LVLH State Uncertainty History from PDI

(Baseline : With Doppler)

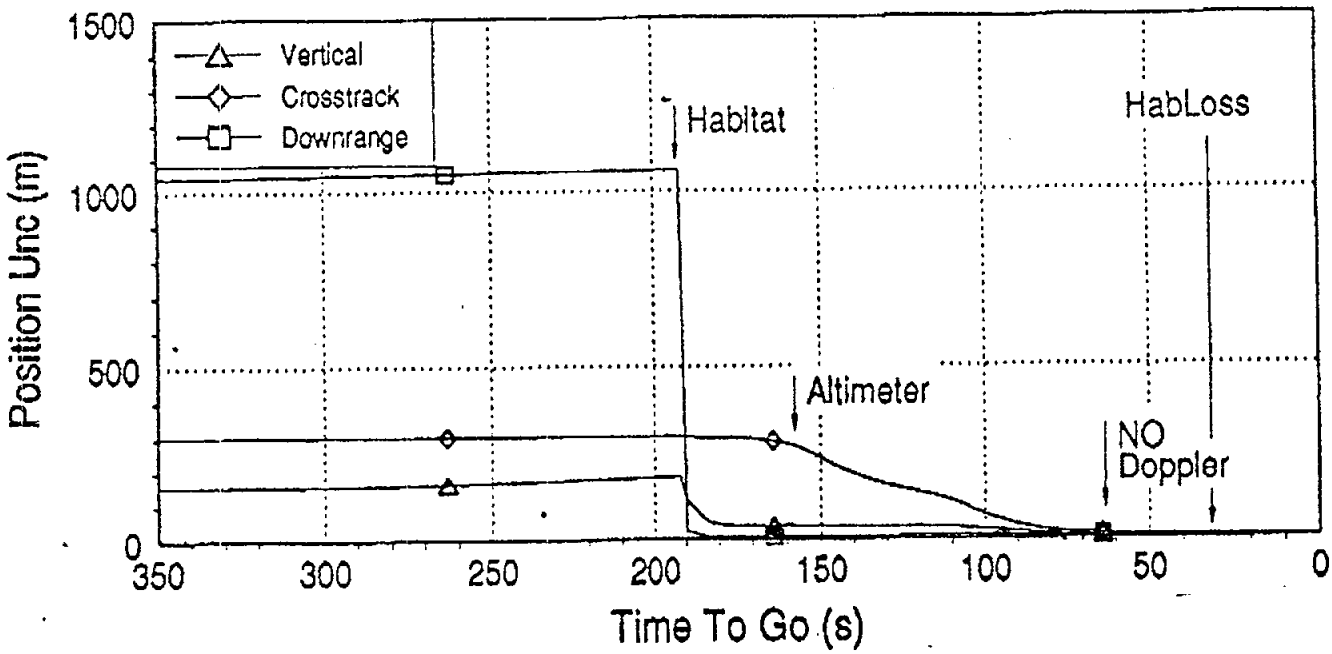


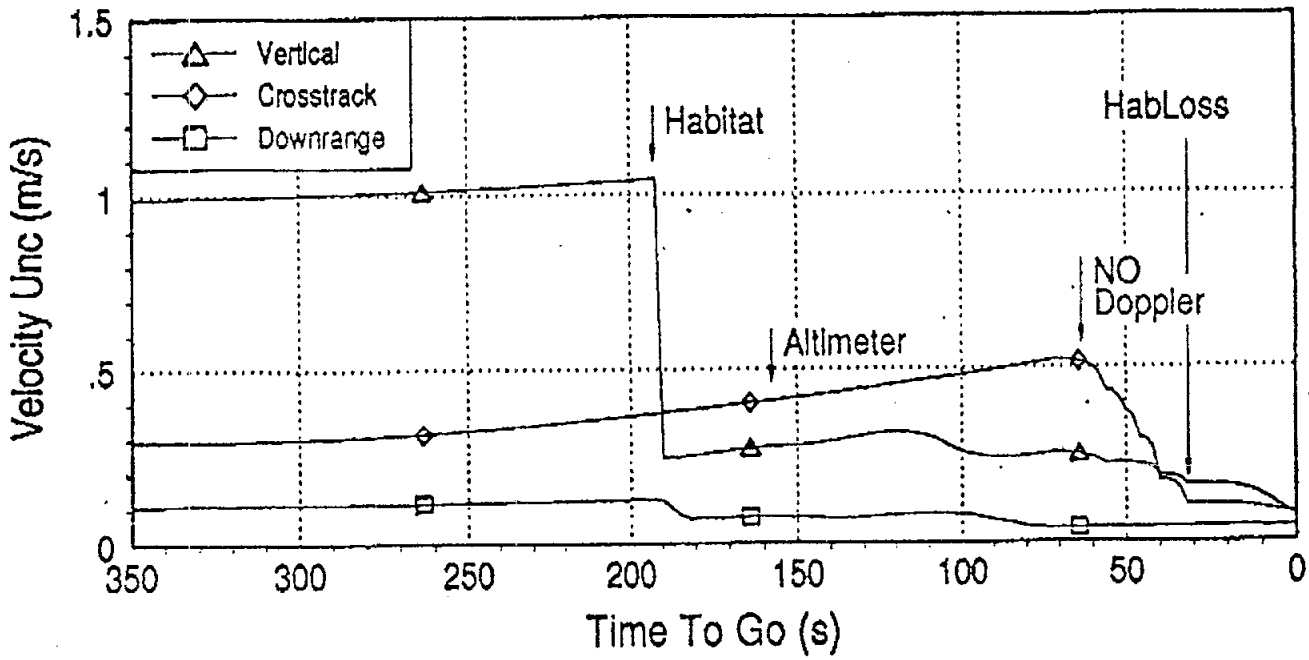


Lunar Deorbit to Landing Trajectory

LVLH State Uncertainty History from PDI

(No Doppler Case)





CONCLUSIONS

- MARKOV TERRAIN NOISE MODEL IS QUITE PESSIMISTIC
- 100 M MAGNITUDE IS PROBABLY TOO HIGH FOR THESE CASES
- HOWEVER, REAL PROBLEM IS THAT 1ST ORDER MARKOV MODELS ARE VERY NOISY (RELATIVELY SPEAKING) OVER SHORT HORIZONTAL DISTANCES
- A POSSIBLE SOLUTION IS TO USE A 2ND ORDER MARKOV PROCESS. THE AUTOCORRELATION FUNCTION FOR THIS PROCESS IS FLAT OVER SHORT DISTANCES AND SHOULD PROVIDE A MUCH MORE REALISTIC MODEL.
- WE WILL IMPLEMENT A 2ND ORDER MARKOV TERRAIN NOISE MODEL IN THE NEAR FUTURE

INTRALAB MEMORANDUM

ETB-92-145

To: Tim Brand

From: John Vytal

Date: March 26, 1992

Subj: Impact Resistance of Electronic Hardware for Lunar Drop

cc: P. Blasche, P. Konop

Proximity fuze circuits found in artillery shells routinely survive impacts of several thousand g's as do the batteries needed to power these circuits. (The US Army maintains a facility in New Jersey to test these devices at maximum impact.) So it is possible that an S-band transponder could be built to survive a drop with little or no vertical deceleration from lunar orbit. There are, however, other complications which make this approach unattractive.

1.) A solar array would be needed to charge the storage (accumulator) battery. It does not appear that any solar cells have been built to withstand this level of impact. Development would therefore be needed.

2.) An S-band transponder which could survive the impact would also need to be developed, but the technology exists which would permit this.

3.) A strong casing is necessary to protect the electronics using proven methods. Steel is the primary material used in artillery-shell casings.

   a.) The antenna needs radio transparency.
   b.) The solar array needs optical transparency.

This means two things:

i.) Development of a suitable protective casing with attributes a&b would be needed.

ii.) A hard protective casing will likely cause the transponder to be buried below the lunar surface rendering the antenna and solar array ineffective.

It seems clear that some degree of vertical deceleration is needed to successfully accomplish the drop of the lunar navigation transponders and have them operate effectively. It also appears that some level of control would be needed to ensure correct positioning of the devices (i.e. antenna, solar array). For all of the above considerations, the cost and complexity of such a system may be great enough to make this approach unattractive.

There is another approach, however, which mitigates some of the stated problems. In discussions with Phil Konop, he mentioned that a passive corner reflector was considered for a Mars navigation system but was rejected because:

1.) The large distances involved (Path loss varies as $\frac{1}{R^4}$ instead of $\frac{1}{R^2}$ as with active transponders).

2.) Dust storms on Mars could cause further losses which might make the system inoperative for long periods of time.

In the case of the moon.

1.) Distances are much less.

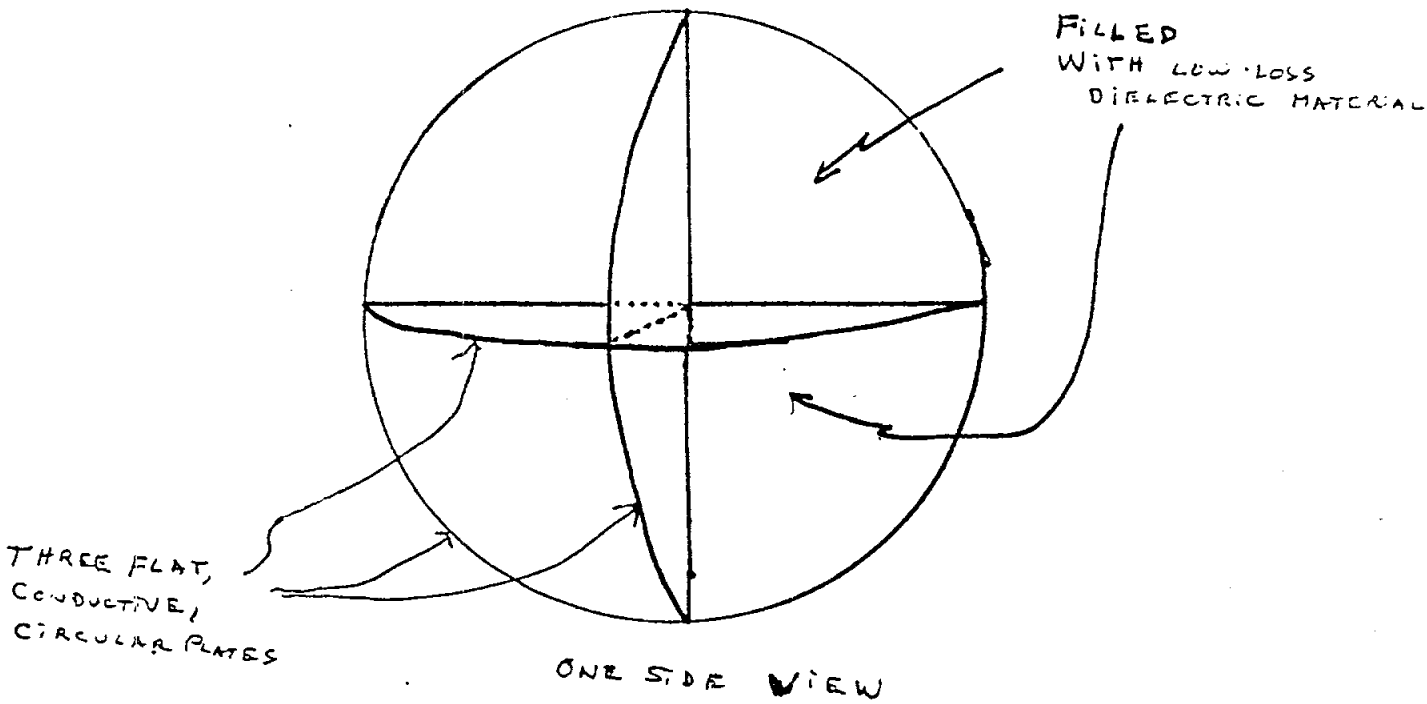
2.) There is no atmosphere and hence no dust storms.

Further, an omni-dimensional corner reflector could be built which would provide nearly full hemispherical omni-directional coverage whatever the orientation. This reflector would be similar to those currently used by small boats to make themselves visible to large radar-equipped ships. (A rough sketch is attached.) Some deceleration would still be needed but active control required would be considerably less since orientation is not important.

This note is intended as a "sanity check" on concepts we discussed. Further work is needed for a true feasibility analysis.

Attachment A

Omni-Directional Passive Corner Reflector



Attachment B

Typical Solar Array Performance

Mean value of total solar radiation at normal incidence (no atmosphere):

140 mW/cm^2

(Source: American Institute of Physics Handbook, 3rd ed.)

Typical Solar Cell

efficiency:	15%
area:	12 cm^2
solar radiation:	140 mW/cm^2
current at peak power:	330 mA
voltage at peak power:	440 mV
peak power:	252 mW

(Solar radiation and therefore power output decreases with incidence angle near the beginning and the end of each lunar day.)

A typical system composed of an array of solar cells and an accumulator battery will have an efficiency in the range of 60%-70% (Source: Energy Technology Handbook 1977).

Peak Sys Power output = (Sys Efficiency) x (No. of Cells in Array) x (Power out per cell)

HABITAT AND MANNED LANDERS DEORBIT TO LANDING NAVIGATION STUDY

(REVISION 1)

T. BRAND
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S. SHEPPERD
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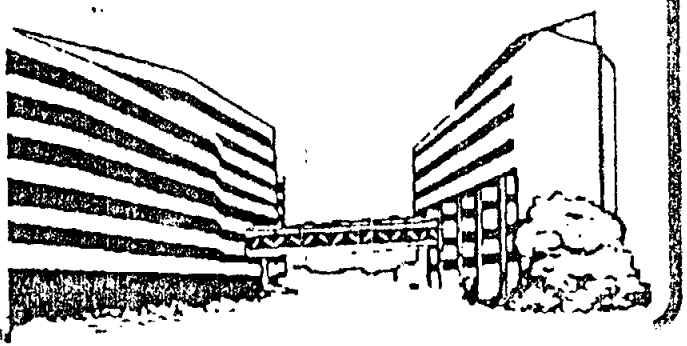
MARCH 27, 1992



THE
CHARLES STARK DRAPER
LABORATORY, INC.

ESB-92-149/REV 1
LMI-92-013/REV 1

PART 1

HABITAT LANDER NAVIGATION STUDY

STUDY OVERVIEW

- OBJECTIVE
  - — Define Habitat Lander navigation sensor and initial condition requirements for landing
    - IMU
    - IMU alignment (star tracker)
    - Radars (altitude, Doppler)
    - Initial state vector accuracy
    - Possible Navsat or uprange Navsite navaid(s)

- APPROACH
  - — Linear covariance analysis
    - Statistical measure of navigation performance on reference trajectory
    - Does not evaluate G&N interaction, particularly the timeliness of nav updates

- Trajectory obtained from JSC (Condon and Bryant)
  - — Equatorial approach to landing site on the equator
  - — Hohmann transfer (180 deg) down to 18 km altitude
  - — Powered descent starts a perilune

INTRODUCTORY COMMENTS

- This is preliminary data; models and assumptions must be reviewed carefully

- Key questions
  (1) Do we need Doppler (ground speed) radar?
  (2) When do we need altimeter update (will single-position radar pointing be sufficient)?
  (3) Is good terrain information necessary (or what is our sensitivity to altitude uncertainty under the ground track and at landing)?
  (4) What navigation performance is possible?

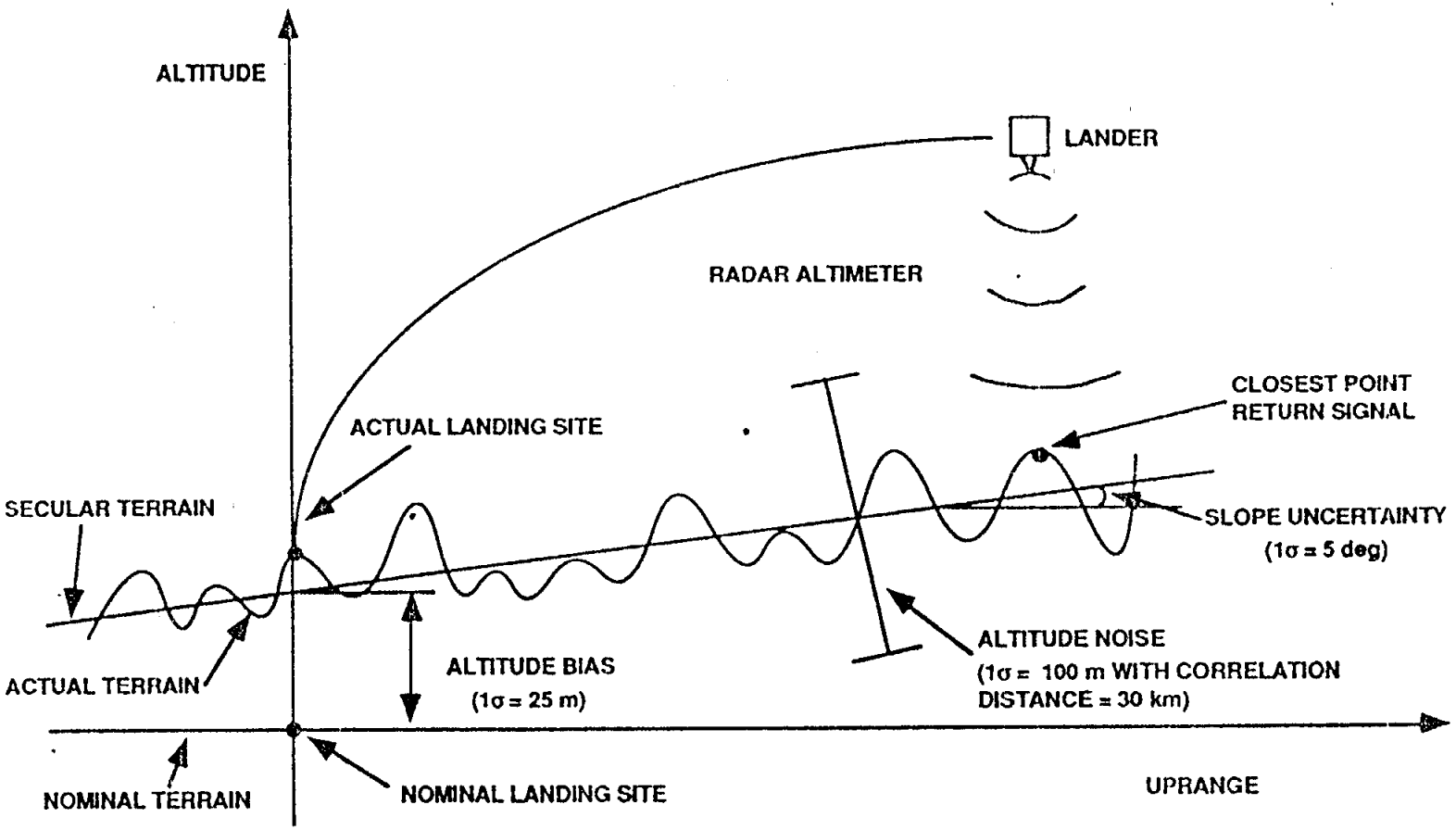
BASELINE SCENARIO ASSUMPTION

- Habitat in 100 km equatorial orbit
- Essentially perfect (< 100 m) DSN state vector update 1 rev prior to deorbit
- Error growth during 1 rev coast prior to deorbit (due to gravity)
  - — Model gravity uncertainty as process noise
  - — Baseline "sized" for 2 km (3σ) downtrack error growth/rev
  - — Crosstrack growth approximately 20% of downrange
  - — LVLH process noise values used:
    $3.18\ 10^{-7}$, $3.18\ 10^{-7}$, $3.97\ 10^{-6}$ m^2/s^3 (DR, VER, CT)
  - — Growth: 667, 136, 133 m (DR, VER, CT) (1σ)
- Deorbit burn (ΔV = 18.6 m/s)
  - — IMU alignment accuracy = 20 arcsec
  - — Small burn has only minor impact on nav

BASELINE SCENARIO ASSUMPTION (cont'd)

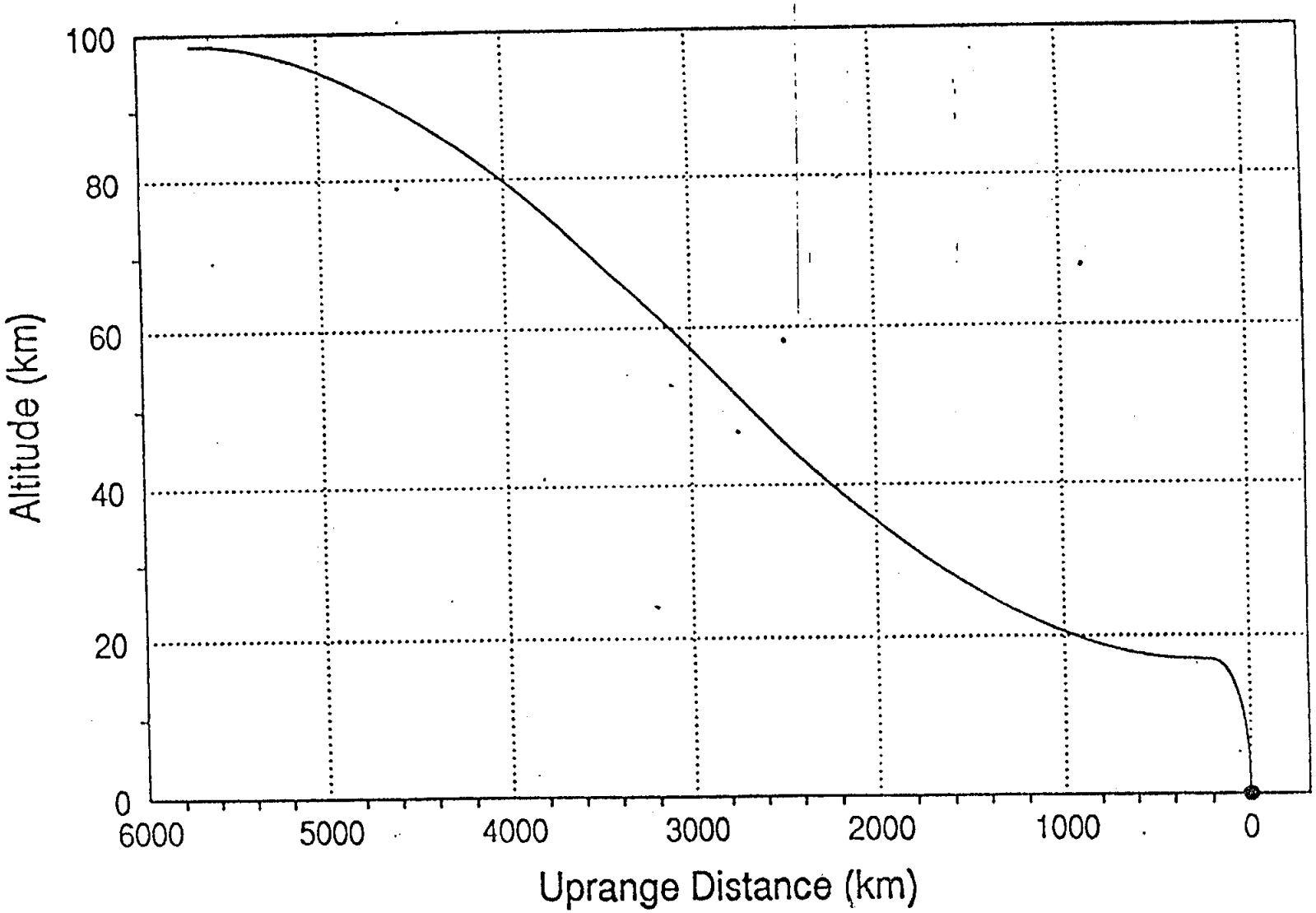
- Landing area terrain model (see next chart)
  - — Local terrain (within 5 km radius) assumed not well known
  - — Site altitude uncertainty = 25 m (may be optimistic)
  - — Terrain slope uncertainty assumed = 1 deg (equivalent to ≈18 m/km)
  - — Approach terrain "noise" = 5 m (Markov process with correlation distance = 30 km)
  - — Operation of radar altimeter over "bumpy" terrain or crater near/at the landing site needs more thought (add spacecraft altitude dependence to combined terrain/radar model)

FINAL APPROACH TERRAIN MODELING



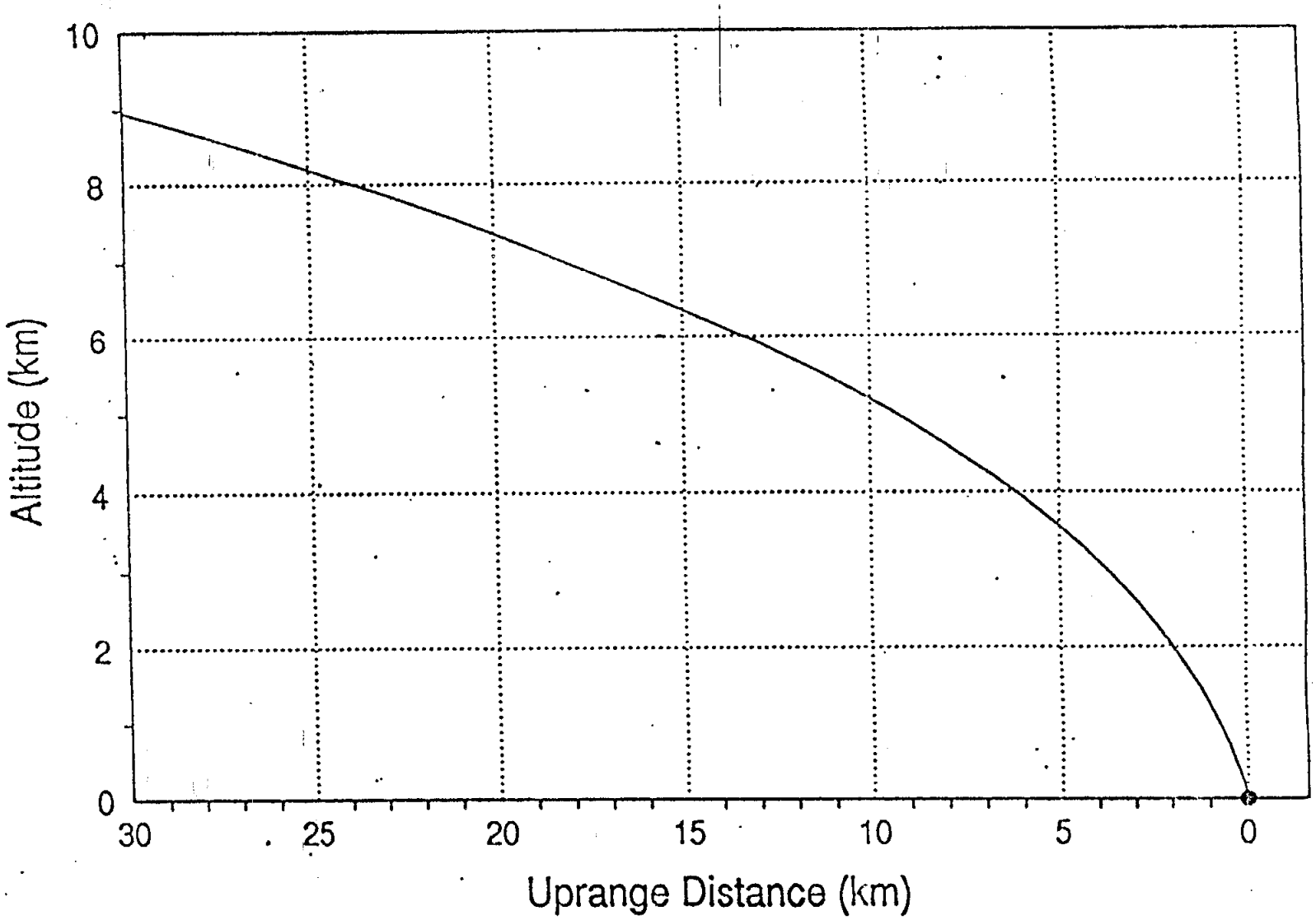
LUNAR DEORBIT TO LANDING TRAJECTORY

FULL TRAJECTORY PROFILE



LUNAR DEORBIT TO LANDING TRAJECTORY

FINAL RELATIVE APPROACH



BASELINE INSTRUMENT ACCURACY ASSUMPTIONS

- IMU Accuracy (1σ)
  - — Initial misalignment at ignition: 20 arcsec (tracker driven)
  - — Misalignment drift rate*: 0.015 deg/hr
  - — Accelerometer bias: 50 μg's
  - — Accelerometer scale factor: 0 ppm (to be added)

- Radar altimeter accuracy** (1σ)
  - — Measurement bias: 0 m (to be added)
  - — Measurement noise: $\begin{cases} 1.3 + 0.05\,H \text{ m } (H < 300 \text{ m}) \\ 9.0 + 0.05\,H \text{ m } (H > 300 \text{ m}) \end{cases}$

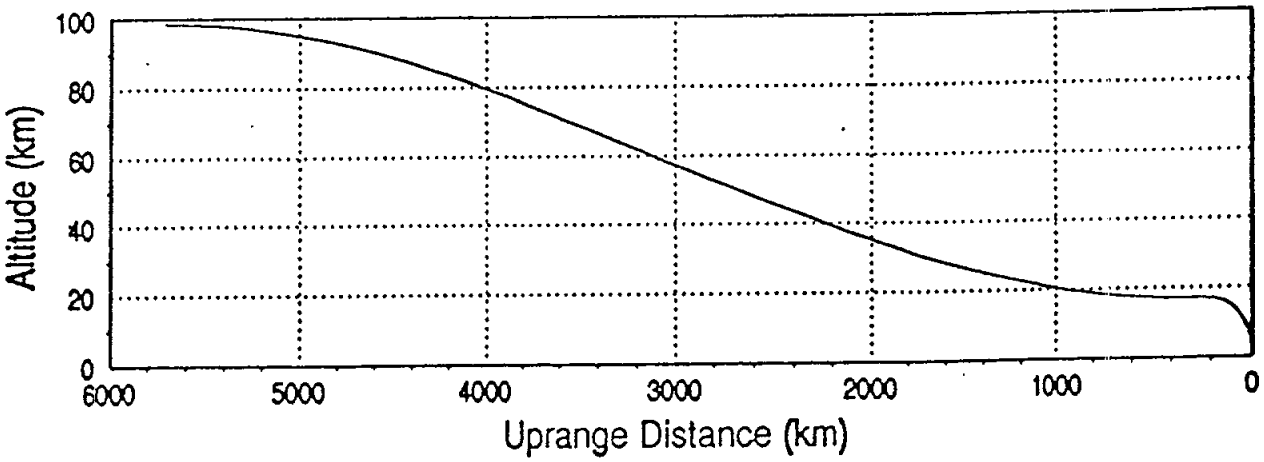
- Doppler radar accuracy** (1σ)
  - — Antenna misalignment: 0.067 deg
  - — Measurement bias ($\tau = 30000$ s): 0.30 m/s
  - — Measurement noise: $\begin{cases} 0.30 + 0.02V \text{ m/s } (V < 200 \text{ m/s}) \\ 0.30 + 0.03V \text{ m/s } (V > 200 \text{ m/s}) \end{cases}$

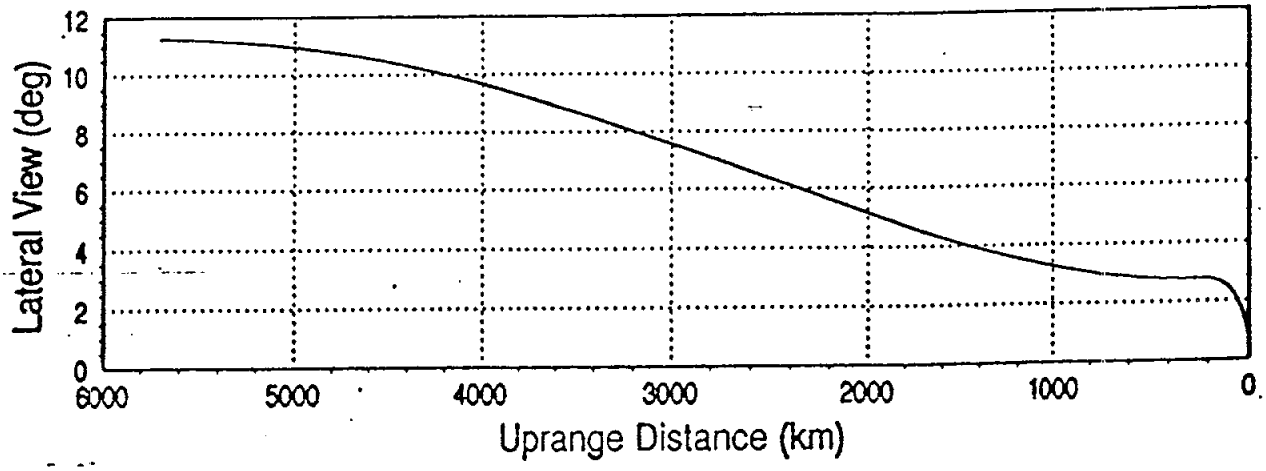
*Consider state (not estimated)
**Acquisition altitude = 10 km
radar altimeter measurement every 1 s (artificially slow)
Doppler radar measurements every 2 s (artificially slow)

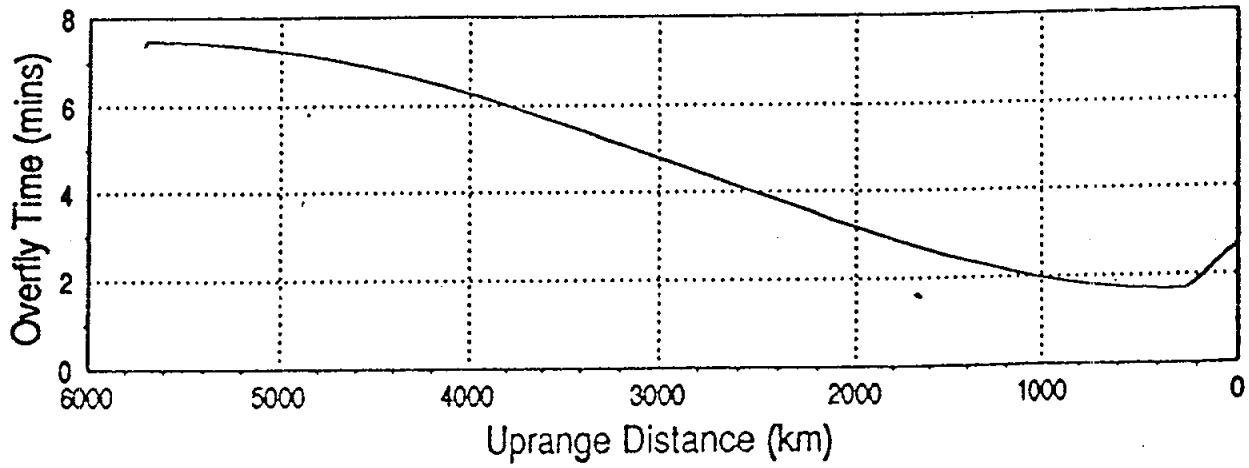
POTENTIAL ADDITIONAL NAVAIDS

- Navsat
  - — 5000 km circular orbit
  - — Located over landing site during powered descent
- Navsite
  - — Located uprange (flyover just before PDI)
  - — 10 deg elevation angle measurement requirement
  - — Yields about 5 measurements at 20 s intervals (see next chart)
- Each provides 2-way range measurements
  - — 3 m bias (due to electronic delays)
  - — 10 m/1000 km measurement noise (low power)

LUNAR DEORBIT TO LANDING TRAJECTORY SURFACE BEACON VISIBILITY ISSUES



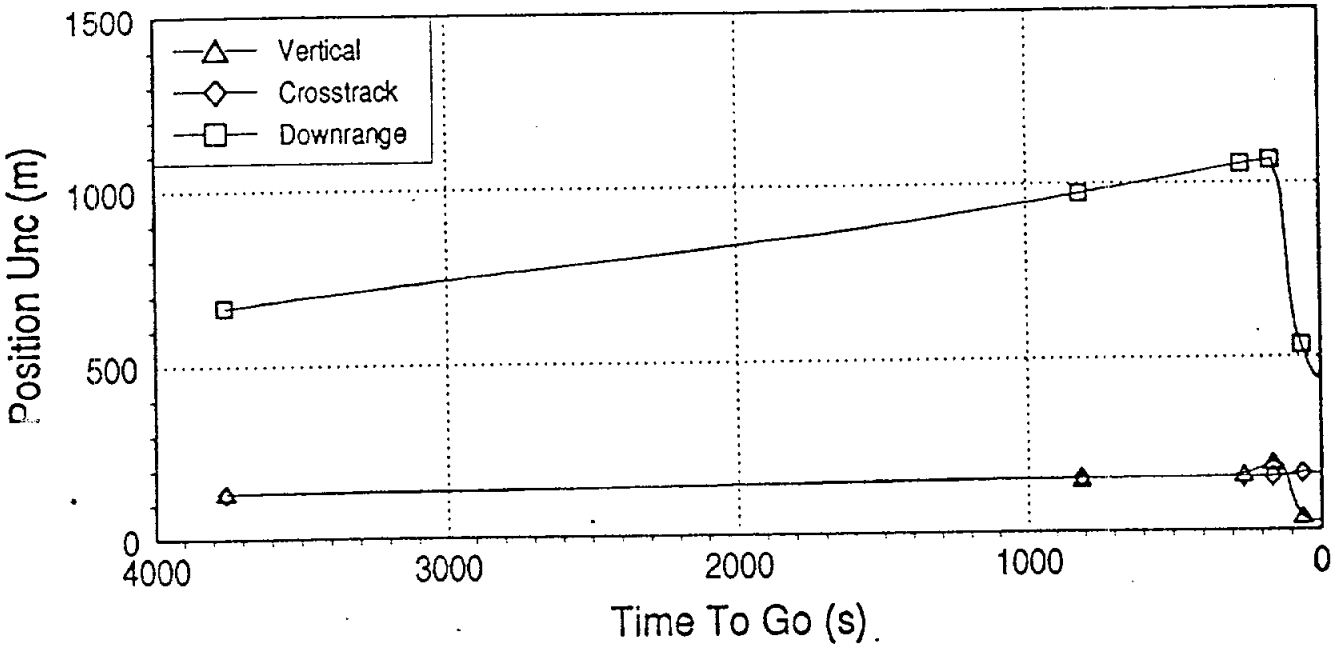


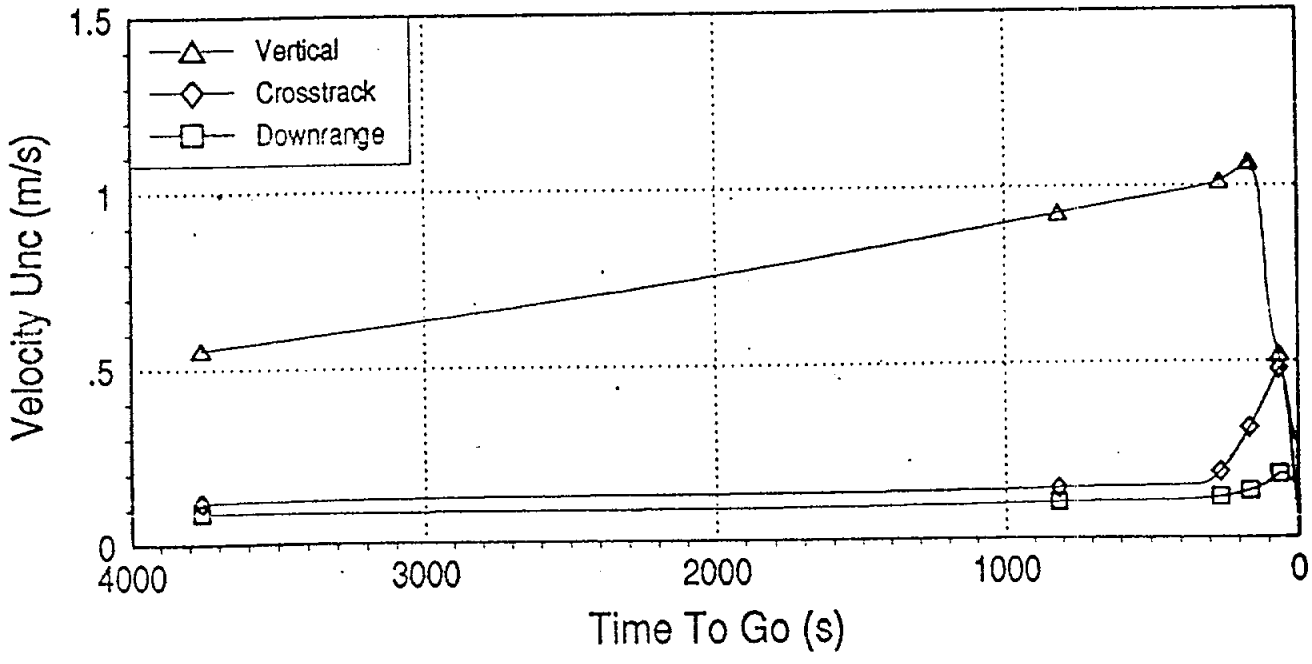


SITE-RELATIVE STATE ERRORS AT LANDING (1σ)

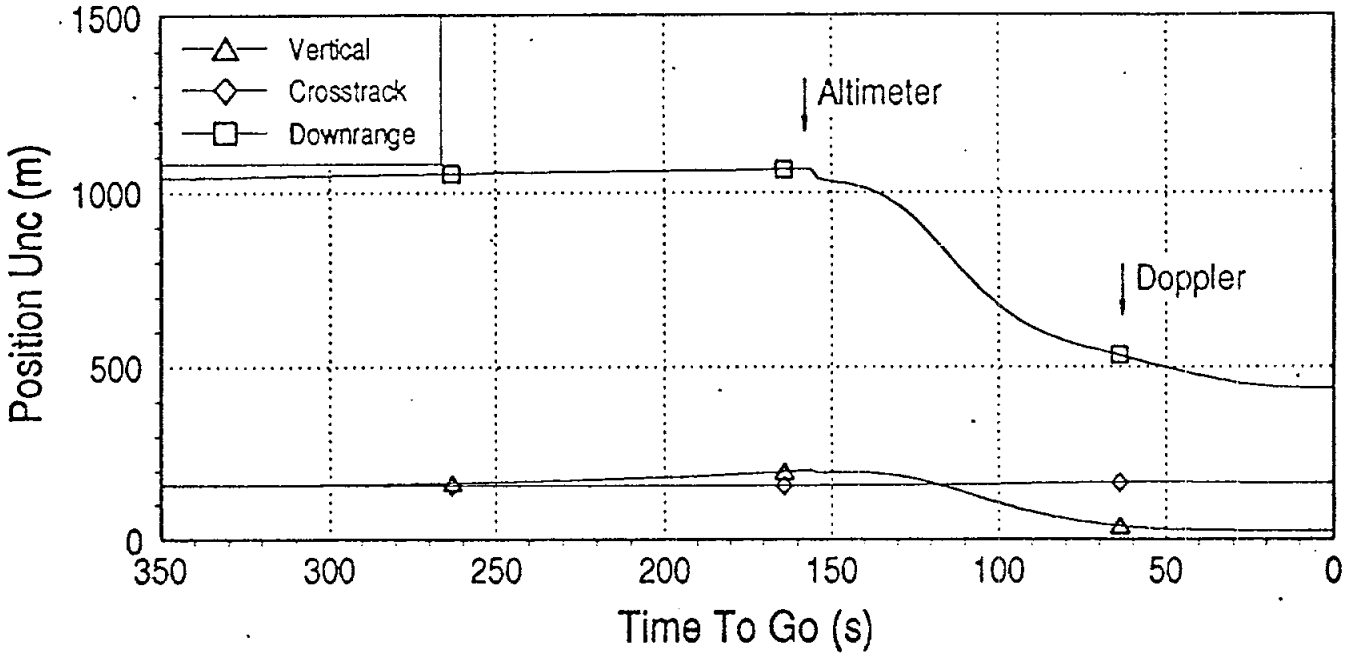
	Position Errors (m)			Velocity Errors (m/s)		
	DR	CT	ALT	DR	CT	ALT
(1) Baseline	437	163	0.55	0.15	0.27	0.06
(2) Isolate Gravity Effects						
(a) Perfect IMU	115	154	0.55	0.04	0.14	0.06
(b) Perfect IMU (no Doppler)	115	154	0.55	0.04	0.16	0.06
(3) Isolate IMU Effects						
(a) Perfect gravity	44	50	0.53	0.15	0.27	0.06
(b) Perfect gravity (no Doppler)	48	90	0.54	0.17	0.50	0.06
(4) Effects of Radar Instruments						
(a) No Doppler radar	438	179	0.55	0.17	0.52	0.06
(b) No Doppler radar, no altimeter	1060	179	341	0.24	0.52	1.21
(c) Lose altimeter at 33 m	440	163	2.94	0.16	0.27	0.15
(d) Lose both radars at 33 m	440	164	2.99	0.16	0.29	0.15
(5) Initial Condition Sensitivity						
(a) 1/2 × Initial state errors	315	93	0.55	0.15	0.27	0.06
(b) 2 × Initial state errors	525	314	0.55	0.16	0.28	0.06
(6) IMU Alignment Sensitivity						
(a) 1/2 × baseline (= 10 arcsec)	425	163	0.55	0.15	0.27	0.06
(a) 2 × baseline (= 40 arcsec)	473	164	0.55	0.15	0.28	0.06
(c) Realign at PDI	253	158	0.55	0.15	0.21	0.06
(d) Realign at PDI (no Doppler)	254	160	0.55	0.17	0.28	0.06
(7) Landing Site Altitude Bias						
(a) 1 km altitude error	465	163	0.55	0.17	0.27	0.06
(b) 1 km altitude error (no Doppler)	466	179	0.56	0.20	0.52	0.06
(8) Additional Navaids						
(a) Navsat over landing site	97	163	0.50	0.15	0.27	0.05
(b) Uprange Navsite (pre-PDI)	35	163	0.54	0.15	0.27	0.06

