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ADVANCED PLANETARY STUDIES
NINTH ANNUAL REPORT

1 February 1981 to 30 April 1982

AND 5-YEAR CONTRACT SUMMARY

SCIENCE APPLICATIONS, INC.

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AND 5-YEAR CONTRACT SUMMARY

by

Science Applications, Inc.
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for

Earth and Planetary Exploration Division
Office of Space Science and Applications
NASA Headquarters
Washington, D.C. 20546

Contract NASW-3035

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N84-22591#

FOREWORD

This report summarizes the results of advanced studies and planning support by Science Applications, Inc. (SAI) under Contract No. NASW-3035 for the Earth and Planetary Exploration Division, Code EL, of NASA Headquarters. In accordance with the established contractual requirements, a detailed summary of the past year's work is provided in this annual report. The term of performance covered here is the fifteen-month period from 1 February 1981 through 30 April 1982; this includes a three-month extension of the contract. During this period, a total effort of 13,073 man-hours (80 man-months) was expended on the general support activities and six study tasks. The total contract cost expenditure for this period of performance was $538,920.

Work concluded on 30 April 1982 marked the end of a total five-year contract initiated in February 1977. The total level-of-effort and cost expenditure during this five-year period was, respectively, 56,005 man-hours and $2,024,958. On the basis of the negotiated CPFF-LOE contract, these performance numbers represent a level-of-effort underrun of 1079 man-hours (1.9%) and a cost overrun of $5 -- essentially zero. The introduction section of this annual report will also summarize the highlights and significant findings of the entire five-year contract.*

Inquiries regarding further information on the contract results reported herein should be directed to the study leader, Mr. Alan Friedlander, at 312/885-6800.

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* Reference: telephone conversation of 6 Aug 1982 with NASA Contracting Officer, Mr. Donald Andreotta, regarding interpretation of reporting requirements stated in Modification 9 of subject contract.

TABLE OF CONTENTS

1. INTRODUCTION

Science Applications, Inc. (SAI) participates in a program of advanced concepts studies and planning analysis for the Earth and Planetary Exploration Division, Code EL, NASA Headquarters. SAI's charter is to perform preliminary analyses and assessments for Code EL planning activities. Specifically, the objective of this support is to ensure NASA of an adequate range of viable future planetary mission options such that its objective of solar system exploration can be pursued in an effective manner within the changing constraints of our Space Program. The nature of the work involved is quite varied, ranging from fast response items to pre-Phase A level mission studies. During the past contract year, a total of fourteen SAI staff members contributed to this effort.

Annual Report

The purpose of this Annual Report is to summarize the significant results generated under this Advanced Studies contract during the fifth and last year, 1 February 1980 through 30 April 1982, of Contract NASW-3035. Progress reports on the task efforts are given at scheduled semi-annual reviews. Task reports are prepared at the completion of each task and presentations of significant study results are given to a wide audience at NASA Headquarters, NASA centers, and at technical meetings. This report, therefore, is necessarily brief. Each of seven contract tasks are presented in the next section. A brief description is given of the analyses performed along with key results and conclusions. The intention is to direct previously uninformed, but interested readers to detailed documentation and to serve as a future reference to completed advanced studies. The final section of the report contains a bibliography of the reports and publications that have resulted from these task analyses.

SAI is presently continuing this program of advanced studies for the Earth and Planetary Exploration Division under a new two-year Contract NASW-3622.

Table 1

CONTRACT TASKS PERFORMED 1977 - 1982

Task	Performance Period
• Continuing Tasks	
1. Advanced Planning Activity	1977 - 82
2. Cost Estimation Research	1977 - 82
3. Planetary Missions Performance Handbooks	1977 - 82
• Directed Tasks	
4. Multiple Discipline Science Assessment	1977 - 78
5. Venus Surface Sample Return	1977 - 78
6. SEP/Sail Discrimination Assessment	1977 - 78
7. Asteroid Exploration Study	1977 - 79
8. Mars Strategy Study	1977 - 79
9. Galilean Satellite Lander Deployment Strategies	1978 - 79
10. Interplanetary Low-Thrust Transport Capabilities	1978 - 79
11. Asteroid Workshop	1978 - 79
12. Flyby Mission Feasibility for New Comets	1979 - 80
13. Uranus/Neptune/Pluto Project Concepts	1979 - 80
14. STS Planetary Applications Assessment	1979 - 80
15. Planetary Mission Opportunities Calendar	1979 - 80
16. ATM Solar Physics Platform Assessment	1979 - 80
17. Future Venus Exploration Mission Concepts	1980 - 81
18. Jupiter Satellite Mission Energy Space	1980 - 81
19. JPL-S02P Cost Estimation Activity	1980 - 81
20. Mission Opportunity Selection Analog	1980 - 81
21. OAST Technology Model Support	1980 - 81
22. Solar System Exploration Committee Support	1981 - 82
23. Mars Program Planning	1981 - 82
24. Galilean Satellite Mission Concepts	1981 - 82
25. Advanced Propulsion Data Base	1981 - 82

Five-Year Summary

In performance of our contractor support role to NASA Headquarters over the past five years, SAI has undertaken a total of twenty-five major tasks. These task areas are listed in Table 1. Note that three of these tasks represented the basic program with continuity from year to year, namely, Advanced Planning Activity, Cost Estimation Research, and Planetary Missions Performance Handbooks. The remaining twenty-two tasks were scoped to have more specific, directed objectives. These comprised varied work in areas of mission concept studies, trajectory and performance analyses, programmatic assessments, and participation/editing of workshops and meetings.

The Advanced Planning Activity task area alone generated a total of 181 subtasks requiring a level-of-effort per subtask varying from as little as 1 man-hour to as much as 3 man-months. These requests by NASA resulted in technical data and information responses reported by phone, telecopier, memoranda, presentations, or complete written reports as the case warranted.

All work performed under this contract was reported in accordance with the requirements of contract deliverables or in other formats. This included 5 annual reports, 26 task reports, 19 conference papers, 10 semi-annual reviews, 30 presentations at meetings, workshops, summer studies, etc., and 3 (editing) workshop proceedings.

As will be noted from Table 1, the work performed was quite broad in scope of application, covering all targets of solar system exploration: the Sun, inner planets, outer planets, comets, asteroids, and natural satellites.

It is difficult to say which areas of study have or will prove most important to NASA's planetary exploration program, or to summarize the most significant findings. The following is an attempt, from our perspective, to list several highlights and major contributions of SAI's technical support role during the past five years.

A. General Support Activities

1) Provided fast response on numerous occasions to NASA HQ. requests for technical data.

2) Prepared various presentation material for NASA Hq. staff staff use in meetings and congressional testimony.

3) Assisted in the formulation and writeup of NASA 5 and 10-Year Plans for planetary exploration, with strong support in areas of cost estimation and option planning.

4) Represented NASA Hq. and made technical presentations at numerous science working groups, steering committees, and COMPLEX meetings.

B. Specific Planning Activities

1) Assisted in the assessment/resolution of the Galileo Mission crises period (1979-80) by providing trajectory and payload margin data relevant to various mission fall-back options.

2) Contributed to the performance assessment of launch vehicle/ upper stage options and requirements for accomplishing planned mission objectives in the 1985-2000 time period.

3) Participated as member of the Solar System Exploration Committee (SSEC) and contributed data on mission options/ performance/cost tradeoffs in support of the Committee's advisory role in developing a long-range exploration strategy.

C. Innovative Suggestions

1) Initiated a preliminary analysis to verify that indirect SEEGA trajectories would significantly increase payload capability for Comet Encke rendezvous missions.

2) Identified the dual-SEEGA flight mode as a means of significantly increasing payload capability for outer planet orbiter missions (Saturn orbiter/two probe example).

3) Identified the analysis of the 1984 Mars Powered Swingby as a fall-back option for the original Galileo mission in 1982.

4) Suggested the use of broken-plane transfers to Jupiter in 1985 as a means of alleviating the high DLA problem, and provided trajectory data to the Galileo Project Office.

5) Conceptualized the design of a 3-Ring jig for on-orbit assembly/storage of IUS stages.

6) Suggested the tandem launch option for Saturn/Titan probe carriers with utilization of Jupiter swingby in the mid-1990's.

D. Mission Concept Studies

1) Investigated the use of balloons in Venus exploration concepts including balloon-assisted ascent to orbit for sample return missions.

2) Conducted a comprehensive study of asteroid mission objectives, implementation and operations; the study report has served as a reference work for subsequent investigations by other researchers.

3) Re-examined the feasibility of flyby missions to first-apparition comets (fall-back option to missing Halley opportunity), and provided detailed statistical data concerning mission opportunity frequency and trajectory requirements.

4) Investigated the launch opportunities, performance tradeoffs, and cost-effectiveness of an integrated, multi-mission project concept for reconnaissance of the far outer planets including atmospheric probes.

5) Examined various (post-Galileo) mission concepts for intensive investigation of the Galilean satellites, including single and multiple target scenarios using orbiter and lander deployments, and addressed the key issue of Jupiter radiation effects and shielding requirements.

E. Evaluations and Assessments

1) Participated in the 1977 NASA assessment of two advanced low-thrust systems, Ion Drive and Solar Sail, as applied to the proposed Comet Halley Rendezvous and other future missions.

2) Undertook a comprehensive, first analysis of the basic characteristics and maneuver strategies for launching planetary missions from a space station in earth orbit. The inherent pros and cons of station-launches vs Shuttle-launches were quantified for a broad spectrum of planetary mission opportunities. Results of this analysis were presented to the NASA Deputy Administrator, COMPLEX, SSEC, and the SSB.

2. TASK SUMMARIES

A total of seven study tasks was planned for the 15-month contract period, 1 February 1981 to 30 April 1982. These tasks are:

1. Advanced Planning Activity
2. Cost Estimation Research
3. Planetary Missions Performance Handbooks
4. Solar System Exploration Committee Support
5. Mars Program Planning
6. Galilean Satellite Mission Concepts
7. Advanced Propulsion Data Base

This section contains summaries of work performed on these tasks during the contract year. Task 1, Advanced Planning Activities, is a general support task designed to provide a budgeted level-of-effort for technical assistance on short-term planning problems which occur daily within the Earth and Planetary Exploration Division. The remaining six tasks are planned efforts with specific objectives of analysis.

A total of 13073 man-hours (80 man-months) was expended on the scheduled tasks during the contract period. A summary description and discussion of key results for the advanced studies effort on each of the seven tasks is presented in the subsections which follow. Specific final reports published or in preparation for each task are noted. These task reports or conference papers contain details on the appropriate study objectives, assumptions and methods of analysis, and also provide more extensive discussion of study results than the summaries which follow.

2.1 ADVANCED PLANNING ACTIVITY

The purpose of this task is to provide technical assistance to the Earth and Planetary Exploration Division on planning activities which arise during the contract period. This type of advanced planning support is a traditional segment of the broader studies work the staff at SAI has performed for Code EL during all past contract periods. Subtasks within this activity range from straightforward exchanges of technical data by phone, through multi-page responses by mail or telecopier, to more extensive memoranda and presentations, and occasionally to complete status reports on subjects of particular interest. The level-of-effort per subtask can vary from as little as 1 man-hour to as much as 3 man-months. A total of 35 reportable advanced planning subtasks, performed during the final 15 months of the present contract, are summarized here. Each of these was the subject of a written submission at the time of its completion.* Descriptive titles of these subtasks are tabulated in chronological order in Table 1-1. A brief summary of each of the subtasks is presented in the subsections which follow.

2.1.1 Revised Galileo/Halley Project Cost Estimate

This costing exercise was a revision to an earlier advanced planning task (Nov. 1980) in which it was assumed that the Galileo launch is slipped to the 1987 opportunity and that the Comet Halley intercept mission launched in 1985 is integrated into the project. Revised guidelines for the HIM development cost include: (1) Voyager 10-bay bus and Viking high gain antenna as no-cost hardware transfers; (2) Galileo subsystems and narrow angle imaging as block buy (recurring cost) transfers; (3) new hardware components related to dust shield, misc. structure and devices, target body tracker, stabilization, solar panels, and battery; and (4) new science related to wide angle imaging neutral mass spectrometer, ion mass spectrometer, dust counter, dust analyzer and magnetometer. Annual funding guidelines include: (1) GLL/85 development and flight project assumed real year $M converted to FY82 $M; and (2) GLL development delayed two years with launch date shift carrying $5M/year allowance and also ground equipment

* The final report for this task, "Advanced Planning Activities -- February 1981 - January 1982", Report No. SAI-1-120-768-M18, is a compilation of the full written submissions.

Table 1-1

SUMMARY OF 1981-82 ADVANCED PLANNING ACTIVITIES

Subtask	Month	Subject Title
1	Feb 1981	Revised Galileo/Halley Project Cost Estimate
2	Feb 1981	Galileo Probe Carrier Assessment
3	Feb 1981	Instrument Technologies Summary
4	Feb 1981	ISPM Launch Option with IUS(II)
5	Feb 1981	Lunar and Planetary Funding History
6	Mar 1981	Planetary Missions Capture Diagrams
7	Mar 1981	Galileo Mission Capability of SIV-C Stage
8	Mar 1981	Approved Planetary Missions Capture Diagram
9	Mar 1981	IUS(II)/SEPS Capability for 1987 Galileo Launch
10	Mar 1981	Consolidated Summary of Launch Vehicle Performance Margins
11	Apr 1981	Minimum Trip Time Saturn Missions
12	Apr 1981	Mission Candidates and Costing for NASA 10-Year Plan
13	Apr 1981	Performance Capability of Shuttle/Advanced Transtage
14	May 1981	Launch Opportunity Data (1980-90) for Jupiter Orbiter and ISPM
15	May 1981	STS Upper Stage Mission Capture Assessment
16	Jun 1981	Vu-Graphs for D. Herman Presentation to COMPLEX
17	Jun 1981	Low-Cost Outer Planet Missions, Presentation to COMPLEX
18	Jun 1981	Pioneer-Class Mercury Mission, Presentation to COMPLEX
19	Jun 1981	Comet Halley Sample Return
20	Jul 1981	Backup Statements for Upper Stage Assessment
21	Jul 1981	Encke/94 Flight Mode Performance Comparison
22	Aug 1981	Revised Upper Stage Assessment (Planetary Program)
23	Nov 1981	Ballistic Comet Rendezvous Performance Trades
24	Nov 1981	Ballistic Comet Rendezvous Performance Trades - Addendum
25	Dec 1981	Bibliography for SSEC Working Groups
26	Dec 1981	NASA 10-Year Plan Writeup
27	Dec 1981	Search for Galileo Low ΔV ΔVEGA Trajectories
28	Jan 1982	Material Transport ΔV Requirements from Moon or Earth
29	Jan 1982	Trajectory/Performance Data for Comet HMP/95 Rendezvous
30	Jan 1982	Supporting Mission Data for 10-Year Plan
31	Mar 1982	Space Platforms and Planetary Program - LPL Lecture
32	Apr 1982	Upper Stage Performance Updates
33	Apr 1982	Planetary Mission Capture Diagrams - COMPLEX Presentation
34	Apr 1982	Planetary Missions Launched From a Space Station - COMPLEX Presentatio
35	Apr 1982	Earth Orbit Capture of Meteoriods by Atmospheric Drag

transfer to HIM. Project cost elements are comprised of program management/MA&E, science and data analysis, spacecraft hardware and related support activities, launch and mission operations, and APA/Reserve. Results of the analysis shows that the HIM segment of the project has an estimated cost of $281M apportioned as $266M for development and $15M for flight operations. The total GLL/HIM project cost is $988M of which $276M is the incremental cost of slipping Galileo to 1987 and adding the Halley launch in 1985. In real year dollars the incremental cost is $355M.