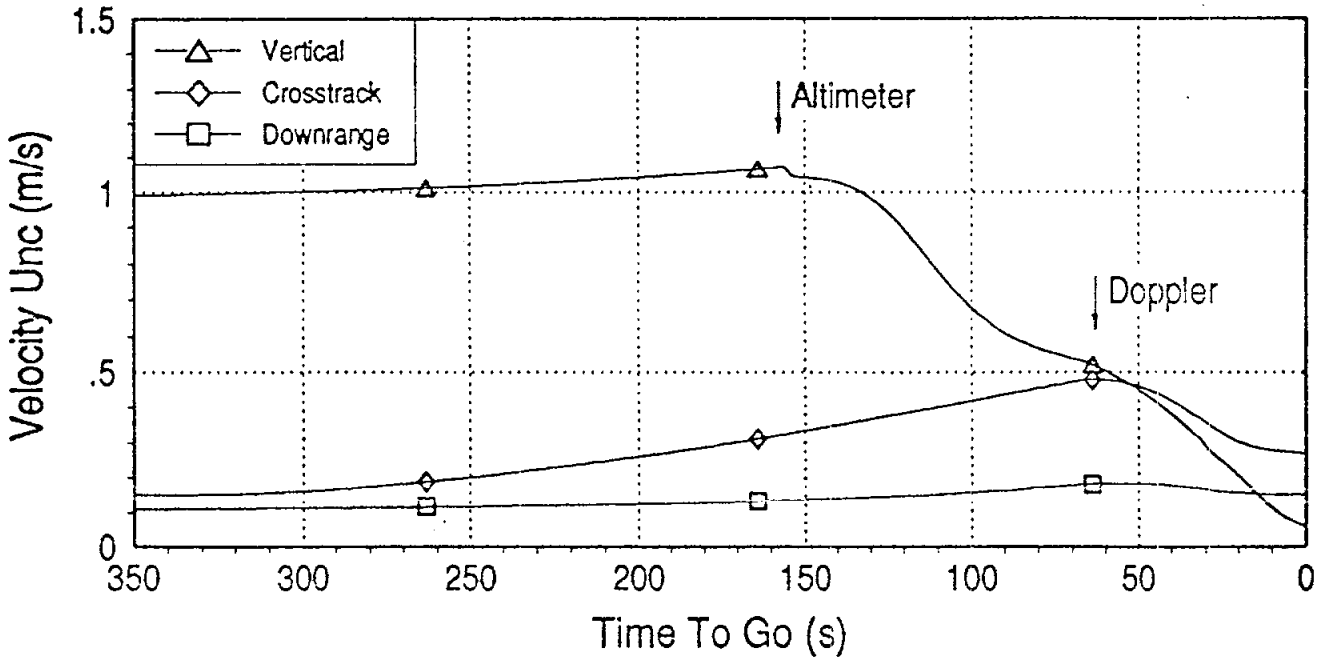
LUNAR DEORBIT TO LANDING TRAJECTORY
LVLH STATE UNCERTAINTY HISTORY





LUNAR DEORBIT TO LANDING TRAJECTORY LVLH STATE UNCERTAINTY HISTORY FROM PDI





CONCLUSIONS

- Performance is primarily driven by gravity and terrain uncertainties
- Altimeter is essential for terminal altitude estimation
- Altimeter also provides terminal vertical velocity
- Stability of IMU inertial reference relatively unimportant because powered descent burn is only 5 min long and velocity errors have little time to integrate into position errors
- Doppler radar primary utility is in estimating horizontal velocity (surface relative) at touchdown. Whether it is needed depends on the horizontal velocity accuracy requirement (tbd) to prevent tipover

PART 2

MANNED LANDER NAVIGATION STUDY

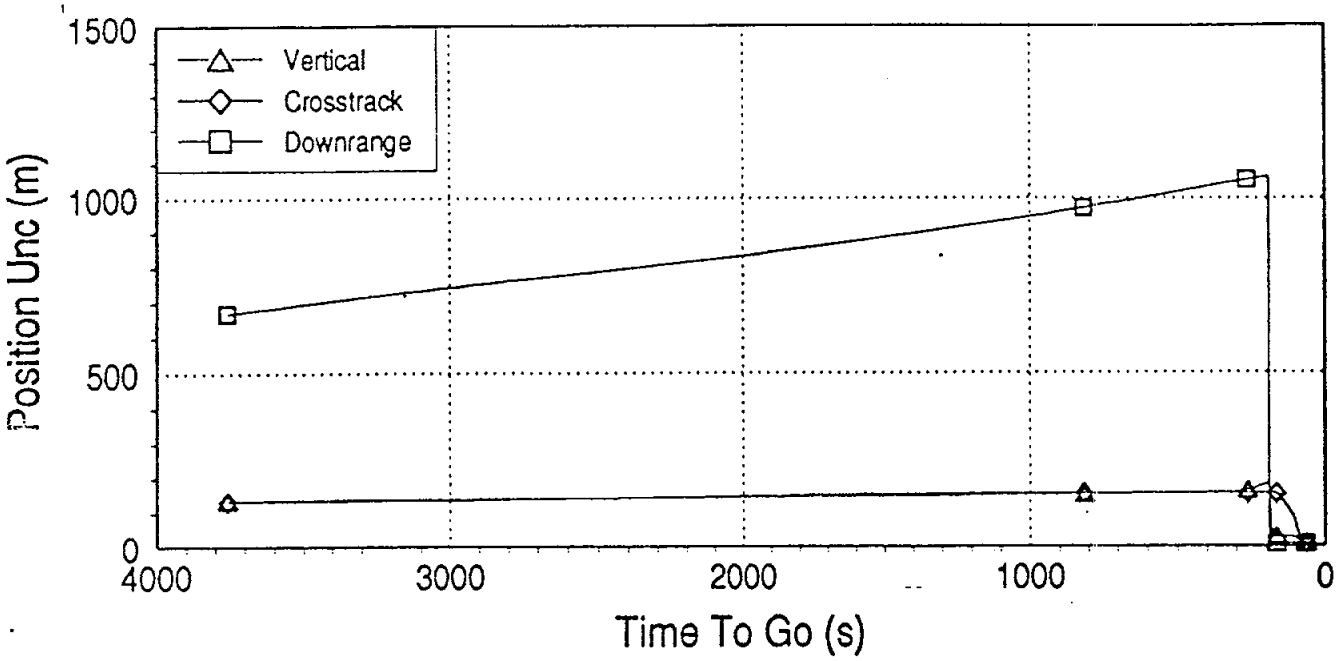
INTRODUCTION

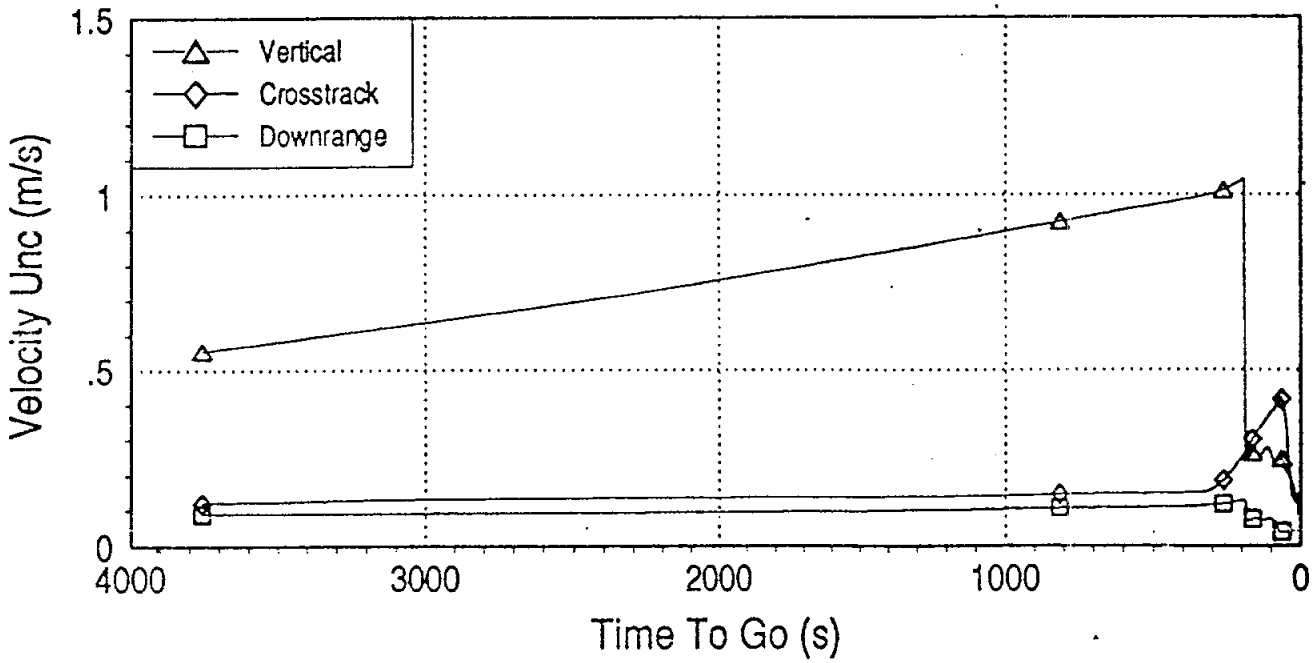
- The manned lander problem is very similar to the habitat lander problem
- The primary difference is that the manned lander must land in the vicinity of the habitat
  - It is assumed that the habitat is 1 km from the nominal landing site
  - The baseline assumes it is 1 km in the crosstrack direction
- As such, the habitat may be used as a radio navaid. The baseline assumes that it provides 2-way range measurements
  - 3 m bias (due to electronic delays)
  - 10 m/1000 km measurement noise (low power)
  - Visibility restricted to 10 deg above the horizon

HABITAT-RELATIVE (HORIZONTAL) AND SURFACE-RELATIVE (VERTICAL) STATE ERRORS AT LANDING (1σ)

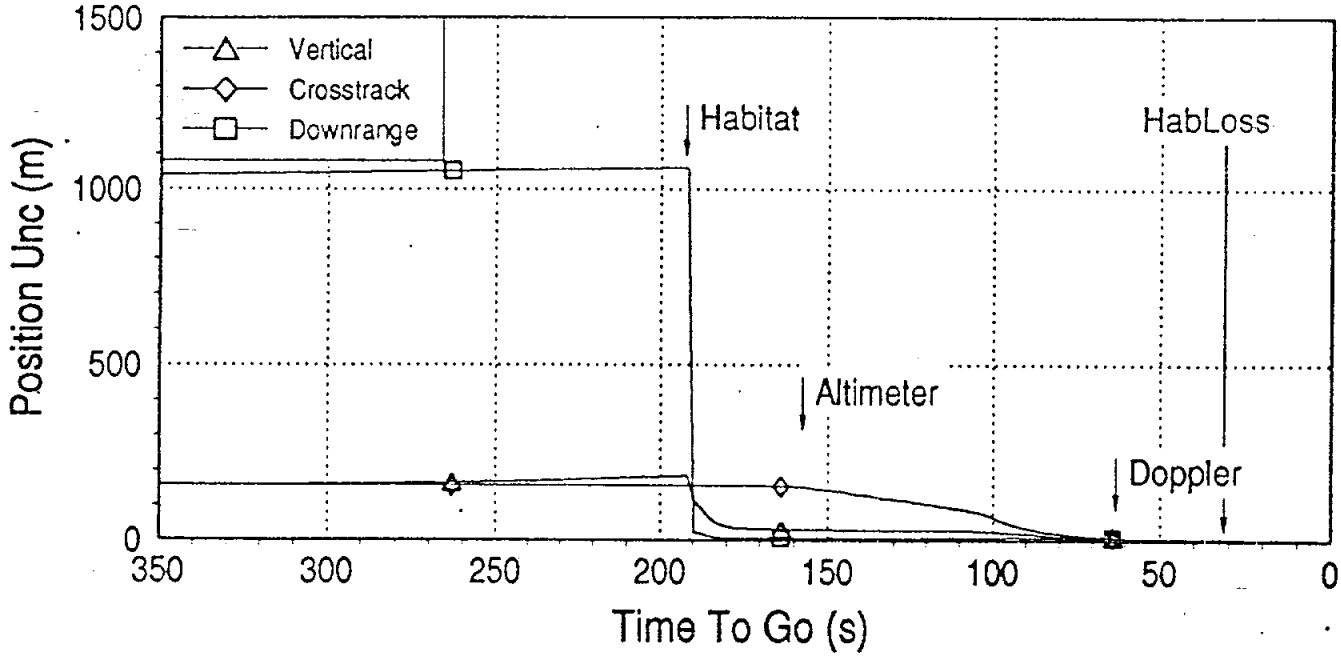
	Position Errors (m)			Velocity Errors (m/s)		
	DR	CT	ALT	DR	CT	ALT
(1) Baseline (Habitat 1 km CT)	5	5	0.54	0.04	0.10	0.06
(2) Effects of Radar Instruments						
(a) No Doppler radar	5	5	0.55	0.05	0.11	0.06
(b) No Doppler radar, no altimeter	8	5	27.19	0.05	0.15	0.16
(3) (a) Habitat 1 km downrange	3	163	0.44	0.02	0.27	0.03
(b) Habitat 1 km downrange, no Doppler	3	177	0.44	0.02	0.50	0.03
(4) IMU Alignment Sensitivity						
(a) 1/2 × baseline (= 10 arcsec)	5	5	0.54	0.04	0.10	0.06
(a) 2 × baseline (= 40 arcsec)	5	5	0.54	0.05	0.11	0.06
(c) Realign at PDI	5	4	0.54	0.04	0.09	0.06
(d) Realign at PDI (no Doppler)	6	4	0.54	0.04	0.09	0.06

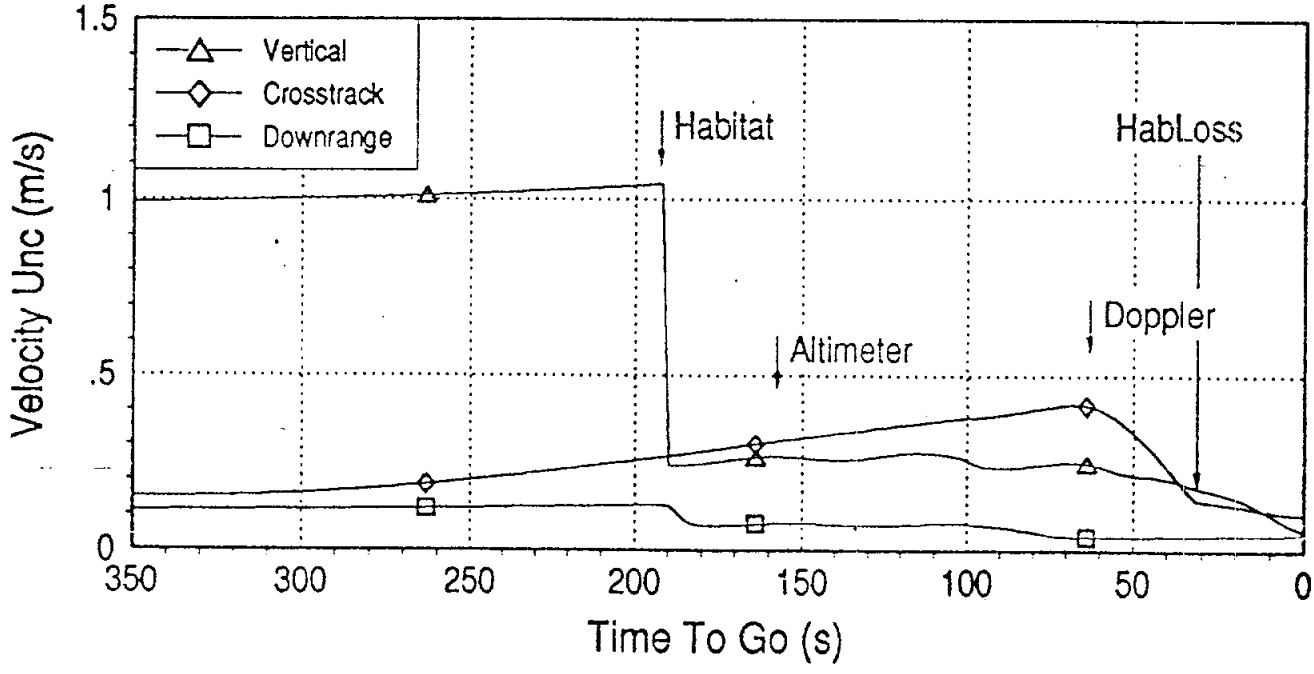
LUNAR DEORBIT TO LANDING TRAJECTORY
LVLH STATE UNCERTAINTY HISTORY



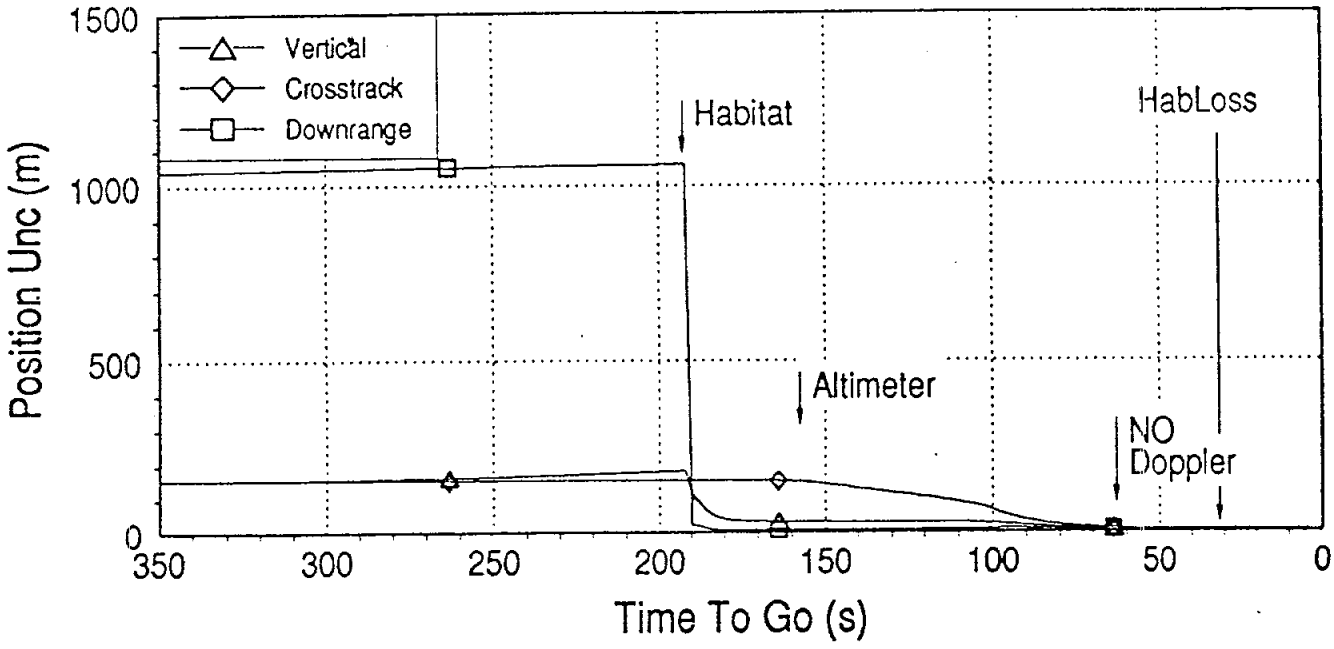


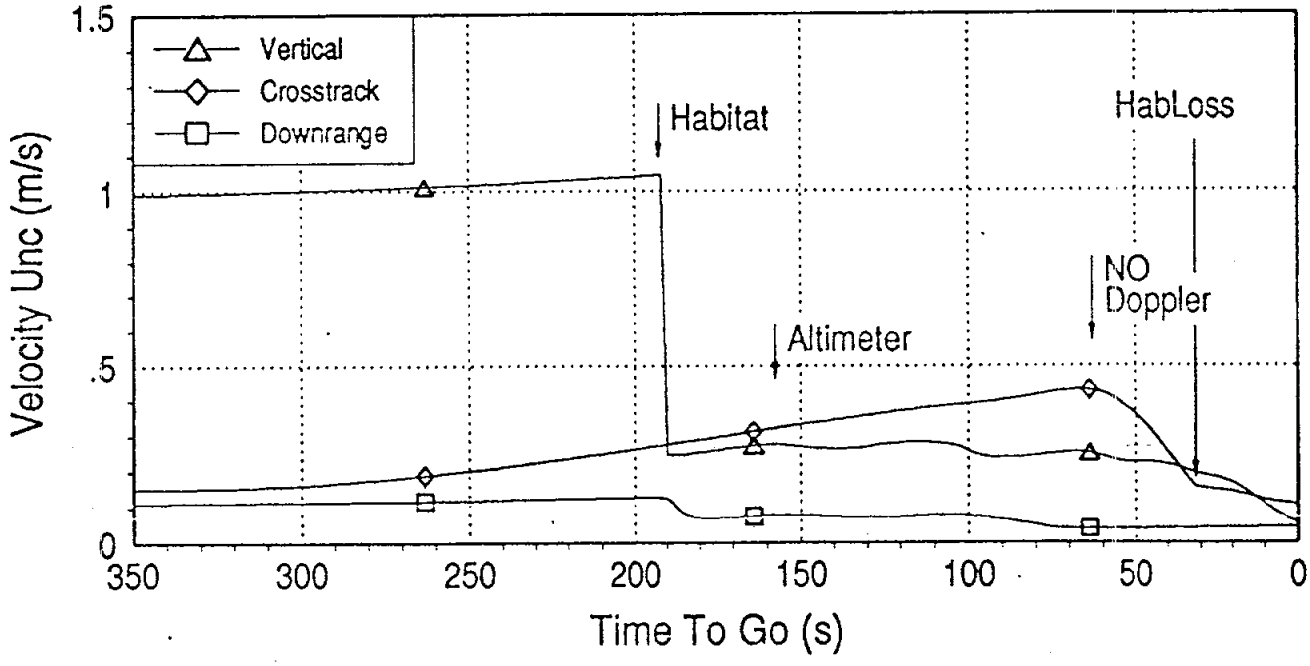
LUNAR DEORBIT TO LANDING TRAJECTORY
LVLH STATE UNCERTAINTY HISTORY FROM PDI





LUNAR DEORBIT TO LANDING TRAJECTORY LVLH STATE UNCERTAINTY HISTORY FROM PDI (NO DOPPLER CASE)





CONCLUSIONS

- The habitat cannot substitute for the radar altimeter because the habitat is offset from the nominal landing site by 1 km

- The habitat may be able to substitute for the Doppler radar depending on the location of the habitat with respect to the landing site
  - — Pure downrange offset provides no crosstrack navigation information
  - — The plots suggest that the crosstrack velocity is derived from the Doppler radar. However, the plot without Doppler radar shows that, in fact, crosstrack velocity information is actually obtained from the habitat

FUTURE WORK

- Review/improve models and assumptions
  - — Particularly gravity and terrain/altimeter models
- Impact of navigation updates on guidance needs evaluation
  - — How 'late' can updates such as radar altimeter be initiated
  - — Can JSC help?
- Include effects of RCS firings on nav during coast
- Consider constraint on radar acquisition based on vehicle attitude (fixed mount radar)
- Model Doppler radar with more fidelity including 3 non-orthogonal beams and correlated measurements

3. Operations

Abstract
Operations Integration Team
First Lunar Outpost (FLO) Study Report

In addition to supporting each of the design teams by providing operations concepts and requirements definition, the operations integration team provided data to the First Lunar outpost (FLO) team consisting of operations assumptions, an integrated operations concept, operations requirements to the First Lunar Outpost Requirements and Guidelines (FLORG), the operations function partitioning analysis, a FLO control center staffing analysis, and a preliminary FLO cost estimate. These studies are included within this section of the report as Attachments 1-6.

Also included in this section are presentations from various team members. "FLO Telecommunications, Information Systems, and Navigation (TISN) and Operations" (D. Hei/GSFC) contains information on the different communication architecture alternatives being considered; "FLO Operations" (Mission Operations Directorate/JSC) contains evolution strategy, operations considerations and risks, and an operations functional breakdown. The "Summary of KSC Activities" (J. Blum/KSC) contains information on the development of facility requirements for various launch vehicle options.

Questions concerning this section should be directed to XE/Barbara Pearson (713/283-5495) or XI/Dave Lamar (713/283-5468).

OPERATIONS INTEGRATION TEAM FLO STUDY REPORT

In addition to supporting each of the design teams by providing operations concepts and requirements definition, the operations integration team provided the following data to the First Lunar Outpost (FLO) team:

1) First Lunar Outpost Operations Assumptions

2) Integrated Operations Concept

3) Operations Requirements to the First Lunar Outpost Requirements and Guidelines (FLORG)

4) Operations function partitioning analysis

5) FLO Control Center staffing analysis

6) FLO Control Center and training outfitting analysis.

The FLO Operations Assumptions document (attachment 1) was distributed in January to the design teams to provide operations considerations to be discussed for inclusion in their design concepts. This document was also used by the operations team as the basis for developing the integrated operations concept, operations requirements and operations cost estimate.

Although the operations team supported both the planetary surface and space transportation segment design teams in the development of operations concepts, an integrated operations concept (attachment 2) was produced to serve as the basis for operations requirements and cost estimate development. The FLO operations concept assumes that highly automated vehicles and surface elements will eliminate the need for routine command and control from the ground. Ground based assistance will only be required to resolve dilemmas resulting from multiple onboard failures. The concept also assumes that operations cost will be reduced by automating much of the ground based planning and monitoring done manually today.

The operations requirements (attachment 3) were provided as an input to the FLORG. These requirements are based on the operations assumptions and concepts. However, additional analysis is required to define the evolution of operations from primarily earth based to space based. As this analysis matures, the operations requirements will be updated to reflect the resulting operations evolution.

An operations function partitioning analysis (attachment 4) was conducted in support of the operations concept and requirements development. Although very immature, this analysis has served to define the FLO operations job and will form the basis for the operations evolution study. Other analyses performed include an estimate of FLO control center positions (attachment 5). This analysis was performed in order to get an estimate of the number of workstations required to support FLO real time operations. The FLO operations cost estimate resulting from the above analysis is provided in attachment 6.

Attachment 1 First Lunar Outpost Operations Assumptions

1.0 GENERIC ASSUMPTIONS

1.1 Safety and Reliability

1.1.1 Safety

Crew safety is our top priority.

All flight and surface elements shall be compliant with crew safety as specified in NASA-STD-3000 with appropriate (TBS) partial gravity amendments.

System robustness should be provided by multiple ways of accomplishing the same function. Systems should be designed such that, when all failure modes and effects are considered, the probability of safe mission completion is better than To Be Determined (TBD) (99 E) percent and the probability of meeting mission objectives is better than TBD percent. The planned response to control any hazard will not depend on Extravehicular Activity (EVA), and fault tolerance will not include planning EVA to replace failed components.

1.1.2 Manrating of Elements

Flight, surface, and ground support elements and robotics providing direct or indirect crew support will be manrated. This rating will be in terms of the vehicle, system or facility as a whole and will be determined according to the effect a failure will have on the safety of the crew.

1.1.3 Safe Mission Termination

The capability will exist for safe mission termination.

1.1.4 Abort Mode Priority

Abort to earth; if possible, has priority over abort modes to the moon.

1.1.5 Crew Rescue

The capability will be provided for the rescue and safe return of the crew from the lunar surface. This capability will include the ability to rescue crewmembers on the surface who may be stranded outside the habitat. This capability must also include the ability to safely return sick or injured crewmembers to earth.

1.2 Crew Health Maintenance

1.2.1 Crew Health Guidelines

Guidelines for maintaining long-term crew health and performance in space and lunar surface environments will be established from human research activities. Implementation of these guidelines is a shared responsibility between the design of the space vehicle, lunar surface elements, and mission operations.

Crew health and performance shall be maintained to the maximum extent possible as

follows:

- Implementation of the guidelines for long term crew health and performance through planned use of countermeasures, vehicle/planet surface equipment, and crew activities.

- Real time monitoring and evaluation of current and projected crew health and performance. Bio-performance data will be monitored locally and remotely.

- Development of mission rules and decision points for continuance, termination, or modification of mission and activity.

- Contingency implementation of available alternate options to the planned countermeasures as required to increase countermeasure effectiveness.

1.2.2 Radiation Protection

There will be a radiation dosage management plan developed premission that covers allowed dosage levels for all flight phases (low-Earth-orbit (LEO) and beyond). The radiation management plan should consider the following:

- A radiation shielded area (storm shelter) will be provided for any area the crew occupies more than TBD (7) days.

- No planned EVA during interplanetary coast,

- Real time monitoring and evaluation of crew exposure,

- Contingency EVA will be based on real time evaluation and impact assessment predictions based on statistical radiation exposure expectations.

1.3 Operations Evolution

Operations capabilities will evolve over the life-time of the program. In particular, as the capability of the Lunar outpost increase, operations will shift from primarily earth based to surface based.

1.4 Maintenance

Flight and surface element design should minimize crew involvement in maintenance. Where crew involvement is required, the design should emphasize intravehicular activity (IVA) maintainability. Hardware not accessible from the inside must be designed for ease of extravehicular activity (EVA) maintenance.

1.5 Automation

Nominal and routine operations will be automated where feasible and desirable. Selected manual and/or ground operations intervention features shall be provided. As appropriate, autonomous operations capability will be provided to permit in-situ control.

1.6 Synergism With On-going Programs

Operations processes utilized by on-going programs such as Space Shuttle and Space Station Freedom (SSF) will be utilized to the maximum extent possible as the platform for the development of Lunar return program operations processes.

1.7 Commonality

To the maximum extent possible consistent with program cost, the Lunar return program elements will utilize common hardware, software, procedures, etc. This concept will reduce the cost of crew and ground support personnel training.

1.8 Verification

All mission support elements will be verified as operational prior to initial required use.

1.9

Vehicle command and control functions initiated by the operations support team will be performed from a centralized control center.

Rationale: Operations cost can be reduced if only one NASA Center is required to perform vehicle control functions. Since the payload and manned portions of this mission will both use the same lander, JSC flight controllers will be responsible for all command and control functions.

1.10

The crew will have some amount of vehicle monitoring capability of critical systems in order to make go/no-go decisions prior to committing to critical maneuvers (TLI, LOI, Landing, etc.)

Rationale: Since the crew personnel will be risking their lives on this mission, they have the right to make critical go/no-go decisions. They will need a certain amount of system data in order to evaluate the vehicle system status.

1.11

The crew will have the capability to initiate or halt the critical auto sequences.

Rationale: Although Earth base flight controllers will have command and control capability, the crew will have the capability to initiate or halt autosequences or manually update maneuver targets. This capability is required to insure the crew's safety and the success of the mission in the event that command and control is lost from the Earth.

1.12

Existing hardware components and systems which are used for the First Lunar Outpost mission will be appropriately modified to allow system checkout, and maintenance by either the crew or by robotic means.

Rationale: Existing hardware may be adequate to support the technical function for which it was designed but it may not be properly designed to allow manual or robotic checkout and maintenance. This capability is essential for FLO to be an evolving and maintainable outpost.

2.0 EARTH TO ORBIT LAUNCH ASSUMPTIONS

2.1 Launch Window

A launch window of TBD (3) hours will be provided daily for TBD (3) consecutive days. At least one launch opportunity will be provided each month.

2.2 Piloted Mission Planning

The piloted mission will be planned for two lunar opportunities after the cargo mission to allow time for cargo vehicle launch slips and habitat checkout.

The piloted mission will not be launched until the habitat has been activated and checked out.

2.3 Disposal of Flight Elements

All expended flight hardware will have the capability to be deorbit by ground command.

3.0 TRANS-LUNAR OPERATIONS ASSUMPTIONS

3.1 Trans-lunar Injection (TLI)

An earth orbit loiter capability of TBD (2) revs will be provided to enable,

- Final system and communications checkout

- Navigation/guidance updates

-Final go/no go decision. 3.2

Earth Based Flight controllers will check out all lander vehicles and payload systems prior to TLI.

Rationale: System checkout is required to insure the proper operation of the vehicle and payload. Probability of mission success can be significantly improved if systems can be checked out in LEO where subsequent maintenance can be performed if required. This assumption implies that vehicle systems must be designed (where practical) for zero G maintenance.

4.0 LUNAR LANDING ASSUMPTIONS

4.1 Prepared Landing Site

Flight vehicle design will not require a prepared lunar landing site.

5.0 LUNAR SURFACE ASSUMPTIONS

5.1 Site Selection

A formal site selection process will be responsible for establishing recommendations for surface sites for landing, permanent outpost, and exploration. The following operational criteria will be considered in the evaluation of potential lunar landing sites:

Capability to conduct voice, data and video communications back to Earth without significant concerns for obstruction by surface obstacles or the lunar horizon.

Outpost is located in an area which allows efficient and safe access to desirable areas for scientific or resource study.

Outpost is located in an area which will promote the outpost physical and technical growth by minimizing site preparation and maximizing the use of in situ resources.

5.2 Surface Stay

The nominal mission will be open ended with a lunar surface stay time not to exceed 45 days plus TBD (7) days for contingencies. Mission continuance decision will be made each day. The capability will be provided to launch within 24 hours of the decision to abort the mission.

5.2

Landing of the first payload lander will be without the help of navigational aids except for surface geometries which were obtained from a precursor mapping mission.

Rationale: Surface coordinates can be selected and programmed into the first payload lander for its use during landing to provide a 1000 meter accuracy. No navigational aids will be required. The payload lander will carry a navigational aid which will be used by the manned lander to improve its accuracy in landing to within 100 meters.

5.3 Equipment Location

Remote scientific and outpost infrastructure equipment will be placed at strategic locations which will allow acceptable access by an EVA crewman or robot such that efficient maintenance and routine manual operations can be performed.

All equipment essential to the safe operation of the habitat must be located within TBD km to allow two EVA crewman the time to reach a failed system, repair it, and return to the habitat. This assumes that the rover may not be available for the crewman's use in a contingency situation.

Rationale: In order to minimize EVA overhead, and maximize crew safety, most of the crews activities associated with maintaining the critical outpost systems or with the performance of periodic tasks must occur near the habitat. The outpost must be strategically located near the area of most scientific interest or which offers the best use of local resources.

5.4

The crew will have the capability to perform a checkout of vital launch systems immediately after landing. This checkout of vital systems will insure that they have the capability to return to Earth.

Rationale: If problems with the return vehicle exist, the crew will want to know them immediately so that appropriate corrective actions can take place. The crew will checkout these systems immediately after landing in the event that the habitat can not sustain them for their stay on the Moons surface.

5.5

Lunar base habitat modules and other safety critical hardware delivered to the surface will

be remotely checked out prior to crew launch from Earth.

Rationale: System checkout is required to insure the proper operation of the habitat prior to the crews arrival. Probability of mission success can be significantly improved if systems can be checked out so that proper maintenance can be performed on any hardware problems that occurred after TLI but prior to the crews arrival.

5.6

The cargo lander will carry a radio beacon and some form of visual cue to enable the following crew to locate its position.

Rationale: Crew vehicle will carry the necessary guidance equipment to detect and follow a beam on the payload lander. A visual cue would be used by the crew to orient their manual traverse to the habitat. This visual cue is especially necessary since the payload lander will be difficult to find since it will be at least 1 Km from the crew when they arrive and will be absorbed by the black and white lunar landscape. Crew may also be able to pin point lander coordinates from lunar orbit before committing to the landing.

5.7

The piloted lander will be powered down to a safe house keeping mode while the crew is on the surface. Health monitoring systems, keep alive power, and the capability to remotely command the vehicle will be maintained.

Rationale: During the crews stay on the surface, they will not have enough time to continuously monitor their return vehicles health. The Earth based flight controllers will perform this function. They will need the capability to remotely monitor all critical systems and to perform remote commanding during system checkout of the unmanned vehicle.

5.8

The use of cryogenics will be minimized on the surface.

Rationale: Cryogenics are difficult to store long term and transfer. With the exception of oxygen for use in the habitat, cryogenic storage will not be a requirement for the outpost.

5.9

The habitat will contain enough consumables and critical spares for a 4 man crew to perform the required IVA and EVA tasks such that the objectives of the 45 day mission can be achieved.

Rationale: The crew must have an adequate supply of consumable to allow the performance of their duties during the 45 day mission. Enough critical spares must be provided to allow maintenance of critical equipment required for the crews safety and for the successful accomplishment of the higher priority objectives.

5.10

The habitat must be designed such that it provides for the evolution of the outpost.

Rationale: Operations time can be significantly reduced by leaving the habitat on the lander since no emplacement or assembly tasks will need to be performed. The habitat must be

removed from the lander if subsequent habitat modules are to connected on future missions. The first habitat module may remain on the lander but it must be designed so that it can be lowered and emplaced on the surface when future outpost expansion is desired.

5.11

Dual airlocks should be provided to give the crew redundant means of accessing a pressurized habitat.

Rationale: Due to the abrasive environment of the lunar dust, it is conceivable that airlock seals will need to be replaced periodically. This type of operation cannot be performed unless the crew has an alternate means of accessing the habitat without requiring its depressurization and forcing all crew members into pressure suits during the maintenance.

5.12

The crew module and habitat must be provided with systems which aid the crew in the transport of equipment and samples to and from the airlock.

Rationale: Both the crew module and the habitat will be left on their landers when they arrive. The crew will be required to carry essential equipment up and down a set of ladder steps each time they leave or enter these facilities. System handling equipment such as pulley systems, lifts slides etc. will be required to help a suited crewman with limited mobility and visibility and to free his hands for his own safety when climbing the ladder.

5.13

Adequate supplies must be provided on the crew module to allow the crew to operate out of the module in the event that the habitat cannot sustain them when they land.

Rationale: When the crew arrives, they may discover a problem with the habitat such that it cannot support them during their surface stay. They will then be required to operate out of the crew module. The crew module must provide adequate life support consumables to give the crew at least one or two EVA's to the habitat in order to repair it if possible. If the habitat cannot be repaired on this mission, the crew must have at least one or two additional two person EVA's available to them to allow some of the mission objectives to be accomplished (i.e. science package emplacement, collect soil samples etc.). A subset of the mission objectives will be labeled as crew module activities which they will perform in case the habitat cannot support them.

5.14

The crew module and the habitat must be designed with common fittings to allow the transfer of life support consumables between the two facilities.

Rationale: System problems may occur on the crew module or the habitat which may cause the early depletion of a critical life support consumable (oxygen, water, power). If common fittings and transport facilities are provided, the crew can share consumables between the two life support systems. For example, cryogenic oxygen could be carried in specially designed transport tanks to the habitat or lander.

5.15

Adequate radiation protection shall be provided by habitat system packaging and geometry

such that the personnel and systems (radiation sensitive avionics, pressure seals etc.) are protected during the manned phase and the systems are protected during the unmanned phase when consumable mass is no longer available to give radiation protection.

Rationale: The consumables which the crew will store in the habitat tanks will also provide them and the avionics systems with radiation protection. After the 45 day mission is completed the crew will have depleted their consumables supply and returned to Earth. The remaining avionics systems will have less protective mass around them to provide radiation protection. These systems must still be protected in order to allow this habitat to be revisited. The remaining structural shell and equipment must provide this protection.

5.16

One, two person, 6 hour EVA will be performed each day during the 45 days on the surface. No inflight EVA will be performed.

Rationale: Enough surface activities will be scheduled such that an EVA activity will be planned for each day.

5.17

The habitat and crew module will be pressurized to the same pressure requirements as the EVA suit in order to minimize EVA preparation time and hazards.

Rationale: Pressure compatibility must be a design goal in order to minimize the risk of "bends" to an EVA crewman and to reduce the EVA preparation time, thus optimizing the EVA operations.

5.18

Provisions for the removal of abrasive lunar dust shall be provided outside and inside the crew module and habitat pressure volumes.

Rationale: Dust build up inside the pressurized areas of the crew module and the habitat will be a severe problem over a 45 day mission unless the proper precautions are made to remove the dust. Some means of removing a substantial amount of dust off of EVA suits should be provided immediately outside the airlocks. The remaining dust can then be vacuumed off of the suits after the crew enters the pressure volume.

5.19

Manned rover excursion will not go beyond a circle which has a 15 km radius as measured from the habitat.

Rationale: In the event of a rover failure, the crew must be within 15 km of the habitat to allow them to walk back. A 15 km radius will give the crew approximately a circle of 95 km in circumference plus two 15 km radial traverses.

5.20

Direct communications between the habitat and an EVA crew member will be maintained at all times.

Rationale: In order to optimize their activities and provide for there safety, direct

communications needs to be maintained with an EVA crewman. based upon this assumption, the EVA voice and data transmitters must have a range of at least 15 Km which is the walk back assumption.

6.0 LUNAR ASCENT ASSUMPTIONS

6.1 Launch Opportunity

The lunar transportation vehicle and operations elements will provide the capability to support launch from the lunar surface within TBD (24) hours of notification.

7.0 EARTH ENTRY AND LANDING

7.1

When possible, return launch windows will be designed to provide for a sun light landing on Earth near the appropriate recovery operations. However, a night landing is acceptable in an emergency situation.

Rationale: In order to insure a safe return and recovery, the crew return vehicle will need to land in sunlight if at all possible. The recovery of a floating crew module will be difficult to accomplish unless satisfactory lighting is available.

8.0 Post - Mission Operations Assumptions

8.1 Mission Evaluation

8.1.1
The operations team will evaluate mission technical and scientific data, procedures, etc. in order to incorporate lessons learned into subsequent missions.

Rationale: A cooperative effort between all technical teams will be required to insure that the lessons learned during flight and surface operations are properly analyzed and incorporated into future designs and operations. The data gathered during the mission will be used to develop design improvements and to reduce operational complexities.

8.1.2
Study samples which are returned from the Moon will be returned to Earth and isolated in simulated lunar atmosphere in order to prevent contamination of these samples.

Rationale: The scientific value of lunar samples brought to Earth facilities for study could be easily lost if they are not properly protected against contamination.

8.2 Operations Evolution

The capability shall evolve to give the flight crews the ability to perform autonomous operational tasks, including but not limited to, space system monitoring, trouble shooting, procedure development, and daily timeline planning.

Rationale: Operations capabilities will evolve over the lifetime of the program. In particular, as the capability of the lunar outpost increases, operations will shift from primarily Earth-based to surface-based. This evolution is required in order to reduce operations overhead and provide autonomy for the surface crew.

Attachment 2 Integrated Operations Concept

OPERATIONS CONCEPT ELEMENTS

The following is a listing of the data elements which should be included in any operations concept. These elements are listed as study inputs and outputs. Each listing is presented in a manner which represents the logical flow of effort or consideration for the concepts development.

Study Inputs

Study inputs are those data elements which are given before the study begins. These may be hard facts or assumptions and should be labeled as such.

1. Guidelines and priorities defined by management.

2. Estimated cost, and schedule constraints.

3. Special technical or programmatic constraints (i.e. only existing technology will be used, minimize EVA overhead, etc).

4. High level assumption about risk (safety risk, risk to program continuance, risk to adhering to cost and schedule). Will operations be conducted to maximize safety regardless of cost or schedule impacts, or will it be conducted to provide some type of safety but with understood safety risks to maintain cost and schedule?

5. Operational objectives to be accomplished (programmatic, scientific, engineering, etc.)

6. Physical Environment (component or human physical environment). (Temp., press., radiation, dust, noise, vibration, G-level)

7. Assumed or specified user needs from crew, science community, engineering, Earth based operators, whom ever.

8. Interfaces (elect. or hydraulic power, thermal control, voice, data, video communications, comm and control between system and base, system and Earth, system and crew, manual, robotic etc.)