2.1.2 Galileo Probe Carrier Assessment

The purpose of this subtask was to provide trajectory/performance data for launch year options 1984 and 1985, and to derive funding requirement spreads in real year dollars. Total injected mass required is 882 kg comprised of the probe, carrier bus, propellant, and adapter. Direct, Type II flights to Jupiter were determined to have 10-day window C_3 requirements of 90 (km/sec)^2 in 1984 and 86 (km/sec)^2 in 1985. Injected mass margins for the Shuttle/IUS(II) with STAR-48 kick stage are (-2) kg in 1984 and 68 kg in 1985. If a Minuteman kick stage were employed, the margins increase to 168 kg in 1984 and 243 kg in 1985. Funding requirements for the Galileo Probe Carrier mission totalled $157M and $166M, respectively, for the two launch year options.

2.1.3 Instrument Technologies Summary

For purposes of a NASA Hq. presentation to COMPLEX, a request was made to prepare cogent material which addressed: (1) the motivation for improved and new science instrument development; (2) an overview of development requirements as measured against the planetary 20-year mission plan and categorized by measurement types and environments; and (3) a one-page development summary for fields and particles, comet coma, planetary rings, atmosphere, surface, and interior measurement applications. Data sources used for preparation of this material included JPL, ARC, the NASA technology model, and other mission studies.

2.1.4 ISPM Launch Option with IUS(II)

Mass performance capability of the Shuttle/IUS(II) with kick stage was determined for separate launches of the ESA and NASA spacecraft components of the ISPM mission. In this option, the ESA payload would be launched in 1985 and the NASA payload in 1986. Both would employ a Jupiter gravity-assist, and the trajectories designed so that both payloads arrive at the Sun at approximately the same time with priority given to maximum latitude and time overlap. Launch energy requirements with a 10-day window are C_3 = 119 $(km/sec)^2$ in 1985 and C_3 = 122 $(km/sec)^2$ in 1986. The IUS(II)/STAR-48 capability provided injected mass margins of 95 kg for the ESA mission and (-30 kg) for the NASA mission. Use of the Minuteman stage instead of the STAR-48 allowed both missions to be captured with respective margins of 220 kg and 95 kg.

2.1.5 Lunar and Planetary Funding History

A retrospective picture of funding for the lunar and planetary exploration program over the time period 1960 - 1982 was prepared. The data format was a graph of annual expenditures (normalized to FY'82 $) versus fiscal year with "waterfall" curves identifying the program support base, early lunar missions, Mariner block, Pioneer block, Viking, Voyager, and the early portions of Galileo and ISPM. This historical picture clearly showed the two peak funding periods, 1963-67 and 1973-75, with sharp dropoff in flight project activity in the interim years before Viking buildup and again following the Voyager development.

2.1.6 Planetary Missions Capture Diagrams

A set of planetary missions with launch dates from 1984 to 1999 were defined in terms of flight mode options and injected mass requirements. These missions represented exploration initiatives for the inner planets, outer planets, comets and asteroids. The purpose of this subtask was to develop a visual display which indicated the performance capability of Shuttle-based upper stages, namely the Wide Body Centaur and IUS(II), against this mission model. Two capture diagrams were prepared in the usual format of injected mass versus injection C_3 with mission requirement regions shown for direct visual comparison against the launch vehicle capability curves. The first of these diagrams showed the capture status of planetary missions including Venus orbiter, Mars orbiter/penetrator, Saturn orbiter, and outer planet flyby/probes employing Jupiter gravity-assist. The second capture diagram encompassed small body rendezvous missions including near-Earth asteroids, main belt asteroids, and short-period comets.

2.1.7 Galileo Mission Capability of SIV-C Stage

Related to the issue of Centaur alternatives for Galileo payload injection, this subtask addresses the performance capability of the SIV-C (upper stage of Saturn launch vehicle) and possible impacts of the procurement schedule on Galileo launch date. Specifications on the SIV-C were obtained from NASA Headquarters and the injected mass versus C_3 curve was generated using SAI's computer program STAGE. In the C_3 region of interest for Galileo launches (87 km^2/sec^2 in 1986 and 82 km^2/sec^2 in 1987), the SIV-C provided about 200 kg more injected mass than did the Wide Body Centaur. For the nominal Galileo mass requirement, the SIV-C allowed injected mass margins of 340 kg in 1986 and 730 kg in 1987. Regarding the question of programmatics, an accelerated competitive procurement of the SIV-C stage indicated that a 1986 launch would be possible, whereas a longer procurement cycle would necessitate a launch in 1987.

2.1.8 Approved Planetary Missions Capture Diagram

This subtask involved preparation of visual material for use in congressional testimony by NASA Headquarters staff. The point to be illustrated was how well various launch vehicle/upper stage options would match capability against the injected mass requirements of near-term, approved planetary missions. The mission set comprised: (1) various Galileo options such as combined orbiter/probe, separate orbiter and probe carrier, launch opportunities in the years 1984 through 1987, and different flight modes such as Mars powered swingby, direct and ΔVEGA; (2) various ISPM options including combined NASA/ESA and separate NASA payloads implemented by direct and ΔVEGA in the years 1985 through 1987; and (3) VOIR launched in 1988 employing all-chemical and aerobrake-assisted propulsion options. The first mission capture diagram prepared showed these mission requirements as measured against the capabilities of the IUS(II), IUS(II)/STAR 48, IUS(Twin), and Wide-Body Centaur upper stages. The second diagram added the capability curves of the Transtage family of upper stages including proposed advanced configurations and also the effect of Shuttle cargo mass improvement.

2.1.9 IUS(II)/SEPS Capability for 1987 Galileo Launch

As part of the fallback option planning for the Galileo mission, one hypothesis was that the mission launch would be delayed to 1987 but that neither the IUS(Twin) or Wide-Body Centaur would be available. SAI was asked to ascertain whether or not the mission could be performed using a SEP low-thrust stage injected by the IUS(II). The approach taken to answer this question was to contact personnel at MSFC and JPL who were working this problem independently, query them regarding their data source and assessment of capability, and then to conduct a brief trajectory/performance analysis to verify the results. An indirect, low-thrust trajectory of the 1^+ year SEEGA type is required with IUS(II) injection at $C_3 \approx 1$ km^2/sec^2. Launch would occur on 24 May 1987, Earth return/swingby on 14 August 1988, and Jupiter arrival on 25 December 1990 at $V_\infty = 5.687$ km/sec. Total flight time is 1311 days or about 400 days longer than direct ballistic flight to Jupiter. Mass delivery performance appeared to be adequate but somewhat marginal. Assuming the nominal Galileo system mass of 2135 kg, nominal SEPS dry mass of 1450 kg, and a 100 kg launch adapter, the injected mass margin was determined to lie in the range 35 - 130 kg depending on the degree of conservatism assigned to IUS(II) performance capability. Mass growth of Galileo or SEPS, or degraded IUS performance, could easily wipe out this margin. The assessment of MSFC was that the SEPS option could be made to work out with a safe mass margin realized by either: (1) performance improvements in SEPS solar cell and power processor efficiencies; or (2) improved grain design of IUS solid propellant resulting in injected mass gains of 300 to 400 kg.

2.1.10 Consolidated Summary of Launch Vehicle Performance Margins

The objective of this subtask was to construct a data matrix showing the injected mass margins (±) for Galileo, ISPM, VOIR and Halley Intercept mission options as measured against various launch vehicle/upper stage options. The Shuttle-launched stages included Centaur(WB), IUS(II), IUS(Twin) and IUS(III). The Titan-launched stages were the Centaur and Transtage. Data entries for this consolidated summary were obtained directly from the data base developed for previous analyses of this kind.

2.1.11 Minimum Trip Time Saturn Missions

The feasibility and flight time requirements were determined for several ballistic mission concepts utilizing Shuttle/Centaur(WB)/STAR 48 launches. Trajectory data was obtained for the 1990 and 1994 Saturn-direct opportunities and the 1998 Jupiter swingby opportunity. This data effectively represented the range of performance between the best and worst launch opportunities in the 1990 decade. The five mission concepts comprising Saturn flyby and orbiter modes with probe delivery were: (1) Saturn probe carrier with no science on carrier; (2) Titan probe carrier with line scan imaging on carrier; (3) Saturn and Titan probe carrier with imaging on carrier; (4) Saturn probe carrier/orbiter with imaging and ring particle experiment on orbiter; and (5) Titan probe carrier/orbiter as above. A spin-stabilized spacecraft configuration (Hughes design for Galileo probe mission) was employed for both the flyby probe carrier and probe carrier/orbiter concepts. Mass definitions for the various system elements were presented in the format of a hardware utilization matrix. One purpose of the hardware matrix was to delineate system elements of each concept or mission option which could be divisible between the Japanese and the U.S. on a joint project. The second output format of this subtask was a tabulation of injection energy C_3 and minimum flight time (nominal mass delivery) for each mission option and launch opportunity. Results may be summarized as follows: (1) the single probe flyby mission, Saturn or Titan probe, has minimum trip times of 2.75 years in 1990, 2.85 years in 1994, and 2.10 years in 1998; (2) the combined Saturn and Titan probe mission requires minimum trip times of 3.75 years in 1990, 2.30 years in 1998, but cannot be accomplished in 1994; (3) the Saturn (or Titan) probe/orbiter mission cannot be accomplished by direct flights in either 1990 or 1994, but is captured in 1998 via Jupiter swingby with a minimum trip of 3.4 years.

2.1.12 Mission Candidate and Costing for NASA 10-Year Plan

In preparation for the solar system exploration plan development, a document of working papers was compiled for use by NASA Headquarters and SAI staff. This document included: (1) relevant planning material from the published Plan of the previous year; (2) various SAI advanced planning tasks reported in 1980-81, (3) a multi-page listing of launch/encounter date options for all targets of exploration for ballistic and SEP flight modes; and (4) working forms for subsequent fill-in of plan description and data collection profiles. After consultation with NASA Headquarters staff, the next step in this analysis was to structure and obtain cost estimates for six alternative plans: (A) Reference Program - Mariner Class Missions; (B1) Pioneer Class Missions Supplement; (B2) Viking Class Missions Supplement; (C1) Augmented Reference Program - Mariner + Pioneer Class Missions; (C2) Augmented Base Program with Viking Class Missions; and (D) Resource Exploitation Objectives. Waterfall charts of annual expenditures were prepared for each of these plans and compiled with other prepared charts which provided plan descriptive summaries and science data collection profiles. This material was delivered to NASA Headquarters for review and later presentation to the Solar System Exploration Committee.

2.1.13 Performance Capability of Shuttle/Advanced Transtage

In response to a briefing by Martin Marietta regarding the potential application of the Transtage family of upper stages, SAI was requested by NASA Headquarters to evaluate this potential for planetary missions. The Transtage family was comprised of: (1) a single-stage version designed for Earth-orbital missions; (2) a two-stage advanced configuration with a modified 3920 Delta as the second stage; and (3) a Star 48 kick stage added to each of the above for high energy missions. Performance specifications (e.g. propellant loading, specific impulse, inert weight, etc.) were obtained from Martin Marietta. Injected mass capability was then calculated over the C_3 range 0 to 141 km^2/sec^2. Performance of the Advanced Transtage launched by an 83K Shuttle was also calculated. These results were then plotted as a planetary missions capture diagram with direct comparison to IUS and Centaur capability for Galileo, ISPM and VOIR missions. The Transtage performance exceeds that of the IUS(II), the Advanced Transtage is somewhat better than the IUS(III), and the Shuttle (83K)/ Advanced Transtage still does not equal the Wide-Body Centaur performance. Several planetary mission options remain uncaptured by the Transtage family of upper stages.

2.1.14 Launch Opportunity Data (1980-90) for Jupiter Orbiter and ISPM

The objective of this subtask was to prepare two tables listing trajectory characteristics of Jupiter orbiter and ISPM missions for each annual launch opportunity in the 1980 decade. Entries included trajectory type, launch date, injection energy C_3, launch declination, Jupiter approach speed, and flight time.

2.1.15 STS Upper Stage Mission Capture Assessment

SAI was asked to participate in a major NASA undertaking, the purpose of which was to determine the ability of various candidate upper stage designs to capture solar system exploration missions. Upper stage options included the Titan 34D/Centaur, and Shuttle-based stages such as the IUS family, Advanced Transtage, and Wide-Body Centaur. Both ballistic and SEP flight modes were considered. This mission set comprised all targets of exploration (planets, comets, asteroids and sun) and was organized in terms of injected mass vs C_3 regions, e.g. inner planets, small bodies, outer planets, and solar mission regions. A total of 52 ballistic and 10 SEP mission examples were defined over a launch opportunity time span from 1984 to 1999. Trajectory characteristics and nominal mass requirements of these examples were compiled from existing data sources (JPL, SAI, NASA HQ), or were generated anew when data was not available. Upper stage performance specification updates were likewise collected from cognizant NASA offices or centers. Injected mass vs C_3 capability curves for these stages were then generated using an SAI computer code. Output results of this analysis were shown in visual display by a set of mission capture diagrams for each upper stage candidate. Vu-graphs and handout brochures describing study results were prepared and a presentation was made to OSS and OSTS at NASA Headquarters.

2.1.16 Vu-Graphs for D. Herman Presentation to COMPLEX

One theme of this presentation was a comparison of ballistic and SEP performance illustrating the enabling capability of the SEP flight mode in solar system exploration. Four vu-graphs were prepared for this purpose. The Shuttle/Wide Body Centaur was the assumed launch vehicle. A tabulated performance comparison for small body rendezvous missions listed the launch year, flight time, retro mass in the case of ballistic flight, and launch margin for an assumed 800 kg mission module. Although Comet Tempel 2 and Comet T-G-K rendezvous could be accomplished ballistically, SEP offered significant advantages in reduced flight time and increased launch margin. SEP was enabling for most main belt asteroid rendezvous (single targets) and, particularly, for multi-target rendezvous. Graphical formats were used for Saturn and Uranus orbiter missions showing net mass capability in orbit versus flight time. Ballistic modes including direct Jupiter swingby, ΔVEGA and ΔVEGA-Jupiter swingby were compared against the SEP-SEEGA flight mode. For Saturn, only the 1998 JSB ballistic opportunity out-performed SEEGA for payload requirements up to 1200 kg and trip time less than 5.5 years. SEEGA was clearly superior to all ballistic modes for Uranus orbiter missions, and was enabling for trip times under 10 years.

2.1.17 Low-Cost Outer Planet Missions, Presentation to COMPLEX

SAI prepared and delivered a presentation to COMPLEX in which lower cost options and strategies for continuation of outer planet exploration were assessed. These options comprised flyby/probe missions to Saturn, Titan, Uranus and Neptune over a 20-year program time frame beginning in 1985. Specific issues addressed in this analysis were: (1) design commonality for both probes and carrier spacecraft; (2) cost effective combinations of propulsion and flight mode; (3) mission duration related to science data returned; (4) cost benefits of a multi-mission project; and (5) cost benefits of international cooperation. A wide range of design options and programmatic strategies were examined with parametric data developed for different launch vehicles, flight modes, carrier science levels, and launch year opportunities. Cost estimate details were described and compared for these cases. First flight project costs are in the range \$175 - \$265M plus operations, and hardware buys reduce this range to \$85 - \$125M for follow-on missions; FY'82 dollars assumed. An example scenario was configured which comprised four missions launched in the period 1991-1994 with all data returned by the year 2000. Assuming a single project with block-buy hardware and shared mission operations, the total project cost exclusive of launch vehicle cost was estimated to be \$603M in FY'82 dollars.

Other significant assessment conclusions were: (1) design commonality with some limitation appears to be feasible, e.g. 20-bar probes; (2) the upper stage of choice is the Wide-Body Centaur with a STAR 48 kick stage; (3) Uranus and Neptune flyby/probes require special Jupiter swingby opportunities; and (4) an international partner for the carrier development can reduce multi-mission project cost by one-third.

2.1.18 Pioneer-Class Mercury Mission, Presentation to COMPLEX

As part of an integrated presentation on low-cost exploration of the inner planets, several vu-graphs were prepared on the subject of a Pioneer-class mission to Mercury. The assumed discipline objective was investigation of the field/particle interaction of the interplanetary medium with Mercury. Two mission options were examined (multi-flyby and elliptical orbiter) in terms of design requirements, and a cost estimate was made for each. The spacecraft bus costing assumed exact repeat inheritance of Pioneer Venus orbiter components. In FY'82 constant dollars, and assuming a 20% APA/Reserve, the total project cost estimates were \$90M for the multi-flyby option and \$137M for the orbiter option.

2.1.19 Comet Halley Sample Return

A preliminary analysis was conducted of a moderate-cost mission concept for returning coma samples collected during a 1986 intercept with Halley's Comet. This mission concept focuses on the sample return objective with limited supporting science. A Pioneer-type spinning spacecraft is assumed, and the launch mass requirement is to be compatible with Shuttle/IUS(II) capability or less. Trajectory data was generated for post-perihelion encounters (March 1986) with round-trip times of 1, 3, 4 and 5 years. Mission design tradeoffs could thereby be analyzed in terms of the key characteristics such as injection energy, minimum perihelion distance, encounter date and geometry, and encounter speed. The sample collection experiment is based on multi-cell vapor deposition of 100 μm - 1mm coma particles intercepted at very high speed at closest approach distances under 1000 km from the comet nucleus. In conceptual design, each cell is of hexagonal shape, 5 cm wide by 20 cm long, and made of either teflon or platinum material. A nominal 3 square meter collector area is configured from a hexagonal array of 200 cells per chamber pair mounted in front of a 4 square meter dust shield which protects the main spacecraft elements. Sample return is to a 150 nm circular orbit about Earth; both aerobraking and all-propulsion orbit capture options were examined. Flight profiles, mass statements, and cost estimates were obtained for three distinct mission options representing variations in collector area, Earth capture mode, and sample return time. Preliminary results indicate the potential feasibility of a Pioneer-class, Halley coma sample return for a cost ceiling of $150M (FY'82 $, excluding recovery and sample analysis costs). Critical issues bearing on such feasibility and needing further study are sizing and design of the sample collector and encounter navigation accuracy. Results of this mini-study were presented to the Space Science Advisory Committee of the NASA Advisory Council.

2.1.20 Backup Statements for Upper Stage Assessment

This subtask concerned the drafting of two groundrule statements in connection with the NASA Assessment of Upper Stages (see subtask 2.1.15). The first of these statements, titled ΔVEGA Exclusion, briefly explained the pros and cons of the ΔVEGA flight mode, the current judgement that it is not the option of choice but rather reserved as a performance "safety net", and should not therefore be used as a baseline mode for purposes of establishing reference propulsion requirements. The second groundrule statement, titled SEPS Inclusion, gave the historical and current rationale for the performance advantage and mission enabling capability of solar electric propulsion, its integral role in any comprehensive solar system exploration program, and therefore the judgement that SEPS availability should be a key assumption in the upper stage assessment.

2.1.21 Encke/94 Flight Mode Performance Comparison

A bar chart was prepared showing net rendezvous mass capability versus launch year for missions to Comet Encke arriving 50 days before its 1994 perihelion passage. Launch vehicle options included the Shuttle/IUS(Twin) or IUS(II) and the Shuttle/Centaur(WB). Flight mode variations included ΔVEGA and direct ballistic, and SEPS flat array and concentrated array. Both Earth-storable and space-storable retro propulsion options were considered for the ballistic flight modes. Altogether, twenty mission design options were represented by this bar chart summary. The superior performance of SEPS was clearly indicated, although sufficient mass capability was also available with the ΔVEGA mode provided the Centaur(WB) injection stage could be used.

2.1.22 Revised Upper Stage Assessment (Planetary Program)

This subtask was performed for Code SL in support of their final report contribution (Planetary Program Requirements Section) to the NASA High Energy Upper Stage Study. SAI wrote the subsection entitled "Assessment of Reference Program Impacts" which provided a quantitative measure of the degree of performance impact that upper stage selection might have on the long-range exploration program. For this purpose a set of ten mission objectives (including Galileo, ISPM and VOIR) representing anticipated accomplishments from 1985 to 2000 was defined as a reference mission set. Beyond the currently approved missions, the set included two small body rendezvous missions, three Mars missions from orbiter to sample return capability, a Saturn orbiter with planet and Titan entry probes, and Uranus/Neptune entry probe missions. The mission requirements and performance analyses conducted previously by SAI as part of this NASA-wide study were extended in scope and detail. Impact evaluations were classified into six specific areas: available launch opportunities, number of launches, payload margin, trip time and/or mission operations, hardware developments needed, and mission objectives accomplished. A baseline or reference upper stage option, namely the Wide-body Centaur with SEPS augmentation, was selected as the "preferred" capability against which seven alternative stage options were compared in the above impact areas. Results of this comparison were presented in a matrix format (mission vs. impact) for each alternative option with the use of multi-color areas of impact measure: red area - severe impact or major performance degradation, yellow area - possible impact or minor performance degradation; green area - no impact or equivalent performance. An overall summary, integrated over all impact areas, was given separately for near-term and future missions for the four upper stage options with and without SEPS augmentation. In conclusion, only the Wide-body Centaur was acceptable to the near-term plan consisting of Galileo, ISPM and VOIR missions. Only the Wide-body Centaur and Interim OTV provide the necessary performance capability in the future program objectives to assure adequate flexibility and resilience for sensible planning. For these two upper stage options, the absence of SEPS augmentation incurs a performance penalty in the range 15% to 25%.

2.1.23 Ballistic Comet Rendezvous Performance Trades

This analysis and presentation material was prepared for Bob Farquhar of NASA-GSFC. Trajectory and payload performance data for ballistic rendezvous with Comets HMP/95 and Encke/97 were calculated for direct and ΔVEGA flight modes over a range of comet arrival conditions. Launch capability of the Shuttle/Wide-Body Centaur was assumed. The data format for each comet mission comprised: (1) a bar chart showing payload at specific arrival dates for different flight modes and launch opportunities; and (2) a table listing trajectory and payload parameters over the arrival date extent. Performance measures of both Earth-storable and space-storable retro propulsion were examined. Direct ballistic missions to HMP seemed to be quite adequate, but ΔVEGA flights to Encke appeared to be necessary to achieve desired payload margins.

2.1.24 Ballistic Comet Rendezvous Performance Trades - Addendum

This continuation of the previous subtask involved the generation and presentation of similar technical data for Comets Tempel 2/94 and Giacobini-Zinner/98 rendezvous missions. Also delivered was a table of trajectory characteristics for a 1990 launched flyby-sample return mission to Comet Encke.

2.1.25 Bibliography for SSEC Working Groups

An extensive bibliography of recent mission studies for Mars, the outer planets, and comets/asteroids was prepared for the information of members of the Solar System Exploration Committee. Category listings included: a) trajectory and orbital analysis; b) mission concepts and design tradeoffs; and c) workshop and committee reports. A total of 83 available reports were listed in the SAI-prepared bibliography; additional bibliography data was prepared by JPL.

2.1.26 NASA 10-Year Plan Workshop

As part of the Office Space Science contribution to the NASA Program Plan: Fiscal Years 1982-1991, we were asked to write Section IV G, <u>Exploration of the Solar System</u>. This section was organized in three parts: (1) Planning Strategy; (2) Status of Solar System Exploration; and (3) Program Content. The first part defined the logical sequence of steps in the overall strategy, namely, reconnaissance, exploration, and intensive study. The implications of current funding limitations on the ability and manner of carrying out this strategy were clarified. The second part of Section G focused on the solar system targets by region - inner planets, outer planets, small bodies - and briefly described the principal science objectives and the extent to which these objectives have been or will soon be accomplished. The detailed content of the program was presented in the third part of Section G with descriptions of the current program through VOIR, new initiatives for FY'1983-87, and future initiatives for FY'1988-92.

It should be noted here that SAI also participated (under a separate contract task) in the preparation of the total Space Science Program Plan by providing services to Code EL related to the coordination, assembly and preparation of the Draft Plan. These services included the collection, editing and compilation of all text material and supporting charts and illustrations, and word processing typing and reproduction of the Draft Plan.

2.1.27 Search for Galileo Low ΔV ΔVEGA Trajectories

With the apparent necessity of employing the Shuttle/IUS(II)/Star 48 for launching Galileo in 1985, the baseline flight mode was shifted to a ΔVEGA trajectory. A mission ΔV budget problem was created since the spacecraft propulsion system could not be changed without major disruption to the project. One way of alleviating the problem was to fly an off-optimal ΔVEGA trajectory trading off higher injection energy C_3 for lower post-injection ΔV; this was allowable since injected mass margin was available at the nominal C_3 value. In support of the Galileo Project Office we conducted a trajectory search for these lower ΔV cases. Starting at the 15 Aug 1985 launch for the nominal 2^- ΔVEGA mission, the launch date was advanced by one day increments through 1 Sep 1985. Trajectory results showed that C_3 increased from 27.1 to 35.5 $(km/sec)^2$ over this range while the midcourse ΔV decreased from 567 to 479 m/sec.

2.1.28 Material Transport ΔV Requirements from Moon or Earth

This subtask was undertaken in response to a request by Jim Arnold of the University of California - San Diego. The ΔV impulse requirement for transfers from the lunar surface and from the Earth surface were calculated for four different destinations: (1) low Earth orbit; (2) high, stationary Earth orbit; (3) the L_5 Earth-Moon Lagrangian point; and (4) the L_2 Earth-Moon Lagrangian point. Lunar surface launch provided a ΔV savings of 30.4%, 70.5%, 79.3% and 78.1%, respectively, for these destinations. These data were used by Dr. Arnold in a presentation to the Fletcher Committee.

2.1.29 Trajectory/Performance Data for Comet HMP/95 Rendezvous

This subtask was undertaken at the request of Bob Farquhar of NASA-GSFC in support of a conference paper being written on the subject of comet rendezvous missions. The work performed involved the compilation and generation of optimal trajectory data, analysis of this data in terms of payload mass calculations, and preparation of charts and illustrations for the paper. The target comet was Honda-Mrkos-Pajdusakova at its 1995 perihelion passage and the rendezvous flight mode was ballistic-direct and ΔVEGA options. New trajectory generation was necessitated in part by a 4-day change in the estimated date of the comet's 1995 perihelion. The other significant factor in this analysis was the systematic search for asteroid flyby targets enroute to H-M-P along the 1990-launched direct ballistic flight path. Thirteen target candidates were found requiring an excess ΔV (over the nominal mission) ranging from only 2 m/sec to 326 m/sec. Furthermore, five cases of 2-asteroid flybys were found with excess ΔV ranging from 238 m/sec to 579 m/sec. Technical data delivered to Dr. Farquhar consisted of summary tables of trajectory/payload characteristics for H-M-P rendezvous with and without asteroid flyby augmentation, example trajectory profile plots, performance comparison bar charts, and computer printouts of all trajectories generated by SAI's MULIMP program.