9. Concept evaluation criteria (must not consume more than TBD watts, or weight more that TBS lbs, use existing technology, satisfy user needs, add to the accomplishment of mission goals and objectives, adhere to program guidelines).

Study Outputs

The following items are the results from the study and will be used to determine the concepts for operations.

Functional Elements - Based upon the needs of the users and the activities to be performed the functional elements will be defined. (Habitats, rovers, power systems, radiators, antenna, airlock, etc.)

Design Constraints - These will be criteria for future tradeoff studies. These constraints can be presented as a table of functional components versus power, data rate,

EVA time etc.

System Evaluation Critique- This should not be confused with concept evaluation criteria. System evaluation critique is used to grade how successfully a system concept meets a functional need. The concept evaluation criteria states how effective a need must be met. The concept evaluation criteria is the standard, while the system evaluation critique is the grade given for the resulting concept.

Description of How the Strategic Plan and Policies are Reflected -

New Technologies are Identified -

Crew Work Load is Determined - Nominal, maintenance, EVA vs IVA tasks, Science etc.

Robotic Work Load -

Autonomous vs manual activities -

Earth Control Center activities -

System operations Modes (prelaunch, launch, entry, orbit, surface, unmanned, manned, autonomous, etc.)

Contingency Protection - FDIR, autonomous, manual action crew rescue.

Required Crew Skill Mix -

Design Reference Mission including detail timeline -

EVA Constraints -

Maintenance Constraints -

SPACE TRANSPORTATION VEHICLE CONCEPT FOR THE FIRST LUNAR OUTPOST MISSION

The following concept describes the operations of both the unmanned cargo vehicle, and the crew transfer vehicle. The interactions between the vehicle systems, the crew, and the operations support personnel during the in-flight and lunar surface phases of the First Lunar Outpost Mission are discussed.

Overview

One of the chief goals of the FLO program is to automate the vehicle systems such that Earth operational support personnel and infrastructure can be reduced without jeopardizing the success of the mission or the crews safety. Obviously Earth based support cannot be eliminated entirely. Certain mission planning functions, as well as remote vehicle operations will be required. Crew training and consultation activities will also be needed.

Attempting to provide vehicle and crew autonomy also forces the program to incur some additional risk. The checks and balances which are currently provided by support personnel watching the vehicle status and the crews implementation of flight critical procedures could be significantly reduced or eliminated. The program may be forced to reevaluate the risk associated with not having complete autonomous systems as well as consider transferring the planned autonomous functions back to Earth based operators if the cost associated with developing these autonomous systems proves to be prohibitive. Reversing these decisions late in the program will significantly add to the programs cost and schedule. In order to reduce this risk, an evaluation of the progress in providing the desired autonomy must be made during the preliminary design review (PDR) time frame. The decision whether to proceed with planned autonomous system developments or to move them to Earth based operations should also be made at this time.

The following operations concept is based upon the assumption that the vehicles and crews will operate as independent of the Earth flight controllers as practical, and that appropriate autonomous flight systems will be effectively developed and appropriate crew and mission operations support training will be provided in order to reduce the dependency on the ground.

OPERATIONS CONCEPT

Launch Vehicle Operations

The space transportation system will be designed to provide a common launch vehicle capable of launching either the unmanned cargo vehicle or the four-person crew transport vehicle into low Earth Orbit. This vehicle will have autonomous flight control systems. Remote operations from the ground will be limited to crew and vehicle health system monitoring, vehicle performance determination, and range safety monitoring and destruct commanding. Responsibility for the launch vehicle will be handed over from the launch processing team to the mission control team after the vehicle has cleared the launch tower.

Appropriate voice and data links will be provided in order to access vehicle status data, crew health monitoring data, and to communicate with the flight crew. Except for range safety functions, only the crew will have the capability to instigate an ascent abort. The crew will be given adequate data with which to make an abort decision independent of ground operators, although the Earth based flight control team will be given the primary responsibility for making flight rule decisions as long as voice and data links are active.

Crew training will be limited to vehicle system training required for launch processing, prelaunch and post abort safety, and launch vehicle systems monitoring associated with making independent abort decisions.

Mission control training will be limited to appropriate vehicle launch processing, maintenance of communications links with the vehicle, determination of vehicle performance and health, and the evaluation of system and control data associated with making flight rule decisions.

Trans-Lunar and Trans-Earth Operations

The unmanned and manned vehicles will both use similar maneuvers to perform their trans-lunar and trans-Earth flights. Trajectory targets and software sequenced burn commands will be developed and remotely implemented by Earth based operators for the manned and unmanned missions. The crew will also have the capability to load and verify the targets and to manually start or over ride the sequences if required. The crew will have the capability to evaluate a limited amount of vehicle status and trajectory data in order to make their own go-for-burn decisions.

During the three to four day coast phases of the unmanned missions, the vehicle will rely upon its own system robustness and redundancy for system survival. Earth based operations will be limited to mission evaluation and planning. Remote commanding will be used for maintaining communications links, and in controlling the auto sequences required for in-flight maneuvers. Switching to any onboard redundant systems will be performed by autonomous onboard failure detection and reconfiguration software and hardware. Resource management will be autonomously controlled by onboard software and hardware. Earth based operators will periodically evaluate these resources as well as the vehicles health status in order to make mission planning and flight rule decisions. Earth based operators will be trained on the appropriate systems and flight procedures which are required to perform these tasks.

During manned flights, the crew will rely on the onboard autonomous systems to maintain the safety of the vehicle. They will continually analyze the onboard resources for detection of problems and for making flight rule and mission planning decisions. The Earth based operators will aid the crew in these types of evaluations during their coast to and from the Moon. Crew and flight controller training will be required for system evaluation, flight procedures implementation and for resource management and planning.

Provisions for performing EVA's will not be provided during the coast phase when returning from the Moon. The capability to perform a contingency EVA on the way to the Moon will exist since the consumables associated with the depressurization and repressurization of the crew cabin on the lunar surface have not been used. If such an EVA were performed, it would subsequently drive the decision to return to Earth without performing a lunar landing.

Earth based mission support will be limited to console positions and personnel required for the performance of mission critical and safety critical functions. Due to the transient nature of these flight phases continuous support will be required to deal with time-limited mission or safety critical activities. In the event of an emergency, precious time cannot be wasted attempting to contact the appropriate personnel whose presence is required to deal with a problem.

Lunar Descent

The command to start the unmanned vehicle landing sequence will be remotely given by the Earth based flight control team. Since there is no capability for a Earth recovery of the unmanned cargo vehicle, there is no need for a lunar descent abort mode back to Earth. Once the auto sequence for lunar landing is given, the onboard vehicle systems will control the landing. The ground will be given the capability to remotely evaluated the landing site and to command a translation away from any surface obstructions during the final 100 meters of the descent.

The manned vehicle will be monitored by the crew down to the lunar surface. The crew will have the capability to evaluate the landing site during their descent and to redesignate their landing site if an obstacle is seen. The crew will also have the capability to perform a lunar landing abort back to Earth using the ascent engines if a failure occurs with one of the descent engines. The need for the abort is dependent upon when the engine failure occurs during the descent profile. Because of the time delay associated with transmittal time between the vehicle near the Moon and the Earth based operators, this type of abort mode will have to be fully automated or the crew will have to make the abort decision completely independent of Earth support. An onboard abort region determinator as well as appropriate training must be provided in order for the manned vehicle to have this abort capability.

Surface Operations

Once lunar landing has occurred, the manned and unmanned vehicles must be safed and the appropriate systems powered down for the duration of their stay on the surface.

The autonomous systems on the unmanned vehicle will attempt to completely safe the vehicle and turn off all vehicle systems which are independent of the cargo. At this time, Earth based operators will remotely monitor and command the vehicle systems as required to insure that all safing processes have been completed. Once safing has been confirmed, Earth monitoring of the descent vehicle will no longer be needed. The mission operators will then concentrate on remote cargo operations.

The flight crew will have the responsibility of safing their vehicle for the 45 day stay on the surface. Immediately upon landing, the crew will also verify that all systems required for the return trip to Earth are operating properly before the vehicle is placed into a quiescent mode for the surface stay. It is essential that this checkout be performed immediately upon landing to insure that the ascent vehicle is ready to support an any-time abort from the surface. Any problems which might be detected must then receive immediate attention both from the crew and from Earth based operators. It should be noted that the crew cabin can only sustain the crew for 48 hours while they are on the surface. This 48 hour capability exists in the event that the habitat cannot support the crew.

At least two other ascent vehicle checkouts should be performed during the surface stay; one half-way through the mission and the last one just prior to returning to Earth. The mid-mission checkout will be performed either remotely from Earth or by the surface crew. This checkout is primarily required to verify the integrity of the propulsion and life support pressure systems. The pre-ascent checkout will be performed by the crew with Earth operator support as part of the vehicle check just prior to leaving the surface. This will be a complete check of all essential ascent vehicle systems.

During the unmanned periods before the crew arrives and after they leave the ground operations team will have the primary responsibility for periodic evaluation of the cargo systems carried on the unmanned cargo flight. Remote deployment and checkout of the

habitat elements will also be a Earth based operations function. The flight control team will periodically monitor the outpost systems status. Any problems which might be found will be completely analyzed in preparation for possible maintenance or work arounds when the flight crew arrives. The problems detected prior to the crews arrival will affect the manifest of equipment and spares which the crew will carry on their flight.

Earth Landing and Recovery

Earth based operators will provide continuous monitoring of the crew and their vehicle during the return trip home. Prior to landing the mission operators will verify the vehicles trajectory to insure that its flight path will allow it to reach the planned landing site. Mid course maneuvers will be planned and performed as required. The mission operations team has the responsibility to mobilize the recovery forces for both land and water type landings. Coordination of the recovery activities will be the responsibility of the recovery team leader at the landing site. Mission operations will keep the recovery team appraised of the vehicles flight path, and the health of the vehicle systems and the crew. As much as possible, data and voice communications will be maintain throughout the entry and recovery phases between the Mission operations team, crew and vehicle, and the recovery forces.

Due to unforeseen problems during entry, it may take several hours for the recovery forces to reach the crew. The return vehicle will therefor be capable of providing life support and communications functions for at least 24 hours after landing on Earth.

Facilities

When at all possible, existing mission control facilities will be used to support the First Lunar Return Mission. Engineering and operations will attempt to share FLO specific facilities which are required for crew training and system verification. An attempt will be made to share databases between engineering and operations. A special crew cabin simulator will be required to train the crew on nominal and contingency situations. This simulator will be integrated into the control center facilities for integrated and stand alone training. Data from the vehicles will be gathered and distributed by the mission support facilities. Software reconfiguration facilities will also be required to support future missions beyond the first return mission.

Ascent Vehicle/Crew Cabin Training on The Lunar Surface

The need for contingency system training on the surface will depend upon the complexity of the ascent vehicle and crew cabin which the crew must operate. A completely autonomous flying machine with redundant systems should not require any crew participation during the actual ascent or entry maneuvers. As a result the crew should only have to deal with system malfunctions and therefore may only need training to maintain their proficiency at failure recognition and system analysis. This type of training will be performed by a computer based training system which will be part of the workstation environment in the habitat. No special crew cabin hardware mockups (switches, panels, hand controllers etc.) or crew cabin training visits would be required.

OPERATIONS CONCEPT FOR THE FIRST LUNAR OUTPOST SURFACE ACTIVITIES

INTRODUCTION:

Experience has shown that one of the key first steps in any successful systems engineering approach is the development of an operations concept. All of the assets and planned activities associated with conducting the proposed program are studied in an integrated fashion to insure that the program objectives can be efficiently achieved. The following operations concept will deal exclusively with the lunar surface based operations.

SCOPE:

The surface operations concept will take a look at the surface systems, earth based operations, crew activities, and programmatic guidelines needed to achieve the proposed activities during the first lunar return mission. Subsequent missions will not be discussed except from an operations evolution perspective.

DISCUSSION:

Pre-mission Preparation:

As discussed in the FLO Design Reference Mission, the unpiloted cargo mission will be launched two months before the crew is launched. The mission operations people will begin preparing for the surface stay before either mission is launched however. Detailed crew procedures as well as ground procedure development will begin approximately two years before the cargo vehicle and the crew are launched. This allows the operations personnel approximately one year to develop the preliminary set of procedures which will be used for the initial set of crew and flight controller training.

Crew and Flight Controller Earth Based Training:

The crew and flight controllers will spend the next year performing integrated and non integrated training tasks. Simulations will be used for both the unmanned cargo mission and the piloted mission. Where possible, training facilities which are currently being used for Space Shuttle and Space Station will also be used to provide FLO inflight and surface training. Specialized mockups may be required to support this training, however. These simulators will be built to support safety related training and to support the crew and flight controllers in dealing with the future evolution of the surface assets. Some of this training will require integrated participation by people from various engineering and science disciplines who have a surface system or payload to be deployed or operated. Detailed systems and maintenance training for the crew and flight controllers who are involved in the remote operation and checkout of surface systems will be performed at the vendors facility.

Crew Training on The Surface:

Some of the techniques developed for training before the mission starts will be applied to training on the surface. While it is desirable to minimize the crews training, it must be recognized that any contingency procedures which will require quick, decisive and efficient implementation by a crew member will also require periodic training. This is especially true for long duration missions such as FLO's 45 day surface stay. Comments from Shuttle flight crews indicates that crew personnel will lose proficiency in performing complicated tasks within two weeks if training is not provided. Also any new tasks which

the crew may be asked to perform will require some limited amount of training. An example of this would be special maintenance tasks which requires a crew member to become familiar with the detailed operation or construction of a surface device. The amount of required training will depend on the complexity of the task and its associated time criticality and safety risks. If the critical systems on the flight vehicles, and the habitat are properly designed for autonomous reconfiguration, and failure tolerance, then crew training can be limited to problem recognition, analysis and system familiarization. Existing computer workstations in the habitat will double as computer based trainers during off duty hours.

Cargo Deployment, Activation and Checkout Prior to Crew Arrival:

Once the cargo mission is landed, the Earth based flight control teams will begin the process of remotely deploying cargo elements. Those elements which cannot be completely automated will be deployed and checked out via remote command and control from Earth. Even fully automated systems will require ground operations personnel to monitor data and evaluate the outpost deployment and activation in preparation for the crews arrival.

Ground operators will use remote command and control functions to checkout potentially failed systems in order to determine the source and mode of the failure. If these problems cannot be corrected remotely, the piloted mission will be replanned and the appropriate crew procedures developed. Hardware will be manifested on the first piloted mission to enable the crew to fix the problems after they arrive. Remote command and control will be used for specialized robotics tasks such as camera and antenna control, unmanned rover control, and surface system reconfiguration for special tests etc.

Once the initial checkout of the surface systems is completed, periodic monitoring and control functions will be performed. These Earth based tasks will not require around the clock participation by a full team of flight controllers. Selected positions may be required to monitor and control critical systems however, especially if a problem is detected.

Operations After Crew Arrival:

A special team of Space Transportation vehicle flight controllers will support the in flight phase of the mission from Earth launch through TLI, LEO and lunar landing of the FLO crew. The crew will safe the piloted vehicle and configure it for surface habitation in the event that they must move back in due to an unforeseen problem with the surface habitat. Also in the initial few minutes after landing, the crew, and the ground support will evaluate the ascent vehicle systems parameters for any possible problems which must be corrected for ascent. If such a problem is detected, immediate action will be required to insure that the vehicle can safely return the crew back to Earth at any time. After the vehicle is secured, all four crew members will walk or ride the unpressurized rover to the waiting surface habitat.

Most of the habitat activation will have been performed by automated systems and by ground command before the crew arrives. Some amount of crew involvement may be required especially if the crew is required to emplace surface assets away from the habitat. or if the crew is required to perform maintenance on a habitat system immediately after they arrive. The crew will verify that the habitat and its associated system elements are ready to safely and effectively support human live and the mission objectives.

Once the habitat is operationally verified, a set of return trips to the crew module will occur. These trips will require the crew to transport the remaining consumables from the crew

cabin to the habitat. At an appropriate point, the crew lander/ascent vehicle will be configured to support a low power surface stay, and the crew will return to the habitat to continue with the remainder of the surface operations.

The ascent vehicles onboard fault detection isolation and reconfiguration hardware and software will monitor, and if possible reconfigure the onboard systems in order to protect the health of critical ascent vehicle systems. Earth based controllers will review vital systems data at least every TBD days (approximately every 14 days) in order to evaluate the health of the ascent vehicle systems while the crew is living in the habitat. If ascent vehicle problems do occur, it is essential that crew action be taken immediately rather than waiting until they have finished their 45 day surface stay. These problems will be fixed using a mix of the crews skills and the advice from ground based systems experts. Only those flight critical systems which can be effectively powered up without worry about potentially harmful start transients will be assessed. The intent here is to catch problems early which may be due to the extreme pressure and temperature environment on the lunar surface and at the same time minimize the risk of introducing any new problems during start transients associated with these periodic checks.

Obviously there will be a category of failures which cannot be dealt with by crew or remote Earth command and control. These types of failure modes will be analyzed during the design phases of the program and will be considered as part of the program safety risk assessments.

Up to this point, the crew has been busy with activation and checkout of the habitat and surface systems, transport of consumables to the habitat from the crew cabin, and reconfiguring the crew ascent vehicle for its surface stay. Once all of these tasks are accomplished the crew will start the daily operational and scientific tasks which will make make up the largest portion of the mission objectives. Typical daily timelines will provide for the following: crew health maintenance, periodic system checkout and maintenance, conducting scientific exploration and experiments, EVA traverses, crew sleep and eat periods. These EVA tasks will consist of two EVA crew members with one IVA crew support. An EVA crew person will not normally perform two EVA days in a row. Based upon flight experience in other programs, each person will perform to an EVA every third day. With two teams of people this equates to 4 two person EVA's within every 6 days.

5.2.3 Earth Based Operations Segment

(330A as currently written)

(Add after 330A)

The Earth based operations segment shall encompass all activities and related services conducted on Earth in direct support of the First Lunar Outpost (FLO) mission and to which resources shall be acquired to meet facility, outfitting, hardware systems, software systems, and manpower requirements.

Rationale: To establish the scope and objective for Earth based operations requirements.

(Place after above)

The Earth based operations segment shall provide capabilities to support the following mission functions: a) Mission Operations, b) Ground Operations, c) Recovery Operations, d) Network Operations, e) Integrated Logistics, f) Special Services, and g) any other specific function subsequently identified and added.

Rationale: The overall mission level operations support requirements need to be established to justify the subsequent implementation.

(464 as written)

(Change 465 to)

The Earth based operations segment shall provide the capability to support 60-day launch centers.

Rationale: no change

(place after 465) The Earth based operations segment shall provide the capability to evolve to eight flights per year and 45-day launch centers through capability augmentation and learning experience by TBD timeframe.

Rationale: The Earth based operations segment must have adequate capability support the required flight rate.

(place after above) The Earth based operations segment shall provide the capability to accommodate any flight/surface mission sequence of cargo and piloted missions.

Rationale: Follow on missions are currently undefined. Therefore, this requirement serves to anticipate cost-effectiveness of implementing capability up front rather than incremental expansion. This requirement also provides follow on mission flexibility.

Requirements for the acquisition and utilization of mission support services from provider/sources external to SEI program shall be in accordance with NASA Management Instructions, Memorandum of Agreements, Memorandum of Understandings, or Program Directives as instructed by the Associate Administrator for Exploration.

Rationale: Associate Administrator for Exploration is responsible for policy and external management interfaces.

Each Earth based operations function, as appropriate, shall develop support concepts (timelines, resources, flows, etc.) in such a manner to isolate the effects of unplanned operations and to eliminate the propagation of schedule impacts for expected contingencies.

Rationale: Providing margin (schedule, resources) for each Earth based operations function is necessary to insure some reserve capacity to accommodate contingencies and still meet the mission schedules.

The Earth based operations segment shall provide communications capability to support efficient operations.

Rationale: The capability to communicate between space and ground assets are required for safe and efficient operations.

The Earth based operations segment shall have the capability to perform end-to-end command and data verification testing.

Rationale: Establishes that all independent links of the communication system are integrated and functional.

The Earth based operations segment shall be implemented such that it is tolerant to TBD (1 E) failure without jeopardizing crew safety. The operations system shall be capable of recovering to full operations capability within TBD minutes.

Rationale: Establishes failure tolerance and down time requirements.

5.2.3.1 Mission Operations Sub-Segment

(change 162b to) The mission operations sub-segment shall provide the capability to provide planning and analysis, simulation and training, mission support and control, facility operations, post-flight support functions for First Lunar Outpost Missions.

Rationale: The level of implementation of this requirement will be based upon the need to provide command and monitor functions in support of First Lunar Outpost missions and upon the crews capability to perform autonomous operations.

(add after above) The mission operations sub-segment shall provide the capability for mission support and control of dormant elements on the lunar surface between crew missions.

Rationale: Dormant surface elements will need to be monitored to determine readiness to support the next crew mission and to maintain on going surface operations which are independent of crew visits.

The mission operations sub-segment shall provide capability for real-time operations to accommodate routine, operationally and unscheduled operations.

Rationale: Provides flexibility and evolution. Routine operations are those that are periodic and well understood. Complex operations are those that are new or difficult. Unscheduled activities are deviations from nominal plans. This capability should allow for near term (e.g. daily) planning and the ability to handle unforeseen situations.

The mission operations sub-segment shall have the capability to transition from primarily Earth based control to flight and surface based control and autonomy.

Rationale: Operations capability will evolve over the life- time of the program with operations shifting from primarily Earth based to vehicle/surface based. This evolution is required to reduce operations overhead.

5.2.3.1.1 Mission Planning and Analysis

(change 467 to) The mission planning and analysis element shall provide the capability for planning each First Lunar Outpost mission.

Rationale: Activities such as manifesting of spares and mission elements, science operations, maintenance actions, crew operations, and consumables management will need to be planned before each mission. The capability to for real-time replanning will also be required.

(place after above) The mission planning and analysis element shall provide for the preparation of operations support systems to support First Lunar Outpost missions.

Rationale: Operations facilities and flight elements will need to be configured to support each First Lunar Outpost mission.

The mission planning and analysis element shall provide capability to develop mission operation data sets for space, earth, supporting systems and interfaces.

Rationale: Data are needed to test, diagnose, validate, certify, train and operate space, earth and supporting systems and interfaces prior to and during the mission.

The mission planning and analysis element shall provide the capability and support required to develop operational flight and surface sequence of events and timelines.

Rationale: Real-time mission operations will require nominal and contingency event sequences and timelines. This must be prepared in advance to optimize system performance and to identify and react to off-nominal situations. The allocation of this capability among earth, flight, surface and automated systems is TBD.

The mission planning and analysis element shall provide support to SEI strategic and tactical planning.

Rationale: TBS

The mission planning and analysis element shall plan for early mission termination such that abort modes to the earth have priority over those to the moon.

Rationale: Abort modes to the moon will require additional action to return the crew to earth.

The mission planning and analysis element shall support the capability for one planned two-person EVA of eight hours suit duration every TBD (2) days. No in-flight EVA's will be planned.

Rationale: Provides a balance between crew time and crew health and safety.

The mission planning and analysis element shall design First Lunar Outpost missions such that adequate windows are provided to accommodate delays in the launch preparation process. The earth return launch window shall be designed to optimize recovery operations for either land or water landings.

Rationale: Allowances must be made to accommodate problems which will occur during the launch preparation process. Earth return trajectories should be designed, if possible, to allow daylight, land landings.

The mission planning and analysis element shall provide for an earth orbit loiter capability prior to TLI.

Rationale: Earth orbit loiter will allow time for final systems checkout before committing to lunar transfer.

The mission planning and analysis element shall design FLO missions to accommodate a lunar surface stay of up to 45 days. The mission will be designed to allow earth return within 24 hours of a decision to abort the mission.

Rationale: The nominal surface stay is 45 days, however, the capability to abort the mission at anytime is required.

5.2.3.1.2 Simulation and Training

The simulation and training element shall provide the capability to train the crew and mission operators.

Rationale: The FLO mission will involve highly complex hardware and software systems. The crew and mission operators will need to be trained to operate, trouble-shoot, and maintain this hardware and software.

Science training shall be accomplished by the prime investigator.

Rationale: Science training needs to be conducted by the organization most familiar with the science equipment and objectives.

5.2.3.1.3 Mission Support and Control

(change 469 to) The mission support and control element shall provide the capability to monitor and control the space transfer and surface elements during First Lunar Outpost missions.

Rationale: Even for highly automated flight and surface elements, the need will exist for ground support as a result of system failures or contingencies.

(place after above) The mission support and control element shall have the capability to capture science telemetry data and route the data to remote science centers.

Rationale: Many scientists will want to operate their experiments from remote science centers. This will require the routing of science data (and mission data) to these centers.

The mission support and control element shall have the capability for permanent archive of mission data and short term archive of science data.

Rationale: Data will be required for post-mission evaluation and science analysis.

The mission support and control element shall have the capability of determining space transportation vehicle and surface element mass properties.

Rationale:

The mission support and control element shall support overall vehicle and surface element configuration management.

Rationale:

The mission support and control element shall have access to onboard, surface, and Earth based navigational data in order to provide navigation updates to the space transfer vehicles, surface rovers, and scientific instruments.

Rationale: Navigation and state vector updates will be required to support the onboard systems.

The mission support and control element shall provided with the capability to monitor solar activity and provide the crew a warning of impending solar events.

Rationale: Early warning of solar events is required to enable the crew to take adequate safety precautions.

The mission support and control element shall have the capability to remotely power, checkout and operate the lunar systems critical to human occupation.

Rationale: The lunar habitat and other surface systems critical to human occupation must be verified operational prior to the crewed mission.

The mission support and control element shall have the capability to remotely operate the lunar rover.

Rationale: Many of the tasks to be carried out on the Moon can benefit from the ability to teleoperate the rover. This capability would decrease crew time requirements and increase science yield.

The mission support and control element shall provide the recovery team with earth landing site information (including location and weather conditions).

Rationale: Abort conditions may result in a change in landing site.

The mission support and control element shall be provided the capability to provide weather predictions for earth landing sites.

Rationale: Trajectory constraints limit the earth landing site capability once TEI has occurred. Accurate landing site weather predictions will be required to plan TEI and mid-course corrections.

5.2.3.1.4 Facility Operations

The facility operations element shall have the capability to maintain operations support systems, hardware, software, and facilities.

Rationale: Operations capabilities must be maintained in order to support the program.

The facility operations element shall have the capability to defer planned, non-mandatory, maintenance.

Rationale: Adds resiliency for unplanned events.

5.2.3.1.5 Post-Flight Support Operations

The post-flight support element shall evaluate mission technical and science data, procedures, etc. and incorporate lessons learned into subsequent mission planning.

Rationale: Data gathered during a mission will be utilized to improve system design, procedures, etc. and will reduce operations complexity.

The post-flight support element shall provide for the protection of samples returned from the Moon.

Rationale: The scientific value of samples returned from the Moon could be lost if they are not properly protected against contamination.

The post-flight support element shall provide for the processing of post-flight products and archived data as required.

Rationale: Mission and scientific products and data must be processed in order achieve maximum benefit from the mission and to prepare for subsequent missions.

5.2.3.2 Launch Site Operations Sub-segment

The launch site operations sub-segment shall consist of those activities/functions to be performed at the launch site. These include:

1) Cargo processing

2) launch vehicle

3) Integrated vehicle processing and transport

4) Pad operations

5) Operations coordination and interfacing

6) Launch site logistics operations.

Rationale: Defines the scope of launch site operations.

Total launch site operations sub-segment processing shall not require launch centers in excess of ninety-one (91) days.

Rationale: The launch site operations sub-segment processing timeline shall be capable of evolving to sixty-day (60) launch centers through capability augmentation and learning by TBD timeframe.

Rationale: Provides a goal for launch site operations evolution.

The launch site operations sub-segment processing timeline shall be based on a work schedule of five (5) days per week and up to three (3) shifts per day availability.

Rationale: This provides a reference which is consistent with recommendations made by KSC. This reference will provide a twenty-five (25) percent manpower margin to mitigate schedule slip due to contingencies.

5.2.3.2.1 Cargo Processing

The cargo processing element shall provide the capability to perform the following services and related accommodations:

1) transport of individual payload elements of the flight manifest from the point of arrival to the cargo integration location,

2) Organizational level maintenance, including storage of mission spares, tools, and other mission support equipment.

3) Integration of the following, as required:

   a) user and/or mission specific equipment into the mission elements,

   b) mission elements into an assembled cargo.

4) Hazardous material handling, i.e., storable propellant tanking and installation of ordinance.

5) Transport of the integrated payload configuration from the cargo integration location to the integrated vehicle processing location.

Rationale: Defines the scope of cargo processing operations.

5.2.3.2.2 Launch Vehicle Processing

The launch vehicle processing element shall provide the capability to perform the following services and accommodations:

1) Transport of individual launch vehicle core and booster elements from the point of arrival to the vehicle processing location.

2) Perform stacking, mating, integration, and testing of the launch vehicle.

3) Perform hazardous material handling, i.e., installation of ordinance.

5.2.3.2.3 Integrated Vehicle Processing

The integrated vehicle processing element shall provide the capability to perform the following services and accommodations:

1) Mating of payload configuration to launch vehicle.

2) Integrated checkout of the launch configuration.

3) Transport of the integrated vehicle to the launch pad.

Rationale: Defines the scope of pad operations.

5.2.3.2.4 Launch Control

The launch control element shall provide and execute the command and control procedures for the launch countdown that:

1) Test and checkout the launch vehicle and supporting systems.

2) Sequence the vehicle from ground support to internal functioning.

3) Follow the launch commit criteria.

The launch control function shall terminate when the vehicle has cleared the pad.

Rationale: Defines the scope of launch control operations.

5.2.3.2.5 Launch Coordination and Interfacing

The launch coordination and interfacing element shall provide the capability to coordinate launch activities and to interface with the following:

1) Mission operations for launch coordination and handoff.

2) Recovery operations for activity coordination in the event of a pad abort off the launch vehicle.

3) Network operations for tracking and data acquisition.

4) Department of Defense for weather forecast and analysis data and for range safety.

5.2.3.3 Recovery Operations Sub-segment

The recovery operations sub-segment shall encompass the services, accommodations, and equipment for physically locating, securing, and delivering the crew, flight hardware, and mission products from the Earth landing location. This includes:

1) Capability to locate and return flight crew from the Earth landing location to permanent habitable facilities.

2) Capability to retrieve returned flight hardware from the Earth landing location and deliver to continental U.S. facilities or other logistics transportation depots.

3) Capability to recover and protect mission samples/data until delivery to the end user.

Rationale: Defines the scope of the recovery operations sub- segment.

Recovery capability and support shall be provided for both planned and unscheduled (early) lunar return and credible abort conditions.

Rationale: The capability must exist to recover the crew under all credible conditions.

5.2.3.4 Network Operations Sub-segment

5.2.3.5 Integrated Logistics Sub-segment

The integrated logistics sub-segment shall provide the capability to manage the (Earth-based) logistics activities for SEI FLO elements. This activity shall include:

1) Terrestrial packaging, handling, storage, and transportation,

2) Terrestrial off-line maintenance, storage, training, etc. facilities as appropriate,

3) Inventory management,

4) Support equipment,

5) Technical data and documentation system,

6) Procurement support.

Rationale:

Attachment 4 Operations Functional Partitioning

FLO Ops Functions	Activities	LMCC Role	SOC Role	Crew Role	On Board Systems	Function Automated	Function Eliminated
Logistics	Inventory/stowage management Resupply/return planning	Prime for resupply		Prime for surface logistics in real time		Prime - Tool for crew support (Inventory management)	
Training	Lesson development Crew/operator training In-situ training	Prime		Prime - Involved		Onboard or surface trainer	
Medical							
Network	Scheduling	Prime					
Engineering Support		Prime					
Natural Environment	Weather prediction Radiation monitoring	Prime					

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FLO Ops Functions	Activities	LMCC Role	SOC Role	Crew Role	On Board Systems	Function Automated	Function Eliminated
Surface System Ops	LTV System Monitoring	Prime	N/A	Backup	N/A	Prime (crew not available for quick response)	N/A
	LTV System Data Analysis	Prime	N/A	Backup	N/A	Advisory Role	N/A
	LTV System Commanding	Prime for time critical	N/A	Backup - crew not available for quick response	N/A	Prime for vehicle safeing	
	Performance Analysis	Prime	N/A	Backup	N/A	Advisory Role	N/A
	Failure Analysis	Prime	N/A	Backup	N/A	Advisory Role	N/A
	Anomaly Responses	Prime	N/A	Backup	N/A	Prime for vehicle safeing	N/A
	Resource Management	Prime	Provide inputs	Prime	Provide inputs	Tool for crew use	N/A
	Configuration management	Backup	N/A	Backup	N/A	Prime	N/A
	Procedures development	Prime	N/A	Review & provide inputs	N/A	Tool available for crew use	N/A
Science Ops	Experiment monitoring Experiment data analysis Experiment commanding Experiment performance and failure analysis Experiment anomaly response Experiment servicing Resource management Experiment deployment Procedures development/revision Experiment activation and checkout Robotic ops	Prime		Prime - Crew and LMCC share load		Backup - Crew Tool	
Maintenance	Procedures development Workaround development Maintenance Robotic ops EVA ops IVA ops	Backup withprocedure development support		Prime		Backup - Tool for crew support	

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FLO Ops Functions	Activities	LMCC Role	SOC Role	Crew Role	On Board Systems	Function Automated	Function Eliminated
LTV System Ops	LTV System Monitoring	Backup	N/A	Backup	N/A	Prime	N/A
	LTV System Data Analysis	Prime	N/A	Backup	N/A	Advisory Role	N/A
	LTV System Commanding	Prime	N/A	Prime for time critical functions	N/A	Prime for vehicle safeing	N/A
	Performance Analysis	Prime	N/A	Backup	N/A	Advisory Role	N/A
	Failure Analysis	Prime	N/A	Backup	N/A	Advisory Role	N/A
	Anomaly Responses	Prime	N/A	rime for time critical functions	N/A	Prime for vehicle safeing	N/A
	Resource Management	Prime	Provide inputs	Backup	Provide inputs	Advisory role	
	Configuration management	Backup	N/A	Backup	N/A	Prime	N/A
	Procedures development	Prime	N/A	Review & provide inputs	N/A	Tool available for crew use	N/A
Flight Dynamics Ops	Attitude control Navigation and structure update Debris avoidance	Prime Prime Prime		Backup Backup Backup		Advisory Backup Backup if radar available	
Mission Planning	Activity (long term) Plan Execution Plan Abort Planning Mission Replanning	Prime Backup Prime Prime		Backup Prime Backup Backup		Backup Prime Backup Backup	

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Attachment 5 FLO Control Center Positions

PROPOSED CONSOLE POSITIONS FOR LMCC

LOCR	LSSCR	MISSION PHASE						
		EA	TLI	LL	SURF	LA	TEI	ELAND
OPS DIR - 1		U/M	U/M	U/M	U/M	M	M	M
CAPCOM - 1		M	M	M	M	M	M	M
PAO - 1		M	M	M	M	M	M	M
SURGEON - 1	1	M	M	M	M	M	M	M
GC - 1	4	U/M	U/M	U/M	U/M	M	M	M
OPS PLAN - 1	4	U/M	U/M	U/M	U/M	M	M	M
DATA PROC - 1	1	U/M	U/M	U/M	U/M	M	M	M
COMM/TRACK - 1	1	U/M	U/M	U/M	U/M	M	M	M
EPS - 1	1	U/M	U/M	U/M	U/M	M	M	M
ENV/TCS - 1	1	U/M	U/M	U/M	/M	M	M	M
FLT DYN - 2	2	U/M	U/M	U/M	U/M	M	M	M
TRAJECTORY	2	U/M	U/M	U/M	U/M	M	M	M
GNC/PROP - 1	1	U/M	U/M	U/M	U/M	M	M	M
*ASC/DSC ENG -1	1	U/M	U/M	U/M	M	M	M	
*EVA - 1	1				M			
MECH/ACT/ROB-1	2	U/M	U/M	U/M	U/M	M	M	M
PAYLOADS - 1	1	U/M	U/M	U/M	U/M			

TOTAL FRONT ROOM WORKSTATIONS	16
SUPPORT ROOM WORKSTATIONS	22
TOTAL WORKSTATIONS	38

Other Facilities In The LMCC

OPS/ENGR INTF - 2
ENGR DATA ROOM - 3
MULTIPURPOSE VIDEO

TOTAL CONSOLES OUTSIDE OF FLT CONTROL AREA = 5

PHASE DESCRIPTION

EA - EARTH ASCENT
TLI - TRANS-LUNAR INJECTION
LL - LUNAR LANDING
SURF - SURFACE OPS
LA - LUNAR ASCENT
TEI - TRANS-EARTH INJECTION
ELAND - EARTH LANDING

* The Ascent Entry Specialist monitors the Earth Ascent engines and the Earth Entry. This position may also be responsible for the Lunar descent and ascent. During surface activities, this console will be used by the EVA control team. The Ascent/Entry Specialist Support room consoles will be manned by the EVA team as well during the surface activities.

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Attachment 6 FLO Operations Cost Analysis

FLO MISSION OPERATIONS COST
(FY92 $M, No Reserve)

	FY93	FY94	FY95	FY96	FY97	FY98	FY99	FY00	TOTAL
Operations Capability Development									
Training Facility Outfitting	19	50	85	91	86	62	4	0	$397M
Mission Control Facility Outfitting	15	55	85	107	107	97	4	0	470
Operation Analysis/Supporting Development	1	4	8	10	12	10	0	0	45
Mission Operations									
Mission Planning & Analysis	0	0	0	0	0	2	40	40	82
Simulation & Training	0	0	0	0	0	2	10	10	22
Command & Control	0	0	0	0	0	2	7	7	16
Facility Operations	0	0	0	0	0	20	80	80	180
Recovery Operations	0	0	0	0	0	0	10	10	20
Postflight Support	0	0	0	0	0	0	2	2	4
TotaB	35	109	178	208	205	195	157	149	1236

U.S. Gov't

FLO TISN & Operations

Data Systems & Communications Architecture

Dr. Donald J. Hei, Jr./GSFC

Elaine Stephens/JSC

April 2, 1992

FLO Communications Architecture Alternatives Considered

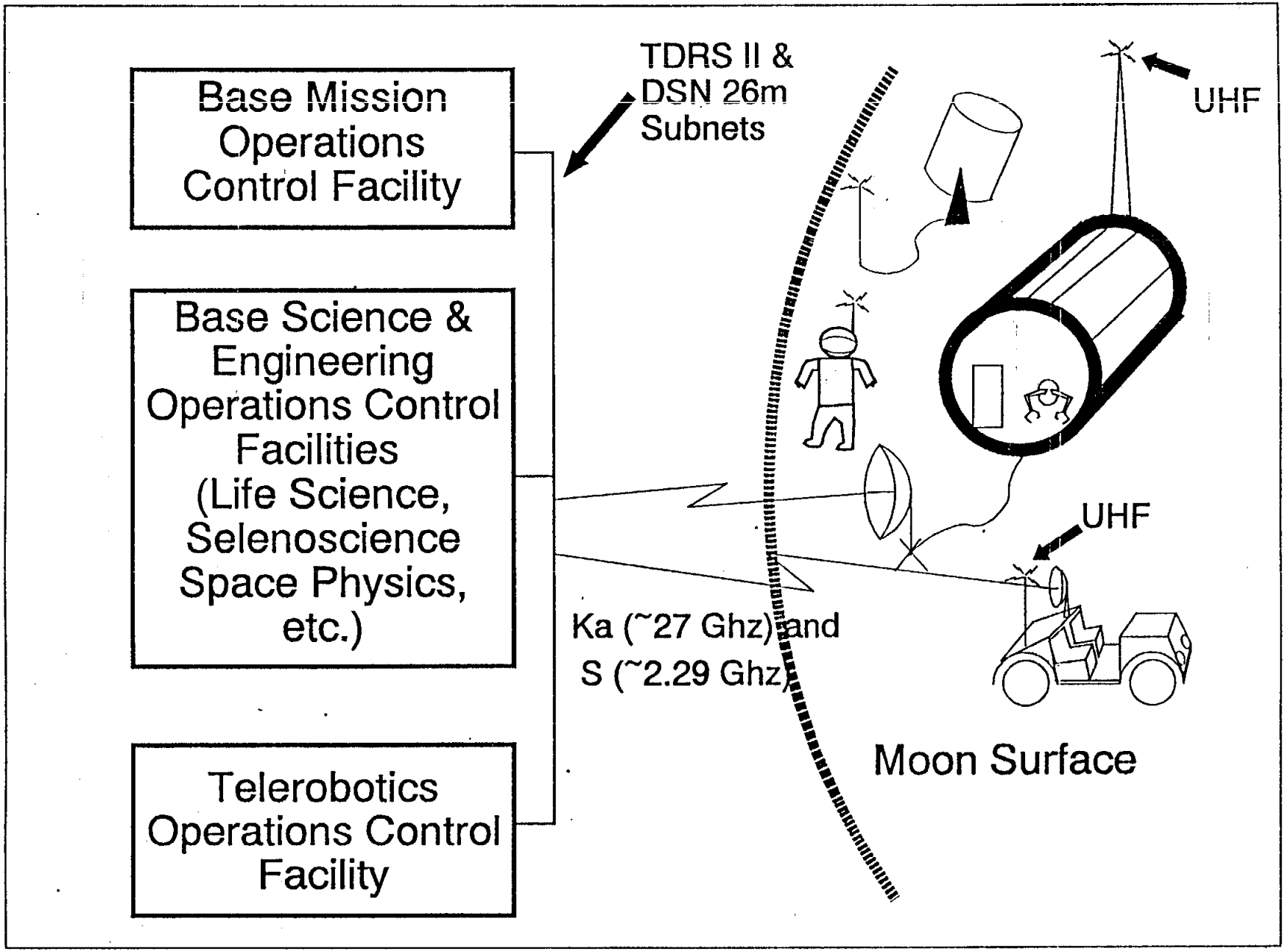
EXPLORATION PROGRAMS OFFICE

- ALL ARCHITECTURES CONSIDERED SUPPORT NEARSIDE LUNAR COMMUNICATIONS ONLY
  - FARSIDE COMM REQUIRES RELAY SATELLITES AT L-2
  - CONTINUOUS POLAR AND LIMB COMM DIFFICULT
- Exclusive DSN 34m Subnet per 90 day study Option 5
  - Substantial costs from '95-'99
  + Ample forward & return telemetry

FLO Communications Architecture Alternatives (continued)

EXPLORATION PROGRAMS OFFICE

- Exclusive TDRSS II service
  - Current loading model depicts planned TDRSS II constellation as subscribed
  - Substantial cost (less than DSN 34m), schedule risk in adding additional TDRS
  - \+ Ample return telemetry
- Project dedicated & operated ground network
  - Substantial cost from '95-'99, schedule risk
  - \+ Ample forward & return telemetry


Base Mission Operations Control Facility
Base Science & Engineering Operations Control Facilities (Life Science, Selenoscience Space Physics, etc.)
Telerobotics Operations Control Facility
TDRS II & DSN 26m Subnets
UHF
UHF
Ka (~27 Ghz) and S (~2.29 Ghz)
Moon Surface

FLO Communications Architecture Alternatives (continued)

EXPLORATION PROGRAMS OFFICE

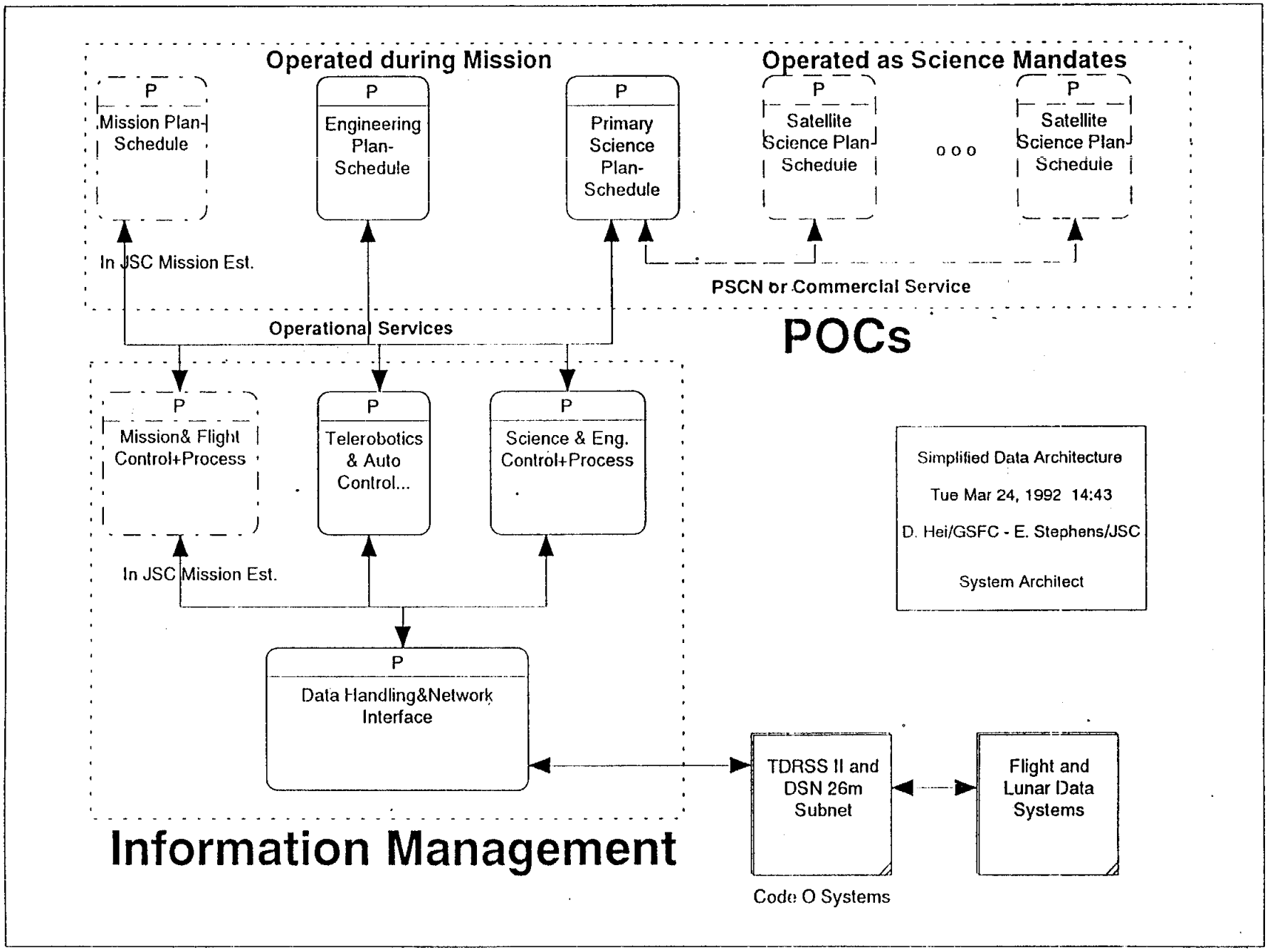
✔• Combining TDRS II Ka-band high data rates, DSN 26m S-band low data rates

- \- Additional operations complexity of combined networks
- 0 Needs feasibility & implementation scope study
- \+ Low probable implementations cost from '95-'99
- \+ No substantive Code O networks assets increase
- \+ Low recurring cost overhead if EXP schedule slides
- \+ Ample return telemetry

FLO Operational Data Systems Architecture

EXPLORATION PROGRAMS OFFICE

- Developed by top-down analysis of preliminary FLORG
- Constructed so that costs could be derived by analogy to existing data systems
  - Decomposed to detail level sufficient to develop cost estimate
  - Architecture shown "rolled up" to detail of 92-1 POP submittal
    - * POCs
    - * Information Management
    - * TDRSS II & DSN 26m network


Operated during Mission
Operated as Science Mandates
P
Mission Plan-Schedule
P
Engineering Plan-Schedule
P
Primary Science Plan-Schedule
P
Satellite Science Plan-Schedule
o o o
P
Satellite Science Plan-Schedule
In JSC Mission Est.
PSCN or Commercial Service
Operational Services
POCs
P
Mission& Flight Control+Process
P
Telerobotics & Auto Control...
P
Science & Eng. Control+Process
Simplified Data Architecture
Tue Mar 24, 1992 14:43
D. Hei/GSFC - E. Stephens/JSC
System Architect
In JSC Mission Est.
P
Data Handling&Network Interface
TDRSS II and DSN 26m Subnet
Flight and Lunar Data Systems
Code O Systems
Information Management

FLO Operational Data Systems Architecture (Continued)

EXPLORATION PROGRAMS OFFICE

- Hidden line blocks indicate costs developed and included within the JSC Mission Operations Estimates
- Dashed line blocks denote POC elements with special non-continuous operations status that reduces systems development and recurring cost
- Data system architecture modular anticipating :
  - Re-use of EDOS in "Data Handling & Network Interface"
  - Economies of scale in "Planning and Scheduling" & "Control and Processing"
  - Next generation Networks Control Center for scheduling TDRS II & the DSN 26m Subnet
  - Future Lunar Outpost operations extensions of FLO capability

FLO TISN Studies & Results

EXPLORATION
PROGRAMS OFFICE

Study	Results
Communications architecture	Implementation Estimates
DSN 34m upgrade	~$1B based on 90 day study
Additional TDRS II	~$500M wedge & schedule bind
Project 20m network	~$300M based on 1991 study
Combined TDRS & 26m DSN	~$50M (needs refinement)
Operational Data Systems	Implementation Estimates (with good engineering)
POCs	~$150M (excludes JSC mission)
Information Management	~$350M (excludes JSC mission)

FLO TISN Studies & Results (Continued)

EXPLORATION PROGRAMS OFFICE

Study	Results	
Very preliminary TDRSS II & DSN 26m feasibility	Rover	Ka return rate of ~24Mbps for 1m antenna and 20W transmitter S TDRS II return rate of 100Kbps for 1m-20W (S 26m return rate of 1Mbps)
	FLO	Ka return rate of ~100Mbps for 2m antenna and 20W transmitter S TDRS II return rate of 500Kbps - 2m-20W transmitter (S 26m return rate of 5Mbps)
	Science	S 26m => 100Kbps with 0.5m & 10W transmitter

FLO Operational Communications and Data Systems Architecture

Dr. Donald J. Hei, Jr./GSFC & Elaine Stephens/JSC - April 2, 1992

FLO Communications Architecture Alternatives

Only lunar neared service was studied for costing purposes. If fared communications is required, one or more relay satellites will be needed orbiting the Lagrangian L2 point. (While low altitude lunar orbiters are technically possible as communications satellites, operations and satellite complexity, and satellite visibility from various locations preclude the feasibility of this option.) These satellites can be less complex and massive that the existing TDRS (weighing about 3/4t and needing about 750KW each based on "Lunar Communications System: Trades Between Ground-Based and GEO-Based Architectures", CSC/TR-91/6084, May 1991 special report for GSFC). Continuous polar and limb communication is more difficult. The solutions depends on the details of the surface site visibility from Earth and from possible relay satellite orbits.

1. The first alternative discussed was the DSN 34m subnet as proposed for the "90 day study Option 5". This is the option that was most probably used as the basis for the Code O budget lines. Implementation would require substantive cost from 1995-1999. However, this option would supply ample forward and return telemetry with very low mass and power needs for users COMM.

2. The second alternative discussed was TDRSS II lunar service. The current TDRS-II workload model projects TDRSS to be fully subscribed during the First Lunar Outpost (FLO) period of operations. Adding an additional satellite to the planned constellation should cost less that implementing a 34m DSN. However, the TDRS implementation schedule does not currently include and additional TDRS II launch for about 2000. Consequently, this option would introduce schedule risk. TDRS II should provide ample return telemetry at Ka band for reasonable user mass and power requirements. (Based on the reference cited above, <150kg and consumed power <300W would accommodate outpost needs of about 100Mbps.) Forward telemetry support was not studied, but may be a limitation due to limited TDRS-II transmitter power.

3. The next alternative discussed was the construction of a project dedicated 20m ground station network. Based on the reference cited above, this option would cost slightly less to implement than launching an additional TDRS. It would require substantive cost from 1995-1999, and particularly the international arrangements could present schedule risk. However, this option would provide ample return telemetry for about the same user COMM subsystem requirements as TDRSS-II, and would supply ample forward telemetry.

4. The forth and last alternative is the teams' recommendation for the FLO. This alternative would (1) make use of the existing DSN 26m subnet for needs which can be practically served by that system. And (2) utilize TDRS II capabilities to serve the occasional high rate user on a TDRS prioritized schedule basis. This alternative needs additional study. It will introduce operational complexity due to the mixed network, and schedule uncertainty for TDRS II time. However, this alternative has very low implementation cost from 1995-1999. No substantive additional Code O network assets would be required. Consequently this alternative also has the lowest

overhead if the EXP FLO schedule should slip. It will also provide ample telemetry coverage (by the DSN 26m subnet) and an ample FLO return telemetry rate (~100Mbps) as discussed in the TDRS II option above.

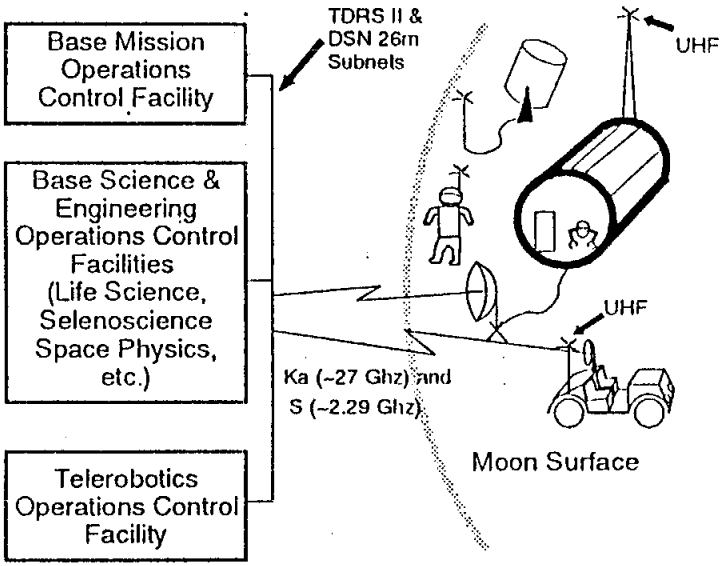


Figure 1: Communications Architecture Concept

Figure 1 illustrates this lunar Communications support concept. Within the vicinity of the FLO, as for shuttle EVA, UHF is utilized to affect voice and data communications. The astronauts' transceivers are limited by power to 1km range, so that relays, such as through the rover, must be utilized whenever the crew is beyond 1km of a UHF tower. It is anticipated that a tower will be included at the habitat and with the piloted vehicle, enabling the crew to transit from the lander to the habitat with a continuum of communications. It is also anticipated that the crew lander will have a high gain transponder capable of serving as an emergency backup to the system on the habitat, reducing mission risk.

The team recommendation (Telemetry would be transferred via the DSN 26m subnet, whenever practical. Intervals of high rate data would be scheduled via TDRS-II) was accepted with clarification. In the course of the discussion, J. Green decided that FLO implementations should plan to use the DSN 26m subnet. If requirements are identified and validated that cannot be reasonably accommodated via the DSN 26m subnet, their means of accommodation will be decided later. The decision to defer accepting the TDRS-II recommendation was based on uncertainty in both user COMM hardware mass estimates, and Congressional support for the current TDRS-II implementation schedule.

Data Systems Architecture

The functional requirements for communications architecture discussed above were derived as part of the operational data systems analysis. This analysis started by analyzing the preliminary First Lunar Outpost Requirements and Guidelines document, available in January, 1992. From this document, a data base of communications and data systems functionality was developed. Then a functional allocation among earth and space systems was created. The functionality and allocation was reviewed with level II and III teams. Then a straw design was created for costing purposes. The design was detailed enough to allow comparison to existing operational data systems. Consequently, the parametrics of the cost model should be reasonably good (for Lines of Code (LOC), and experienced based functionality.) A functional representation of this architecture is shown in Figure 2. The figure is constructed to correspond with the GSFC POP 92-1 submittal. The area identified as POCs represents mission, engineering, and science planning, scheduling, and command validation. This activity precedes and follows mission activities. The section labeled Information Management represents functions that actively control mission activities, process and present information to operators and engineers/scientists, and detect and manage faults. The Communications Infrastructure is labeled as Code O Systems.

Flight and lunar data systems are indicated as a compound external block in the lower right hand corner of the figure. Hidden lines (short - long - short dashes) indicate blocks accounted for in the JSC Mission Operations POP Estimates. Dashed lines indicate satellite science facilities.

These satellite science POCs can be located anywhere, and are not required to be on line continuously. They can be interfaced to the information system via commercial or government owned networks, saving operations cost.

The cost estimates in the GSFC POP 92-1 pop for implementation of the POCs and Information Systems should be regarded as place holders or wedges. It is probable that all of the functions indicated in the fugue should be located at GSFC. For example, the mission functions should probably be with mission control, presumably at JSC. Also, if the DSN 26m subnet serves most of the FLO COMM needs, it is likely that most of the Code O system augmentation cost would be spend through JPL.

The data system architecture indicated in the figure must be (functionally) supported regardless of the FLO implementation. The architecture is modular, and was created to optimize benefit from (1) re-use of modern data systems (e.g. EDOS) and (2) efficient design which utilizes similarity of functions like planning and scheduling to affect economies of scale (modular designs that emphasize re-use of code). The cost model used to develop the GSFC POP 92-1 inputs has taken credit for an overall savings of 1/3 due to these measures. It will be necessary to systems engineer throughout the implementations so that these savings are realized.

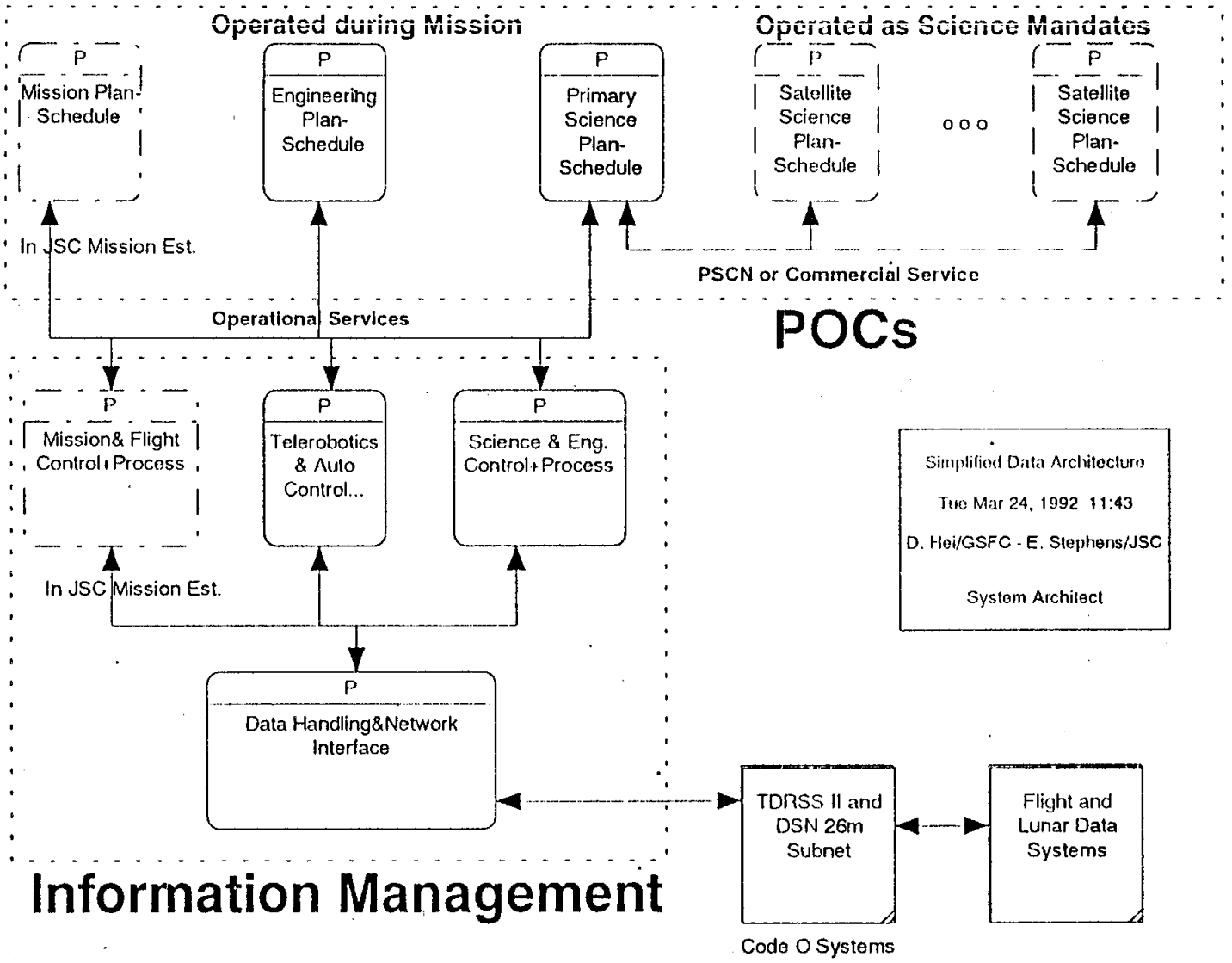


Figure 2: Simplified Data System Architecture

FLO TISN Studies & Results

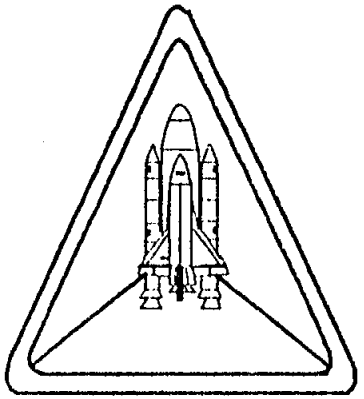
Study	Results
Communications architecture DSN 34m upgrade Additional TDRS II Project 20m network Combined TDRS & 26m DSN	Implementation Estimates ~$1B based on 90 day study ~$500M wedge & schedule bind ~$300M based on 1991 study ~$50M (needs refinement)
Operational Data Systems POCs Information Management	Implementation Estimates (with good engineering) ~$150M (excludes JSC mission) ~$350M (excludes JSC mission)
Very preliminary TDRSS II & DSN 26m feasibility	Rover Ka return rate of ~24Mbps for 1m antenna and 20W transmitter S TDRS II return rate of 100Kbps for 1m-20W (S 26m return rate of 1Mbps) FLO Ka return rate of ~100Mbps for 2m antenna and 20W transmitter S TDRS II return rate of 500Kbps - 2m-20W transmitter (S 26m return rate of 5Mbps) Science S 26m => 100Kbps with 0.5m & 10W transmitter

Table 1: FLO Operational Communications and Data Systems Studies and Results

The communications architectures studies were focused on developing implementation and operations cost estimates. Since Code O chose to provide the JSC/GSFC team little visibility into its cost projections, other than the cost data provided during the 90 day study, that information was used to develop the DSN 34m upgrade cost estimate. These data were reconciled with the May, 1990 budget preview, and were found to agree within ~10%. The cost estimate for an additional TDRS-II was based on a $200M expendable launch cost plus a wedge cost of #300M for another TDRS-II. These numbers are expected to go down after award of the C/D TDRS II contract. The project 20m network estimates, as discussed above, were based on a 1991 study comparing the cost of earth stations to a new dedicated relay satellite network. The implementation cost for the combined DSN 26m subnet and TDRS-II was based on engineering judgment: if by further study, the costs were demonstrated to be more than ~$50M, selection of this alternative ought to be revisited. Operational data systems estimates were based on comparison of LOC for existing operational data systems except for telerobotics and automated systems (these estimates were based on engineering judgment). Credit was taken for reduced hardware cost due to use of modern hardware, and for reuse of EDOS data systems, commercial software where appropriate, and careful implementation of functions so engineered re-use of similar software can be accomplished. Overall savings by this new way of doing business is estimated to be at least 1/3.

MISSION OPERATIONS DIRECTORATE

ADVANCED PROJECTS



SHUTTLE

SPACE EXPLORATION INITIATIVE (SEI)
FIRST LUNAR OUTPOST (FLO) OPERATIONS

April 2, 1992

DA15/T. W. Eggleston

OPERATIONS EVOLUTION VISION

- REDUCE COST AND MANPOWER REQUIRED TO PERFORM MISSION OPERATIONS IN ORDER TO BUILD WEDGE FOR SEI OPERATIONS.

- EVOLVE MISSION OPERATIONS FROM PREDOMINANTLY GROUND BASED TO SEMI-AUTONOMOUS SPACE BASED (REQUIRED FOR MARS MISSIONS).

OPERATIONS CONSIDERATIONS AND ASSOCIATED RISKS

- HOW INDEPENDENT THE CREW WILL BECOME ON THE LUNAR SURFACE IS DEPENDENT UPON THE FOLLOWING FACTORS:
  - THE AMOUNT OF RISK TO CREW SAFETY AND MISSION SUCCESS THAT THE FLO PROGRAM IS WILLING TO SIGN UP FOR AND ADVERTISE TO THE PUBLIC AND CONGRESS.
  - LEVEL OF AUTONOMY AND RELIABILITY THAT CAN BE AFFORDED AND BUILT INTO THE SURFACE AND FLIGHT VEHICLE SYSTEMS.
  - PROPER CREW SKILL MIX WHICH WILL ENABLE THEM TO OPERATE, TROUBLE SHOOT, AND MAINTAIN THE MISSION SYSTEMS.
    - -- SELECT AT LEAST ONE CREW MEMBER WITH NATURAL MECHANICAL SKILLS AND PROVIDE APPROPRIATE TRAINING.
  - AMOUNT OF LOGISTICS TOOLS PROVIDED TO THE CREW WHICH WILL ENABLE THEM TO PLAN DAILY ACTIVITIES, TRACK SPARE EQUIPMENT, DEVELOP PROCEDURES, TRAIN FOR NEW ACTIVITIES OR MAINTAIN PROFICIENCY IN CONTINGENCY OR COMPLEX PROCEDURES.
  - IMPROPER IMPLEMENTATION WHEN PROVIDING CREW AUTONOMOUS OPERATIONS WILL PLACE THE FLO PROGRAM AT RISK.
    - -- COST - NEW TECHNOLOGY DEVELOPMENTS.
    - -- SCHEDULE - IF COST FOR NEW TECHNOLOGIES BECOMES PROHIBITIVE, THEN PROGRAM MUST SIGN UP TO ADDITIONAL RISK OR BASED UPON A PREDEFINED PROGRAM MILESTONE DECIDE TO PROVIDE GROUND MISSION SUPPORT.
    - -- SAFETY - AUTOMATION MAY NOT PROPERLY PROVIDE CHECKS AND BALANCE WHICH CURRENTLY EXISTS BETWEEN CREW AND GROUND CONTROLLERS.

VISIC 2005:
AN OPERATIONS CONCEPT

- MULTIPURPOSE CONTROL CENTER ENABLES CONTROLLERS TO HANDLE SPACE ELEMENTS FOR SEVERAL PROGRAMS; ROUTINE, PREDICTABLE FUNCTIONS HANDLED AUTOMATICALLY.
- COMPLEX, MULTIPERSON PROBLEM SOLVING (E.G., FAULT RECOVERY, MISSION PLANNING) PERFORMED OR AIDED BY ARRAY OF SUPPORT TOOLS FOR FASTER EXECUTION.
- MULTI-MISSION ARCHITECTURE IN PLACE THAT PERMITS EFFICIENT, EVOLUTION OF RELIABLE OPERATIONAL SYSTEMS AS TECHNOLOGY AND THE MISSION MODEL CHANGE.
- UTILIZE DISTRIBUTED, POWERFUL WORKSTATIONS OPERATED BY GROUND OPERATORS. SCIENTIST PARTICIPATE IN OPERATIONS FROM REMOTE SITES.
- DETAILED FLIGHT AND GROUND DESIGN DATA CAPTURED IN ON-LINE DATA BASE TO SUPPORT ALL ASPECTS OF OPERATIONS. (MISSION PLANNING, PROCEDURE DEVELOPMENT,AND TRAINING).
- SPACECRAFT AUTONOMY ELIMINATES NEED FOR ROUTINE COMMANDING FROM THE GROUND, AND REDUCES IMPACTS TO CREW TIMELINE.
- INTERACTIVE TRAINING SYSTEMS INTEGRATE DESIGN DATA BASE AND HIGH SPEED MODELS; HIGH-FIDELITY TRAINERS ACHIEVED WITHOUT REPLICATING ACTUAL FLIGHT VEHICLE ENVIRONMENT (CURRENTLY VERY EXPENSIVE).
- CREW USES SURFACE BASED TRAINING TOOLS FOR PROCEDURE DEVELOPMENT AND SKILL MAINTENANCE.
- CREW IS RESPONSIBLE FOR THEIR OWN INVENTORY MANAGEMENT.
- CREW HAS ACCESS TO ENGINEERING AND OPERATIONAL DATA BASES FOR USE IN SYSTEMS ANALYSIS AND PROCEDURE DEVELOPMENT.

U.S. Gov't

FLO Operations Strategy

- EARLY OPERATIONS ANALYSIS OF SYSTEM CONCEPTS AND DESIGNS. DESIGN FOR OPERABILITY OVER ARCHING PRINCIPLE OF ALL FLO SYSTEMS (FLIGHT AND SURFACE).
- MAXIMIZE SHARING OF EXISTING SHUTTLE AND SPACE STATION CONTROL CENTERS AND TRAINING SYSTEM HARDWARE/SOFTWARE WHERE FEASIBLE.
- MAXIMIZE SYNERGY BETWEEN SHUTTLE, SPACE STATION, AND SEI INCLUDING USE OF COMMON SYSTEMS WITHOUT BUILDING DEPENDENCE ON EXISTING ON-GOING PROGRAMS.
- MAXIMIZE USE OF COMMON ENGINEERING AND OPERATIONS FACILITIES FOR HARDWARE/SOFTWARE INTEGRATION, VERIFICATION, AND TRAINING. SHARING OF FACILITIES ENCOURAGED.
- UTILIZE AUTOMATION, KNOWLEDGE BASED SYSTEM, VIRTUAL REALITY ENVIRONMENT, WORK STATION BASED TRAINING WITH SIMULATED VIDEO GRAPHICS, ETC. TO PROVIDE AUTONOMOUS SPACE BASED CREW TRAINING.
- HIGHLY AUTOMATED SPACE TRANSPORTATION VEHICLES WITH ONBOARD VEHICLE HEALTH MANAGEMENT, FAULT.DETECTION AND RECONFIGURATION WITH GROUND AND CREW COMMAND/CONTROL MONITORING AND OVERRIDE CAPABILITY.
- SURFACE HABITAT DESIGNED FOR SELF CHECKOUT/ACTIVATION WITH GROUND MANAGEMENT VIA REMOTE COMMAND AND CONTROL PRIOR TO HUMAN LAUNCH COMMIT.
- OPERATIONS SYSTEMS DESIGNED TO ALLOW TRANSITION AND PHASING FROM LUNAR OPERATIONS TO MARS MISSION SUPPORT.

FIRST LUNAR OUTPOST COMMAND & CONTROL ASSUMPTIONS

EARTH-BASED COMMAND AND CONTROL TASKS INCLUDE:

- MISSION PLANNING AND ANALYSES
- GROUND CONTROL OF AUTOMATED AND CREWED/CREW-TENDED VEHICLES
- ANOMALY OR MALFUNCTION RECOGNITION AND RESOLUTION
- PROCEDURES GENERATION AND VERIFICATION
- TELEOPERATION OF ROVER EXCURSIONS
- LOGISTICS PLANNING
- SPACE COMMUNICATIONS

FLO OPERATIONS STUDIES PERFORMED

- OPERATIONS FUNCTION PARTITIONING BETWEEN MISSION CONTROL, SCIENCE CENTER, AND ONBOARD CREW/AUTOMATED
  - FIRST CUT FOR EARLY FLO MISSION COMPLETED
  - ANALYSIS OF EVOLUTION PATH REQUIRED
- OPERATIONS CONCEPT AND REQUIREMENTS DEFINED
  - BASED ON ABOVE
  - OPERATIONS EVOLUTION NEEDS TO BE CONSIDERED
- CONTROL CENTER STAFFING ANALYSIS
  - ESTIMATE BASED ON ABOVE COMPLETED
    - UNMANNED OPERATIONS (12 OPERATIONS POSITIONS, 19 SUPPORT POSITIONS)
    - MANNED OPERATIONS (16 OPERATIONS POSITIONS, 22 SUPPORT POSITIONS)
- CONTROL CENTER AND TRAINING OUTFITTING ANALYSIS
  - BOTTOMS UP ESTIMATE BASED ON ABOVE STUDIES
  - TRAINING WILL BE REQUIRED FOR TRANSPORTATION VEHICLE AND SURFACE SYSTEMS
  - BASIS FOR POP 92-1
- FACILITY IMPACT ANALYSIS
  - NO NEW BUILDINGS REQUIRED

REMAINI G ACTIVITY

- COMPLETE ANALYSIS OF EVOLUTION TO DISTRIBUTED OPERATIONS
- REVISE OPERATIONS CONCEPT AND REQUIREMENTS TO ACCOUNT FOR EVOLUTION
- WORK WITH CODE R AND CODE D TO PROVIDE SUPPORT FOR REQUIRED OPERATIONS TECHNOLOGIES.
  - ADVANCED TRAINING SYSTEM
  - FAULT DETECTION AND RECOVERY AUTOMATION
  - AUTOMATED PLANNING AND PROCEDURES GENERATION
- REVISE FACILITY REQUIREMENTS FOR NEW OPERATIONS CONCEPT
- MISSION ELEMENT OPERABILITY AND MAINTAINABILITY ANALYSIS.
- HELP PROGRAM IDENTIFY RISKS AS AUTONOMOUS OPERATIONAL CAPABILITY IS DEVELOPED.

OPERATIONS ISSUES

- LEVEL OF CREW AUTONOMY
- AMOUNT OF RISK TO CREW SAFETY AND MISSION SUCCESS THAT IS ACCEPTABLE
- NUMBER OF EVA'S WHICH CAN BE REALISTICALLY PERFORMED DURING THE FLO MISSION
- EVALUATION OF MAINTENANCE REQUIREMENTS

ADDENDUM

OPERATIONS FUNCTIONS BREAKDOWN

FLO Ops Functions	Activities	LMCC Role	SOC Role	Crew Role	On Board Systems	Function Automated	Function Eliminated
[illegible]	Inventory/stowage management Resupply/return planning	Prime for resupply		Prime for surface logistics in real time		Prime - Tool for crew support (Inventory management)	
Training	Lesson development Crew/operator training In situ training	Prime		Prime - Involved		•Onboard or surface trainer	
Medical							
[illegible]	Scheduling	Prime					
Engineering Support		Prime					
External Environment	Weather prediction Radiation monitoring	Prime					

OPERATIONS FUNCTIONS BREAKDOWN

FLO Ops Functions	Activities	LMCC Role	SOC Role	Crew Role	On Board Systems	Function Automated	Function Eliminated
[illegible] System Ops	LTV System Monitoring	Prime	N/A	Backup	N/A	Prime (crew not available for quick response)	N/A
	LTV System Data Analysis	Prime	N/A	Backup	N/A	Advisory Role	N/A
	LTV System Commanding	Prime for time critical	N/A	Backup - crew not available for quick response	N/A	Prime for vehicle safeing	
	Performance Analysis	Prime	N/A	Backup	N/A	Advisory Role	N/A
	Failure Analysis	Prime	N/A	Backup	N/A	Advisory Role	N/A
	Anomaly Responses	Prime	N/A	Backup	N/A	Prime for vehicle safeing	N/A
	Resource Management	Prime	Provide inputs	Prime	Provide inputs	Tool for crew use	N/A
	Configuration management	Backup	N/A	Backup	N/A	Prime	N/A
	Procedures development	Prime	N/A	Review & provide inputs	N/A	Tool available for crew use	N/A
[illegible] Ops	Experiment monitoring Experiment data analysis Experiment commanding Experiment performance and failure analysis Experiment anomaly response Experiment servicing Resource management Experiment deployment Procedures development revising Experiment activation and checkout Robotic ops	Prime		Prime - Crew and LMCC share load		Backup - Crew Tool	
Maintenance	Procedures development Workaround development Maintenance Robotic ops EVA ops IVA ops	Backup withprocedure development support		Prime		Backup - Tool for crew support	

OPERATIONS FUNCTIONS BREAKDOWN

FLO Ops Functions	Activities	LMCC Role	SOC Role	Crew Role	On Board Systems	Function Automated	Function Eliminated
LTV System Ops	LTV System Monitoring	Backup	N/A	Backup	N/A	Prime	N/A
	LTV System Data Analysis	Prime	N/A	Backup	N/A	Advisory Role	N/A
	LTV System Commanding	Prime	N/A	Prime for time critical functions	N/A	Prime for vehicle safeing	N/A
	Performance Analysis	Prime	N/A	Backup	N/A	Advisory Role	N/A
	Failure Analysis	Prime	N/A	Backup	N/A	Advisory Role	N/A
	Anomaly Responses	Prime	N/A	rime for time critical functions	N/A	Prime for vehicle safeing	N/A
	Resource Management	Prime	Provide inputs	Backup	Provide inputs	Advisory role	
	Configuration management	Backup	N/A	Backup	N/A	Prime	N/A
	Procedures development	Prime	N/A	Review & provide inputs	N/A	Tool available for crew use	N/A
Flight Dynamics Ops	Attitude control	Prime		Backup		Advisory	
	Navigation and structure update	Prime		Backup		Backup	
	Debris avoidance	Prime		Backup		Backup if radar available	
Mission Planning	Activity (long term) Plan	Prime		Backup		Backup	
	Execution Plan	Backup		Prime		Prime	
	Abort Planning	Prime		Backup		Backup	
	Mission Replanning	Prime		Backup		Backup	

MISSION OPE[illegible]IONS CONCEPT

* OVERVIEW- Mission operations will consist of three basic steps which will be applied both before the mission and during the mission.

  * Mission Planning - The development of time lines, crew and ground procedures to be used during the mission.

  * Training - Flight crews as well as mission support personnel will require training in any specialized tasks which they may need to perform.

  * Implementation - Actual hands on implementation of the operations associated with the accomplishing the FLO mission objectives.

* Each step will involve differing degrees of ground and crew interaction as well as vehicle system autonomy.

* How independent the crew will become on the lunar surface is dependent upon the following factors:

  * The amount of safety risk that the FLO program is willing to sign up for, and advertise to the public and congress as program risks

  * Level of autonomy that can be afforded and built into the surface and flight vehicle systems.

  * Having the proper crew skill mix which will enable them to operate, trouble shoot, and maintain the mission systems.

  * Amount of logistics tools provided to the crew which will enable them to plan daily activities, track spare equipment, develop procedures, train for new activities or maintain proficiency in contingency or complex procedures.

* MISSION PLANNING

* Pre and post mission planning will consist of taking the program objectives and reference design missions to develop a logical order for accomplishing these goals. Time lines will be built for both remote operations during the unmanned cargo mission as well as any follow on remote operations which will be required after the crew has finished their surface stay.

* Inflight mission planning will require ground support since crew will not have the complete capability to perform all aspects of their mission analysis required for trans-lunar or trans- earth activities.

* Surface activity planning - Crew will use surface based planning tools to develop daily time lines. Ground will provide goals and objectives which the crew will use as input to the planning algorithm.

* EARTH BASED TRAINING

* When practical to do so, existing Shuttle, Station training facilities will be used to train both flight crews and mission controllers.

* When required, and to save cost, specialized systems and science training will be performed at the vendors facilities.

* LUNAR SURFACE TRAINING

* A means of providing the crew with proficiency training on time critical, and safety critical tasks shall be provided.

* A means of training or reviewing new tasks which are complex or which the crew has had no prior training will be provided. (Example, maintenance on unfamiliar systems elements).

* CREW SELECTION

* At least one crew member will have natural mechanical abilities and will have specialized systems and maintenance training.

* Crews which have experience in ~~flew~~ FLO missions will be encouraged to maintain their proficiency and to participate in future lunar missions.

  -- Minimize crew training to experienced crews

  -- Build expertise in ~~flow~~ FLO operations.

* IMPLEMENTATION OF MISSION OBJECTIVES

* Earth based operations will cover the remote monitoring, command, and control of unpiloted vehicles such as the cargo lander and the habitat during unmanned phases.

  * Space transportation operations will require a full time set of operators due to the time criticality of events associated with space flight.

  * A small contingent of operators will be required to coordinate Earth based operations when surface problems are detected.

  * Surface operations will allow a significant reduction of Earth based operators needed for continuous monitoring once cargo activation and checkout is complete.

  * Surface systems data will be transmitted to Earth computerized analysis systems which will look for failure trends and alert the proper support personnel.

* IMPLEMENTATION OF MISSION OBJECTIVES (Continued)

* Earth based operators will support piloted missions during all phases of the mission but to different degrees.

  * Space transportation phases will require full Earth operations support due to the time critical nature of events associated with space flight and the limited operational capabilities on board the transportation vehicle.

  * Manned surface phases will see a significant reduction in the number of operators between the time the crew moves into the habitat and when they prepare for return to Earth.

  * A small number of operators will be required around the clock to act as interfaces between the surface crew and the on-call Earth support personnel.

  * Existing Mission support facilities on the Earth will be used to support the FLO missions.

  * Crew surface activities will be planned and implemented entirely by the crew. Earth support will be available on an on-call bases.

  * Surface systems analysis will be performed by autonomous surface systems which can also perform reconfiguration.

  * Crew will have access to critical systems data via surface based computer analysis tools.

  * The training of at least one crew member will have almost exclusively been dedicated to the analysis and maintenance of surface and vehicle systems. This crew member will have the natural skills associated with system troubleshooting and maintenance.

* MISSION OPERATIONS FACILITIES

* In order to reduce cost and minimize the complexities in mission operation activities, existing JSC training, mission planning, and control center facilities will be appropriately modified to allow their use between existing programs and the new FLO program.

* Specialized simulators will be required for systems training and will be used by both flight crews and mission controllers.

* These simulators will be kept to a minimum and will only cover tasks which require integrated activities between crews and ground or which cover specialized tasks.

* NEW MISSION OPERATIONS TECHNOLOGIES REQUIRED FOR FLO

* Space based failure detection, analysis, and reconfiguration capabilities built into safety and mission critical systems.

* Space based logistics tools for crew use:

  * Computer based trainers
  * Time line and procedure development tools
  * Inventory and spare equipment management tools
  * Resource management and planning

* Capability to uplink video and computer models for crew training, as well as engineering and scientific analysis.

* Continued development of Earth based automated systems analysis and trend analysis tools.

* Automated data gathering systems which select appropriate data for mission evaluation and reporting to upper management. (automated consumables analysis, health monitoring data, selected systems trend data etc.)

* Remote operations simulators which can be used to properly evaluate methods of performing remote operations by surface crew members or by Earth based operators.

FIRST LUNAR OUTPOST

SUMMARY OF KSC ACTIVITIES
JAN. - APR. 1992

J. S. BLUM
KSC ADVANCED STUDIES MANAGER
FOR SPACE EXPLORATION INITIATIVE

KSC FIRST LUNAR OUTPOST

FROM JANUARY THROUGH APRIL 1992, KSC ACTIVITIES, IN SUPPORT OF THE SEI FIRST LUNAR OUTPOST, WERE CONCENTRATED ON THE DEVELOPMENT OF FACILITY REQUIREMENTS FOR MANY LAUNCH VEHICLE OPTIONS. THESE INCLUDED THE SATURN V DERIVED , 2 BOOSTER VEHICLE; SATURN V DERIVED, STRETCHED, 2 BOOSTER VEHICLE; NLS DERIVED, 2 BOOSTER VEHICLES (MANY OPTIONS); NLS DERIVED, 4 BOOSTER VEHICLES; AND SATURN V DERIVED, 4 BOOSTER VEHICLE (MARS CAPABILITY). ADDITIONAL OPTIONS INCLUDED DUAL LAUNCH SCENARIOS. THIS EFFORT RESULTED IN THREE BASIC OPTIONS:

- LUNAR SATURN V DERIVED FACILITIES (2 BOOSTER)
- LUNAR NLS DERIVED FACILITIES (4 BOOSTER)
- SATURN V DERIVED FACILITIES SCARRED TO HANDLE MARS SATURN V VEHICLE (4 BOOSTER, 434' HEIGHT)

KSC FIRST LUNAR OUTPOST

IN CONSONANCE WITH THIS ACTIVITY, TIMELINE ASSESSMENTS WERE DEVELOPED USING AN OPTIMISTIC APPROACH. UNDERLYING ASSUMPTIONS DRIVING THESE TIMELINES INCLUDED:

- FLIGHT HARDWARE OPERABILITY
- SHIP AND SHOOT PHILOSOPHY
- MINIMUM PAD ACCESS
- SIMPLIFIED PAPER SYSTEM

THESE ASSUMPTIONS RESULTED IN TIMELINES SIGNIFICANTLY REDUCED FROM THE AS-RUN APOLLO TIMELINES

KSC FIRST LUNAR OUTPOST

ADDITIONALLY, OTHER EFFORTS INVOLVED:

- ACOUSTICAL ENVIRONMENT PREDICTIONS
- RECOVERY ANALYSIS
- GROUND OPERATIONS CONCEPTS BUDGET ESTIMATES

THE FOLLOWING DATA REPRESENTS THE DETAILED RESULTS OF ACTIVITIES TO DATE (BUDGETARY DATA IS NOT INCLUDED IN THIS PACKAGE)

KSC FIRST LUNAR OUTPOST

ON FEBRUARY 27, 1992 JOEL BLUM PRESENTED A STATUS BRIEFING ON THE SATURN V DERIVED SINGLE LAUNCH, 2 BOOSTER VEHICLE TO THE EXPLORATION CONTROL BOARD. THIS BRIEFING WAS SIGNIFICANT FOR SEVERAL REASONS:

- 21 GROUNDRULES AND ASSUMPTIONS WERE EXPLAINED INCLUDING OUR USE OF ONLY 1 MLT AND A NEW MOBILE SERVICE STRUCTURE (MSS) AND CRAWLER TRANSPORTER (CT).SUBSEQUENTIALLY, DIRECTION WAS RECEIVED TO UTILIZE 2 MLT'S TO SUPPORT FLIGHT RATE/INTERVALS, AND TO ELIMINATE UTILIZATION OF THE MSS AND NEW CT AS PAD ACCESS WOULD ONLY BE REQUIRED THROUGH SWING ARMS FROM THE MLT

- GROUNDRULES ALSO IDENTIFIED "SHIP & SHOOT" APPROACH, CONCERNS FOR FLAME TRENCH CAPABILITY, LACK OF DATA ON RECOVERY OPERATIONS, AND STATED THAT NO FACILITY/MANPOWER WAS INCLUDED FOR OFF-LINE PAYLOAD PROCESSING

- A FUNCTIONAL FLOW DIAGRAM AND TIMELINE STUDIES WERE ALSO PRESENTED.

PRESENTATION TO EXPLORATION CONTROL BOARD

KSC STATUS SEI SATURN V DERIVED SINGLE LAUNCH, 2 BOOSTER

J. S. BLUM
KSC/PT-FLS
2/27/92

KSC COST ESTIMATE
SEI SATURN V DERIVED, SINGLE LAUNCH, 2 BOOSTER LUNAR, STS, NLS COMBINED FLEET
GROUNDRULES/ASSUMPTIONS

1. COSTS ARE IN FY-92 DOLLARS
2. 1ST CARGO LAUNCH IS 10/99; 1ST PILOTED LAUNCH IS 12/99
3. REFERENCE VEHICLE CONFIGURATION IS ATTACHED (MSFC PD24, DATED 1/23/92; 386 FOOT VEHICLE)
4. NLS-1 VEHICLE WITH ASRM'S IS DELETED
5. NO PAYLOAD HANDLING OR CHANGEOUT AT LAUNCH PAD
6. NLS PROCESSED IN VAB HIGH BAY 4, SEI IN HIGH BAY 2, STS IN HIGH BAYS 1 AND 3
7. FLIGHT RATE IS 4 LAUNCHES PER YEAR

GROUNDRULES/ASSUMPTIONS (CONT'D)

8. PER DIRECTION, ONLY UTILIZE 1 MLT (NOTE: INITIAL TIMELINE ASSESSMENTS WITH ONLY 1 MLT DO NOT SUPPORT 90 DAY LAUNCH CENTERS)

9. STS ET, NLS CORE AND SEI STAGES/BOOSTERS CHECKOUT ARE IN SAME FACILITY.

10. SRB STACKING BUILDING IS REQUIRED TO MOVE SRB OPEN GRAIN STACKING OPERATIONS FROM VAB. (REDUCES MULTI-PROGRAM INTERFERENCE AND SAFETY RISKS)

11. FLIGHT HARDWARE IS DESIGNED FOR OPERABILITY TO REDUCE MANPOWER AND GROUND PROCESSING TIMELINES. (FLIGHT ELEMENTS UTILIZE "SHIP AND SHOOT" APPROACH).