2.1.30 Supporting Mission Data for 10-Year Plan

For each of three Plan options (low, intermediate and high level funding), a pair of descriptive data charts was prepared. The first of these charts showed the milestone time line for each mission in the Plan, i.e. new start, launch, encounter and data collection. The second chart listed the launch energy C_3, injected mass requirement, and relevant comments for each mission.

2.1.31 Space Platforms and the Planetary Program - LPL Lecture

A seminar was prepared in response to an invitation by the Lunar and Planetary Laboratory of the University of Arizona. The topic of this seminar was the current status and future plans of the U.S. planetary exploration program. The lecture drew upon existing material of historical perspective, current thinking of NASA Headquarters and various advisory groups, and recent studies performed by SAI. Part 1 of this presentation focused on overall goals of exploration, space mission accomplishments to date, the historical funding profile, and factors contributing to the contraction of the planetary program. Part 2 described the new directions being followed in planning for lower cost mission options, the work of the Solar System Exploration Committee, and the candidate missions, timelines and cost profiles of the evolving long-range plan. Part 3 of this lecture focused on recent space station concepts and the implications for future planetary exploration in terms of performance benefits and penalties.

2.1.32 Upper Stage Performance Updates

The latest performance specifications for the PAM-A, IUS family, Centaur-F (long wide body) and Centaur-G (short wide body) were obtained from cognizant personnel at NASA Centers. Updated performance curves of injected mass versus launch energy C_3 were then generated using SAI's STAGE program. Least-squares function fits of these data were calculated and appropriate changes were made to our launch vehicle data files on the PDP 11-34 computer. Similar work was performed for several new upper stage concepts, namely, the spinning solids (IUS SRM configurations) as proposed by Hughes Aircraft Company. For want of an official name, these stage configurations were designated on our files as IUS(I-S), IUS(II-S), and IUS(III-S).

2.1.33 Planetary Missions Capture Diagrams - COMPLEX Presentation

In support of a Code EL presentation to COMPLEX, a set of vu-graphs was presented showing mission capture performance for both near-term and future planetary missions. Near-term missions included Galileo, ISPM and VOIR. The point made of this capture diagram was to illustrate the transition from Centaur to the IUS(II)/IM upper stage. This transition to a lower performing stage required the following changes in mission design: ΔVEGA flight mode for Galileo orbiter/probe, aerobraking orbit capture for VOIR, and a single ESA-only spacecraft for ISPM. The reference plan for future missions to be launched in the time period 1988-2000 comprised four categories: (1) Pioneer inner planets -- Mars, Venus and the moon; (2) Outer planet probes -- Saturn, Titan, Uranus and Neptune; (3) Mariner Mark II -- asteroid and comet rendezvous, new comet intercept, and Saturn orbiter; and (4) Viking-class missions -- asteroid multi-rendezvous, Mars sample return, Saturn orbiter with two probes, and comet sample return. The injected mass versus launch C_3 requirements for these reference missions were plotted and the category regions were indicated. Upper stage capability curves were provided as vu-graph overlays, one set for the Centaur family and another for the IUS family and PAM-A.

2.1.34 Planetary Missions Launched from a Space Station - COMPLEX Presentation

Interim results of an ongoing study (see Task 7) regarding the performance assessment of planetary missions as launched from an orbiting space station were presented to COMPLEX. Basic characteristics and maneuver strategies for station launches were described. The inherent pros and cons were quantified in terms of injected mass capability of selected upper stages, plane change penalties, and launch timing penalties. Performance measures of injected mass margin for nominal payloads and maximum payload capability were compared for space station launches and standard Shuttle launches for three example missions: Mars sample return, Saturn orbiter/probe, and Anteros rendezvous.

2.1.35 Earth Orbit Capture of Meteoroids by Atmospheric Drag

This subtask was undertaken in response to a request by Jim Arnold of the University of California - San Diego. The question posed was: what is the likelihood (probability per year) that Earth-approaching meteoroids in the size range 10-100 meter diameter will be captured into Earth orbit by atmospheric drag forces? This problem can be separated into two basic components: (1) given that a meteoroid is on an Earth-collision course, determine the atmospheric entry/exit bounds that will result in elliptical orbits with apogee distance above the atmosphere, and determine the capture fraction dependence on velocity and size; and (2) integrate this result with the probability frequency distribution of such objects on collision courses to determine the overall capture probability. The solution to this problem was determined by the use of analytical formulas after verifying their accuracy by numerical integration experiments. Results obtained showed that the conditional orbit capture fraction varied from 4×10^{-3} at $V_\infty = 1$ km/sec down to 3×10^{-4} at $V_\infty = 25$ km/sec. Integration over the size-frequency distribution, allowing for modeling uncertainty, yielded the result that such objects may be expected to be captured (temporarily) with a frequency of once every 2300 years to once every 70,000 years depending on the estimated influx of such bodies.

2.2 COST ESTIMATION RESEARCH

Cost estimation analysis has been an on-going Advanced Studies support task for eight years. Its objective is to develop and implement a methodology for predicting costs of future lunar and planetary flight programs. Its purpose is to provide reasonably accurate cost estimates based on pre-Phase A study definitions to key advanced planning activities within NASA for the U.S. Planetary exploration program. The work on this task has encompassed historical cost data collection and analysis, development and refinement of a cost estimation model based on the historical data and extensive use of the model for predicting costs of future missions. During the past two years, emphasis has been placed on updating and extensively revising the SAI Planetary Program Cost Model so that it now incorporates cost data from the most recent U.S. planetary flight projects and its algorithms more accurately capture the information in the historical cost database. This year's effort has also included the completion of the task report "Cost Estimation Model for Advanced Planetary Programs - Fourth Edition" (Report No. SAI 1-120-768-C9) which documents the database and methodology used in the revised model.

Cost Model Overview

The SAI Planetary Program Cost Model can be characterized by the following features.

- The Model is based on all relevant U.S. planetary projects from Mariner Mars 1964 through Pioneer Venus.
- Inputs to the Model are limited to information generally available at the level of pre-Phase A mission definition. Generally, these consist of estimates of spacecraft subsystem masses, design heritage, flight time and encounter duration.
- The primary output is manpower, expressed in direct labor hours. Total cost is obtained by use of appropriate conversion factors which include inflation indices.

- The Model views a mission program as consisting of two distinct phases: The Development Project, which encompasses all activity through the mission's launch + 30 days milestone and the Flight Project, which includes all activity from L + 30 days through the nominal end of mission.

- At its most detailed level, the Model deals with cost categories which are derived as compromise aggregations of the variety of work breakdown structure definitions found in the cost database.

- The Development Project is further separated into hardware-related cost categories and functional support cost categories. The hardware categories are directly related to the mission spacecraft engineering and science subsystems.

- Hardware categories are further separated into non-recurring costs (design and development) and recurring costs (fabrication and subsystem-level tests). Inheritance is assumed to affect only the non-recurring cost.

- The Model is capable of dealing with a wide variety of spacecraft designs, including inertial or spin stabilized spacecraft, atmospheric entry probes and highly automated soft landers.

Summary of Model Updates, Revisions and Performance

Revision of the cost model was made with a two-fold objective: to increase the flexibility of the Model in its ability to deal with the broad scope of scenarios under consideration for future missions, and to at least maintain and possibly improve upon the confidence in the Model's capabilities with an expected accuracy of $\pm 20\%$.

The model development effort resulted in an udpated and revised Cost Model which adequately meets the objectives stated above. Only the Development Project portion of the model was revised; cost estimates for the Flight Project are generated using algorithms from the previous version of the model.

Historical cost data for thirteen unmanned lunar and planetary flight programs currently comprise the SAI cost model database. Table 2-1

Table 2-1

COST MODEL DATABASE STATUS

PROGRAM	DEVELOPMENT PROJECT (TO L+30 DAYS)	FLIGHT PROJECT (POST L+30 DAYS)	USE IN MODEL REVISION (DEVELOPMENT PROJECT)
MARINER '64	C	C	PARTIAL
SURVEYOR	C	C	PARTIAL
LUNAR ORBITER	C	C	PARTIAL
MARINER '69	C	C	TOTAL
MARINER '71	C	C	TOTAL
PIONEER JUPITER/SATURN	C	C	TOTAL
MARINER '73	C	C	NOT USED
VIKING LANDER	C	I	TOTAL
VIKING ORBITER	C	I	TOTAL
VOYAGER	C	I	TOTAL
PIONEER VENUS	C	I	TOTAL
GALILEO ORBITER	I	X	FUTURE
GALILEO PROBE	X	X	FUTURE
INTERNATIONAL SOLAR POLAR	X	X	FUTURE

C: COMPLETE
I: IN PROGRESS
X: NO DATA YET

summarizes the present status of the database. Cost data for the programs in Table 2-1 up to and including Voyager were used in developing the previous version of the SAI cost model. At the time, however, the Viking Lander, Viking Orbiter and Voyager (then called Mariner Jupiter/Saturn) development projects had not been completed and the cost data used in modeling were based on estimates to complete. Thus, prior to the present model revision effort, it was necessary to analyze and reduce the actual completion costs which had been collected for these three programs into forms useful for modeling.

During the process of examining cost allocations, a decision was made to broaden and redefine the model cost categories. A total of 21 revised cost categories were defined (see Table 2-2), 16 related to flight hardware and five to functional support. Two separate algorithms were derived for each hardware category: one which estimates total direct labor and another which estimates recurring labor. Non-recurring labor can be obtained by differencing the two estimates. The hardware labor algorithms are, in general, power laws or exponential functions of a single independent variable formed by the product of the number of flight units and the subsystem (category) mass.

Statistical analysis of the historical cost data resulted in a conclusion that factors derived as simple ratios can be used to convert category labor hour estimates to total cost.

An extensive error analysis of the Model measured against the programs in the database indicated that the information in the database had been captured with an average error of less than 10%. However, a simulation of the Model's performance, with number of flight units as the parameter, showed that predictions made with the Model would be highly sensitive to the number of flight units. A straight-forward adjustment procedure was devised that effectively eliminates this sensitivity but results in an increased average error of just less than 20% as measured against the database.

The last step in analyzing model performance was to run benchmark tests of the model against project costs from other sources. Three cases were

Table 2-2

REVISED MODEL COST CATEGORIES

DEVELOPMENT PROJECT

FLIGHT HARDWARE CATEGORIES

STRUCTURE & DEVICES
THERMAL, CABLING & PYRO
PROPULSION
ATTITUDE & ARTICULATION CONTROL
TELECOMMUNICATIONS
ANTENNA
COMMAND & DATA HANDLING
RTG POWER
SOLAR/BATTERY POWER
AERODECELERATION
LANDING RADAR/ALTIMETER
LINE SCAN IMAGING
VIDICON IMAGING
PARTICLE & FIELD INSTRUMENTS
REMOTE SENSING INSTRUMENTS
DIRECT SENSING/SAMPLING INSTRUMENTS
NON-IMAGING SCIENCE INSTRUMENTS

SUPPORT CATEGORIES

SYSTEM SUPPORT & GROUND EQUIPMENT
LAUNCH +30 DAYS OPERATIONS & GROUND SOFTWARE
IMAGE DATA PROCESSING
SCIENCE DATA ANALYSIS
PROGRAM MANAGEMENT & MISSION ANALYSIS & ENGINEERING

FLIGHT PROJECT (FUTURE WORK)

MISSION OPERATIONS
DATA PROCESSING & ANALYSIS

examined: (1) Mariner Venus/Mercury 1973, (2) the Galileo Probe and (3) the Venus Orbital Imaging Radar spacecraft. The Mariner '73 project was particularly appropriate to the benchmark tests since no data from it was used during the model development, yet it was accomplished during the same time period as those projects in the model database. The other two cases, representing current and future programs, were also well suited for these tests in that their benchmark costs could be obtained from independent project estimates. Results of the tests are summarized in Table 2-3, which indicates the estimated errors both with and without use of the flight unit adjustment procedure. From those results and the previous analyses, it appears that the revised model performs well for applications involving either two flight units or one flight unit with the adjustment procedure. Firm conclusions cannot be made for applications involving three or more flight units since appropriate benchmark projects are not available to test such cases.

Sample Application

A sample application of the cost model is presented to illustrate its applicability to a variety of project implementation concepts and mission scenarios. This particular example deals with a multi-mission project to deliver atmospheric entry probes to the outer planets; Saturn, Uranus and Neptune.

The following three pages present completed input data worksheets for this mission. The first worksheet defines the project scenario and provides necessary guidelines for generating the cost estimate. The next two worksheets present subsystem mass estimates for the probe and probe carrier flight hardware together with estimates of the inheritance classifications for each subsystem. As indicated, the project implementation scenario is based on an assumption that all six spacecraft (three probes and three probe carriers) are developed under a single hardware system contract. The probe design is assumed to rely heavily on the current Galileo Probe with suitable modifications for use at the other giant outer planets. The carrier

Table 2-3

Summary Results of Benchmark Tests

Development Project/ Source	Flight Units	Benchmark Cost	Model Cost Estimate	Estimated Error	Adjusted Model Cost Estimate	Estimated Error
Mariner '73 SAI Database	2	$130M FY77	$137M FY77	-5%	$137M FY77	-5%
Galileo Probe POP 80-2	1	$108M FY82	$ 75M FY82	31%	$109M FY82	-1%
VOIR JPL Cost Review	1	$218M FY80	$157M FY80	28%	$226M FY80	-4%

design is assumed to benefit from the contractor's experience in designing low-cost spinning spacecraft. Other guidelines include a presumption that the project is charged for RTG units, 15% contingency is to be applied, and only the development cost, i.e. costs to launch + 30 days, is to be estimated.

Figure 2-1 presents the cost model output for the Outer Planet Probes Development Project. Hardware cost categories are shown separately for the two different spacecraft. Functional categories, however, are shown as totals for the overall development effort and not prorated to each spacecraft. The total development is estimated at approximately $277 Million without contingency. Table 2-4 summarizes the development cost estimate for this project. "Science Development" refers only to the probe science since the carrier spacecraft has no science hardware. "Probe System" and "Carrier System" each include a portion of the System Support category prorated on the basis of total hardware cost. With contingency, the total development cost is estimated at $353 Million.