12. DEDICATED SEI/NLS PAD AND STS PAD (UNLESS RETAINING STS DUAL PAD CAPABILITY CAN BE DONE FOR MINIMAL COST)

13. NORTH/SOUTH VEHICLE ORIENTATION ON LAUNCH PAD.

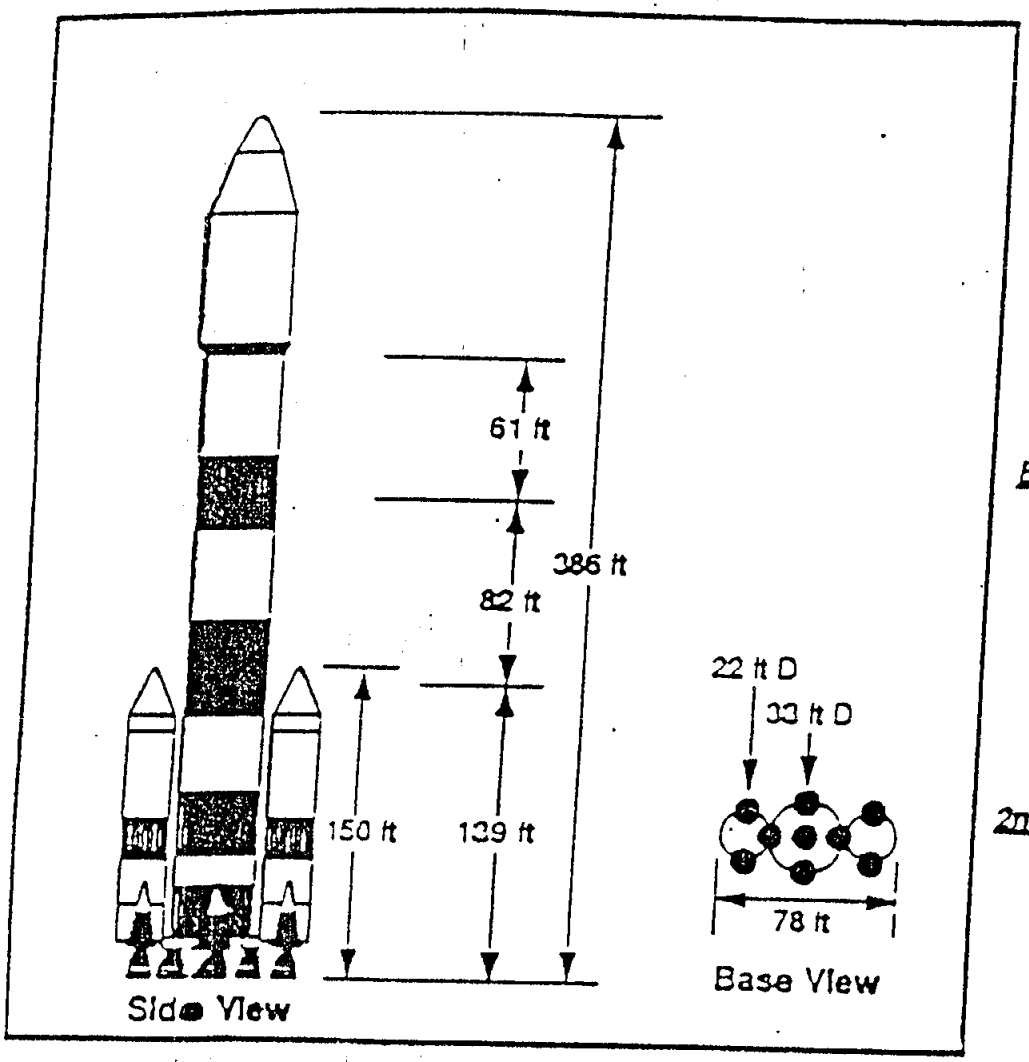
14. TOWER ON MLT.

GROUNDRULES/ASSUMPTIONS (CONT'D)

15. LAUNCH PAD ACCESS REQUIREMENTS DRIVE NEED FOR MOBILE SERVICE STRUCTURE AND, THEREFORE, NEW CRAWLER-TRANSPORTER.
16. ASSUME FLAME TRENCH IS CAPABLE OF HANDLING EXHAUST FROM 9 FI-A ENGINES (NEED MSFC SCALE MODE L TESTING).
17. NO COSTS DEVELOPED/INCLUDED FOR RECOVERY OPERATIONS.
18. NO OFF-LINE PAYLOAD PROCESSING MANPOWER OR FACILITY COST INCLUDED.
19. SHIFTS ARE GENERALLY 3 SHIFTS/DAY, 5 DAYS/WEEK
20. EXISTING TECHNOLOGY IS UTILIZED.
21. NO PROGRAM MANAGER'S RESERVE INCLUDED.

Lunar Saturn V Derived HLLV w/ 2 LOx/RP Boosters

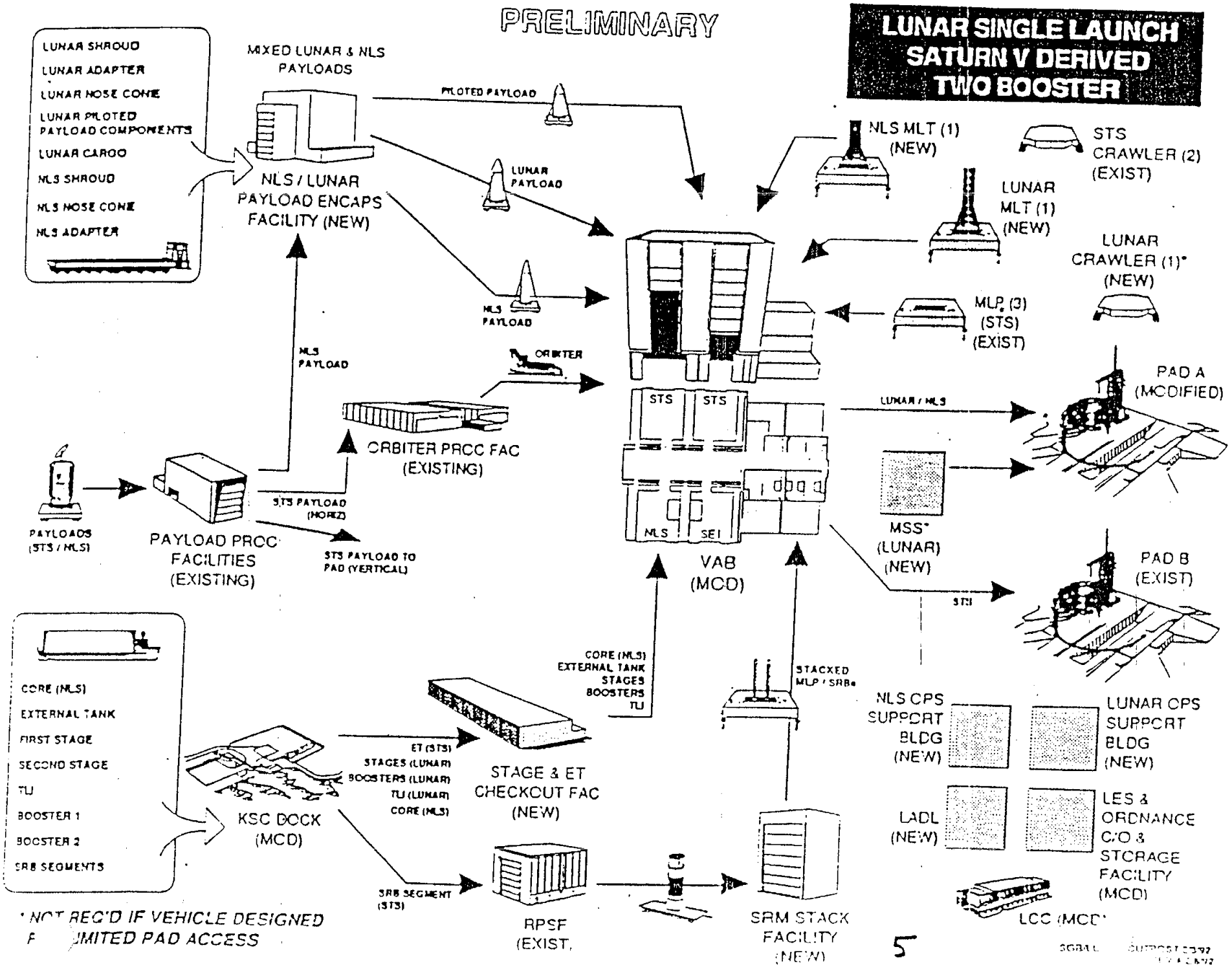
Single Launch - Cargo



Payload:	177 lb (80.5 t) / 487 klb
Final Position:	TLI / LEO Cutoff
GLOW:	10.2 Mlb
Engine Out:	None
1st Stage:	
Inert Mass:	365 klb
Propellant Mass:	4.58 Mlb
Propellant Type:	LOX/RP
Engine Type/#:	F-1A/5
Vac /SL Thrust (Ea):	1.74/1.522 Mlb
Vac ISP:	304.2 s
Engine Exit Dia:	144 in
Length / Diameter:	138 / 33 ft
Booster:	
Number/Type:	2/New
Inert Mass:	135 klb
Propellant Mass:	1.6 Mlb
Propellant Type:	LOX/RP
Engine Type/#:	F-1A/2
Vac /SL Thrust (Ea):	1.74/1.522 Mlb
Vac ISP:	304.2 s
Engine Exit Dia:	144 in
Length / Diameter:	150 / 22 ft
2nd Stage:	
Inert Mass:	90.3 klb
Propellant Mass:	995 klb
Propellant Type:	LOX/LH2
Engine Type/#:	J-2S/5
Vac Thrust/Vac ISP (Ea):	265 klb/436 s
Engine Exit Dia:	80 in
Length / Diameter:	82 / 33 ft
TLI Stage:	
Inert Mass:	49.3 klb
Propellant Mass:	400 klb
Propellant Type:	LOX/LH2
Engine Type/#:	J-2S/1
Vac Thrust (Ea):	265 klb
Vac ISP:	436 s
Engine Exit Dia:	80 in
Length / Diameter:	61 / 33 ft

Structure:	Al 2219
Shroud - Usable Volume:	33 x 60 ft
Mass:	35,500 lb / 7400 lb Adapter

Notes:
- 1st Stg/Boosters are Ignited on Pad w/Core [redacted] Holdown
- 1st & 2nd Stg. Are Sat. V S-IC/S-II
- Adapter Mass: 1st/2nd Stg. = 9900 lb / 2nd Stg./TLI = 8200 lb
- F-1A is 75% Step Throttlable / Calibrated to 1.522 Mlb
- Use Throttle Shutdown for Loads
- Max G = 4.0 / Max q = 890 psf


PRELIMINARY
LUNAR SINGLE LAUNCH
SATURN V DERIVED
TWO BOOSTER
LUNAR SHROUD
LUNAR ADAPTER
LUNAR NOSE CONE
LUNAR PILOTED PAYLOAD COMPONENTS
LUNAR CARGO
NLS SHROUD
NLS NOSE CONE
NLS ADAPTER
MIXED LUNAR & NLS PAYLOADS
NLS / LUNAR PAYLOAD ENCAPS FACILITY (NEW)
PILOTED PAYLOAD
LUNAR PAYLOAD
NLS PAYLOAD
NLS PAYLOAD
ORBITER
ORBITER PROC FAC (EXISTING)
PAYLOADS (STS / NLS)
PAYLOAD PROC FACILITIES (EXISTING)
STS PAYLOAD (HORIZ)
STS PAYLOAD TO PAD (VERTICAL)
NLS MLT (1) (NEW)
STS CRAWLER (2) (EXIST)
LUNAR MLT (1) (NEW)
LUNAR CRAWLER (1)* (NEW)
MLP (3) (STS) (EXIST)
STS
STS
NLS
SEI
VAB (MCD)
LUNAR / NLS
PAD A (MODIFIED)
MSS* (LUNAR) (NEW)
STS
PAD B (EXIST)
CORE (NLS)
EXTERNAL TANK
STAGES
BOOSTERS
TU
STACKED MLP / SRBs
NLS OPS SUPPORT BLDG (NEW)
LUNAR OPS SUPPORT BLDG (NEW)
LADL (NEW)
LES & ORDNANCE C/O & STORAGE FACILITY (MCD)
LCC (MCD)
CORE (NLS)
EXTERNAL TANK
FIRST STAGE
SECOND STAGE
TU
BOOSTER 1
BOOSTER 2
SRB SEGMENTS
KSC DOCK (MCD)
ET (STS)
STAGES (LUNAR)
BOOSTERS (LUNAR)
TU (LUNAR)
CORE (NLS)
STAGE & ET CHECKOUT FAC (NEW)
SRB SEGMENT (STS)
RPSF (EXIST.)
SRM STACK FACILITY (NEW)
* NOT REQ'D IF VEHICLE DESIGNED F LIMITED PAD ACCESS

SATURN V - DERIVED, TWO LOX/RP-1 BOOSTERS

LAUNCH PROCESSING TIMELINE (5/3 SHIFTS)

PRELIMINARY

CORE / BOOSTER STAGE CHECKOUT FACILITY

34 — CORE / BOOSTERS RECEIPT & CHECKOUT

25 | 9

5/1 SHIFT

LAUNCH VEHICLE COMPONENTS ARRIVAL AT KSC DOCK

93* DAYS

* INITIAL RECEIPT OF FLIGHT HARDWARE TO LAUNCH

VAB HIGHBAY 2

5/3 SHIFT

20 — CORE / BOOSTERS UPPER STAGES MATE

PAYLOAD MATE / INTEGRATED CHECKOUT / ROLLOUT — 22

LEGEND

X — CALENDAR DAYS

SHIFT CODE
X DAYS PER WEEK
X SHIFTS PER DAY

PAD

5/3 SHIFT
7/3 LAUNCH COUNTDOWN ONLY

PAD OPERATIONS — 26 | 10 / 30

PAD / MLT REFURBISH

LAUNCH

PAYLOAD ENCAPS FACILITY

TBD — CARGO / SHROUD ENCAPSULATION

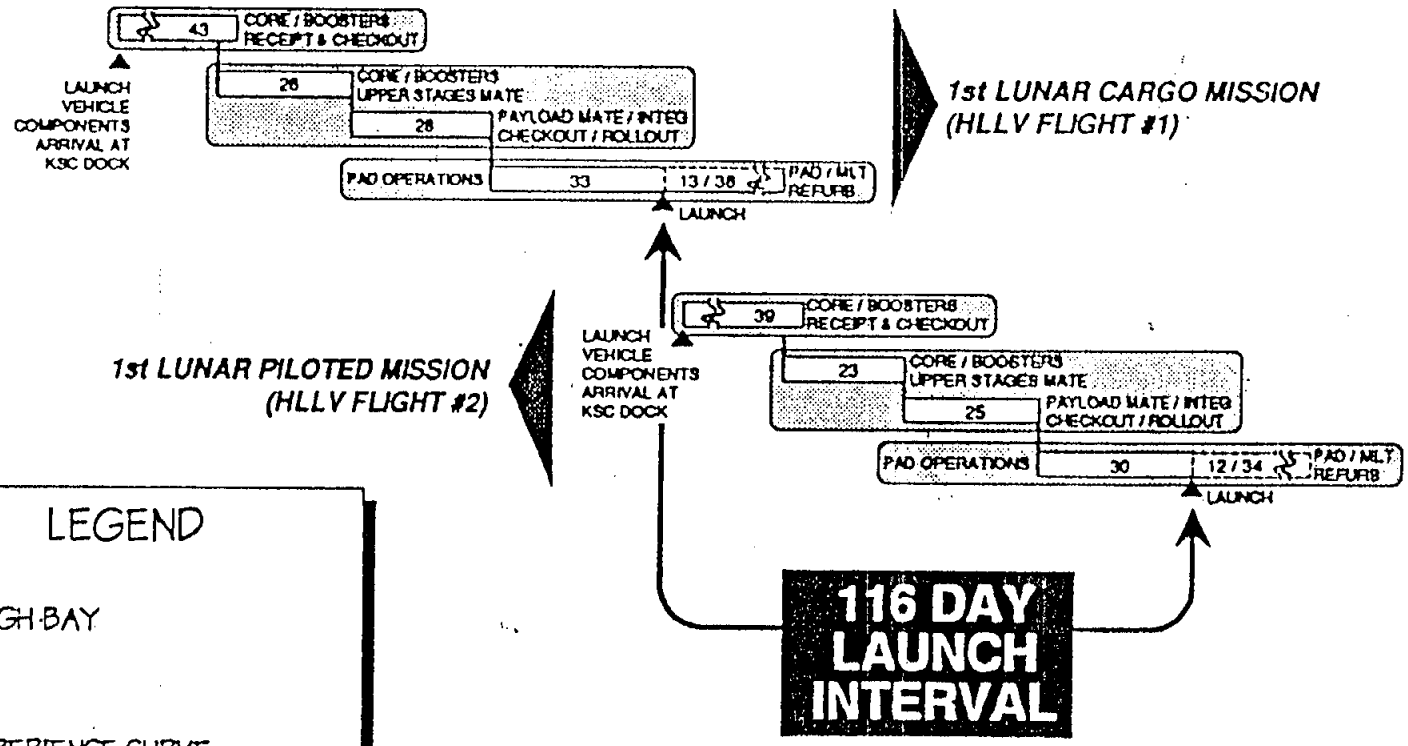
NOTES:
- 98 DAY MLT UTILIZATION
- 5TH FLOW (OPERATIONAL / NO DEV FLIGHT INSTR)
- FLIGHT ELEMENTS ARRIVE "READY TO FLY"

SGR/LUNAR OUTPOST TIMELINE GAMES
REV 10/25/92

FIRST LUNAR OUTPOST DEFINITION STUDY — MULTI-FLOW

SINGLE LAUNCH MISSION CONCEPT
SATURN V-DERIVED HLLV
2 LOX / RP-1 BOOSTERS

MISSIONS 1 & 2



LEGEND

1 HPF
1 VAB HIGH BAY
1 PAD
1 MLT

90% EXPERIENCE CURVE

CALENDAR –
HPF 5/1 SHIFT
VAB 5/3 SHIFT
PAD 5/3 SHIFT
7/3 LAUNCH COUNTDOWN
REFURB 5/3 SHIFT

XX CALENDAR DAYS

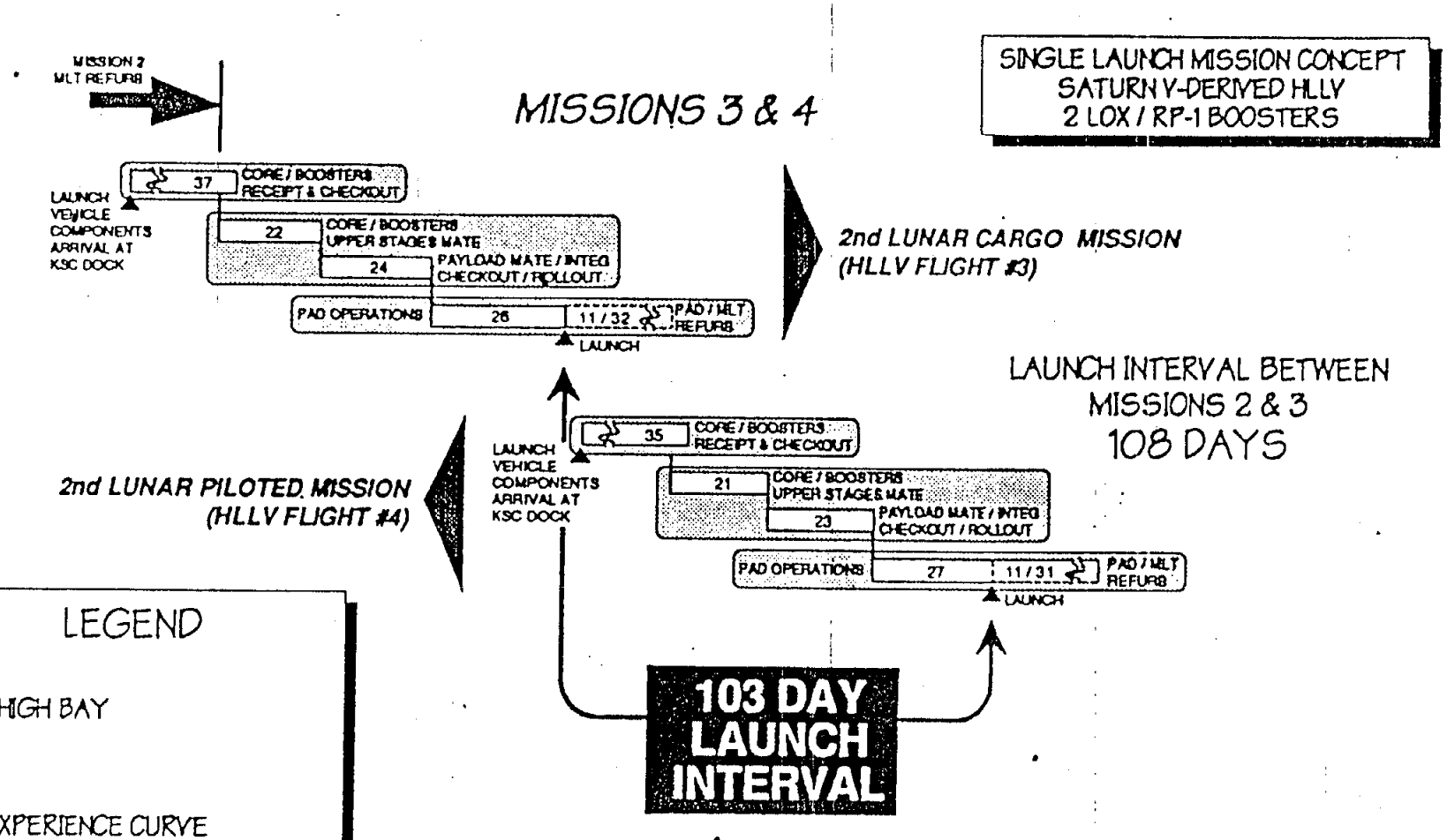
PRELIMINARY

LAUNCH INTERVAL BETWEEN
MISSIONS 2 & 3
108 DAYS

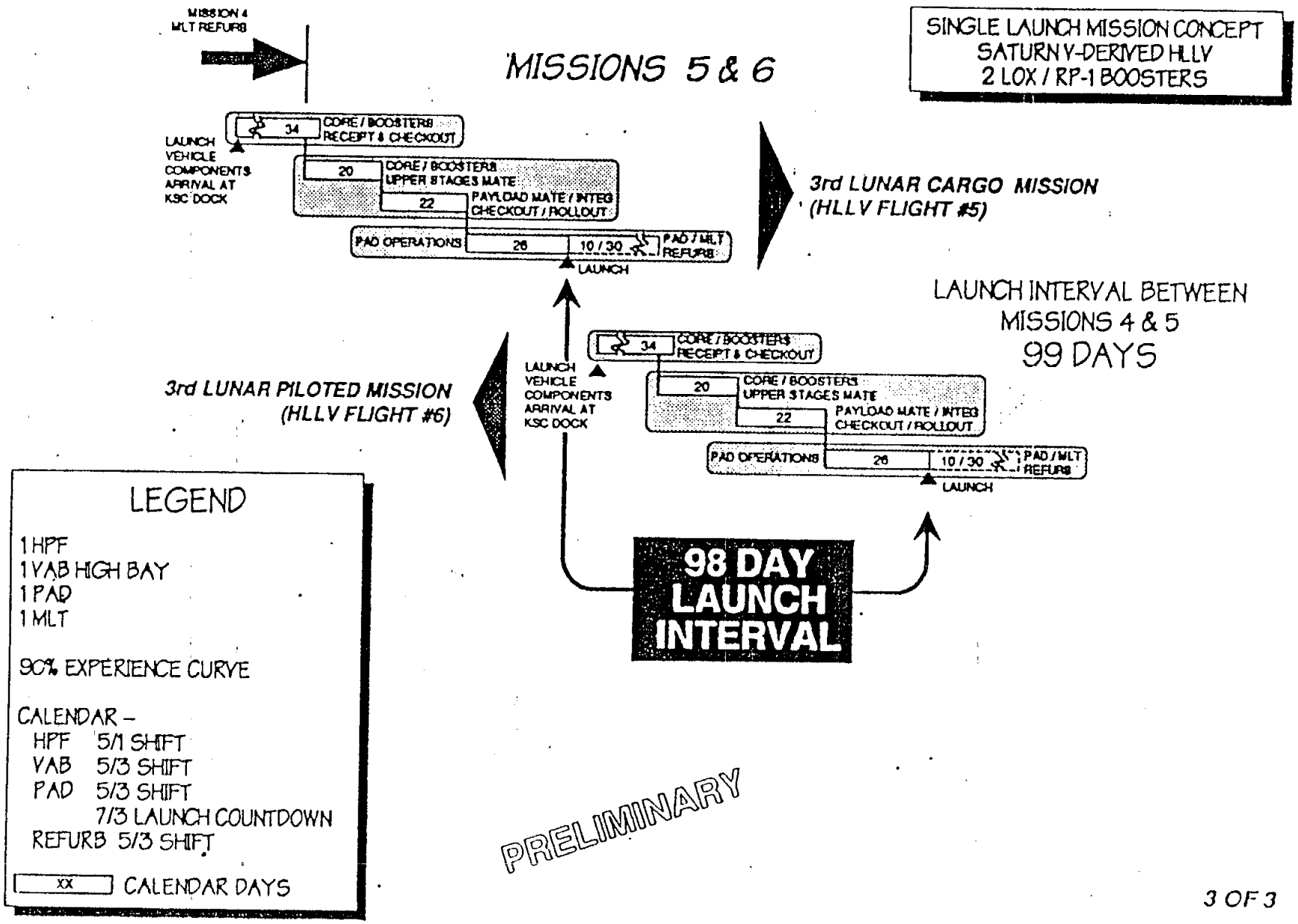
1 OF 3

SGB/LUNAR OUTPOST MORE TIMELINE GAMES
REV A 02/25/92

FIRST LUNAR OUTPOST DEFINITION STUDY — MULTI-FLOW



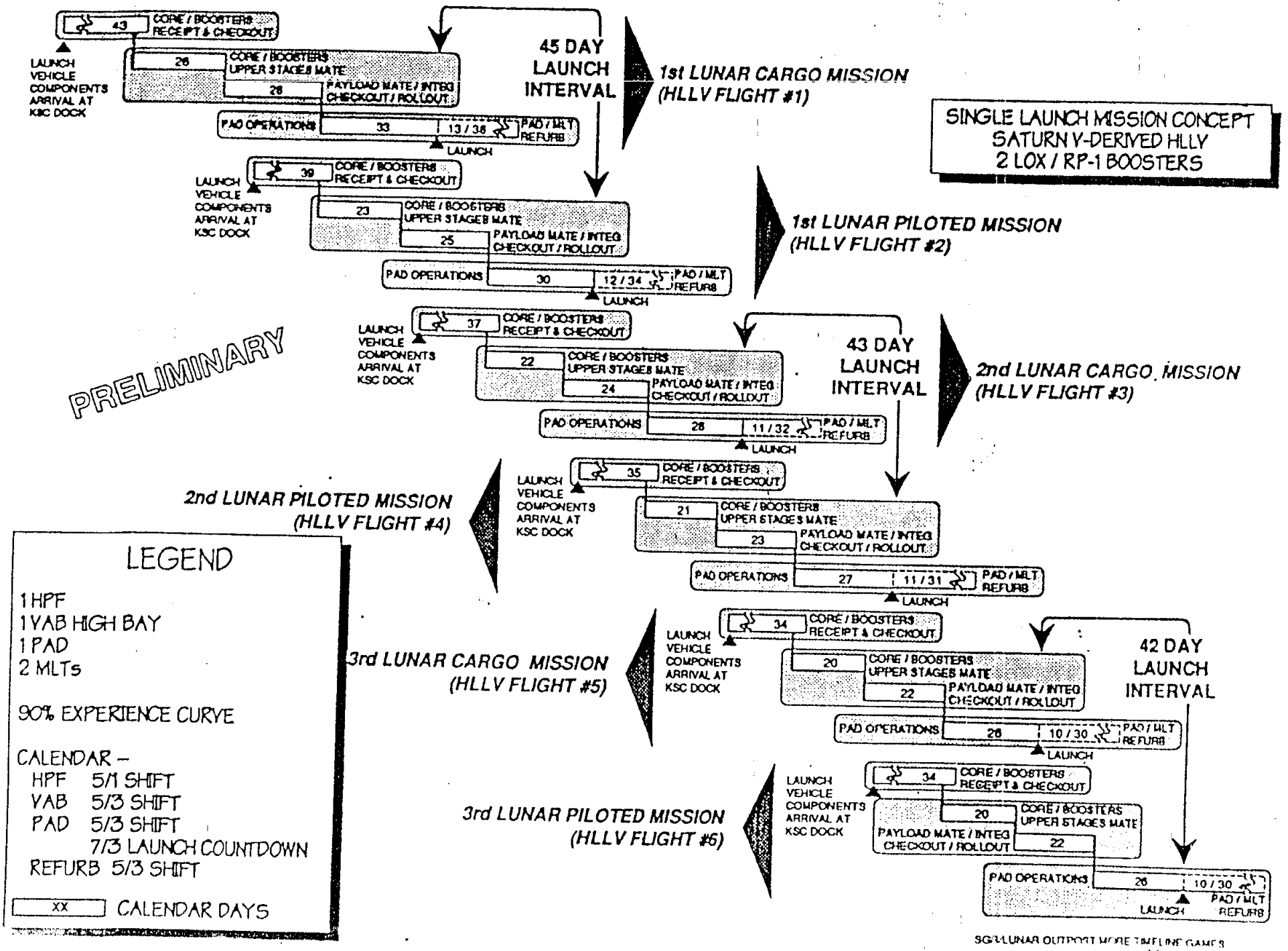
FIRST LUNAR OUTPOST DEFINITION STUDY — MULTI-FLOW



3 OF 3

SGB/LUNAR OUTPOST MORE TIMELINE GAMES
V A 02/25/92

FIRST LUNAR OUTPOST DEFINITION STUDY — MULTI-FLOW



SGR/LUNAR OUTPOST MORE TIMELINE GAMES
REV A 02/25/92

KSC FIRST LUNAR OUTPOST

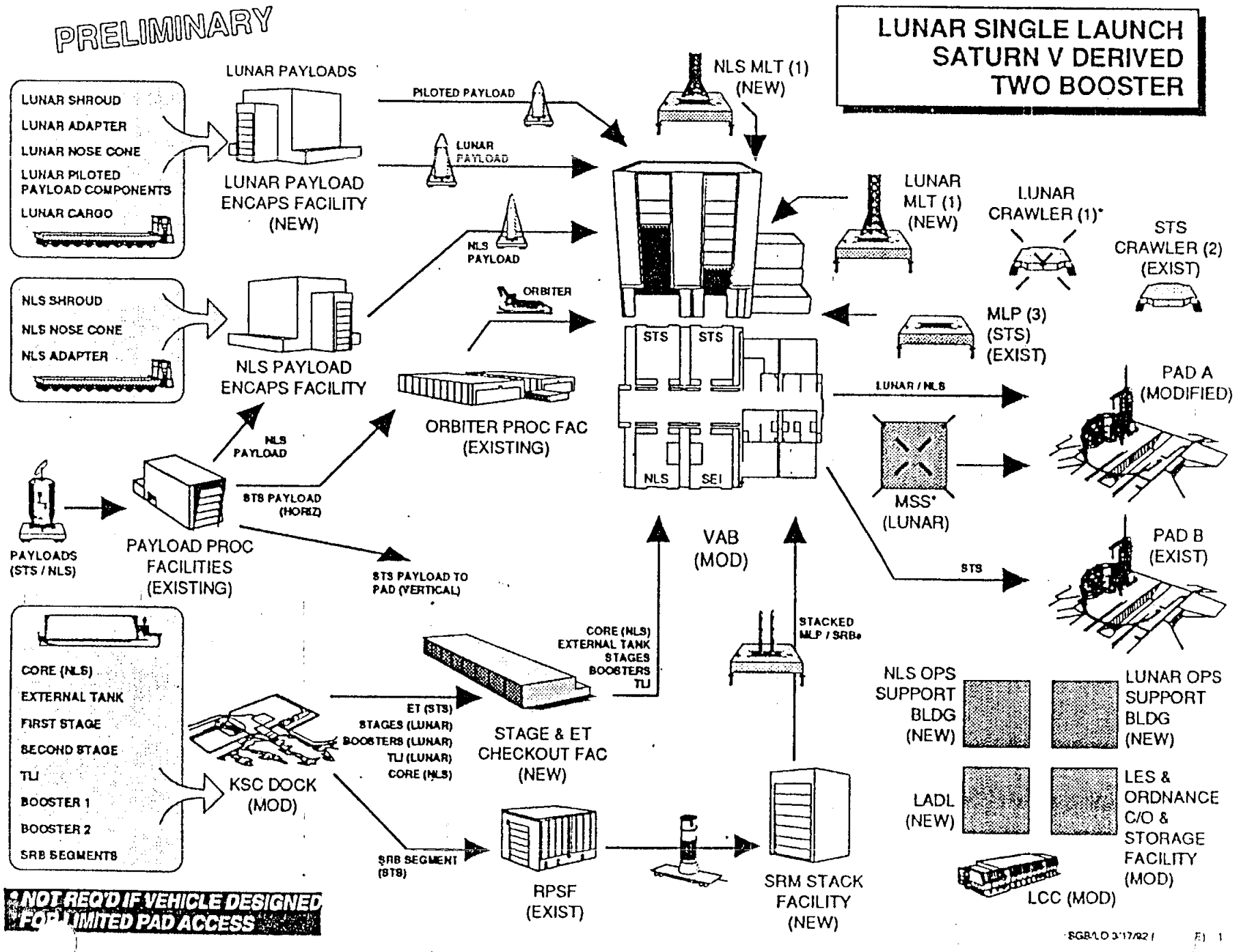
FUNCTIONAL FLOW DIAGRAMS ARE UTILIZED BY KSC TO AID IN THE UNDERSTANDING OF FACILITY REQUIREMENTS AS DRIVEN BY LAUNCH VEHICLE CONFIGURATION, AFFECT BOTH ON AND DUE TO CO-EXISTENCE WITH OTHER PROGRAMS, AND EVOLUTION POTENTIAL. IN THIS REGARD, A SERIES OF FUNCTIONAL FLOW DIAGRAMS WERE GENERATED.

- LUNAR SINGLE LAUNCH, SATURN V DERIVED, TWO BOOSTER
- LUNAR/MARS SINGLE LAUNCH, SATURN V DERIVED, TWO AND FOUR BOOSTER
- LUNAR/MARS SINGLE LAUNCH, SATURN V DERIVED, TWO AND FOUR BOOSTER, EARLY START
- LUNAR SINGLE LAUNCH, NLS DERIVED, TWO BOOSTER
- LUNAR/MARS SINGLE LAUNCH, NLS DERIVED, TWO AND FOUR BOOSTER
- LUNAR/MARS SINGLE LAUNCH, NLS DERIVED, TWO AND FOUR BOOSTER, EARLY START
- LUNAR DUAL LAUNCH, NLS DERIVED, TWO LOX/LH2 BOOSTERS
- LUNAR/MARS DUAL LAUNCH, NLS DERIVED, TWO LOX/RP-1 BOOSTERS
- LUNAR/MARS SINGLE LAUNCH, SATURN V DERIVED, TWO AND FOUR BOOSTER, REVISED CONCEPT

KSC FIRST LUNAR OUTPOST

IT SHOULD BE NOTED THAT, ON APRIL 29, 1992 AT THE HLLV TECHNICAL INTERCHANGE MEETING #5, DIRECTION WAS GIVEN BY PHIL DEANS AND GENE AUSTIN FOR KSC TO UTILIZE THE REVISED CONCEPT OF A NEW COMBINED VEHICLE INTEGRATION BUILDING/ PAYLOAD INTEGRATION AND ENCAPSULATION FACILITY, AND THE EXTENDED LAUNCH PAD WITH FOUR BOOSTER CAPABILITY, AS THE BASELINE FOR CofF PLANNING PURPOSES.

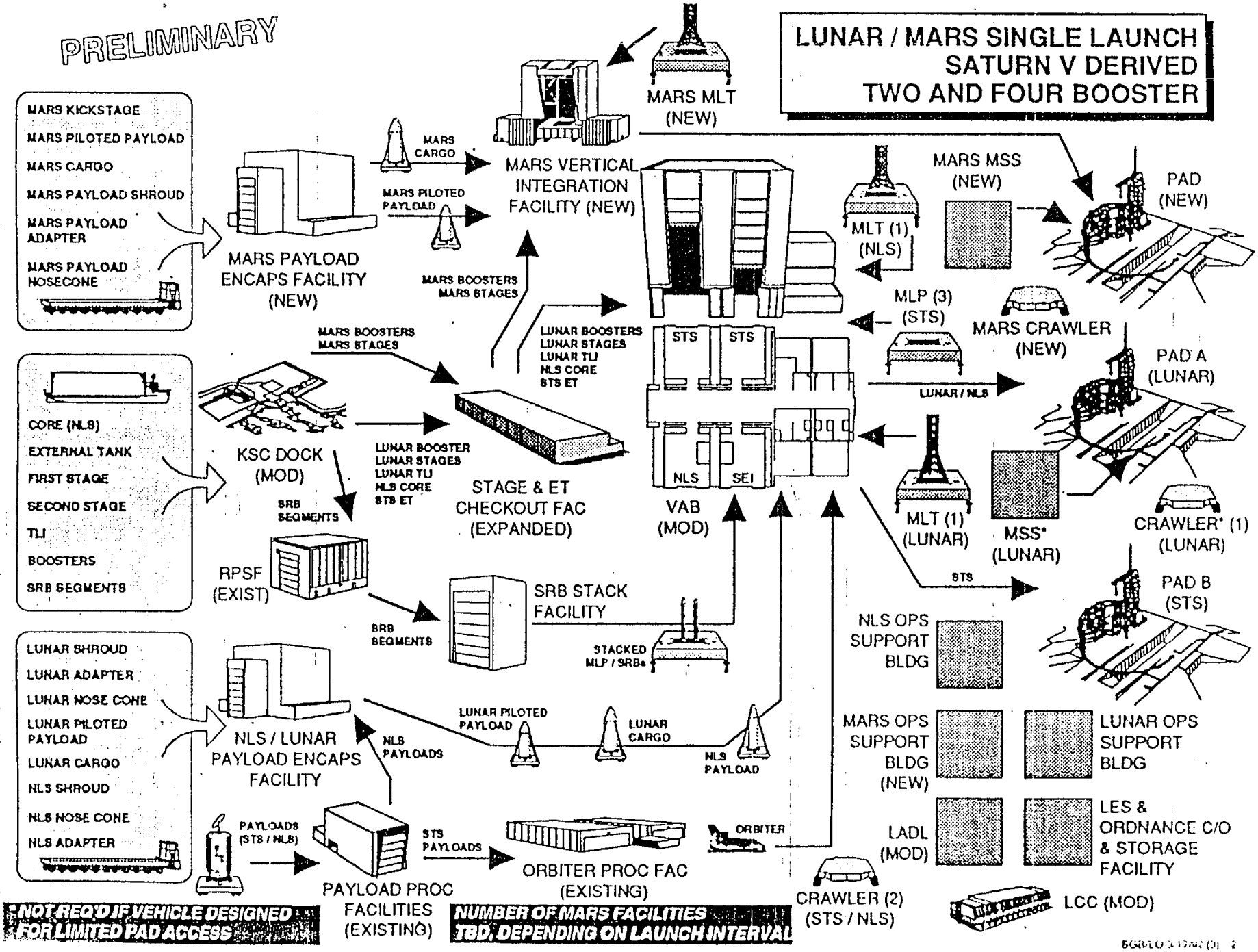
LUNAR OUTPOST



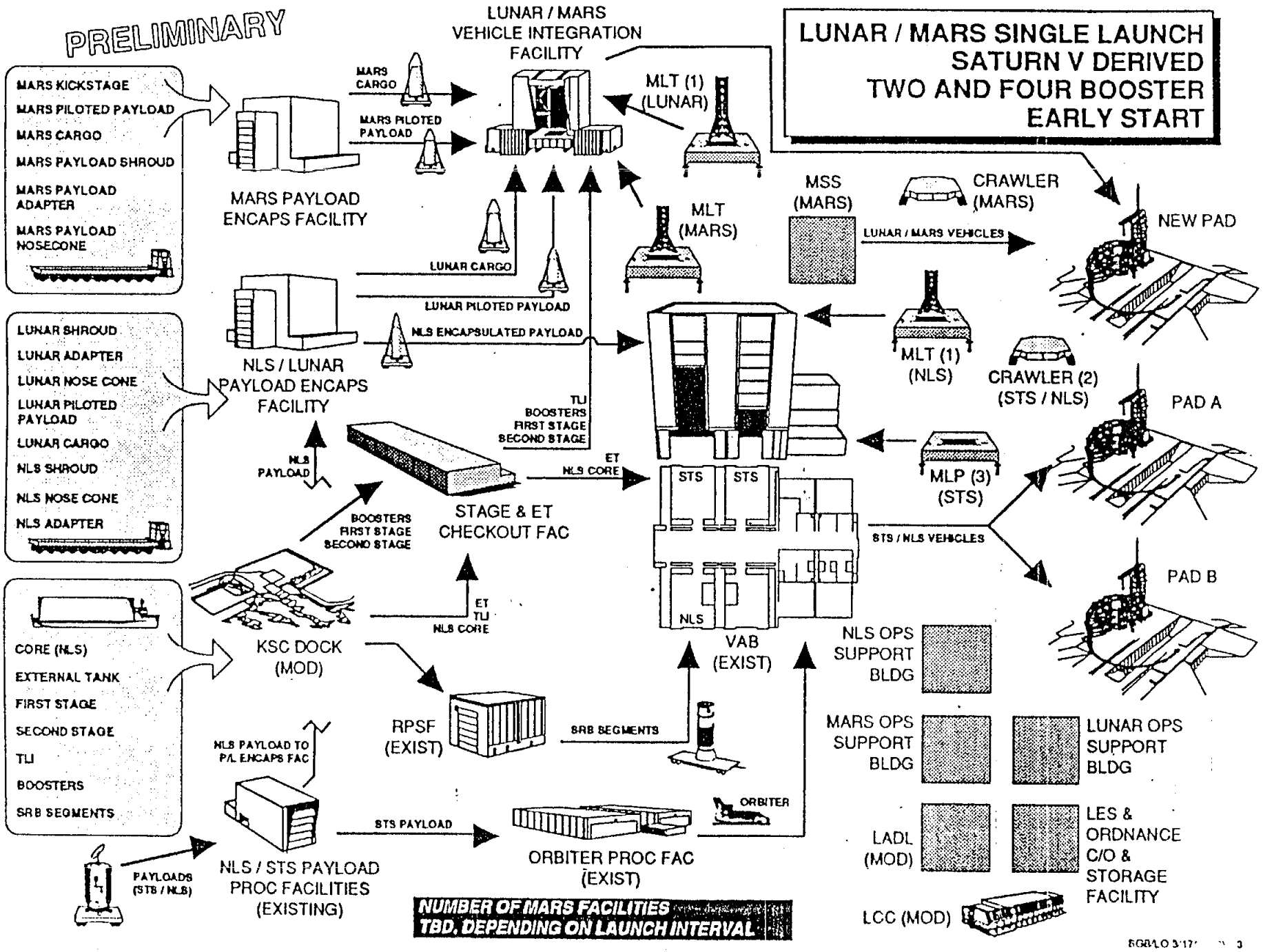
LUNAR OUTPOST

PRELIMINARY

LUNAR / MARS SINGLE LAUNCH SATURN V DERIVED TWO AND FOUR BOOSTER



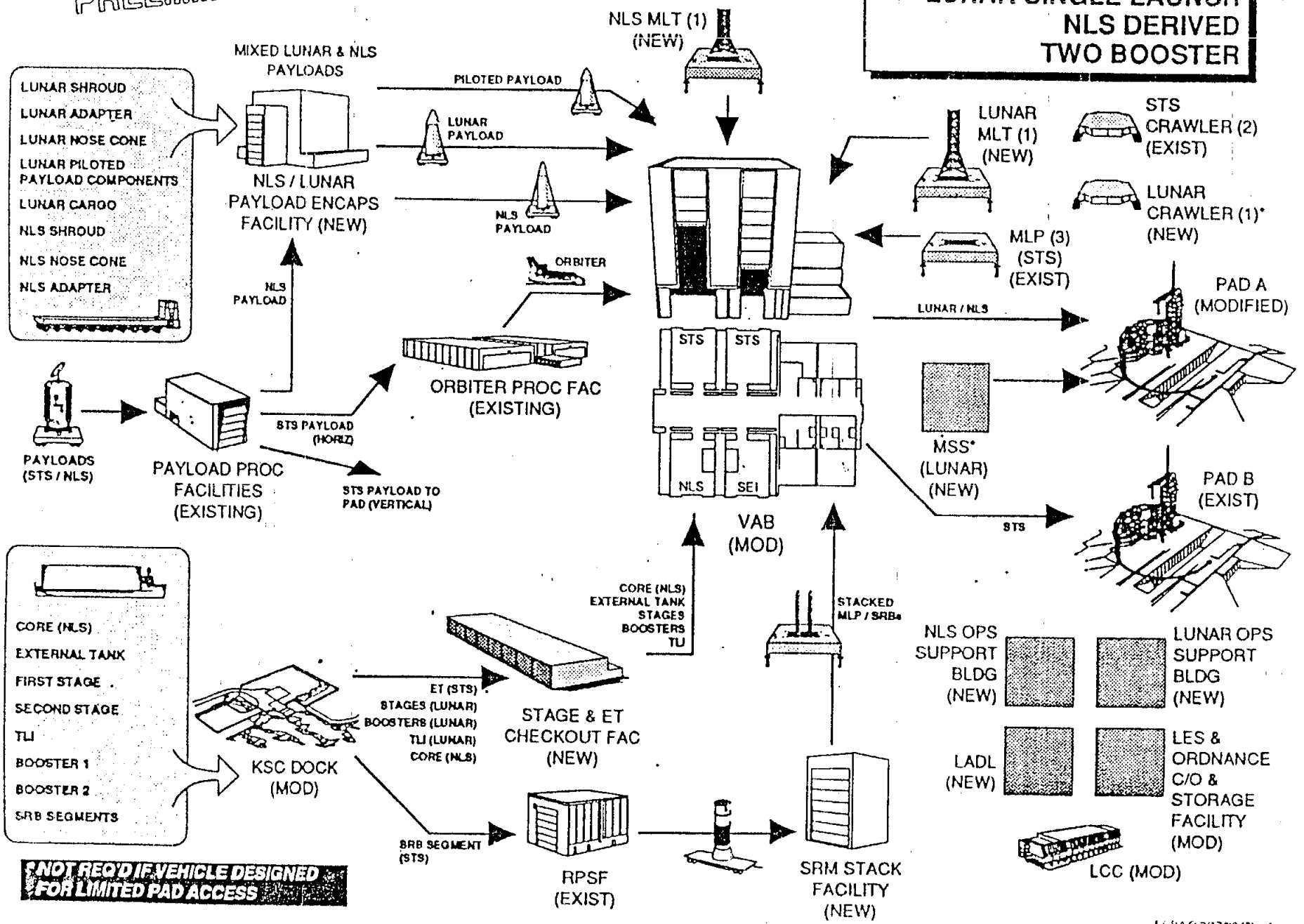
LUNAR OUTPOST

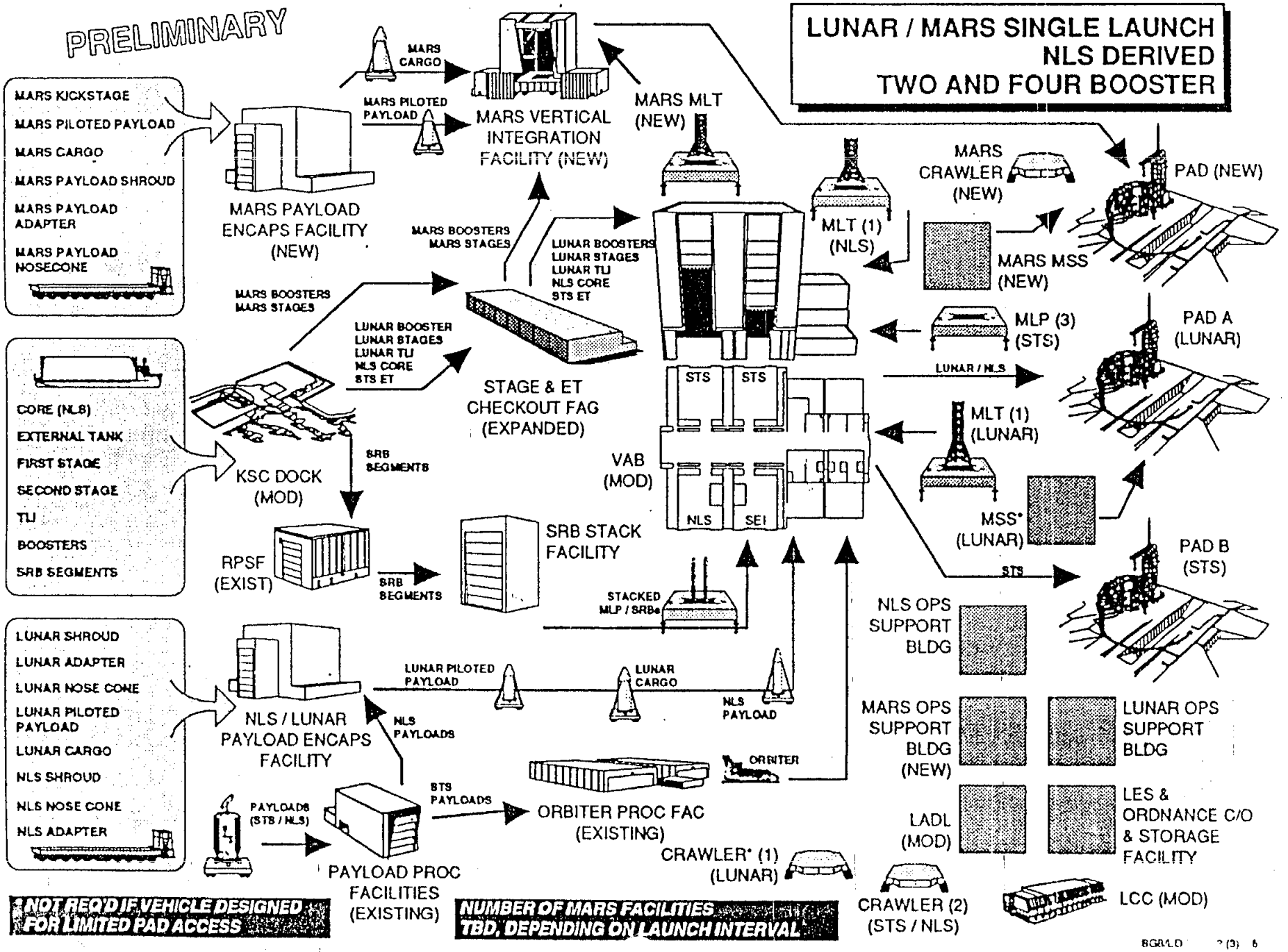


LUNAR OUTPOST

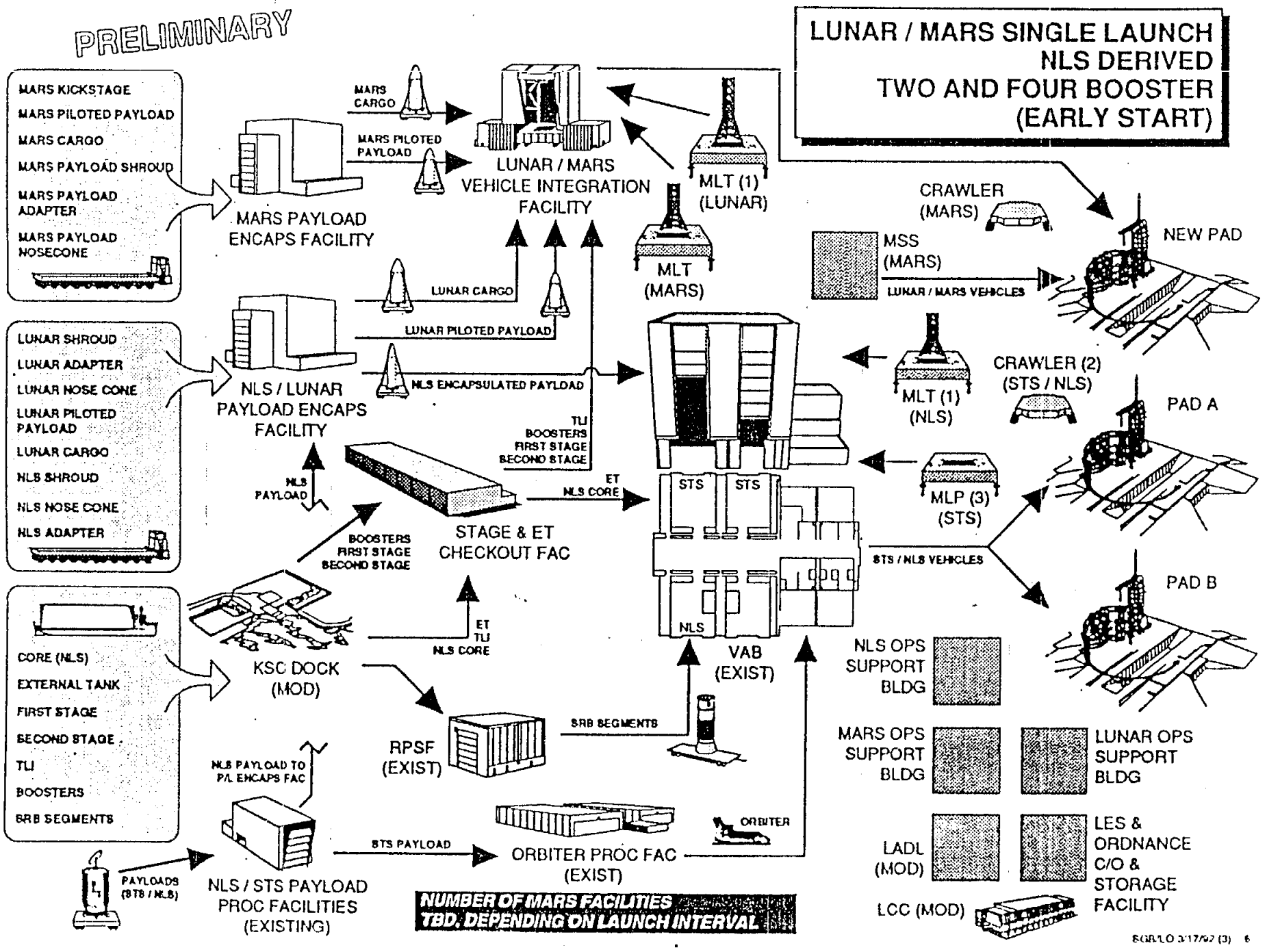
PRELIMINARY

LUNAR SINGLE LAUNCH NLS DERIVED TWO BOOSTER




PRELIMINARY
LUNAR / MARS SINGLE LAUNCH
NLS DERIVED
TWO AND FOUR BOOSTER
MARS KICKSTAGE
MARS PILOTED PAYLOAD
MARS CARGO
MARS PAYLOAD SHROUD
MARS PAYLOAD ADAPTER
MARS PAYLOAD NOSECONE
MARS CARGO
MARS PILOTED PAYLOAD
MARS VERTICAL INTEGRATION FACILITY (NEW)
MARS MLT (NEW)
MARS PAYLOAD ENCAPS FACILITY (NEW)
MARS BOOSTERS MARS STAGES
LUNAR BOOSTERS LUNAR STAGES LUNAR TLI NLS CORE STS ET
MLT (1) (NLS)
MARS CRAWLER (NEW)
PAD (NEW)
MARS MSS (NEW)
MARS BOOSTERS MARS STAGES
LUNAR BOOSTER LUNAR STAGES LUNAR TLI NLS CORE STS ET
MLP (3) (STS)
PAD A (LUNAR)
LUNAR / NLS
CORE (NLS)
EXTERNAL TANK
FIRST STAGE
SECOND STAGE
TLI
BOOSTERS
SRB SEGMENTS
STAGE & ET CHECKOUT FAC (EXPANDED)
STS
STS
MLT (1) (LUNAR)
KSC DOCK (MOD)
SRB SEGMENTS
VAB (MOD)
NLS
SEI
MSS* (LUNAR)
RPSF (EXIST)
SRB SEGMENTS
SRB STACK FACILITY
STACKED MLP / SRBs
STS
PAD B (STS)
NLS OPS SUPPORT BLDG
LUNAR SHROUD
LUNAR ADAPTER
LUNAR NOSE CONE
LUNAR PILOTED PAYLOAD
LUNAR CARGO
NLS SHROUD
NLS NOSE CONE
NLS ADAPTER
NLS / LUNAR PAYLOAD ENCAPS FACILITY
LUNAR PILOTED PAYLOAD
LUNAR CARGO
NLS PAYLOAD
NLS PAYLOADS
MARS OPS SUPPORT BLDG (NEW)
LUNAR OPS SUPPORT BLDG
ORBITER
PAYLOADS (STS / NLS)
STS PAYLOADS
ORBITER PROC FAC (EXISTING)
LADL (MOD)
LES & ORDNANCE C/O & STORAGE FACILITY
CRAWLER* (1) (LUNAR)
PAYLOAD PROC FACILITIES (EXISTING)
CRAWLER (2) (STS / NLS)
LCC (MOD)
* NOT REQ'D IF VEHICLE DESIGNED FOR LIMITED PAD ACCESS
NUMBER OF MARS FACILITIES TBD, DEPENDING ON LAUNCH INTERVAL

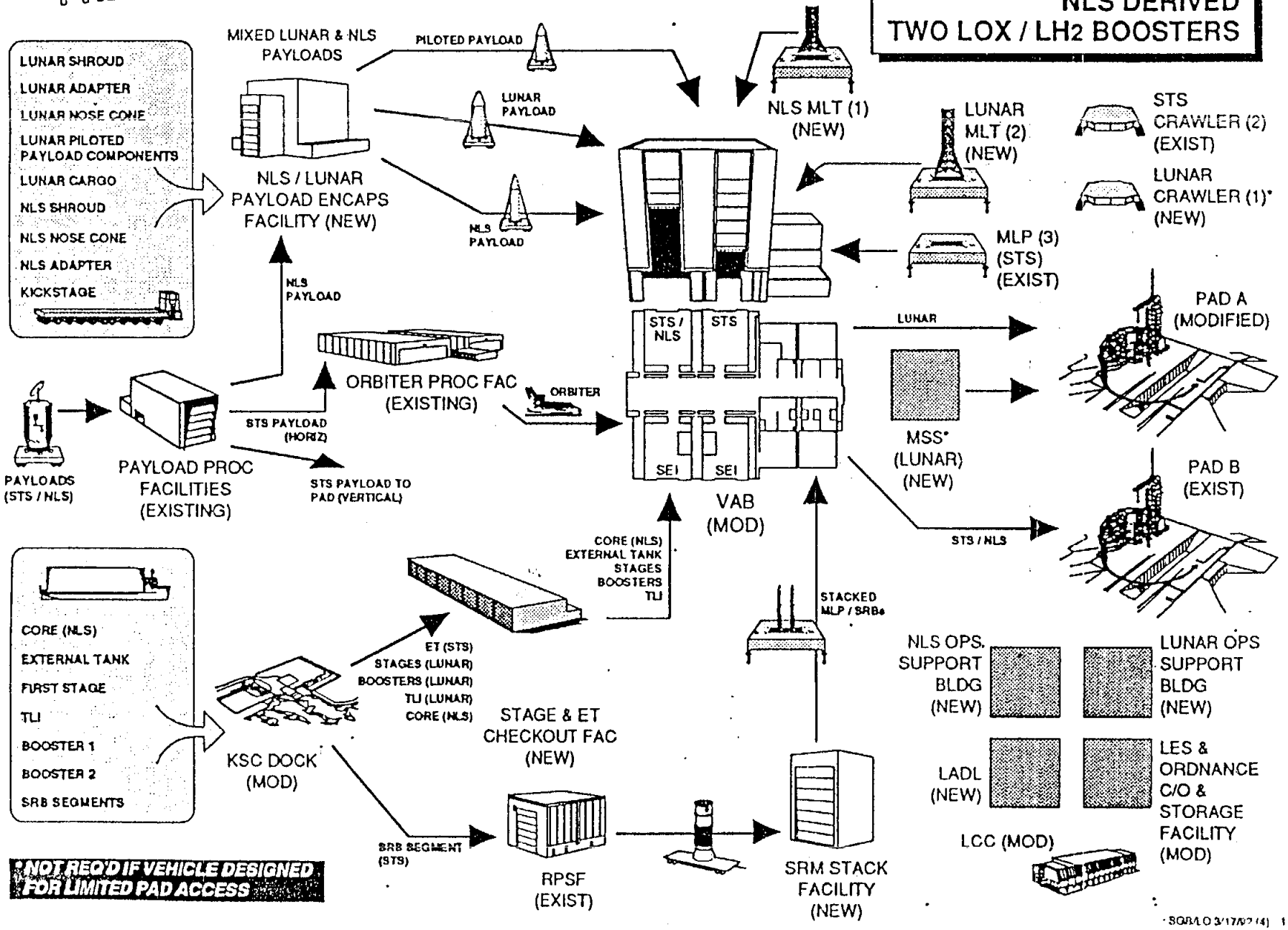
LUNAR OUTPOST



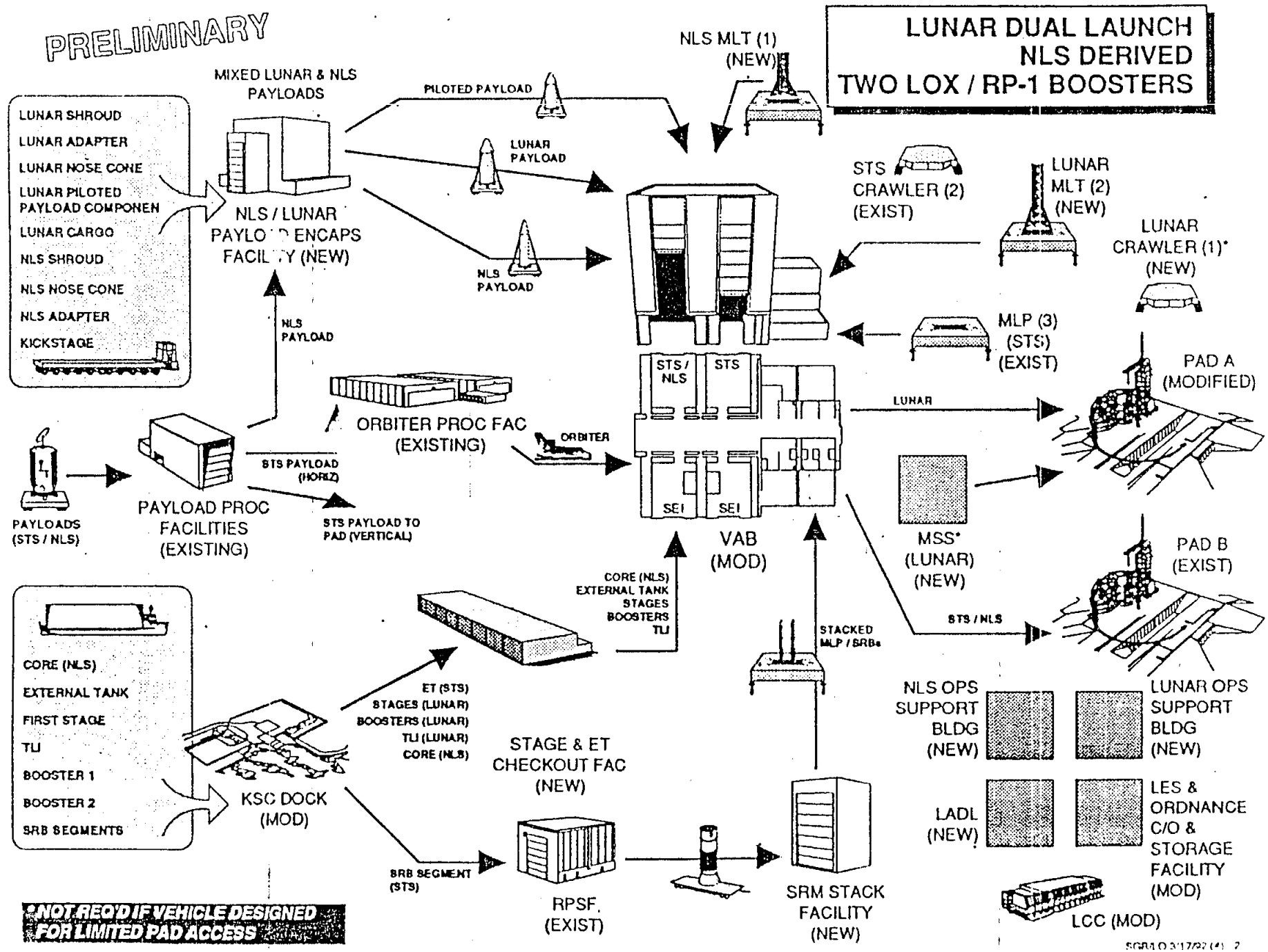
LUNAR OUTPOST

PRELIMINARY

LUNAR DUAL LAUNCH NLS DERIVED TWO LOX / LH2 BOOSTERS



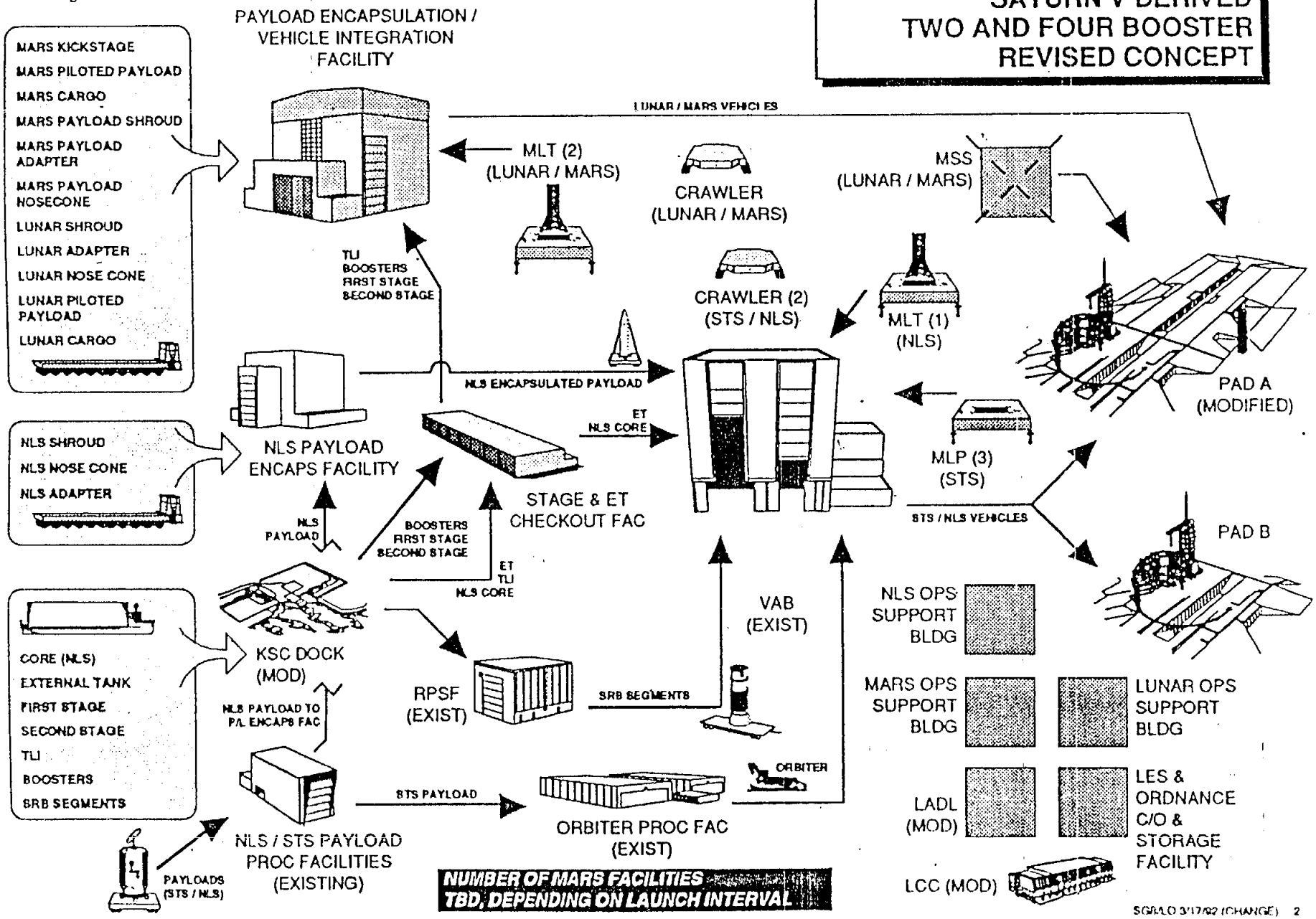
SQRA/LO 3/17/92 (4) 1


PRELIMINARY
LUNAR DUAL LAUNCH
NLS DERIVED
TWO LOX / RP-1 BOOSTERS
MIXED LUNAR & NLS PAYLOADS
LUNAR SHROUD
LUNAR ADAPTER
LUNAR NOSE CONE
LUNAR PILOTED PAYLOAD COMPONEN
LUNAR CARGO
NLS SHROUD
NLS NOSE CONE
NLS ADAPTER
KICKSTAGE
NLS / LUNAR PAYLO D ENCAPS FACIL Y (NEW)
PILOTED PAYLOAD
LUNAR PAYLOAD
NLS PAYLOAD
NLS PAYLOAD
NLS MLT (1) (NEW)
STS CRAWLER (2) (EXIST)
LUNAR MLT (2) (NEW)
LUNAR CRAWLER (1)* (NEW)
MLP (3) (STS) (EXIST)
STS / NLS
STS
SEI
SEI
VAB (MOD)
LUNAR
PAD A (MODIFIED)
ORBITER PROC FAC (EXISTING)
ORBITER
PAYLOADS (STS / NLS)
PAYLOAD PROC FACILITIES (EXISTING)
STS PAYLOAD (HORIZ)
STS PAYLOAD TO PAD (VERTICAL)
MSS* (LUNAR) (NEW)
PAD B (EXIST)
CORE (NLS)
EXTERNAL TANK
STAGES
BOOSTERS
TLI
STS / NLS
STACKED MLP / SRBs
CORE (NLS)
EXTERNAL TANK
FIRST STAGE
TLI
BOOSTER 1
BOOSTER 2
SRB SEGMENTS
ET (STS)
STAGES (LUNAR)
BOOSTERS (LUNAR)
TLI (LUNAR)
CORE (NLS)
KSC DOCK (MOD)
STAGE & ET CHECKOUT FAC (NEW)
NLS OPS SUPPORT BLDG (NEW)
LUNAR OPS SUPPORT BLDG (NEW)
LADL (NEW)
LES & ORDNANCE C/O & STORAGE FACILITY (MOD)
SRB SEGMENT (STS)
RPSF (EXIST)
SRM STACK FACILITY (NEW)
LCC (MOD)
*NOT REQ'D IF VEHICLE DESIGNED FOR LIMITED PAD ACCESS

LUNAR OUTPOST

PRELIMINARY

LUNAR / MARS SINGLE LAUNCH SATURN V DERIVED TWO AND FOUR BOOSTER REVISED CONCEPT



SGALO 3/17/92 (CHANGE) 2

KSC FIRST LUNAR OUTPOST

AS PREVIOUSLY STATED, THE PRIMARY EMPHASIS OF THE KSC EFFORT WAS TO DEVELOP FACILITY REQUIREMENTS BASED UPON THE VARIOUS VEHICLE CONFIGURATIONS DEVELOPED DURING THIS PERIOD. THIS CULMINATED WITH A DETAILED PRESENTATION BY DON PAGE TO THE ASSOCIATE ADMINISTRATOR FOR EXPLORATION ON APRIL 2, 1992. THIS DATA WAS ALSO PRESENTED AT THE HLLV T.I.M. #4 ON APRIL 15, 1992. THE PRESENTATION WAS STRUCTURED TO DELINEATE FACILITY REQUIREMENTS FOR THREE OPTIONS:

- LUNAR SATURN V DERIVED FACILITIES (2 BOOSTER)
- LUNAR NLS DERIVED FACILITIES (4 BOOSTER)
- SATURN V DERIVED FACILITIES, SCARRED TO HANDLE MARS SATURN VEHICLE (4 BOOSTER, 434 SCARRED HEIGHT)

ALSO INCLUDED IN THE APRIL 15 PRESENTATION WAS AN EXPLANATION OF THE CofF PROCESS INDICATING APPROXIMATE PERIODS OF TIME REQUIRED FOR VARIOUS CONSTRUCTION RELATED TASKS.

BLUM

SPACE EXPLORATION INITIATIVE SEI

KSC FACILITIES

DON PAGE
DE-PMO
APR 15, 1992

NASA
National Aeronautics an
Space Administratio
John F. Kennedy Space Center

KSC FORM 4-697 (REV. 1/88)

SPACE EXPLORATION INITIATIVE

KSC COST ESTIMATE

- TASK
  - DEVELOP THE COST FOR LAUNCH FACILITY CAPACITY TO SUPPORT LUNAR VEHICLE REQUIREMENTS OF 4 LAUNCHES PER YEAR.

- QUESTION
  - WHAT IS THE COST DIFFERENCE TO SCAR THE LUNAR LAUNCH FACILITY CAPACITY TO HANDLE A MARS SIZE VEHICLE?

SPACE EXPLORATION INITIATIVE

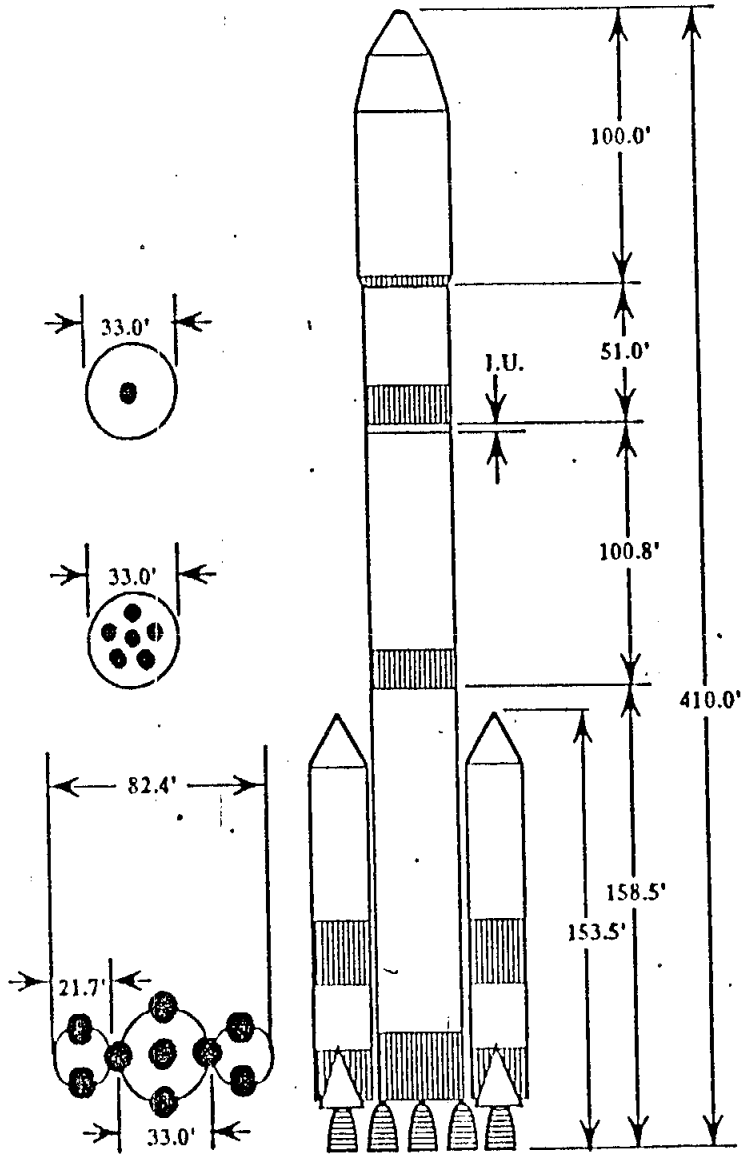
KSC FACILITY COSTS

AGENDA

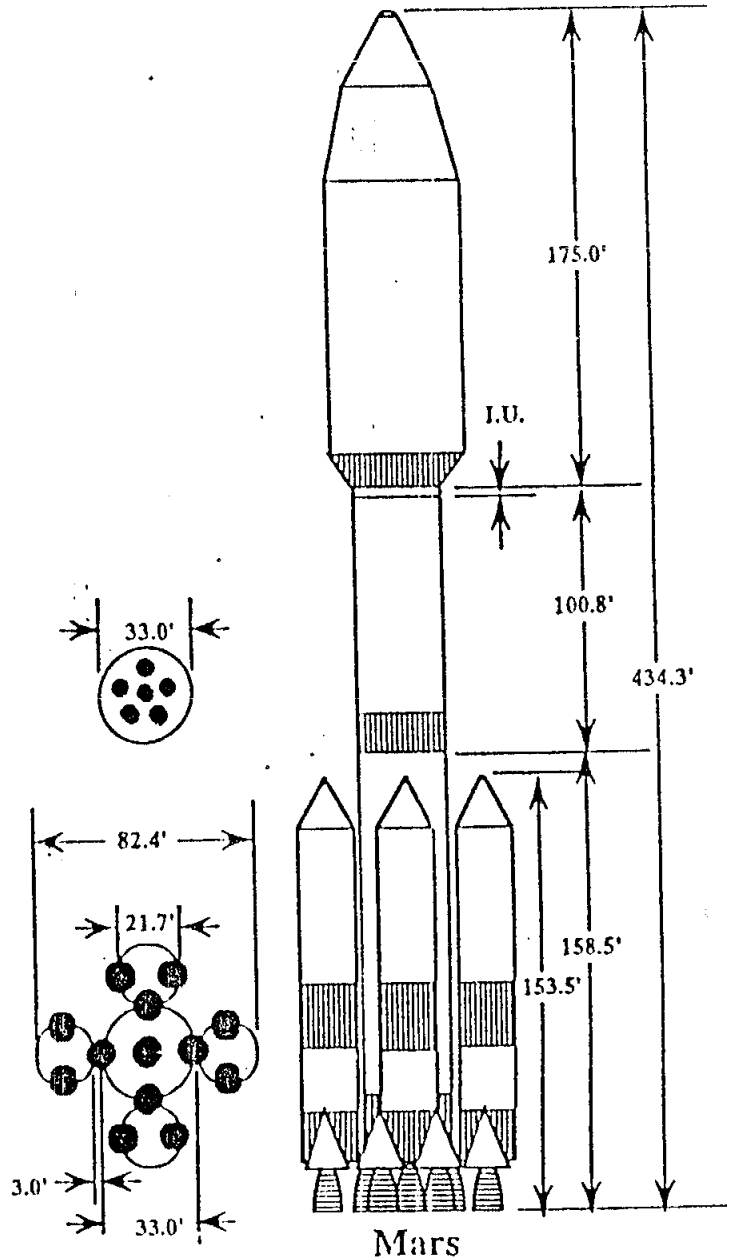
- LUNAR SAT V DERIVED FACILITY REFERENCE
- LUNAR NLS DERIVED FACILITY OPTION
- SCAR COST FOR LUNAR SAT V FACILITY TO HANDLE SAT V DERIVED MARS VEHICLE
- NLS DERIVED MARS VEHICLE UNDEFINED
- OTHER
  - C of F CYCLE
  - FAR FIELD ACOUSTIC STUDY

Reference Satur. V Derived HLLV

EXPLORATION PROGRAMS OFFICE



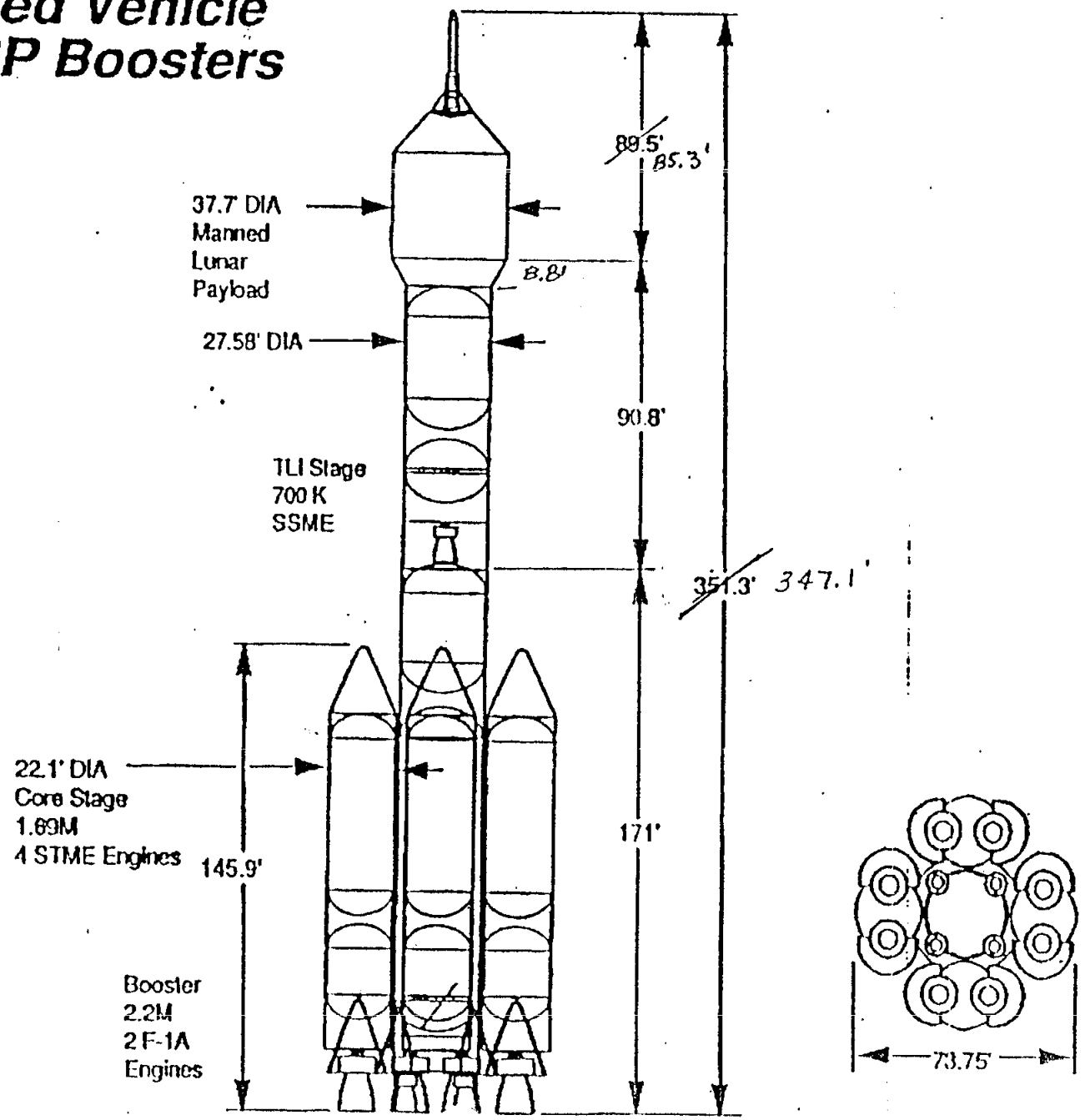
First Lunar Outpost



Mars

FROM CEMC AUSTIN 3/12/

NLS Derived Vehicle W/4 LOX/RP Boosters



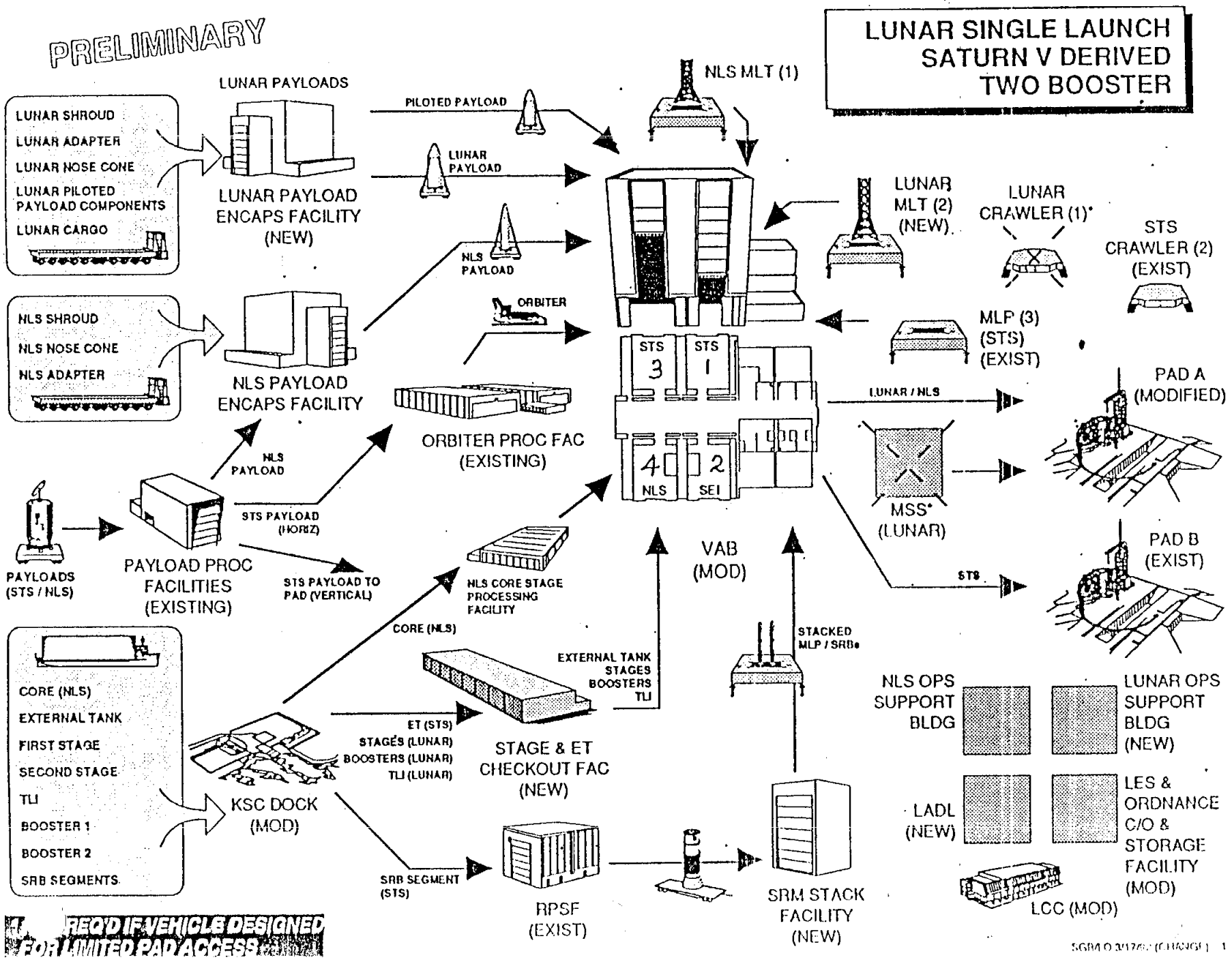
KSC FACILITY COSTS
ASSUMPTIONS ABOUT NLS

SEI COSTING ASSUMES NLS HAS ALREADY FUNDED THE FOLLOWING:

- NLS PAYLOAD ENCAPSULATION FAC
- NLS MLT
- NLS PAD MODS
- NLS VAB MODS HB4
- NLS CORE STAGE C/O FAC
- NLS OPS SUPPORT BLDG
- NLS CTV CHECKOUT FAC
- NLS LPS II ADDITIONS
- NLS TRANSP & HANDLING EQUIP

PRELIMINARY

LUNAR OUTPOST



SPACE EXPLORATION INITIATIVE

KSC FACILITY COST ASSUMPTIONS
LUNAR REFERENCE
SAT V DERIVED, 2 BOOSTER, 410 FT, 9 F-1A

FACILITIES & GSE

- 2 MOBILE LAUNCHERS (NEW)
- 1 VAB HB CELL (MOD)
- 1 PAD (MOD)
- 1 LUNAR PAYLOAD ENCAPSULATION FAC (NEW)
- 2 MLP PARK SITES (NEW)
- 1 SRB STACKING FACILITY (NEW)
- 1 STAGE/ET CHECHOUT FACILITY (NEW)
- 1 LAUNCH ASSESSORIES DEVELOP LAB (NEW)
- 1 OPERATIONS SUPPORT BUILDING (NEW)
- 1 LES & ORD STORE (MOD)
- TRANSP & HANDLING (NEW)
- LPS II ADDITIONS

PRELIMINARY

MOBILE LAUNCHERS

SAT V DERIVED, 2 BOOSTER, LUNAR VEHICLE

NASA
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John F. Kennedy Space Center

KSC FORM 4-607 (REV. 1/88)