Table 2-4. Cost Summary for Outer Planet Probe Development Project

	FY1982 $M
Program Management/MA&E	18.4
Science Development	39.6
Probe System	56.4
Carrier System	144.0
RTG's	30.0
Launch + 30 Days Operations	18.4
Subtotal	306.8
APA Reserve (@ 15%)	46.0
TOTAL	352.9

SAI PLANETARY PROGRAM COST MODEL

Input Data Worksheet Page 1
Project Scenario Definition

MISSION*: *Outer Planet Probe Project*

HARDWARE CONFIGURATION:

SPACECRAFT ELEMENT*	NO. OF UNITS*	DESIGN HERITAGE
Probe	*3*	*Galileo Probe*
Probe Carrier	*3*	*Contractor's design base*

LAUNCH VEHICLE: *Shuttle/IUS*

FLIGHT MODE*: *Jupiter Swingby*

MISSION PROFILE:

LAUNCH NO.	LAUNCH DATE*	FLIGHT TIME*	ENCOUNTER TIME*
1	*April, 1992 (Saturn)*		
2	*Jan, 1994 (Uranus & Neptune tandem launch)*		

BASE FISCAL YEAR*: FY *1982*

COST SPREAD OPTION PROJECT START DATE:

SPECIAL COST GUIDELINES:

System contract for six spacecraft
Costs to Launch + 30 days only
3 RTG's @ $10M/unit
15% APA/Reserve

*Necessary Information

SAI PLANETARY PROGRAM COST MODEL

Input Data Worksheet Page 2

Flight Hardware Definitions

SPACECRAFT ELEMENT: *Probe*

			INHERITANCE CLASS PERCENT BY MASS				
			BLOCK BUY	EXACT REPEAT	MINOR MOD	MAJOR MOD	NEW DESIGN
ENGINEERING:							
Structure & Devices:	58.4	kg	0.0	100.0	0.0	0.0	0.0
Thermal, Cabling & Pyro:	23.9	kg	0.0	100.0	0.0	0.0	0.0
Propulsion Inerts:	-0-	kg					
Att & Articulation Control:	-0-	kg					
Telecommunications:	12.9	kg	0.0	65.0	35.0	0.0	0.0
Antennas:	-0-	kg					
Command & Data Handling:	15.6	kg	0.0	65.0	35.0	0.0	0.0
Power*: Solar ✓ RTG ___	13.4	kg	0.0	100.0	0.0	0.0	0.0
Aerodeceleration:	91.9	kg	0.0	65.0	35.0	0.0	0.0
Landing Radar/Altimeter:	-0-	kg					
SCIENCE							
Imaging Mass:	-0-	kg					
Imaging Resolution:	-0-	PPL	Vidicon ___	CCD ___	Fax ___		
Particle & Field:	-0-	kg					
Remote Sensing:	6.9	kg	0.0	100.0	0.0	0.0	0.0
Direct Sensing:	21.4	kg	0.0	62.8	37.2	0.0	0.0

*For RTG Power Systems, do not include mass of RTG units

SAI PLANETARY PROGRAM COST MODEL

Input Data Worksheet Page 2
Flight Hardware Definitions

SPACECRAFT ELEMENT: *Probe Carrier*

			INHERITANCE CLASS PERCENT BY MASS				
			BLOCK BUY	EXACT REPEAT	MINOR MOD	MAJOR MOD	NEW DESIGN
ENGINEERING:							
Structure & Devices:	183.6	kg	8.2	1.0	0.0	90.3	0.0
Thermal, Cabling & Pyro:	52.6	kg	0.0	0.0	0.0	100.0	0.0
Propulsion Inerts:	16.2	kg	87.0	1.8	0.0	11.2	0.0
Att & Articulation Control:	21.1	kg	34.1	65.9	0.0	0.0	0.0
Telecommunications:	31.0	kg	22.2	77.8	0.0	0.0	0.0
Antennas:	9.1	kg	11.0	17.6	71.4	0.0	0.0
Command & Data Handling:	50.3	kg	24.6	75.4	0.0	0.0	0.0
Power*: Solar ___ RTG ✓	35.5	kg	64.8	0.0	0.0	35.2	0.0
Aerodeceleration:	-0-	kg					
Landing Radar/Altimeter:	-0-	kg					
SCIENCE							
Imaging Mass:	-0-	kg					
Imaging Resolution:	-0-	PPL	Vidicon ___	CCD ___	Fax ___		
Particle & Field:	-0-	kg					
Remote Sensing:	-0-	kg					
Direct Sensing:	-0-	kg					

*For RTG Power Systems, do not include mass of RTG units

LABOR & COST PREDICTION RUN -- 08-APR-82

Program: Outer Planet Probe Project　　First Launch: April 1992　　Fiscal Year 1982 $M

DEVELOPMENT PROJECT

Element 1: Probes
3 Flight Unit(s)

Category	MASS kgs	DLH khrs	RLH khrs	NRLH khrs	COST $M	INHERITANCE CLASS (%) BB	ER	MIN	MaM	DLHI khrs	NRLHI khrs	COSTI $M
STRUCTURE & DEVICES	58.4	144.8	72.6	72.2	8.20	0.0	100.0	0.0	0.0	87.1	14.4	4.93
THERMAL, CABLING & PYRO	23.9	115.3	58.8	56.5	6.44	0.0	100.0	0.0	0.0	70.1	11.3	3.91
TELECOMMUNICATIONS	12.9	215.1	116.1	98.9	11.82	0.0	65.0	35.0	0.0	155.0	38.8	8.51
COMMAND & DATA HANDLING	15.6	171.1	61.8	109.3	8.59	0.0	65.0	35.0	0.0	104.7	42.9	5.26
SOLAR/BATTERY POWER	13.4	75.8	25.7	50.1	4.07	0.0	100.0	0.0	0.0	35.7	10.0	1.92
AERODECELERATION MODULE	91.9	311.7	94.8	216.9	18.09	0.0	65.0	35.0	0.0	180.0	85.1	10.44
REMOTE SENSING INST	6.9	215.1	96.0	119.1	12.35	0.0	100.0	0.0	0.0	119.9	23.8	6.88
DIRECT SENSE/SAMPLE INST	21.4	842.4	311.0	531.4	45.59	0.0	62.8	37.2	0.0	526.0	215.0	28.46
TOTAL HARDWARE	244.4	2091.2	836.9	1254.3	115.14					1278.3	441.4	70.32
(Percent Cost Reduction												38.9%)

Element 2: Probe Carriers
3 Flight Unit(s)

Category	MASS kgs	DLH khrs	RLH khrs	NRLH khrs	COST $M	INHERITANCE CLASS (%) BB	ER	MIM	MaM	DLHI khrs	NRLHI khrs	COSTI $M
STRUCTURE & DEVICES	183.6	396.7	170.4	226.3	22.46	8.2	1.0	0.0	90.8	366.0	195.6	20.73
THERMAL, CABLING & PYRO	52.6	167.3	95.0	72.3	9.34	0.0	0.0	0.0	100.0	163.7	68.7	9.14
PROPULSION	16.2	250.8	34.5	216.4	15.68	87.0	1.8	0.0	11.2	58.3	23.8	3.64
ATT & ART CONTROL	21.1	360.0	122.3	237.7	21.01	34.1	65.9	0.0	0.0	153.6	31.3	8.97
TELECOMMUNICATIONS	31.0	584.9	329.2	255.7	32.14	22.2	77.8	0.0	0.0	369.0	39.8	20.27
ANTENNAS	9.1	194.4	91.2	103.2	11.01	11.0	17.6	71.4	0.0	150.1	58.9	8.50
COMMAND & DATA HANDLING	50.3	940.1	409.7	530.4	47.22	24.6	75.4	0.0	0.0	489.6	80.0	24.60
RTG POWER (W/O RTG'S)	35.5	380.1	249.0	131.0	18.84	64.8	0.0	0.0	35.2	292.9	43.8	14.52
TOTAL HARDWARE	399.4	3274.2	1501.2	1773.0	177.71					2043.2	542.0	110.36
(Percent Cost Reduction												37.9%)

	DLH khrs	COST $M	DLHI khrs	COSTI $M
SYSTEM SUPRT & GRND EQPT	1655.9	88.16	1034.4	55.07
LAUNCH+30DAYS & GRND S/W	526.2	29.72	325.8	18.40
SCIENCE DATA DEVELOPMENT	83.6	6.98	51.0	4.26
PROGRAM MANAGEMENT/MA&E	573.7	29.24	361.4	18.42
TOTAL DEVELOPMENT	8204.8	446.95	5094.1	276.84
(Percent Cost Reduction				38.1%)

Figure 2-1 Cost Model Output for Outer Planet Probe Project

Future Work

In addition to continuing collection and analysis of cost data from on-going projects, three major development efforts have been identified to complete the Model and further enhance its capabilities.

The first task is to complete the model revision effort by developing new algorithms for estimating costs of mission operations and data analysis. Although historical costs will be used as guidelines, these new algorithms must take into account the expected effects of planned cost reducing procedures such as the multi-mission end-to-end information system and reduced cruise phase activity.

The second effort would involve updating the inheritance algorithm with a more systematic determination of the numerical weighting factors used in the algorithm. The factors currently in use represent best estimates of appropriate values. The update will be accomplished by analyzing cost data from past projects which are known to have benefitted from hardware design heritage.

The final task, related to capabilities enhancement, would be to develop a new, analytical model to transform a point cost estimate into annual funding levels. Such a model should account separately for the different phases of a project, e.g. hardware development versus flight operations. It must also be capable of dealing with the wide variety of project implementation scenarios, which can range from relatively simple Pioneer-class projects to highly complex Viking-class projects.

2.3 Planetary Missions Performance Handbooks

The purpose of the Planetary Missions Performance (PMP) Handbook series is to provide analysts and program planners with a compendium of the basic performance data essential to the preliminary stages of mission selection and design. In the past, two types of NASA handbooks have been prepared, each presenting a different type of information: (1) trajectory data handbooks such as the NASA SP-35 series, and (2) propulsion system capability handbooks such as the NASA Launch Vehicle Estimating Factors document. To make use of these data in performance calculations, the analyst is required to do additional work to arrive at an optimum launch date, to explore a window about that date, to budget propellant for midcourse trajectory corrections, and to compute velocity impulse requirements for orbit capture at the target. Such a computational burden inhibits the broad overviews and parametric studies characteristic of preliminary mission planning exercises. The PMP Handbook series carries desk-ready performance analysis one step further by combining the two basic groups of data and addressing these computational chores. The result is mission performance data in a form which is immediately useful in planning exercises.

During the past contract year, work has been performed on three separate volumes of the Handbook series. The status of these tasks is outlined below.

Volume III --- The first edition of Volume III - Comets and Asteroids contains payload calculation results for a wide selection of missions to small bodies. Mission Profiles incorporate the most recently obtainable propulsion system definitions, timely interplanetary transfer techniques, and currently prevailing design and exploration guidelines. Recognizing that continuing research and near-term exploration achievements are constantly revising these assumptions, and that the basic performance data are sensitive to such changes, the Handbook has been organized and assembled in such a manner as to permit ready incorporation of future revisions and additions.

Payload performance results and basic trajectory data are organized by target and mission type in the sections which follow. There are five of these sections:

Comet Flybys
Comet Rendezvous
Comet Sample Return
Asteroid Rendezvous
Asteroid Sample Return

Each of these sections is tabbed and has its own pagination scheme for referencing convenience. Within each section, a consistent pattern of organization is followed. It begins with an introductory overview of the missions presented and a summary of payload performance by mission type and launch opportunity. General descriptions are provided for each mission type, to include any specific assumptions made regarding module masses and the operations sequence. Then, for each launch opportunity, the section includes a Mission Profile which details trajectory data and mass performance, an ecliptic plane projection plot of the trajectory, and ancillary data as appropriate. Colored pages are used within each section to set off logical subsections of the opportunity data.

The work on Volume III was completed during the first months of the contract year, and the Handbook was distributed in May 1981.

Following the completion of the first edition of Volume III, parallel work was conducted on revisions to both Volumes I and II. The status of this work is described in the following paragraphs.

Volume I --- Work on the 3rd revision (4th edition) of Volume I-Outer Planets has been ongoing during the past contract year, and is continuing at the present time. The major revisions for this new edition are summarized below.

- Extend launch opportunities thru 2000
- Add new flight modes for all targets
- Add Neptune orbiter missions to data base

- Improve computer automated generation of Handbook graphs and tables
- Include performance calculations for Shuttle/Centaur

The scope of the missions to be included in the new edition of Volume I are presented in the following table.

VOLUME I --- OUTER PLANETS SCOPE OF MISSIONS

TARGET	MISSION	FLIGHT MODE			
		BALLISTIC	ΔV-EGA	SEEGA	$SEEGA^2$
JUPITER	FLYBY	DIR.			
	ORBITER	DIR.	DIR., VEGA	DIR.(1+ & 2+)	
SATURN	FLYBY	DIR., J/S			
	ORBITER	DIR., J/S	DIR., J/S	DIR., J/S	DIR.
URANUS	FLYBY	DIR., J/U	J/U ?		
	ORBITER	DIR., J/U	DIR., J/U	DIR., J/U	DIR.
NEPTUNE	FLYBY	J/N, J/U/N	DIR., J/N	DIR. J/N, J/U/N, $J{<}^{N}_{P}$	
	ORBITER		J/N	DIR., J/N	DIR., J/N
PLUTO	FLYBY	J/P	DIR., J/P	DIR., J/P, $J{<}^{N}_{P}$	

LAUNCH OPPORTUNITIES: 1984-2000

Launch opportunities for all outer planet missions are extended thru the 1990's decade and encompass the period 1984-2000 inclusive. Some of the new flight modes included in this revision are the ΔVEGA and SEEGA Jupiter Orbiter missions, the J/S opportunity in the mid-1990's, the $SEEGA^2$ missions to Saturn, Uranus, and Neptune, and the ΔVEGA with and without Jupiter swingby to Uranus and Pluto.

The work on this revision of Volume I of the Handbook can be categorized into 3 tasks: trajectory generation, software development, and performance calculations/publication.

The generation of new trajectory data for the 1990's decade comprises the work performed under the first task. This includes the addition of the aforementioned new flight modes to all targets and the inclusion of Neptune orbiter missions to the data set, as well as extending the launch opportunities for trajectory types in the present edition thru the year 2000.