RS&H®

SUBJECT: S E I FACILITIES AT KSC
SAT. V - 2 BOOSTER VEHICLE

DESIGNER: FOSTER

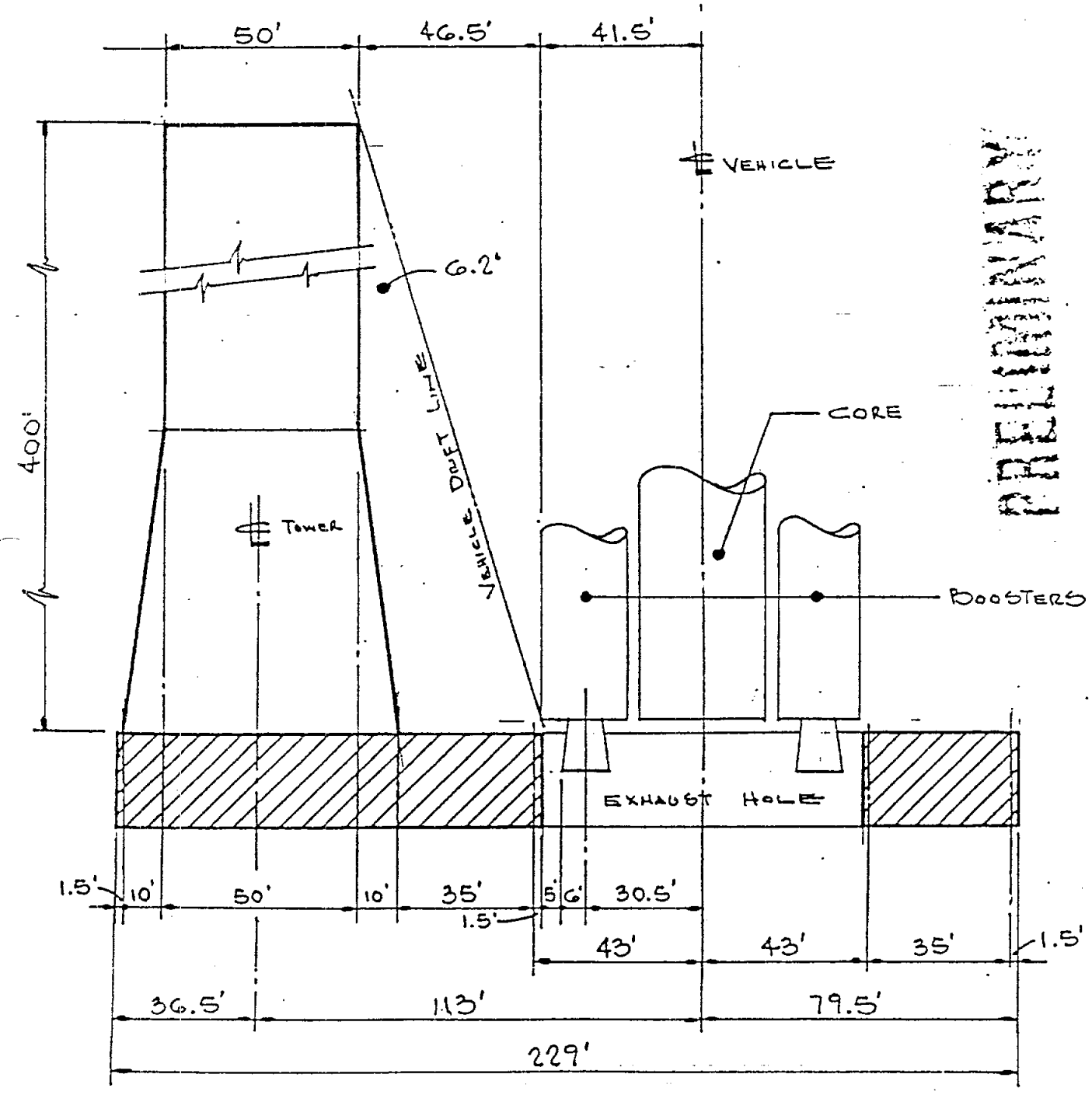
CHECKER:

AEP NO: 290-0520-006T

SHEET: OF:

DATE: MAR. 1992

DATE:



ESTABLISH LAUNCHER LENGTH

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INCORPORATED

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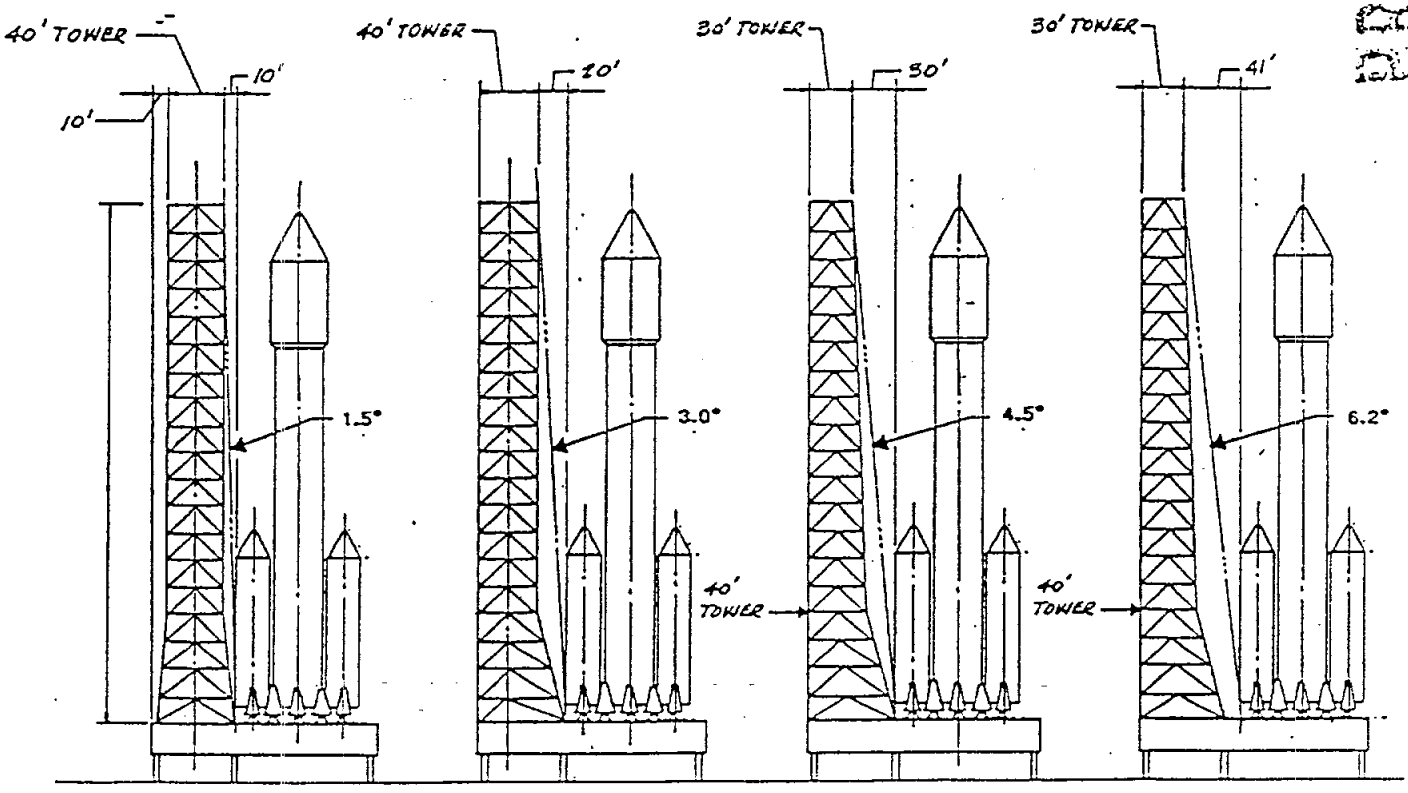
DESIGNER FOSTER
CHECKER
SHEET OF
DATE MAR. 1992
DATE

- LUNAR MLUT -
N-S VEHICLE ORIENTATION
TOWER LIFT-OFF DRIFT CLEARANCE OPTIONS

(SATURN V CLEARANCE WAS 6.2°)



SATURN V TOWER CONFIGURATION (REF)

MOVE TOWER 10' BACK ON LAUNCHER

MOVE TOWER 10' AND TAPER 10'

MOVE TOWER 10' AND TAPER 10' AND MOVE VEHICLE 11'

TASK I

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SUBJECT: S E I FACILITIES AT KSC
SAT. V - 2 BOOSTER VEHICLE
DESIGNER: FOSTER
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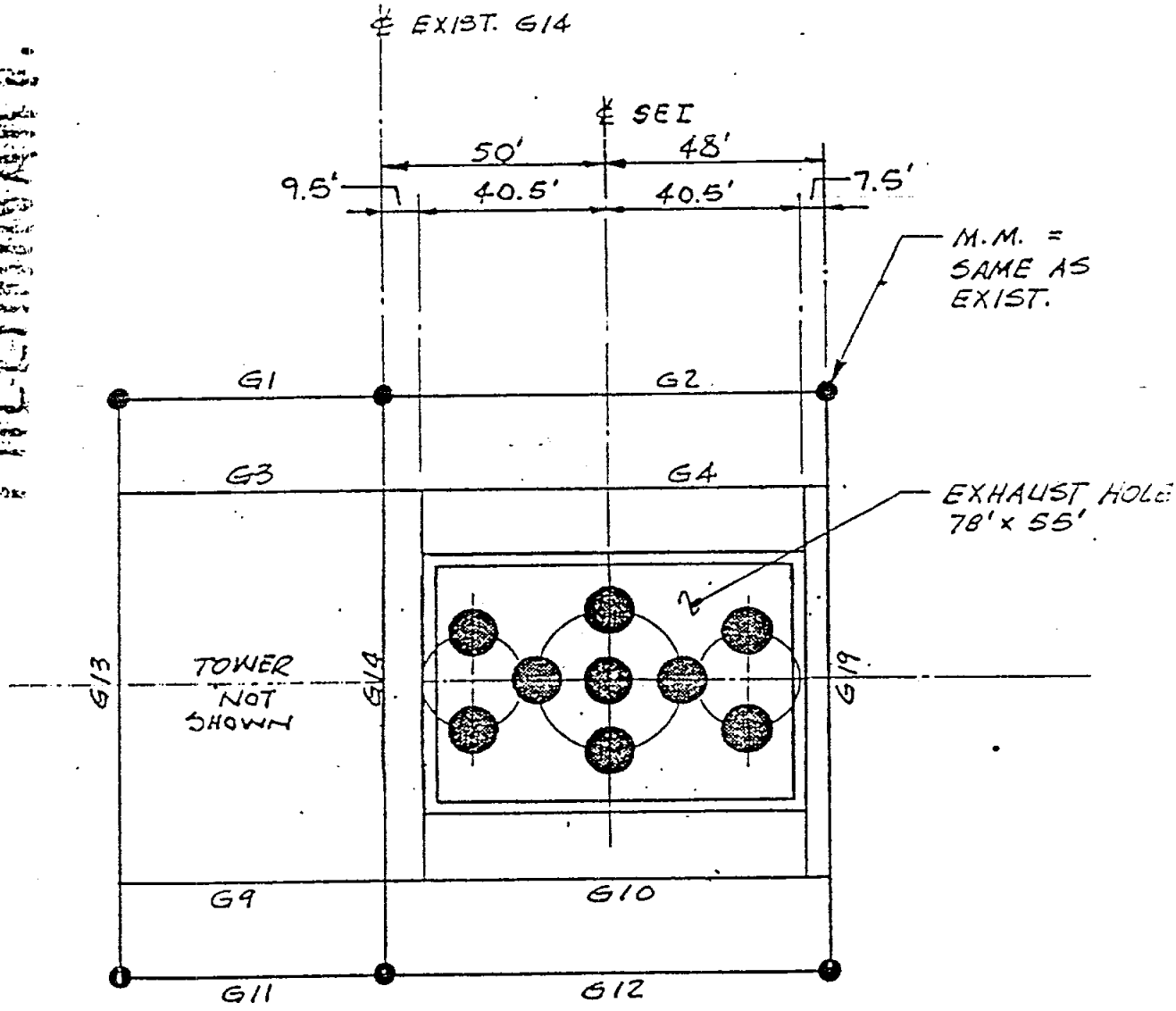
DATE: MAR. 1992
DATE:

PRELIMINARY



GIRDER NUMBERS SHOWN ARE SAME LOCATION AS STS.

←N— AT PAD

PLAN - LUNAR SATURN V - 2 BOOSTER LAUNCHER

1" = 40'

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SAT. V - 2 BOOSTER VEHICLE

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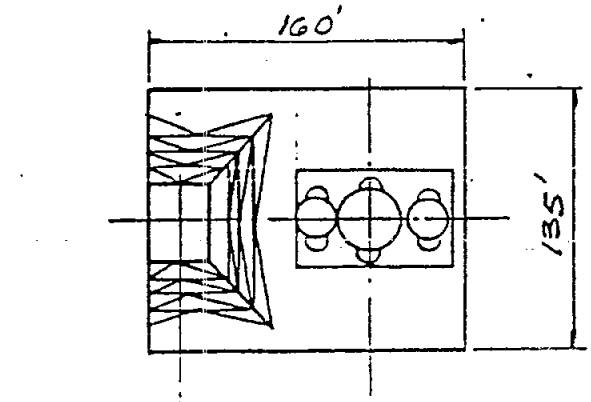
DATE: MAR. 1992

DATE:

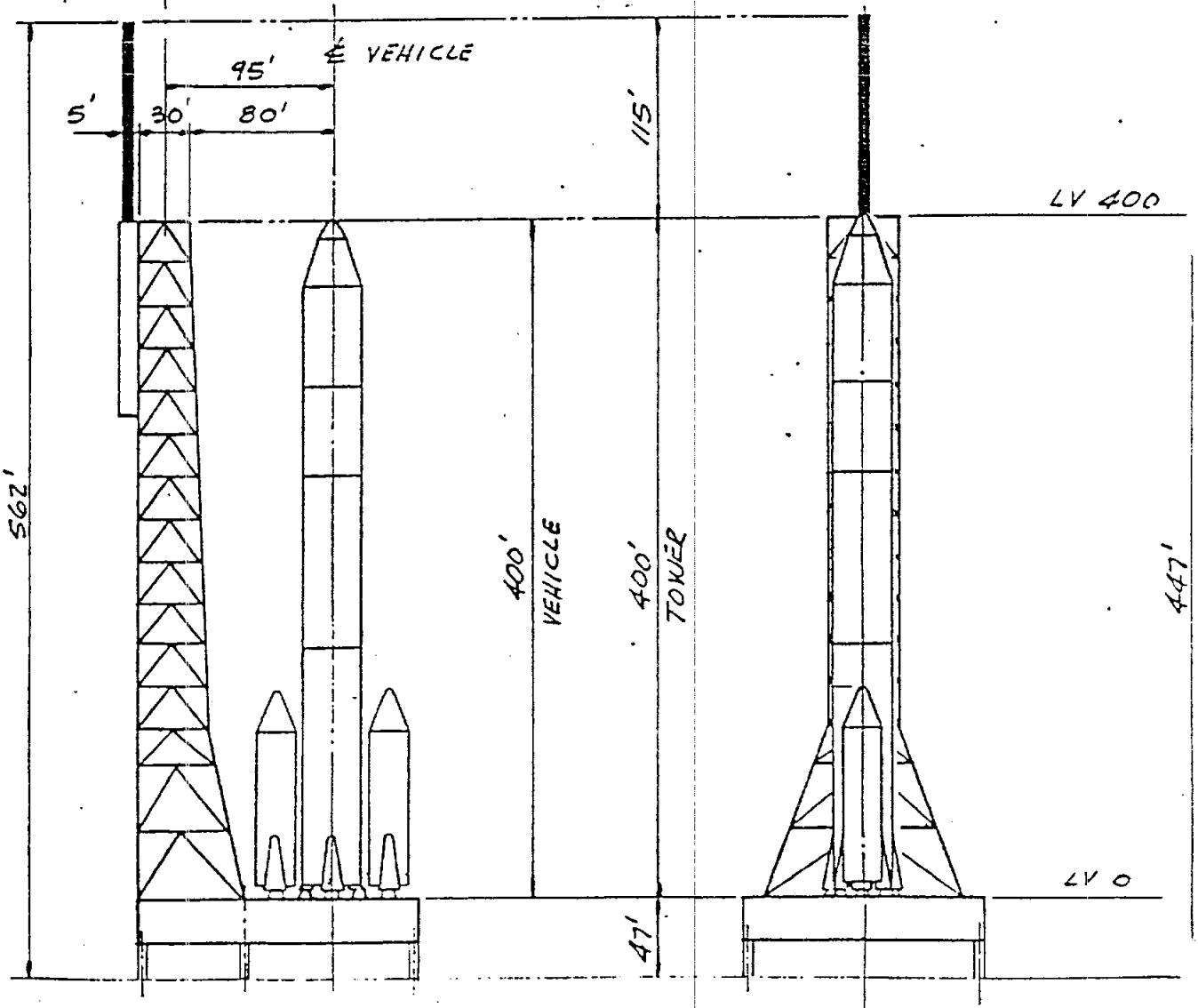
LUNAR SATURN V DERIVED HLLV SINGLE LAUNCH 2 BOOSTER VEHICLE

1" = 100'

- MLUT & VEHICLE



PLAN



SIDE VIEW

END VIEW

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SAT. V - 2 BOOSTER VEHICLE
DESIGNER: FOSTER
CHECKER:

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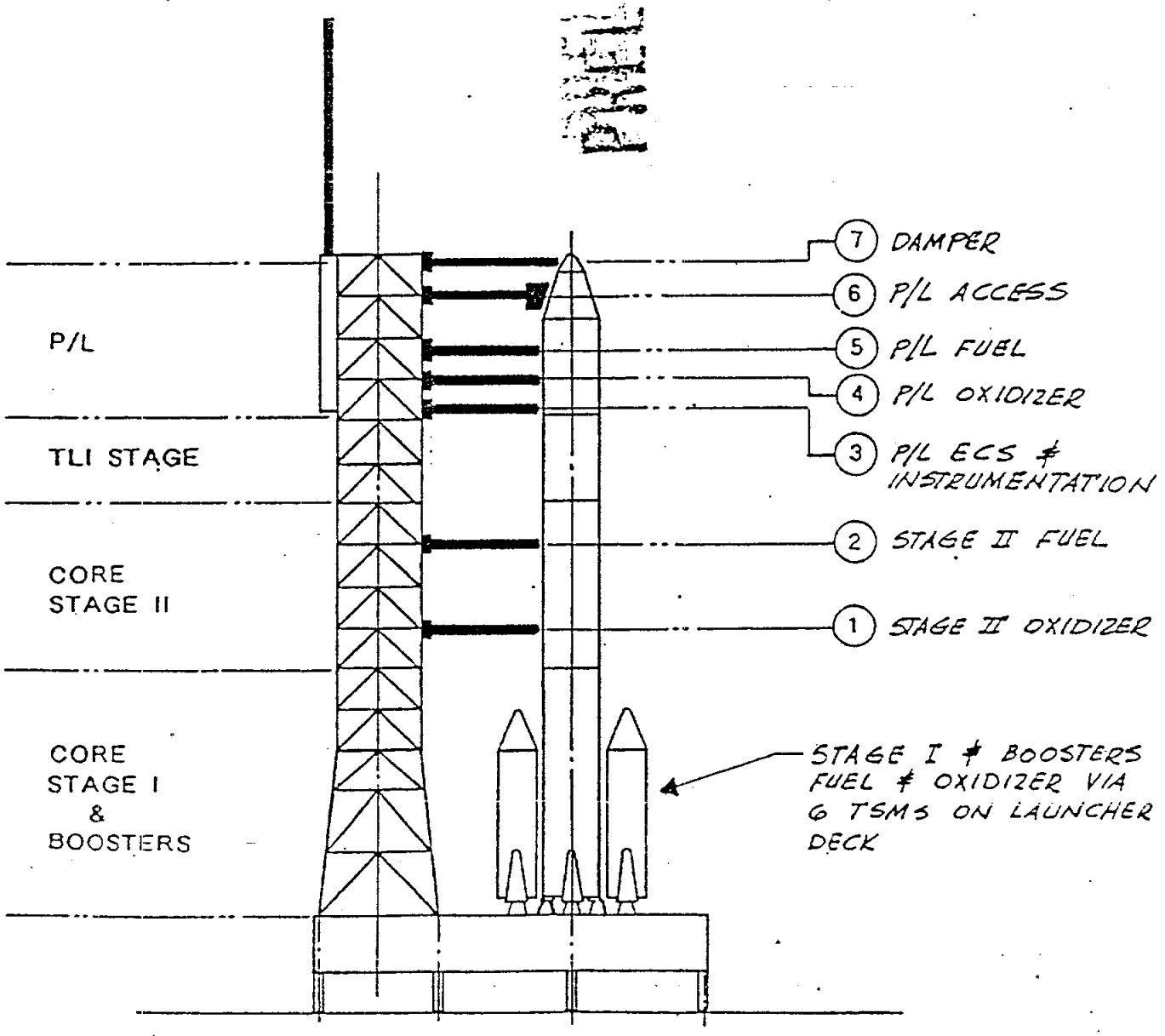
SHEET: OF:
DATE: MAR. 1992
DATE:

LUNAR SATURN V DERIVED HLLV SINGLE LAUNCH 2 BOOSTER VEHICLE

1" = 100'

- SERVICE ARMS -

PRELIMINARY



VAB MODS

SAT V DERIVED, 2 BOOSTER, LUNAR VEHICLE

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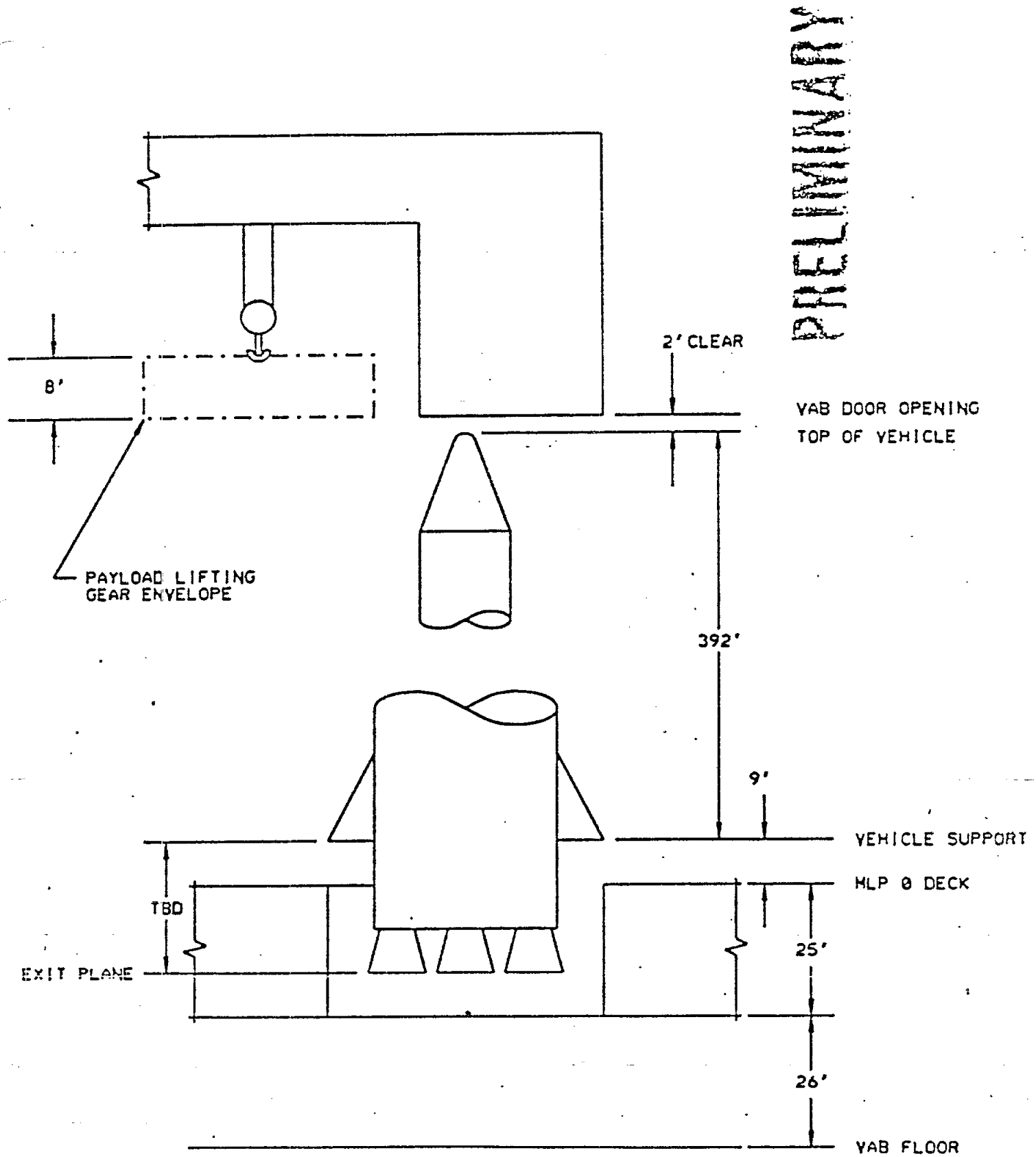
KSC FORM 4-507 (REV. 1/88)


PRELIMINARY
2' CLEAR
8'
VAB DOOR OPENING
TOP OF VEHICLE
PAYLOAD LIFTING
GEAR ENVELOPE
392'
9'
VEHICLE SUPPORT
MLP 0 DECK
TBD
25'
EXIT PLANE
26'
VAB FLOOR

VAB MODIFICATIONS

- MODIFY ONE VAB HIGHBAY CELL (HB2)
- ADD VAB BASIC SERVICES
- ADD HIGH BAY EXTENSIBLE PLATFORMS
- ADD AUXILIARY PLATFORMS
- ADD HAZARDOUS PAYLOAD CLEANROOM

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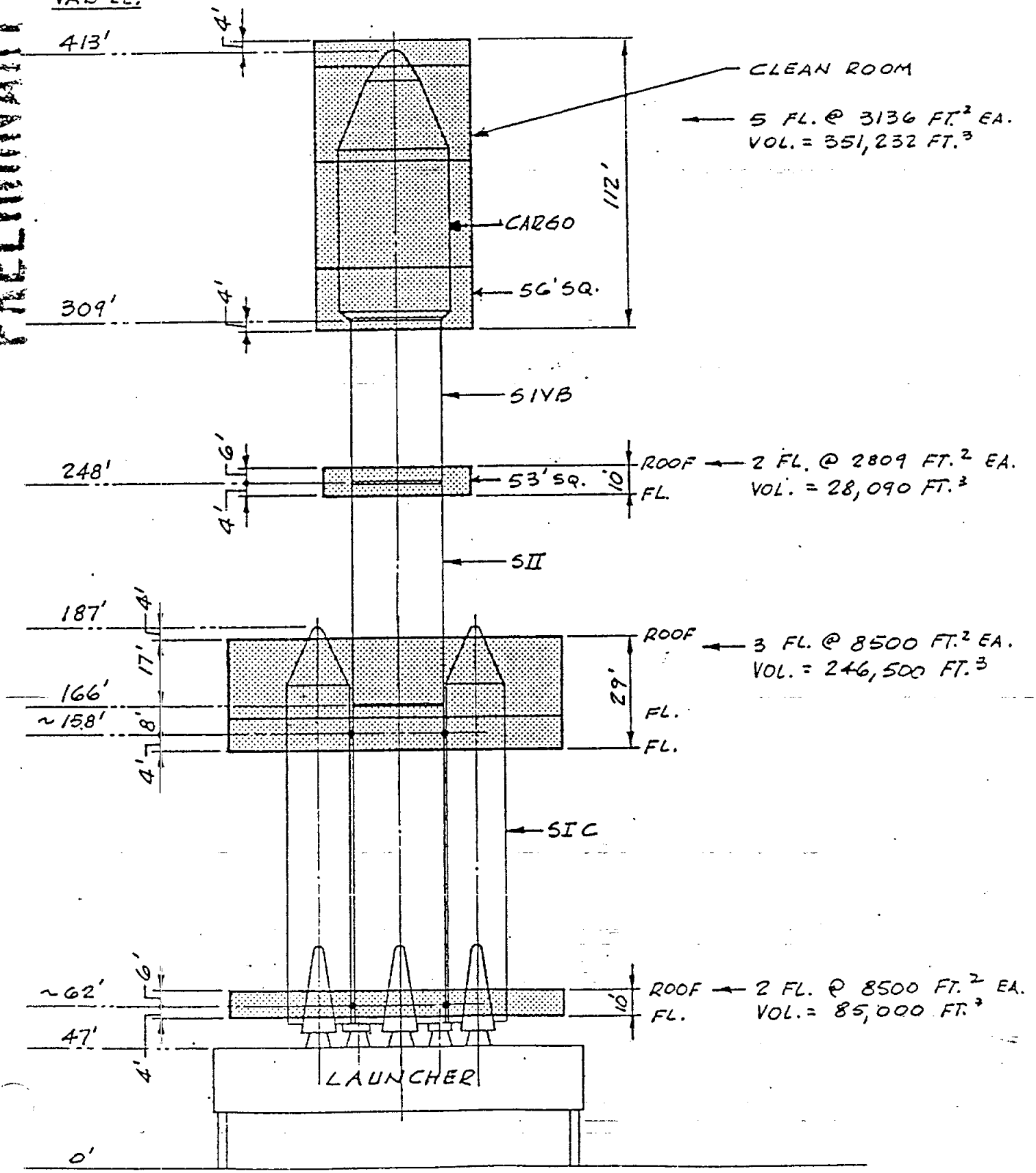
DATE FEB 1992
CHECKER
DATE

VAB PLATFORMS FOR SEI LUNAR VEHICLE

PRELIMINARY



TASK I

PAD MODS

SAT V DERIVED, 2 BOOSTER, LUNAR VEHICLE

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SPACE EXPLORATION INITIATIVE

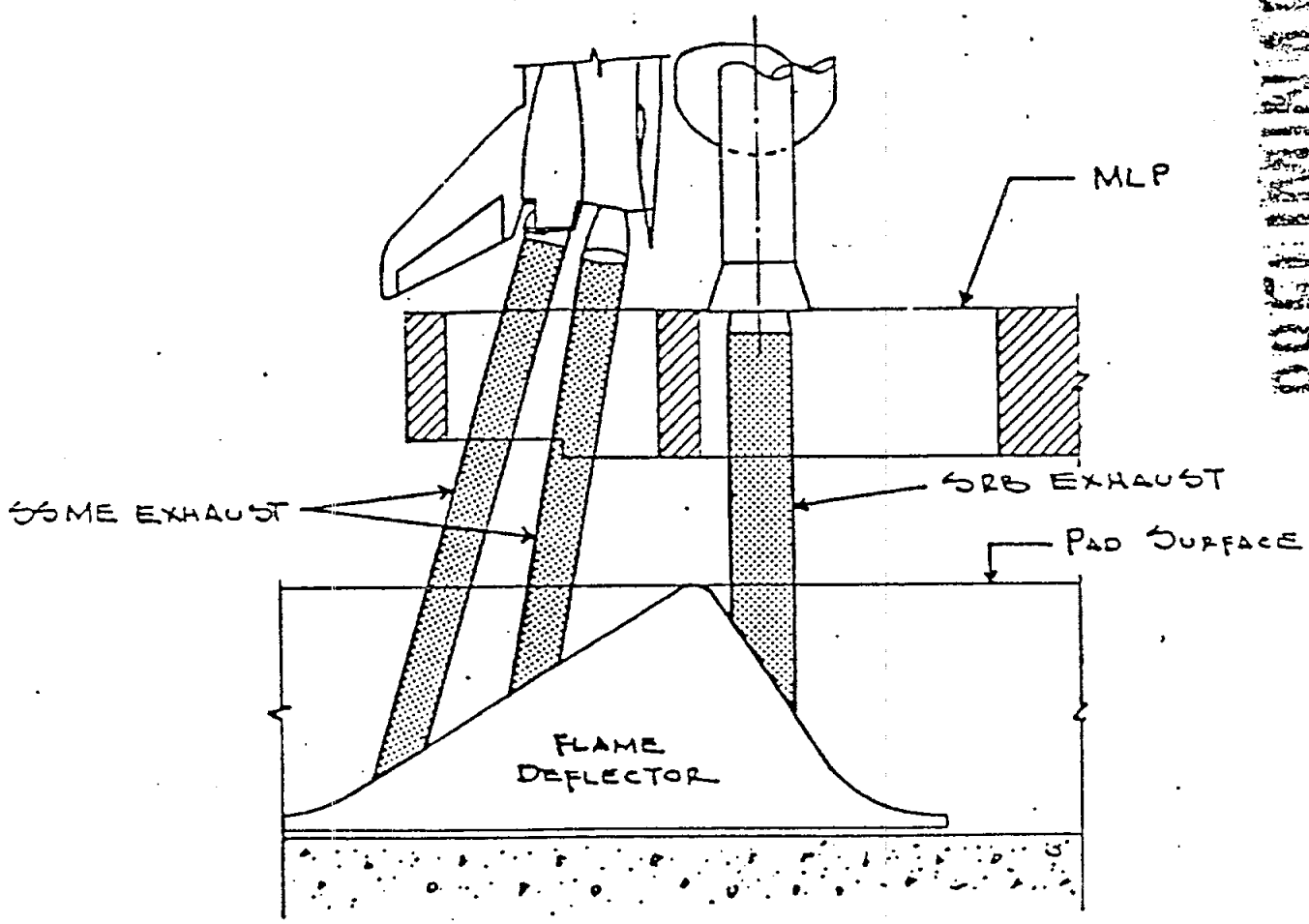
KSC FACILITY COSTS
LUNAR SAT V, 2 BOOSTER
PAD MODIFICATIONS

- MODIFY ONE PAD
- INCREASE STORAGE AND SERVICING CAPACITY OF EXISTING LOX, LH2, HYPERGOLIC, GASES, AND OTHER SYSTEMS
- ADD RP-1 STORAGE AND LOADING SYSTEM
- MODIFY FLAME DEFLECTOR
- ADD LUNAR PAYLOAD SUPPORT REQUIREMENTS

PRELIMINARY

EXISTING STS EXHAUST PATTERNS

INCLUDED FOR REFERENCE



ELEVATION - LOOKING WEST

1/32" = 1'-0"

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DATE FEB 1992

DATE

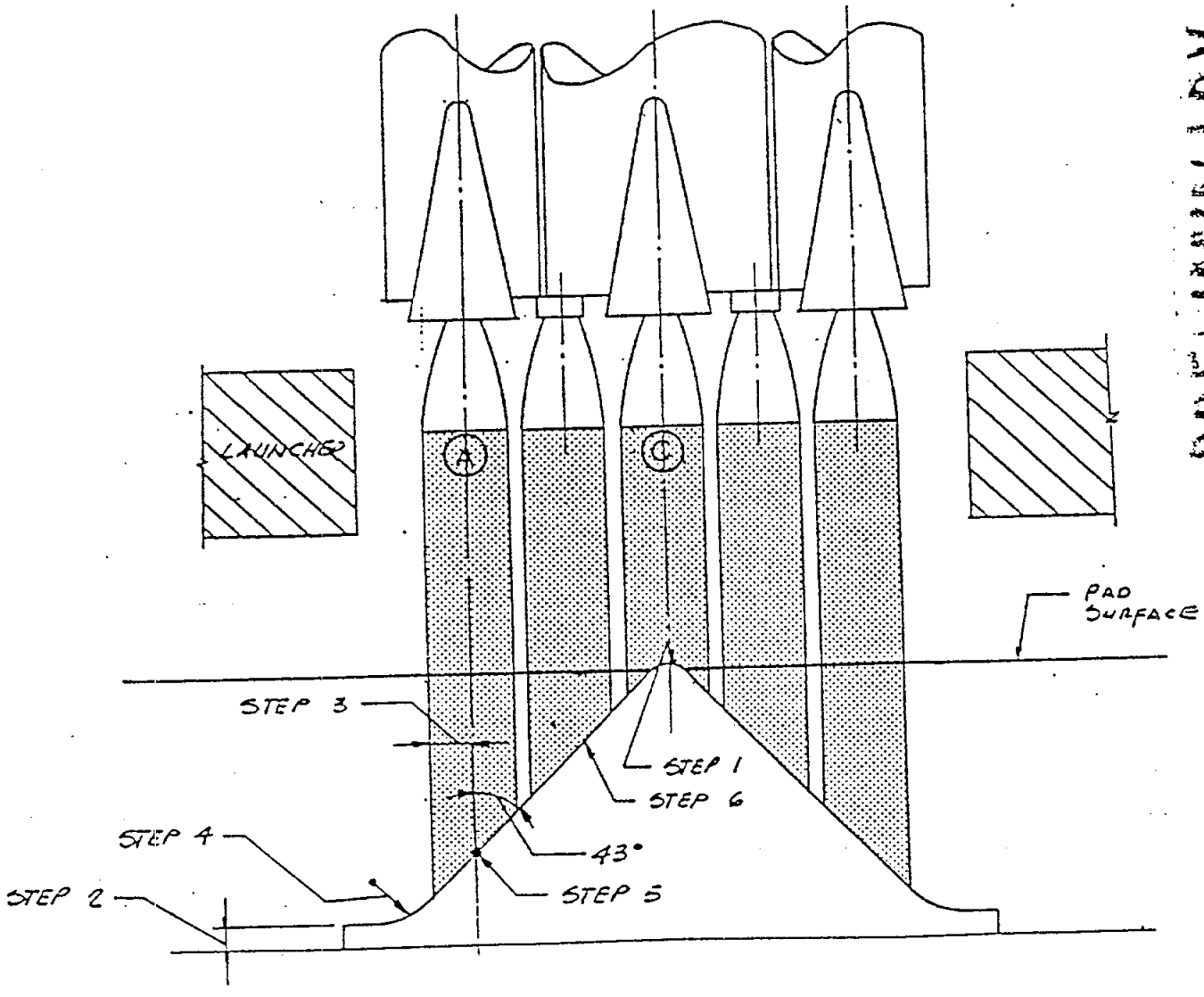
VEHICLE = N-S

CONSTRUCT DEFLECTOR

MAKE IT WORK

STEP 1 - ESTAB. APEX AT 36' BELOW NOZZ. (C).
STEP 2 - LAY IN DEFLECTOR EXIT LIP THK.
STEP 3 - LAY IN 1/2 NOZZ DIA. DIMENSION.
STEP 4 - EXIT RADIUS = 1.7 x 12' ø = 20' R - LAY IN RADIUS.
STEP 5 - CONNECT RADIUS WITH ℄ ENG. (A).
STEP 6 - COMPLETE SURFACE TO APEX.

RESULTS - 11.96 x 4 x 1.6 = FLAME TRENCH NEEDS TO BE 76.5' W. = 32% WIDER
IMPINGEMENT ANGLE = 43° (13° MORE THAN CRITERIA ALLOWS).