The software development task consists of updating and converting existing data reduction software to the PDP-11 computer, and also developing new programs to further enhance the automation of the Handbook production process. This work is nearing completion at this writing, with only minor modifications necessary to handle all mission types remaining to be finished.

When the software development is completed and all the trajectories have been generated, the performance calculations will be made as the first step in the Handbook publication process. Distribution of the 4th Edition of Volume I is anticipated by Spring 1983.

Volume II --- The 2nd Edition of Volume II-Inner Planets of the PMPH series is expected to be released before the end of this calendar year. This new edition greatly expands the amount of basic trajectory data tabulation in addition to the mass performance charts and tables. The major revisions are summarized below.

- Extend Launch opportunities thru 1999
- Add Mercury mission concepts to data base
- Add aerobraking for Venus and Mars missions
- Include performance calculations for Shuttle/Centaur

Launch opportunities for all inner planet missions are extended through the 1990 decade and will encompass the period 1984-1999. Only Type I and Type II direct ballistic trajectories to Mars and Venus are included; neither one-rev ballistic trajectories nor low-thrust SEP trajectories are considered for either planet. In the case of Mercury, both ballistic (multi-Venus swingby) and direct SEPS flight modes are included for all attractive opportunities through the end of the century.

The scope of the data presented on Inner Planet missions in Volume II is presented in the table below.

VOLUME II --- INNER PLANETS SCOPE OF DATA

SCOPE	CURRENT EDITION	REVISED EDITION
• MISSIONS	VENUS FLYBY & ORBITER MARS FLYBY & ORBITER MARS SAMPLE RETURN	VENUS FLYBY* & ORBITER MARS FLYBY* & ORBITER MARS SAMPLE RETURN MERCURY ORBITER/LANDER
• LAUNCH OPPORTUNITIES	VENUS (1981-89) MARS (1981-88) MSR (1981-99)	VENUS (1984-99) MARS (1984-99) MSR (1984-99) MERCURY (1984-99)
• FLIGHT MODE TRAJECTORY	BALLISTIC (TYPE I & II)	BALL. (I & II), VENUS & MARS BALL. MVS] MERCURY SEP]
• LAUNCH VEHICLES	ATLAS/CENTAUR SHUTTLE/IUS (I, II & III) SHUTTLE TUG (R & E)	SHUTTLE IUS (II, STAR 48) SHUTTLE/WIDE BODY CENTAUR SHUTTLE/OTV (R & E)
• RETRO PROPULSION	EARTH & SPACE STORABLE (ORB) ALSO SOLID MONO/(MSR)	SOLID/EARTH STORABLE SPACE STORABLE AEROBRAKING AEROCAPTURE (MSR) FUEL PRODUCTION (MSR)

*TRAJECTORY DATA ONLY FOR VENUS AND MARS FLYBYS

Orbiter mission modes at Venus and Mars consider both all-propulsion and aerobraking/propulsion options for orbit capture. At Mercury, both single and dual orbiter concepts are examined with propulsion provided by combined solid/earth-storable retros or space-storable systems. Mars sample return missions will show performance data for various implementation options such as all-propulsion (i.e. impulsive), aerobraking, aerocapture, and in-situ surface fuel production. Two examples of Handbook data formats are shown in Figure 3-1 for ballistic Mercury orbiter missions and Figure 3-2 for Mars sample return missions.

At this writing work is still in progress in generating the Volume II revision. All of the basic trajectory data has been obtained with the exception of SEPS Mercury orbiter data which is currently available for only a few launch opportunities. The existing computer software for automated generation of Handbook graphs and tables has been modified as needed to incorporate data arrays for the extended launch opportunities, mission modes, propulsion options, and launch vehicles considered. All of the graphs and tables for Volume II have been generated with the exception of the Mars Sample Return missions. These final tasks are to be completed this Fall, with distribution before year-end.

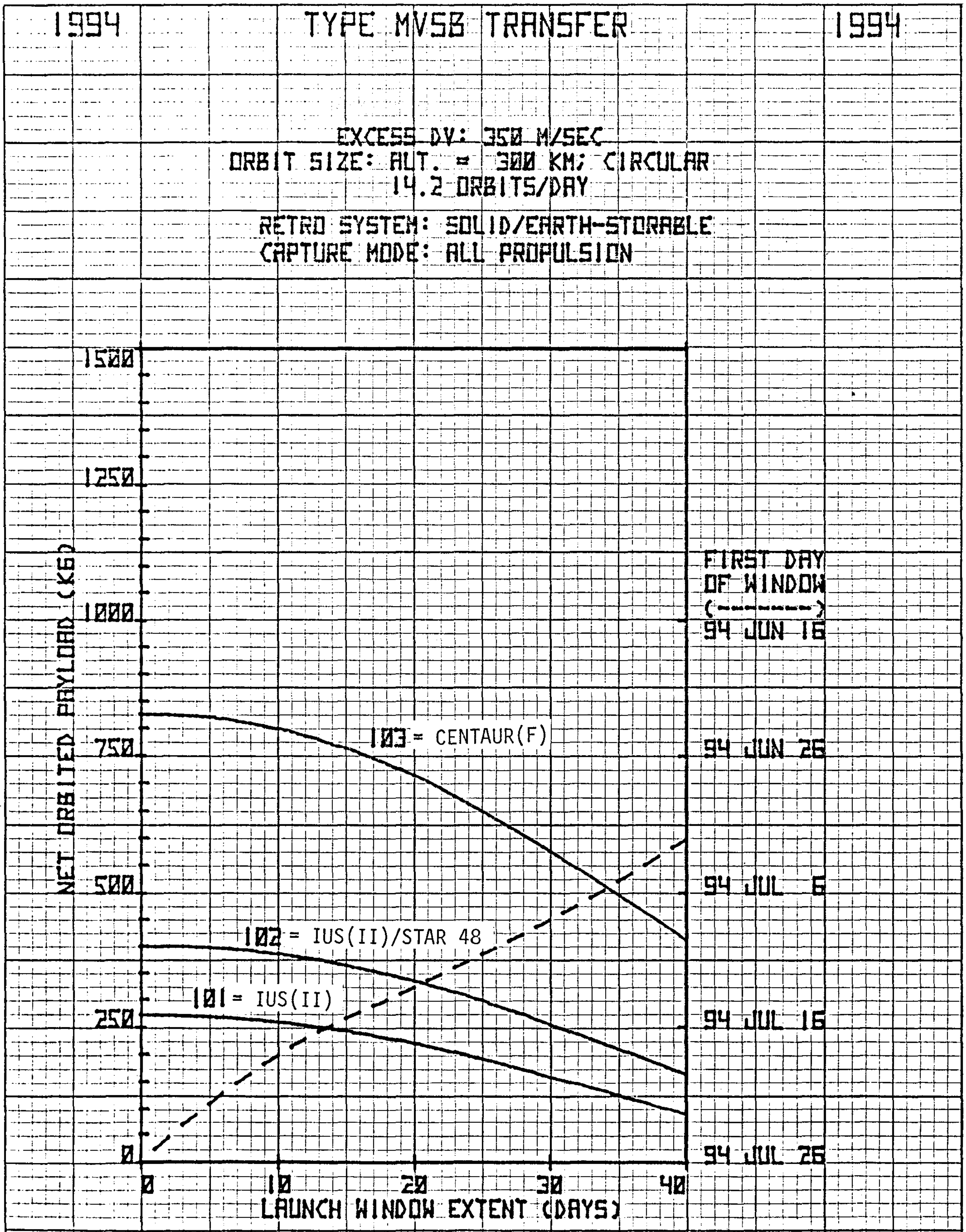


FIG. 3-1

1994 MERCURY ORBITER
MISSION PERFORMANCE SUMMARY

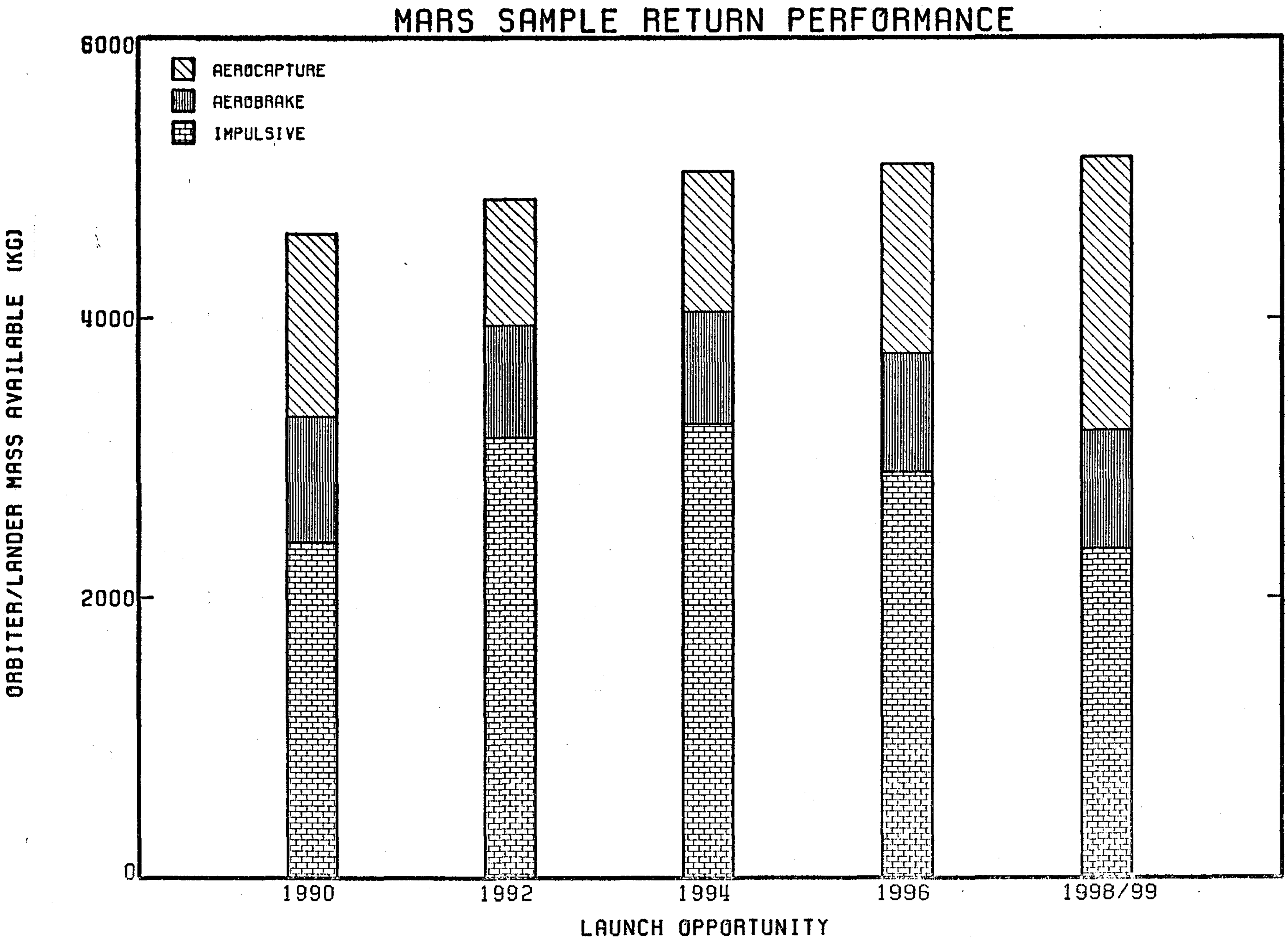

MARS SAMPLE RETURN PERFORMANCE
AEROCAPTURE
AEROBRAKE
IMPULSIVE
ORBITER/LANDER MASS AVAILABLE (KG)
6000
4000
2000
0
1990
1992
1994
1996
1998/99
LAUNCH OPPORTUNITY

Figure 3-2

2.4 Solar System Exploration Committee Support

The Solar System Exploration Committee (SSEC) was formulated by NASA as an ad hoc committee of the NASA Advisory Council to "translate the scientific strategy developed by the Space Science Board of the National Academy of Sciences into a realistic, technically sound sequence of missions consistent with that strategy and with resources expected to be available for solar system exploration. (The SSEC is) to focus its initial efforts on those missions planned for initiation in FY'84 and then extend its considerations as far into the future as possible." Science Applications, Inc. (SAI) was selected as the Earth and Planetary Division Advanced Studies Contractor to provide technical support to the SSEC during its tenure and participate in its proceedings. Mr. John Niehoff, a key person in the advanced studies effort, was, therefore, appointed as a member of the SSEC. The SSEC was formed in October 1980 and is expected to stand as an Advisory Council committee for three years, i.e. through September 1983.