PRELIMINARY

TASK 1

SPACE EXPLORATION INITIATIVE

LUNAR PAYLOAD ENCAPSULATION FACILITY

SAT V DERIVED, 2 BOOSTER, LUNAR VEHICLE

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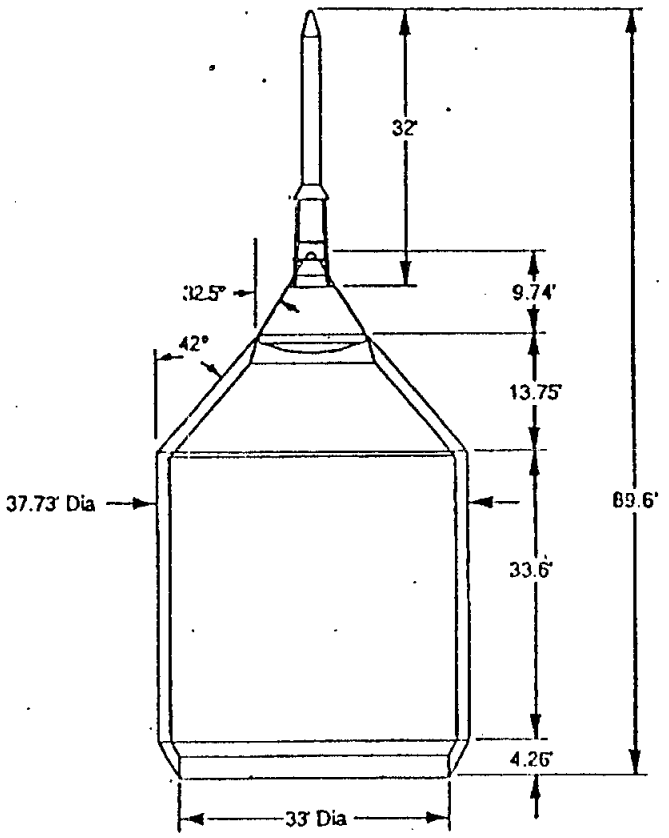
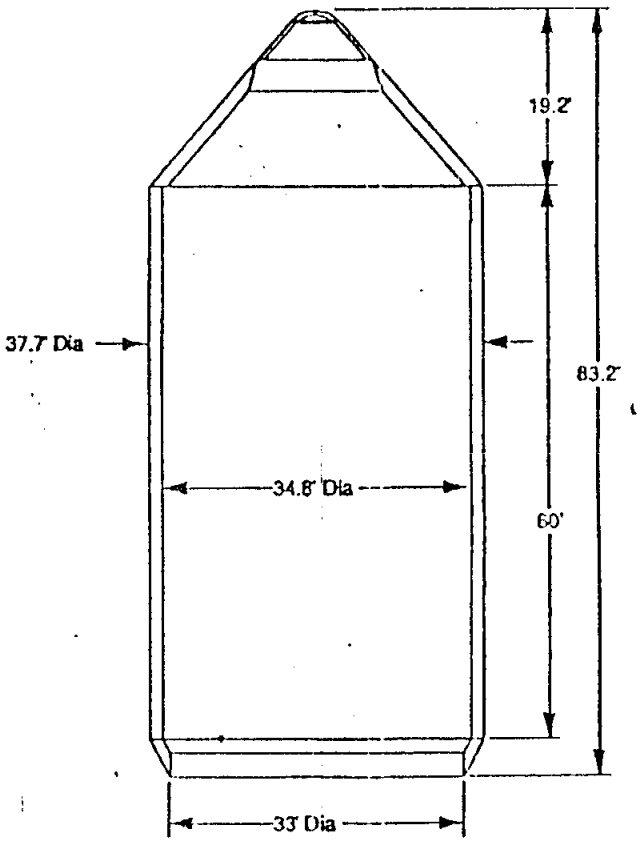
Lunar Manned Payload Fairing for Saturn V Derived Vehicle



Lunar Cargo Payload Fairing for Saturn V Derived Vehicle



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SUBJECT S E I FACILITIES AT KSC
SAT. V - 2 BOOSTER VEHICLE

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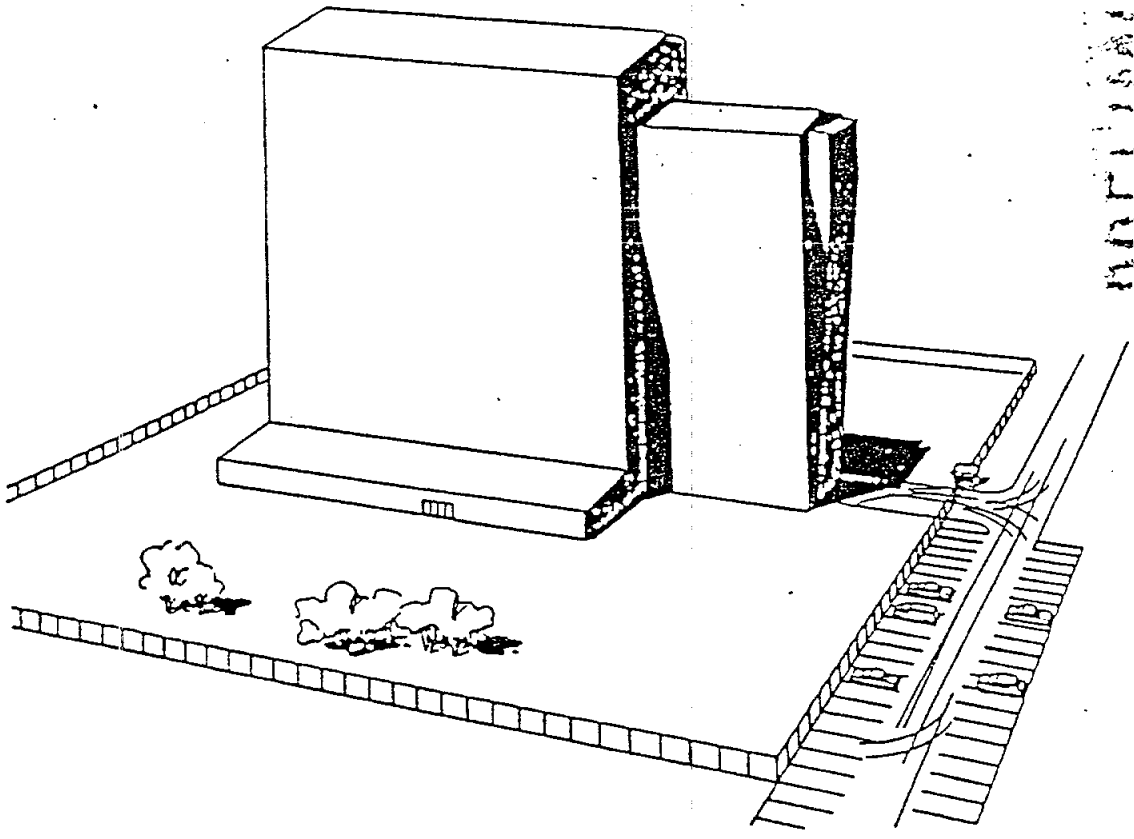
CHECKER

DATE



SEI - PAYLOAD ENCAPSULATION FACILITY

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SUBJECT: S E I FACILITIES AT KSC
SAT. V - 2 BOOSTER VEHICLE

DESIGNER: FOSTER

CHECKER:

AEP NO: 290-0520-006T

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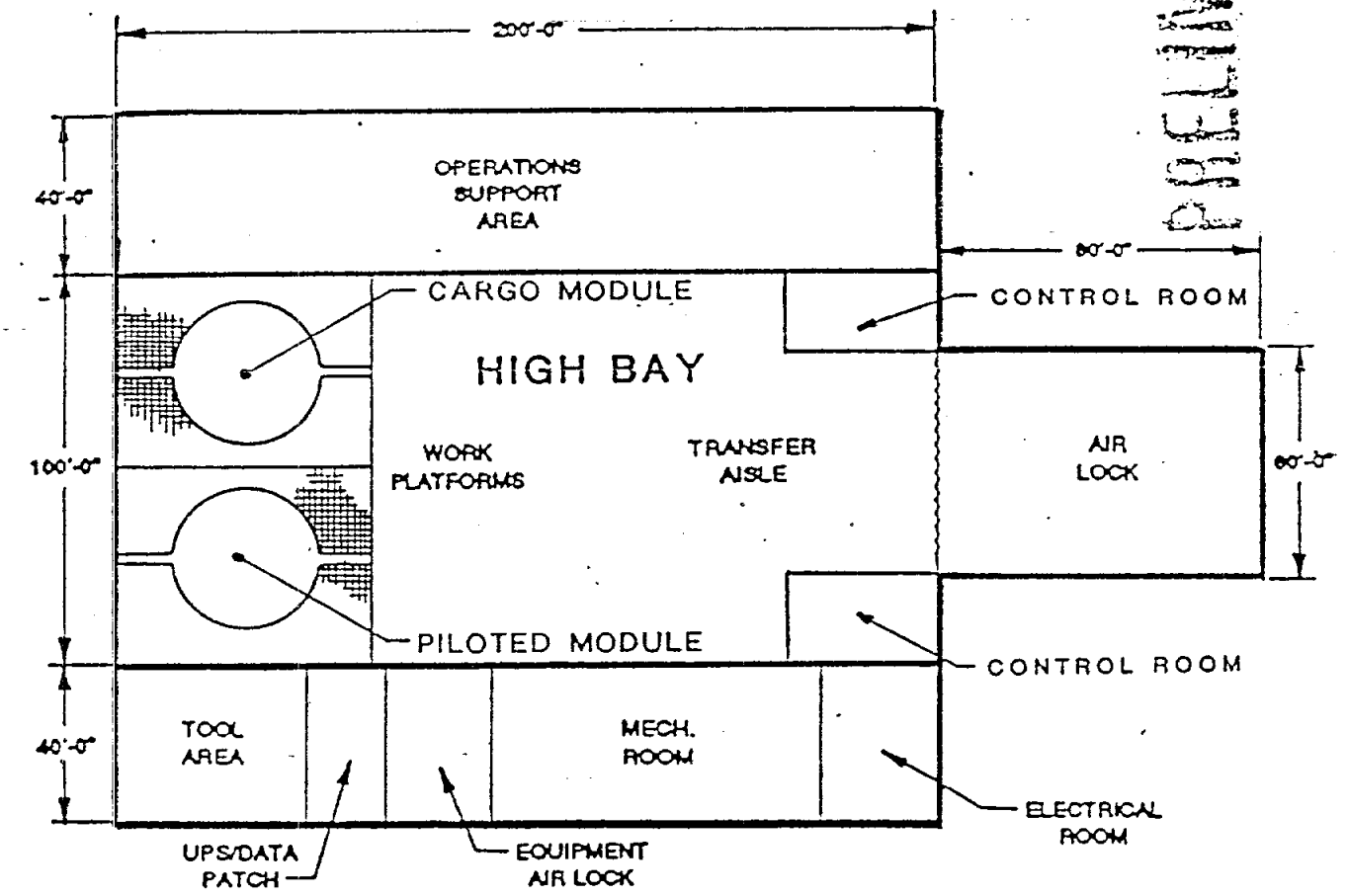
DATE: MAR. 1992

DATE:

PRELIMINARY



N

FLOOR PLAN

SEI - PAYLOAD ENCAPSULATON FACILITY

SRB

STACKING

FACILITY

SAT V DERIVED, 2 BOOSTER, LUNAR VEHICLE

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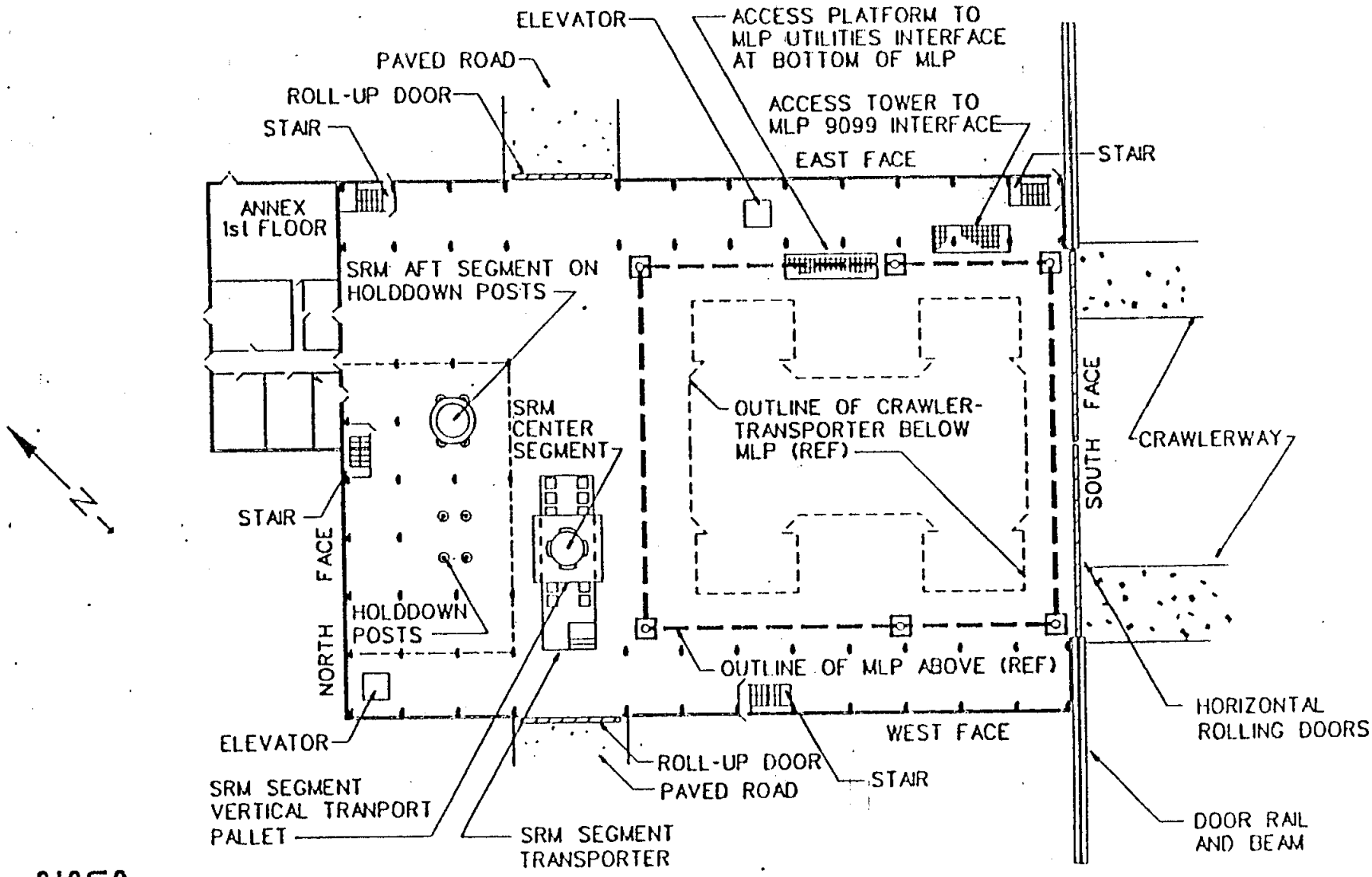
SRB BUILDUP AI) STACKING FACILITY

BUILDING PLANS



PLAN AT GROUND FLOOR - EL. 0'-0"

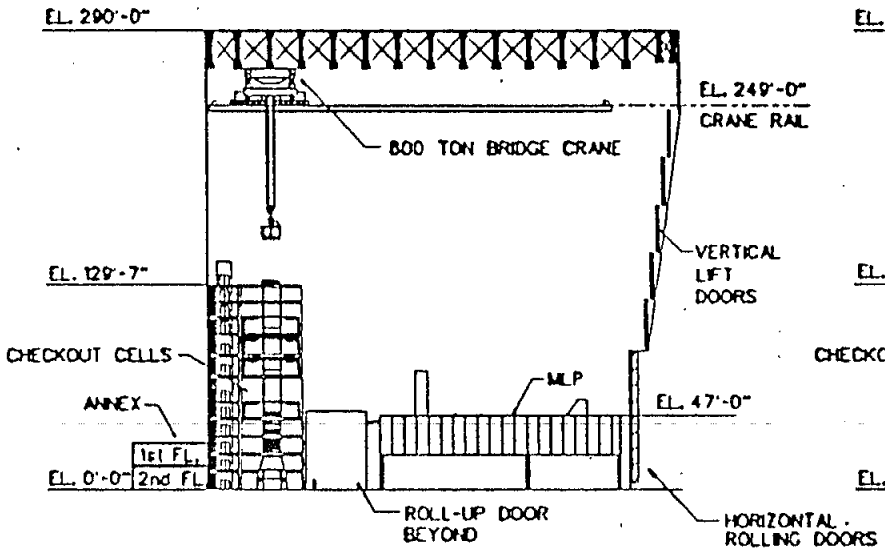
NASA

KSC FORM 4 607 (REV. 1/88)

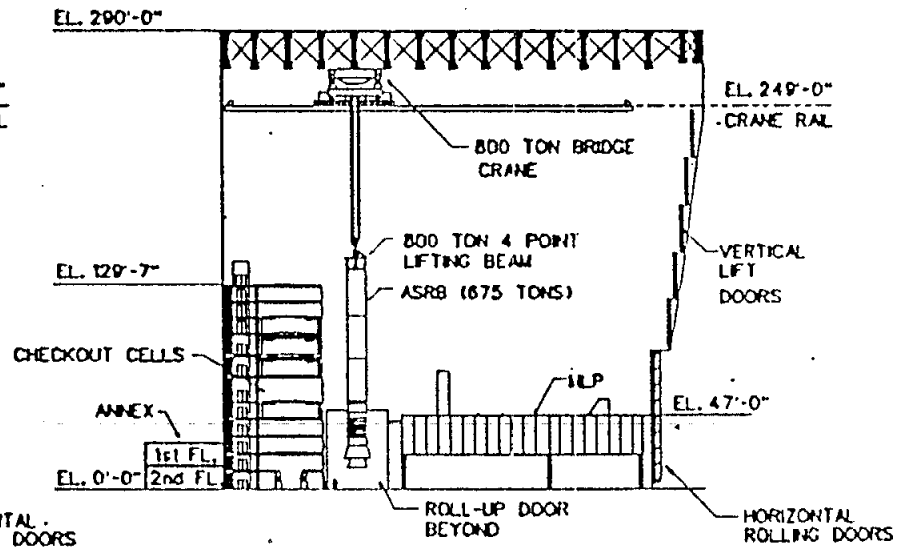
PRELIMINARY

SRB BUILDUP AND STACKING FACILITY

SECTION VEIWS



SRM IN CHECKOUT CELLS



LIFTING SRB OUT OF CHECKOUT CELL

PRELIMINARY

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SPACE EXPLORATION INITIATIVE

STAGE / BOOSTER / ET

CHECKOUT

FACILITY

SAT V DERIVED, 2 BOOSTER, LUNAR VEHICLE

NASA
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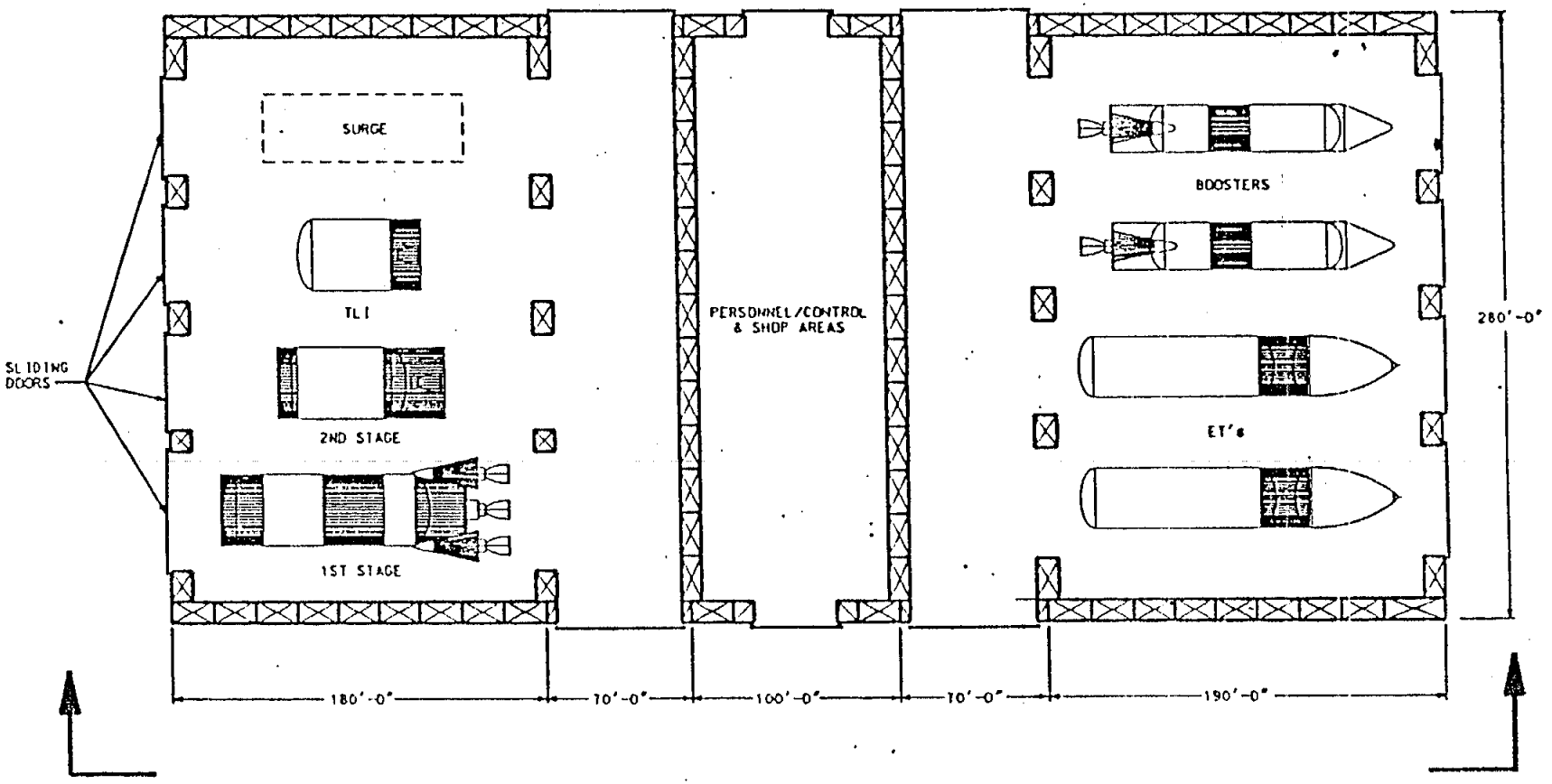
John F. Kennedy Space Center

KSC FORM 4-607 (REV. 1/88)

LUNAR SINGLE VEHICLE CONFIGURATION
SATURN V DERIVED
2 BOOSTER VEHICLE
STAGE/BOOSTER/ET CHECKOUT FACILITY



PLAN

PRELIMINARY

SPACE EXPLORATION INITIATIVE

LAUNCH ASSESSORIES DEVELOPMENT LAB (LADL)

SAT V DERIVED, 2 BOOSTER, LUNAR VEHICLE

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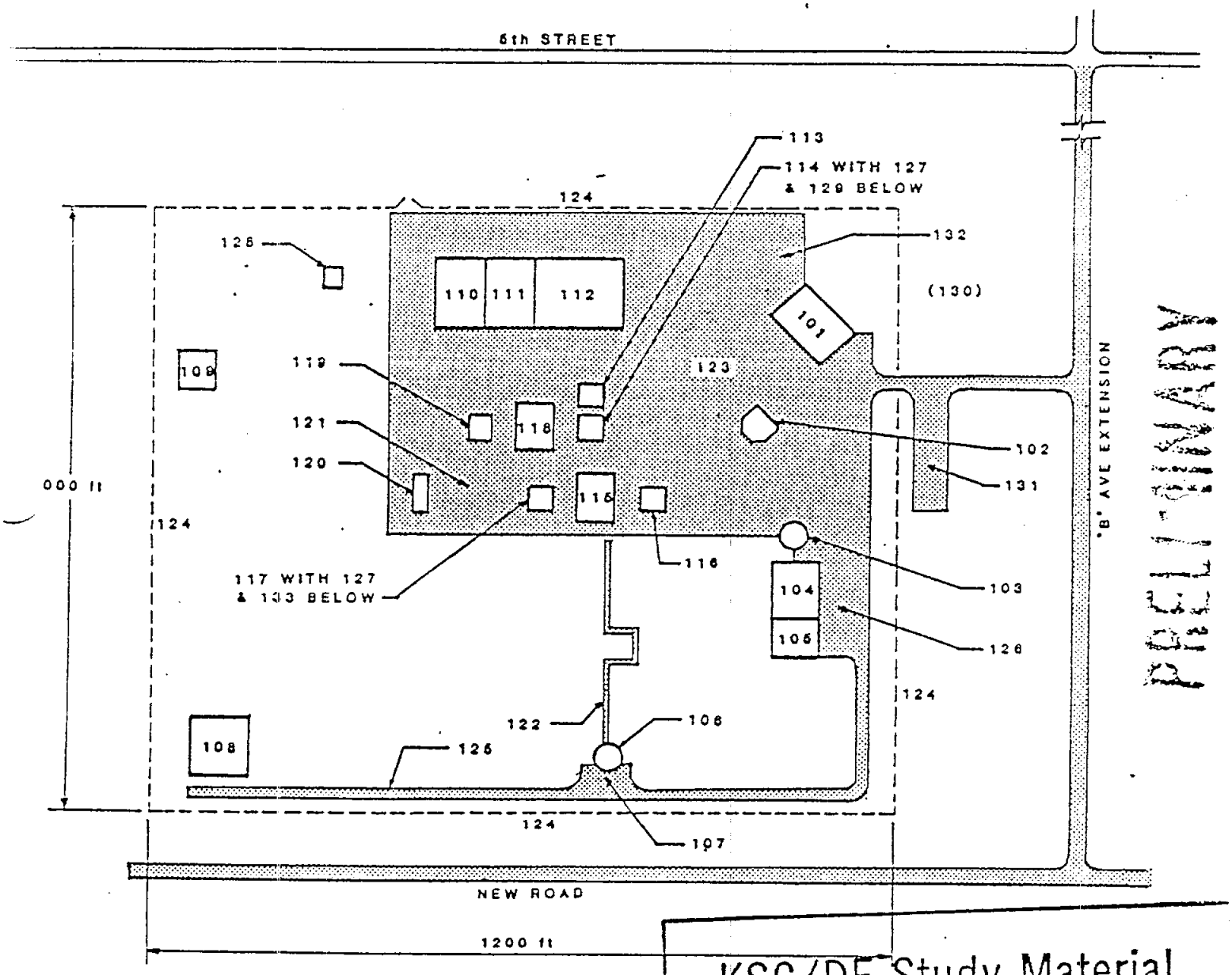
DESIGNER _____ DATE _____

CHECKER _____ DATE _____



SITE PLAN –
1" : 200'

PRELIMINARY

KSC/DE Study Material
PRELIMINARY
Reference Not Authoriz

I.D.	NAME
101	FACILITY OFFICE BUILDING
102	CONTROL ROOM
103	LIQUID NITROGEN (LN_2) TANK
104	PNEUMATIC TEST AREA
105	GASEOUS HELIUM (GHe) STORAGE AREA
106	LIQUID HYDROGEN (LH_2) TANK
107	LH_2 OFF-LOAD AREA
108	HYPERGOL TEST AREA
109	DESTRUCTIVE TEST AREA
110	CLEAN HIGH BAY
111	HIGH BAY
112	LOW BAY
113	HOLDDOWN POST TEST AREA
114	TEST TOWER 114
115	RANDOM MOTION SIMULATOR 115 (RMS-115)
116	TEST STAND 116
117	TEST TOWER 117
118	RANDOM MOTION SIMULATOR WITH LIFT-OFF SIMULATOR (RMS/LOS-118)
119	TEST STAND 119
120	WATER (H_2O) TANK
121	H_2O TEST AREA
122	LH_2 CROSS COUNTRY LINES
123	TEST AREA PAVEMENT
124	PERIMETER FENCE AND GATES
125	PERIMETER ROAD AND ENTRANCE ROAD
126	PNEUMATIC OFF-LOAD AREA
127	HYDRAULIC SYSTEM AND STORAGE
128	POL
129	MECHANICAL BUILDING
130	FACILITY UTILITIES
131	PARKING LOT
132	GOVERNMENT VEHICLE PARKING SPACES
133	MECHANICAL BUILDING

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SUBJECT _____ AEP NO 290-0520-006

SHEET ____ OF ____

DESIGNER _____ DATE _____

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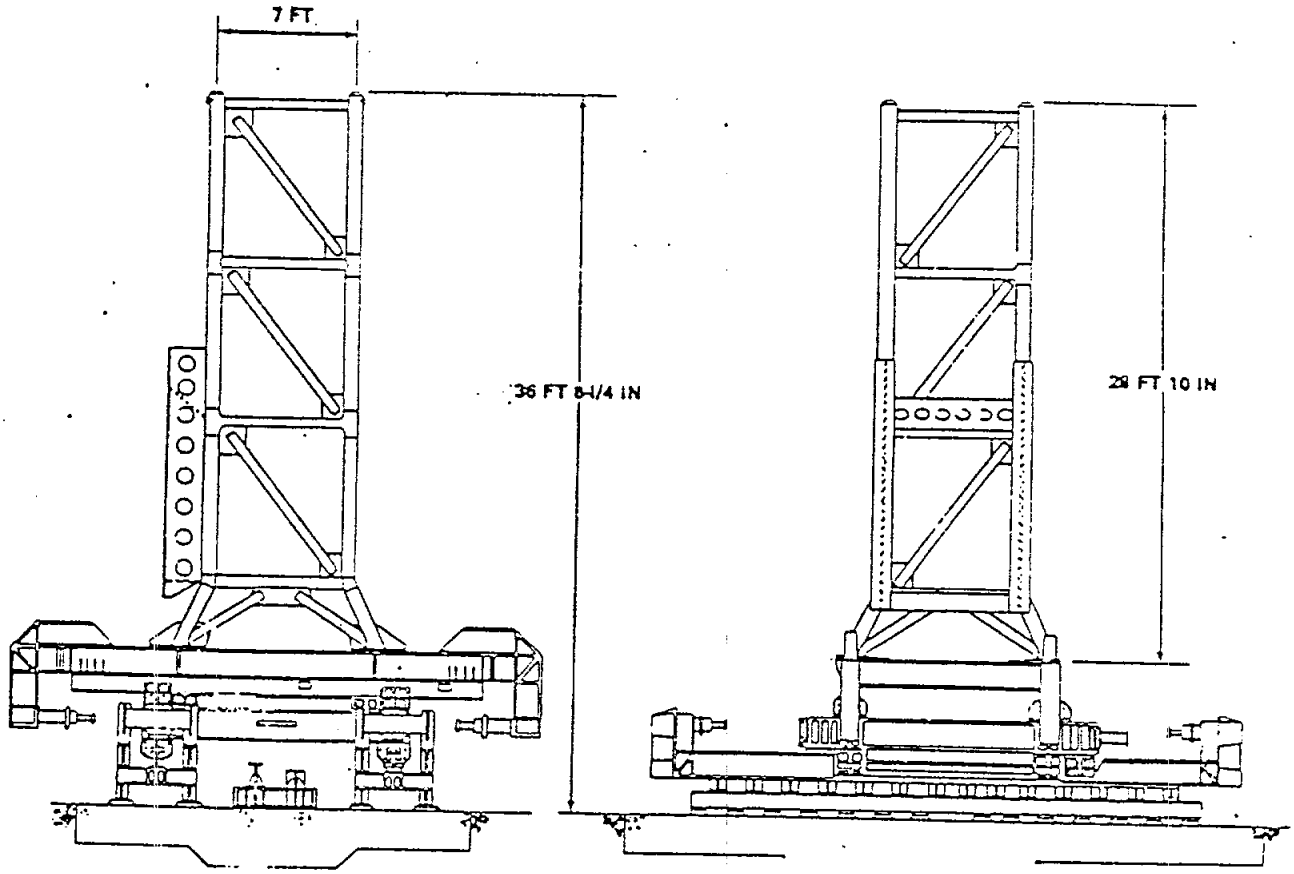
KSC/DE Study Material

PRELIMINARY

Reference Not Authorized



Random Motion Simulator

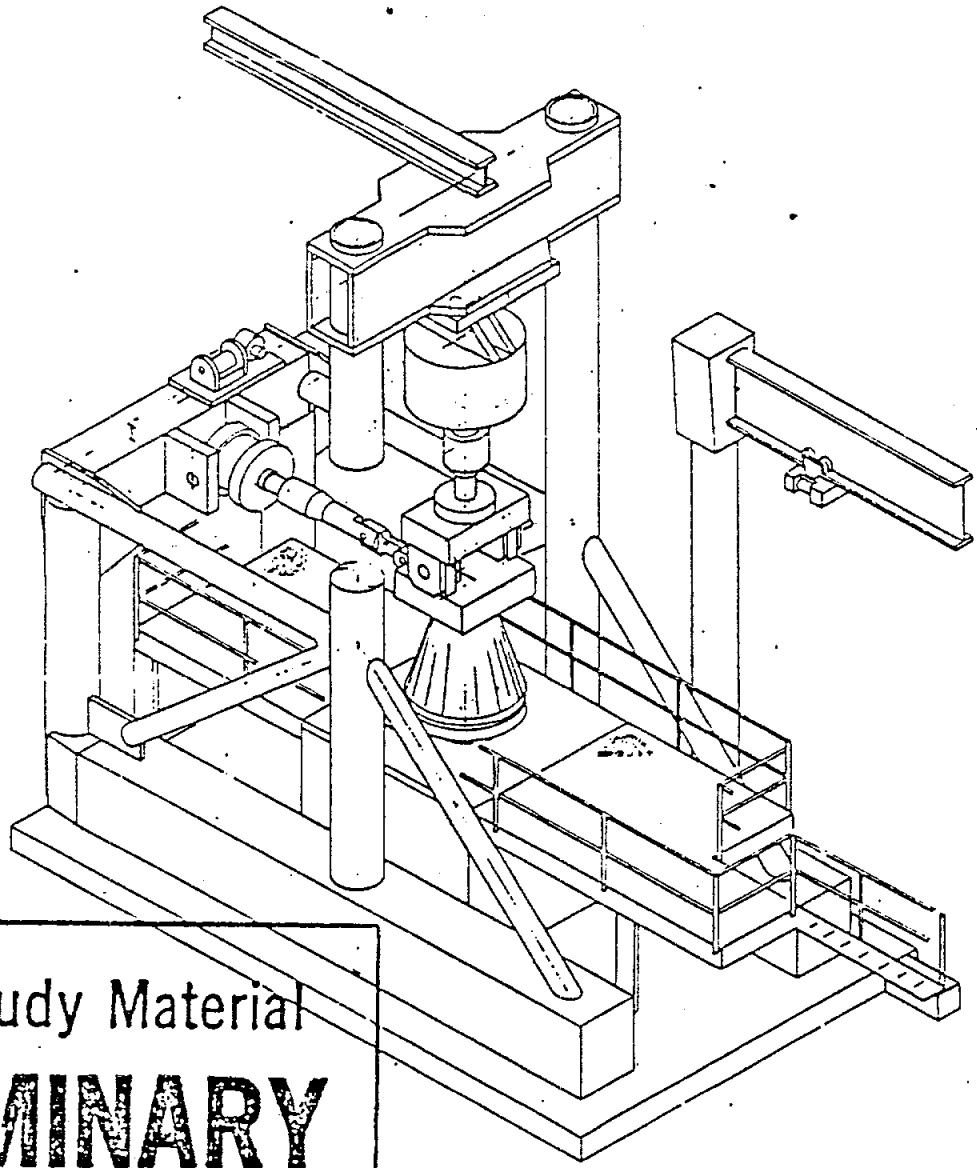
RS&H®

SUBJECT _____ AEP NO 290-0520-006 F

SHEET ____ OF ____

DESIGNER _____ DATE _____

CHECKER _____ DATE _____



KSC/DE Study Material

PRELIMINARY

Reference Not Authorized

SRB Holddown Test Stand

SPACE EXPLORATION INITIATIVE

KSC FACILITY COST ASSUMPTIONS
LUNAR OPTION
NLS DERIVED, 4 BOOSTER, 351 FT, 8 F-1A, 4 STME

FACILITIES & GSE

- 2 MOBILE LAUNCHERS (NEW)
- 1 VAB HB CELL (MOD)*
- 1 PAD EXTENSION (MOD)*
- 1 LUNAR PAYLOAD ENCAPSULATION FAC (NEW)
- 2 MLP PARK SITES (NEW)
- 1 SRB STACKING FACILITY (NEW)
- 1 STAGE/ET CHECKOUT FACILITY (NEW)
- 1 LAUNCH ASSESSORIES DEVELOP LAB (NEW)
- 1 OPERATIONS SUPPORT BUILDING (NEW)
- 1 LES & ORD STORE (MOD)
- TRANSP & HANDLING (NEW)
- LPS II ADDITIONS

PRELIMINARY

NASA
National Aeronautics an
Space Administratio
John F. Kennedy Space Center

MOBILE LAUNCHERS

NLS DERIVED, 4 BOOSTER, LUNAR VEHICLE

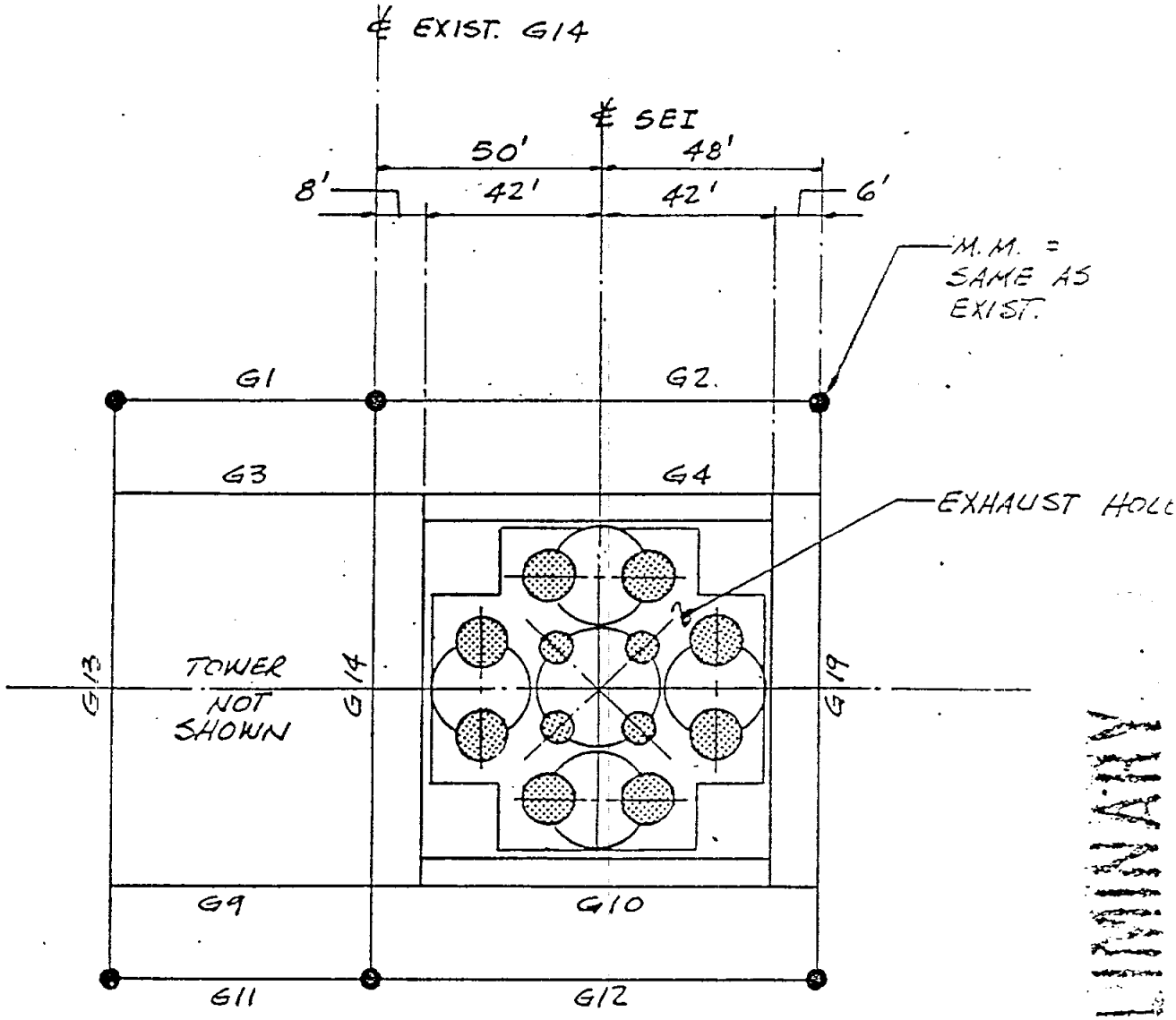
NASA
National Aeronautics an
Space Administratio
John F. Kennedy Space Center

[illegible] 914 507 (REV. 1/88)

RS&H®

SUBJECT: S E I FACILITIES AT KSC
NLS - 4 BOOSTER VEHICLE
DESIGNER: FOSTER
CHECKER:

AEP NO: 290-0520-006T
SHEET: OF:
DATE: MAR. 1992
DATE:



GIRDER NUMBERS SHOWN ARE SAME LOCATION AS STS.

PRELIMINARY

← N — AT PAD

PLAN - LUNAR NLS - 4 BOOSTER LAUNCHER

1" = 40'

VAB

MODIFICATIONS

NLS DERIVED, 4 BOOSTER, LUNAR VEHICLE

NASA
National Aeronautics an
Space Administratio
John F. Kennedy Space Center

FORM 4-607 (REV. 1/88)

RS&H®

SUBJECT: S E I FACILITIES AT KSC
NLS - 4 BOOSTER VEHICLE

DESIGNER: FOSTER

CHECKER:

AEP NO: 290-0520-006T

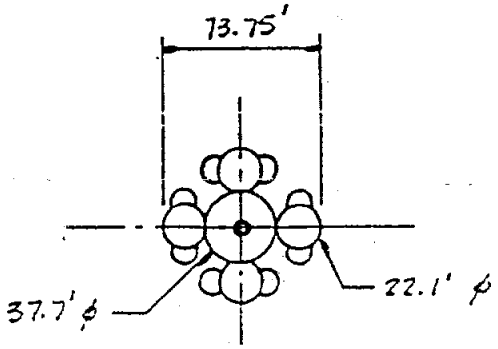
SHEET: OF:

DATE: MAR. 199

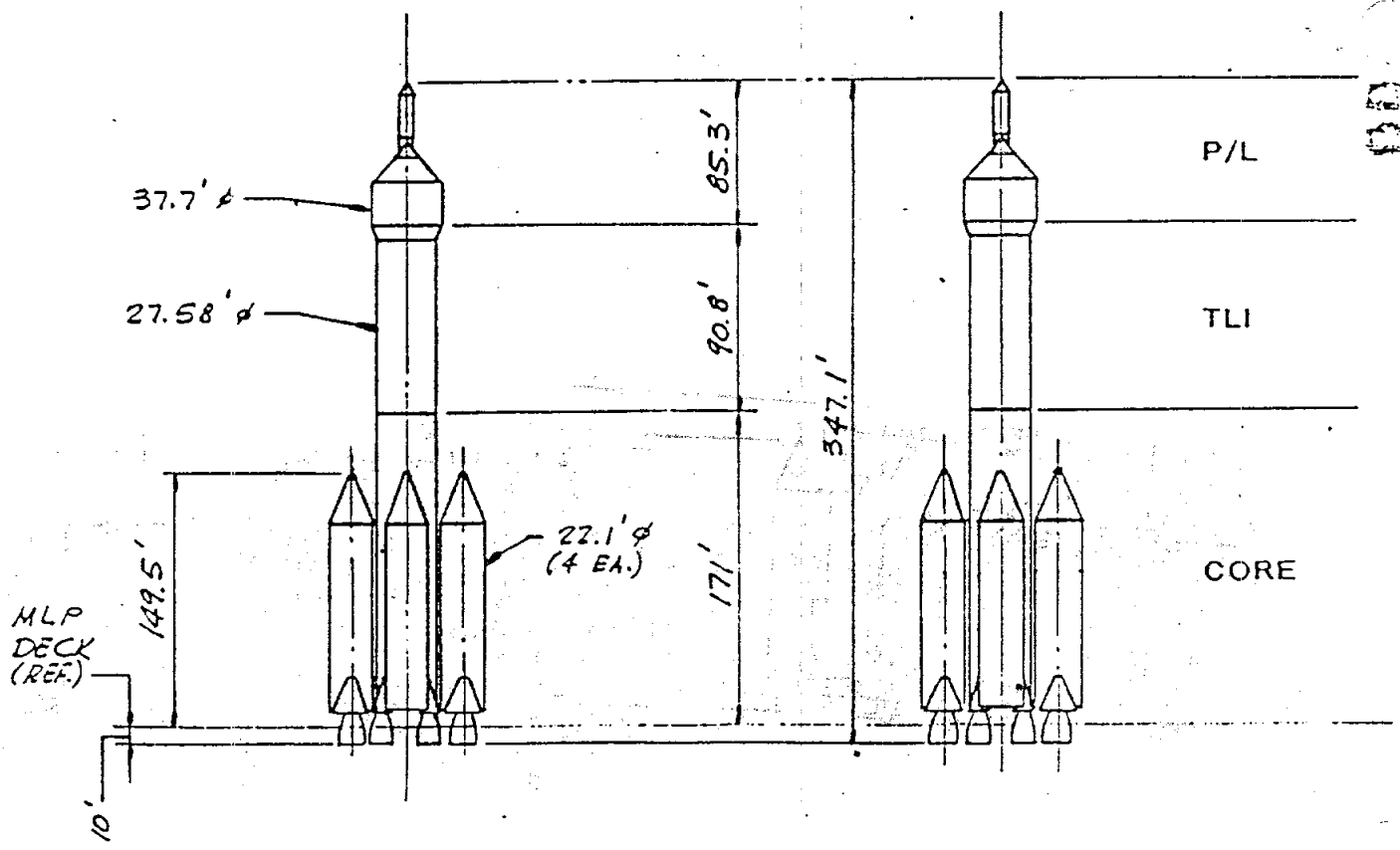
DATE:

LUNAR NLS DERIVED HLLV SINGLE LAUNCH 4 BOOSTER VEHICLE

1" = 100'



ROW : 5.61 MLB
GLOW : 12.4 MLB
8 F-1A ENGINES &
4 STMEs
P/L : 210 KLB



PAD EXTENSION MODIFICATIONS

NLS DERIVED, 4 BOOSTER, LUNAR VEHICLE

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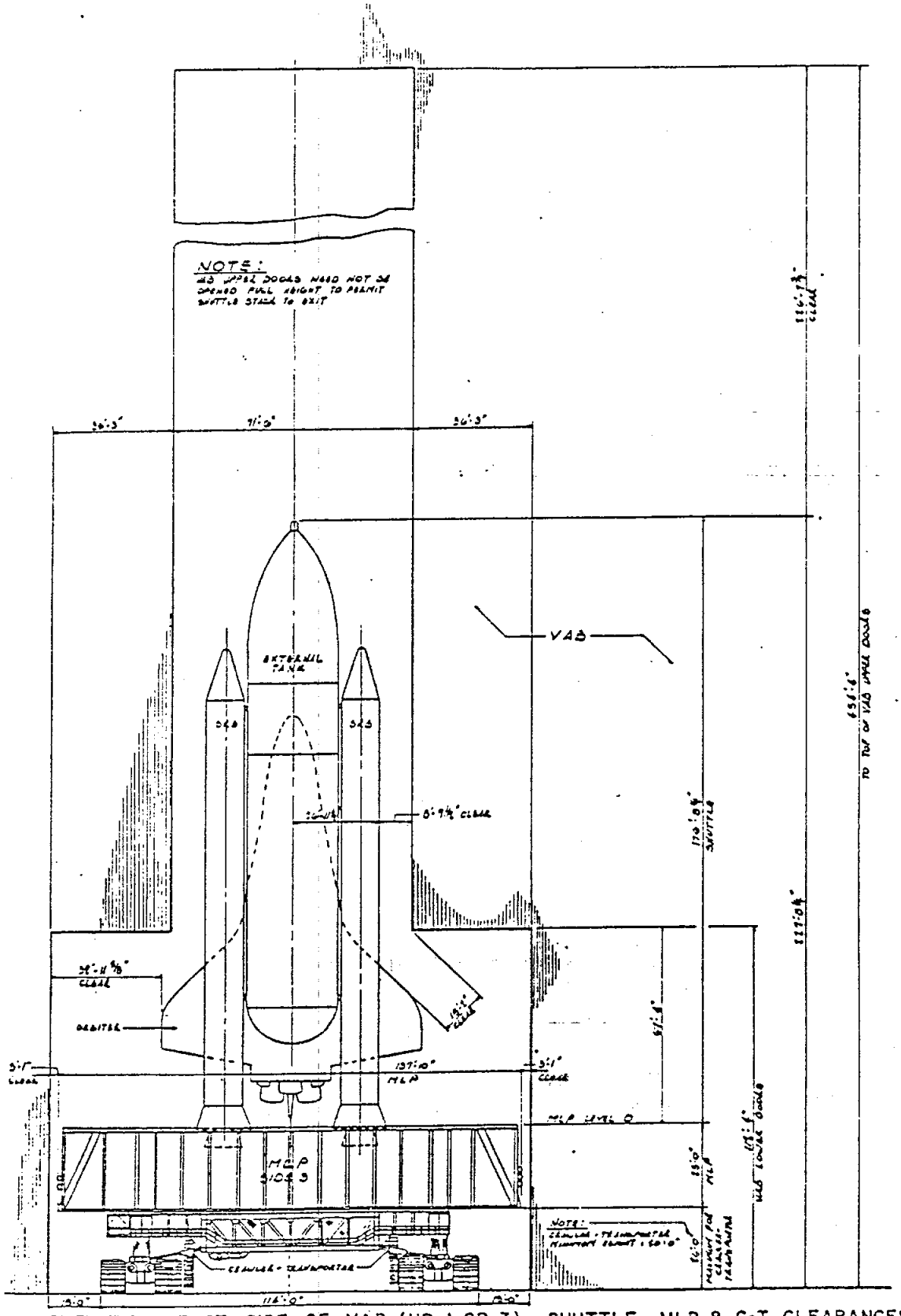

ELEVATION – EAST SIDE OF VAB (HB-1 OR-3) – SHUTTLE, MLP & C-T CLEARANCES