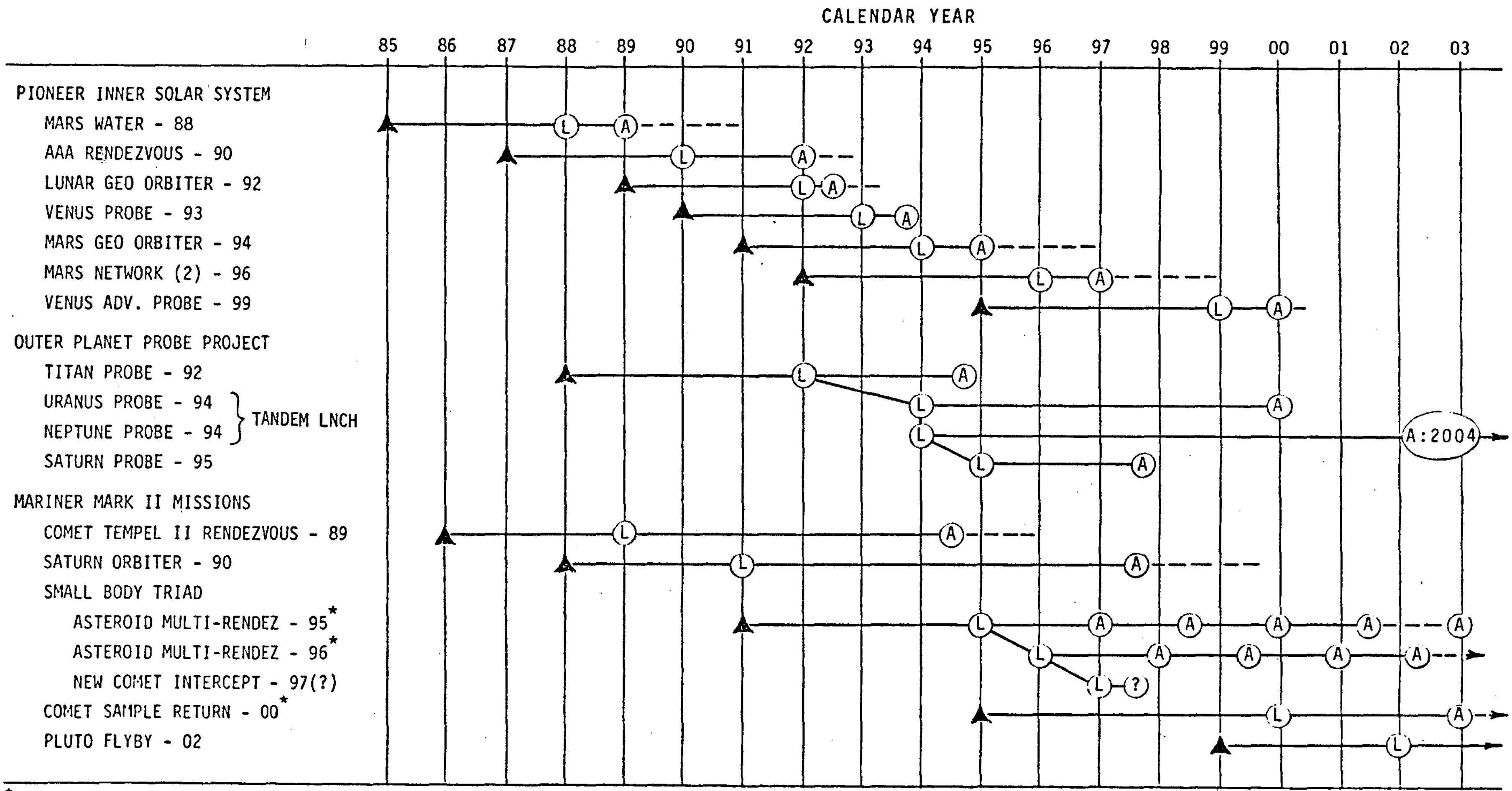
During the current contract period the Committee met eight times including a one-week Summer Study during August 1981. Mr. Niehoff attended all eight meetings and made six presentations to the Committee. Those presentations were:

1) "Constraints to Planning", Cal Tech Meeting, 10 November 1980
2) "Alternative Mission Modes", JPL Meeting, 20 January 1981
3) "Low Activity Program Options", Berkeley Meeting, 23 February 1981.
4) "Mission Sequences, 1982-2002", NAS Meeting, 2 June 1981
5) "SOC-Based Planetary Missions Performance Assessment", University of Arizona Meeting, 9 February 1982
6) "Viking-Class Missions", University of Arizona Meeting, 9 February 1982

In addition to these presentations, considerable effort was expended in preparing and evaluating candidate flight project cost estimates for the Summer Study Activity. Approximately 50 project options were costed and prepared in a cost data base for real-time program planning at the Summer Study. Working with Mr. Daniel Spadoni, Cost Analyst, Mr. Niehoff was able to provide mission sequence schedules and program cost waterfall charts during the Summer Study by using SAI/Schaumburg's computing facilities and a remote Silent 700 terminal. Examples of the schedules and charts produced for the Summer Study participants are shown in Figures 4-1 and 4-2. Technical support and participation by the advanced studies contractor is expected to continue through the remaining tenure of the SSEC.

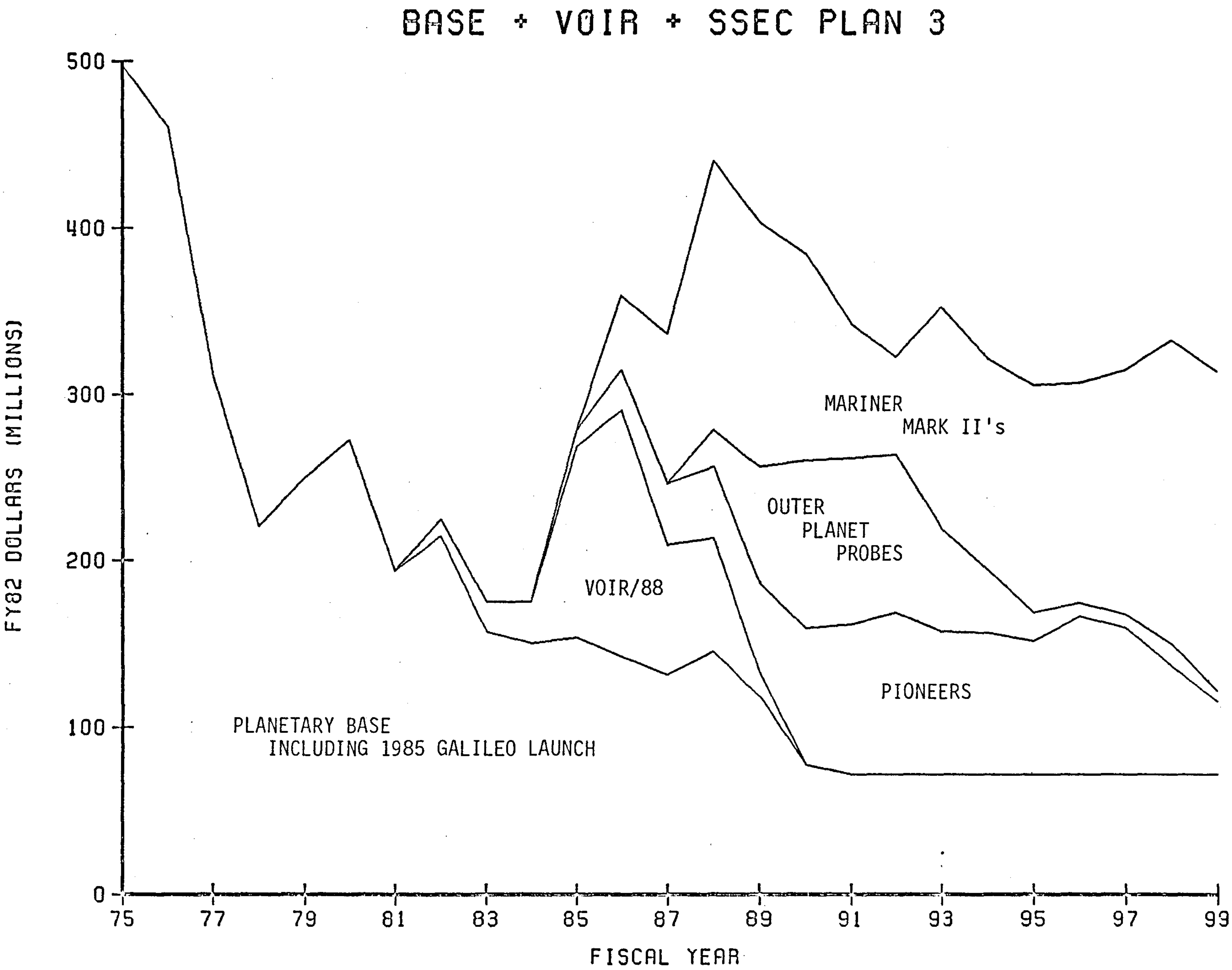
Fig. 4-1

SSEC PLAN #3: DEFERRED SEPS INITIATION



*SEPS MISSIONS

Fig. 4-2


BASE + VOIR + SSEC PLAN 3
FY82 DOLLARS (MILLIONS)
500
400
300
200
100
0
75
77
79
81
83
85
87
89
91
93
95
97
99
FISCAL YEAR
MARINER
MARK II's
OUTER
PLANET
PROBES
VOIR/88
PIONEERS
PLANETARY BASE
INCLUDING 1985 GALILEO LAUNCH

2.5 Mars Program Planning

The purpose of this task was to perform a low but continuing level of effort in Mars program planning in order to maintain the planet's presence in evolving planetary exploration strategies. It was expected that this activity would assure compatibility of a newly developed 10-year plan with long-range Mars exploration objectives at a time when budgetary constraints are limiting many new initiatives. The SAI Advanced Studies Team collaborated with similar groups at JPL and ARC/NASA during the contract period to fulfill this objective. We also participated in a national conference entitled "The Case for Mars" which addressed future strategies for Mars exploration. A brief summary of each of these activities is presented below.

A performance analysis of round-trip Mars missions during the 1990's was completed for JPL support jointly by this task and JPL Contract No. 955096. The study considered six flight options applied to five launch opportunities, i.e. 1990, 1992, 1994, 1996, and 1999. The first two options assumed impulsive capture into Mars orbit with either impulsive or aerobrake returns to earth. The second two options assumed Mars aerobraking with either impulsive or aerobrake returns to earth. The final two options assumed Mars aerocapture and impulsive or aerobrake returns to earth. An example of the performance results is presented in Figure 5-1 for the first two options during the 1994 launch opportunity. Payload performance is plotted as a function of the Mars arrival/departure date. Hence, using the Shuttle/Wide-body Centaur a maximum of 5500 kg could be delivered impulsively into a 24-hour Mars orbit with earth-storable propulsion. The minimum required departure mass, assuming an impulsive return to earth orbit is 2200 kg and occurs 340 days (2450310 - 2449970) after arrival. Thus a net mass of 3300 kg (5500-2200) is available at Mars during its Northern hemisphere fall season for landing, surface exploration, sample collection, and return to orbit.

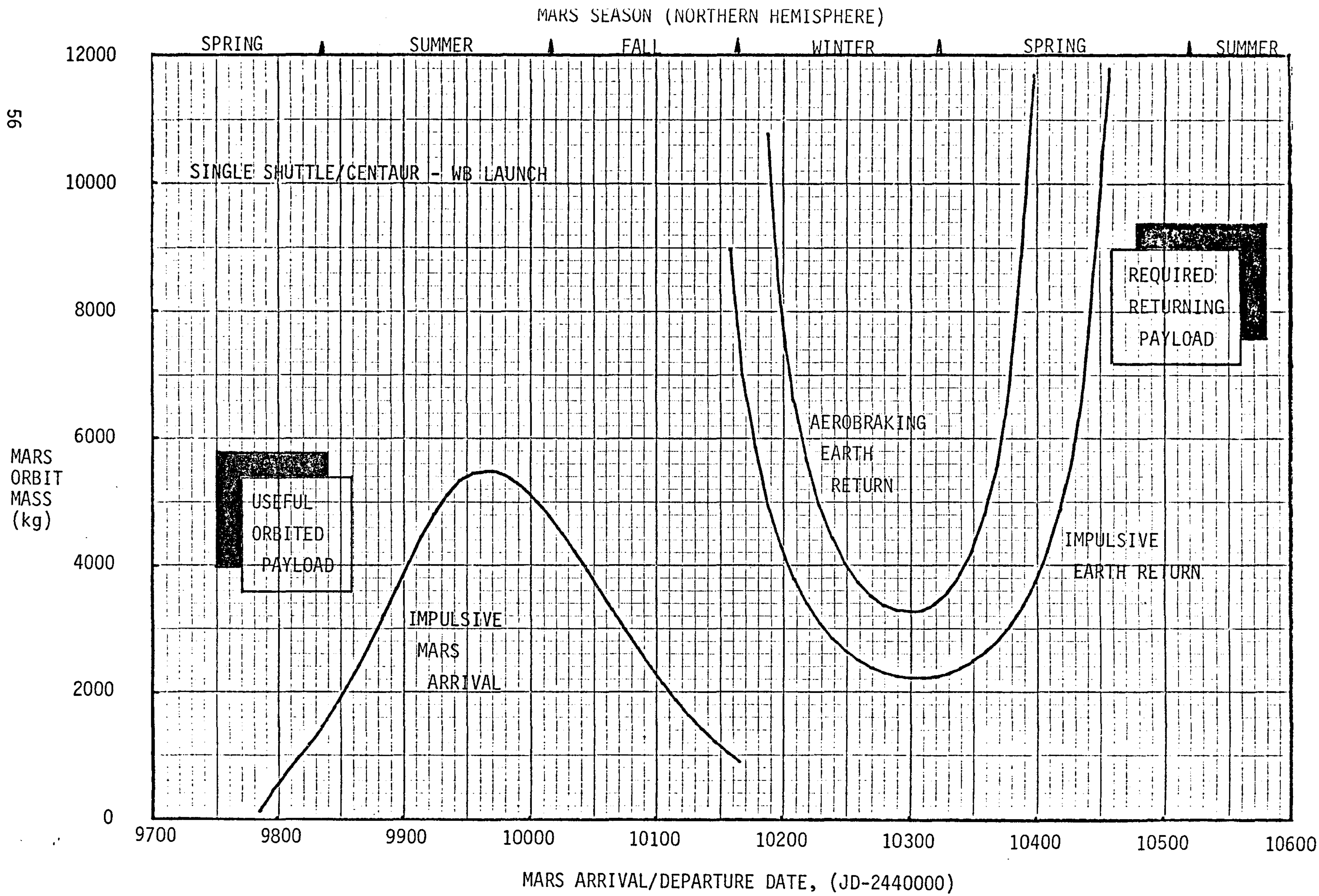


FIG 5-1 MARS ORBIT PAYLOADS FOR 1994 OPPORTUNITY
(with Mars Impulsive Capture)

A total of 15 such performance graphs, i.e. 3 sets of mission options times 5 launch opportunities, were completed in this analysis. From these figures it was concluded that mass performance was most sensitive to launch opportunity characteristics when impulsive Mars capture was employed. Both Mars aerobraking and aerocapture moderate this opportunity dependence. Direct impulsive capture of the earth-returned sample capsule is preferred. Significant differences in the shapes of the performance curves, i.e. staytime and arrival/departure date flexibility, do occur between opportunities. Missions in the later half of the 1990's tend to have broader optimums enabling greater flexibility for mission design. A brief report of these results, including all graphical results and a substantial appendix of supporting trajectory data, was prepared as part of the JPL contract which supported a portion of the analysis. It is available upon request to the interested reader.

A second planning activity was also conducted with JPL concerning the role of In Situ Propellant Production (ISPP) in future Mars missions. During these meetings the evolving planetary exploration strategy was reviewed, the presence of future Mars missions in this strategy was assessed, the potential role of ISPP in these missions was discussed, and a study plan was developed to address system designs and performance results for ISPP-based Mars missions. A schedule was also set to coordinate the completion of study results with the Solar System Exploration Committee's (SSEC) planning effort. Sample returns, re-flyable landers, and airplanes were defined as attractive mission concepts of future Mars exploration interest for evaluation with ISPP augmentation. A joint JPL, Old Dominion University, SAI study was subsequently implemented to assess the Mars ISPP possibilities as a result of this planning effort.

The SAI Advanced Studies Team also worked with mission analysts and planners at ARC/NASA to develop low-cost Mars mission concepts which were of emerging SSEC interest during the contract period. This effort was initiated with a meeting of scientists at the Laboratory for

Atmospheric and Space Physics in Boulder, Colorado. Here the definition of the trend-setting "Mars Water" mission was defined and the constraints of Pioneer-class low-cost projects established. Using guidelines from this meeting mission capability performance graphs were prepared for ARC/NASA mission planners. These graphs presented injected mass requirements as a function of dry orbiter mass for each of four different low-cost missions. The four mission options are 1) an aeronomy orbiter in a 24-hour orbit with a 200 km periapse altitude, 2) an atmospheric mapping orbiter in a 300 km altitude circular orbit, 3) a geochemical mapping orbiter also in a 300 km altitude circular orbit, and 4) a surface network mission deployed from a 24-hour orbit with a 300 km periapse altitude. Performance results were prepared for four launch opportunities, i.e. 1988, 1990, 1992, and 1994. An example of these results is presented in Figure 5-2 for the 1988 Mars launch opportunity. Note the injected mass capabilities of two candidate earth escape stage configurations - the PAM-A, and the PAM-A/PAM-D motors. From the figure it can be seen that the maximum orbiter dry mass that can be successfully launched by the PAM-A motor to accomplish the Mars Water Mission (i.e. Mission 2: Atmospheric Mapper) is just over 300 kg.

During this study period ARC/NASA had also contracted with four aerospace firms, i.e. Ball Aerospace Systems Division, Martin Marietta Corp., Hughes Aircraft Corp., and RCA, to investigate the feasibility of adapting existing earth-orbiter spacecraft designs to the lower cost Mars mission concept. SAI attended the final presentations of these contractor studies to collect the relevant spacecraft system design data. Using these data with our Planetary Cost Estimation Model we developed, at ARC/NASA's request, Class-C estimates of flight project cost for the four Mars mission options cited above. These cost results are presented in Table 5-1. They were provided to ARC/NASA for assessment and became the basis from which subsequent cost estimates of Pioneer-class missions were developed by ARC/NASA in support of the SSEC planning activity.

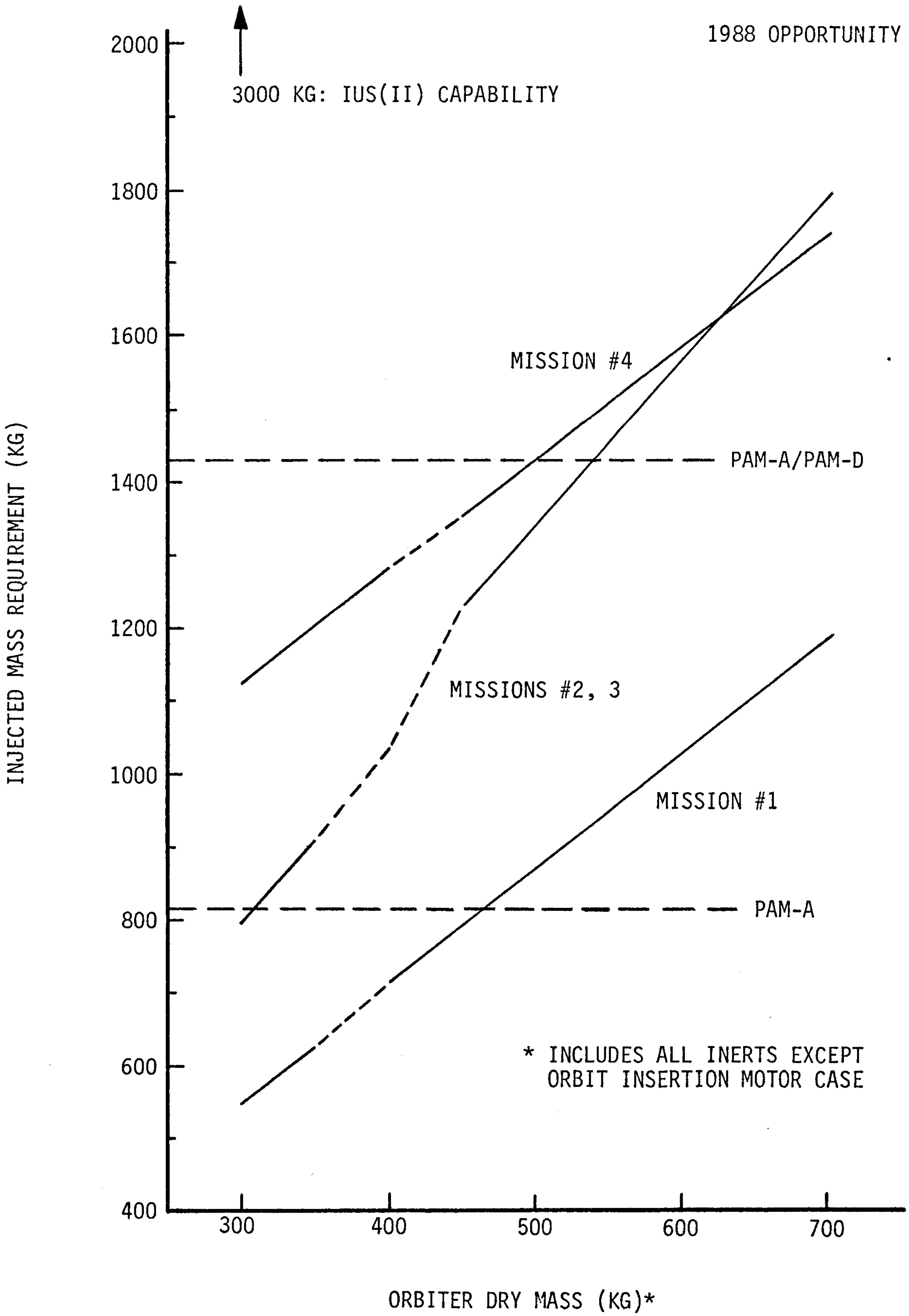


FIGURE 5-2 LOW COST MARS ORBITER MISSION CAPABILITIES (1988)

Table 5-1

LOW-COST MARS ORBITER MISSION COST ESTIMATES (FY'82 $M)

	MISSION/LY			
	Water/88	Aeron/90	Comp/92	Ntwk/94
Pgm Mgmt/MAE	4.1	3.3	3.5	7.2
Science Hdwe	11.4	18.8	19.5	28.6
Spacecraft Hdwe	40.7	23.9	24.2	30.0
Penetrators	—	—	—	56.8
Launch + 30^d Ops	3.9	3.2	3.3	6.8
Mission Ops	7.5	7.5	7.5	8.5
Data Analysis	5.7	9.4	9.7	4.5
Subtotal	73.3	66.1	67.7	142.4
APA/Reserve	14.7	13.2	13.5	28.5
Total	88.0	79.3	81.2	170.9

Notes

All spacecraft based on DE-B

Aeronomy: add Mag boom, use Star 24 motor

Composition: delete scan platform & electronics, add YRS boom
science: NIMS 18 kg, YRS 12 kg, μWR 17 kg

Network: delete scan, add pen support hdwe (40 kg) and relay comm. (3.1 kg), use Star 27 (?)

Penetrators: 4 flt + 1 spare;
release pens on approach & jettison 40 kgs;
science costs escalated from MARS '84 cost review

Orbits

Water	300km circ	Comp	300km circ
Aeron	200km × 24^h	Ntwk	500km circ

A final activity under this Mars Program Planning Task was attendance and participation in the "Case for Mars" conference co-sponsored by the Planetary Society, The Viking Fund, The National Space Institute, The Boulder Center for Science and Policy, and the Rocky Mountain Chapters of the American Astronautical Society and the American Institute of Aeronautics and Astronautics. Two presented papers benefitted from efforts performed under this task. The first was entitled "Mars Missions of Opportunity" and was co-authored by James Murphy, Bruce Pitlman, Chris Leidich, Ray Reynolds, and James Pollack of ARC/NASA, and John Niehoff of SAI. The second paper was entitled "Scientific Activities on the Martian Surface" and was presented by James Cutts of SAI's Planetary Science Institute. John Niehoff, who attended the conference as a representative of the SAI Advanced Studies Team also participated in a Mission Strategy Workshop, and was a panel member in the conference's concluding panel discussion of the question: "Should Man on Mars be the Next Major Goal of the Space Program?"

2.6 GALILEAN SATELLITE MISSION CONCEPTS

The initial exploration of Jupiter's four large Galilean satellites has been accomplished with Voyager and will continue with the Galileo mission in the mid-to-late 1980's. Scientific interest in these bodies would argue for a future, dedicated satellite mission, perhaps in the 1990 decade, whose goal is intensive investigation through close remote sensing and in situ measurements. Several mission concepts that could accomplish this goal were examined in this task. These include single and multiple target scenarios with orbiter and lander deployments. Mass requirements, including propulsion, are significantly greater for such missions than was the case for Voyager and Galileo. However, mission capture capability does or will exist with anticipated developments in launch vehicle upper stages and spacecraft propulsion systems. Payload delivery options include both ballistic flight modes (chemical propulsion) and low-thrust flight modes (solar or nuclear-electric propulsion). The scope of the study addresses various aspects and key issues of Galilean Satellite missions. These include: (1) science objectives and instrumentation; (2) payload delivery performance trades; (3) orbital operations; (4) radiation shielding; and (5) reference mission profiles and cost estimates.

Science Objectives and Instrumentation

The massive planet Jupiter has been likened to a miniature solar system with its numerous orbiting bodies, particularly the four large Galilean satellites Io, Europa, Ganymede and Callisto. These satellites orbit Jupiter in near-circular, equatorial paths in the region 5.9 to 26.4 Jupiter radii. Ganymede and Callisto are about the size of Mercury and Europa and Io are each larger than the moon. Based on Voyager data these four bodies seem to have evolved upon different evolutionary paths resulting in large variability of surface characteristics. Detailed studies of the Galilean satellites may yield profound insights concerning early formation processes and useful correlations with the origin and evolution of planetary history.

Satellite science objectives include: (1) morphology, geology, and physical state characterization of surfaces; (2) surface mineralogy and composition distribution; (3) gravitational and magnetic fields determination, and interaction with Jovian magnetospheric particles; and (4) tenuous atmosphere detection and characterization of extended gas/dust clouds that may arise from dynamic phenomena on the surface such as volcanism.

Post-Galileo mission modes applicable to intensive science investigation include: (1) an extended, close flyby satellite tour by an orbiter about Jupiter; (2) a close orbiter about one or more satellites; (3) multiple deployment of surface penetrators or alternate hard landers; (4) a large lander of the Viking class; and (5) sample return to Earth. The first mission concept is not studied here per se but only as part and prelude to satellite orbiter or lander deployment. The sample return mission is also not examined since its new technology requirements and cost are thought to be well beyond the scope and time frame of post-Galileo concepts.

A candidate science payload for an orbiter spacecraft is listed in Table 6-1. Total mass and power for the seven instruments/experiments are, respectively, 93 kg and 85 watts. Data rate requirements are dominated by the imaging experiment, and to a lesser extent by the near IR/visible reflectance spectrometer. Tables 6-2 and 6-3 show candidate science payloads for a subsurface penetrator and a large surface lander. The penetrator experiments as well as the basic penetrator design are based on previous Mars application studies. Total mass of the several instruments is quite small at about 2 kg as necessitated by the limited volume of the slender penetrator body. One advantage of the alternate hard lander concept is that relaxation of volume constraints allows a larger science payload of the order 7 to 9 kg to be placed on the surface. A Viking-class large lander could support a science payload of about 70 kg, including a complement of instruments capable of carrying out sophisticated analyses of the satellites' elemental and mineralogical composition.

GALILEAN SATELLITE ORBITERS - CANDIDATE SCIENCE PAYLOADS & OBJECTIVES

INSTRUMENT/EXPERIMENT	PRIMARY SCIENCE OBJECTIVES	MASS (KG)	POWER (W)	DATA RATE (BPS)	HERITAGE
RADAR ALTIMETER/DOPPLER TRACKING	GEOPHYSICAL MEASUREMENTS, MAP SATELLITE FIGURE, GRAVITY FIELD, SURFACE RELIEF, SURFACE ROUGHNESS. DETERMINE CENTER OF FIGURE OFFSET, SURFACE PROPERTIES. GRAVITY FIELD CORRELATED WITH SURFACE TOPOGRAPHY TO INFER DEPTH OF COMPENSATION, CRUSTAL THICKNESS.	7	18	625	LPO
MICROWAVE RADIOMETER/ THERMAL IR RADIOMETER	MAP INTERNAL HEAT FLOW AND SURFACE TEMPERATURE. MAP AND TIME-MONITOR VOLCANIC ACTIVITY OF IO	15	12	120	LPO
MULTI-SPECTRAL IMAGING	HIGH RESOLUTION OF AREAS DEEMED TO BE OF SPECIAL INTEREST BASED ON VOYAGER, GALILEO RESULTS; TIME-MONITORING OF RAPIDLY VARYING SURFACE REGIONS OF IO.	28	23	>46K	GALILEO
NEAR IR/VISIBLE REFLECTANCE SPECTROMETER	SURFACE MINERALOGY; COMPOSITIONAL MAPPING IN AREAS OF PARTICULAR INTEREST AS SPECIFIED BY GALILEO RESULTS.	18	8-12	12K	GALILEO
MAGNETOMETER	SATELLITE MAGNETIC FIELDS, INTERACTION WITH JUPITER'S MAGNETOSPHERE; TIME VARIATIONS.	6	5	240	GALILEO
ENERGETIC PARTICLES DETECTION	MEASURE ENERGETIC ELECTRONS AND PROTONS AS FUNCTION OF RADIAL DISTANCE AND ANGULAR DISPLACEMENT FROM JUPITER'S MAGNETOSPHERE EQUATOR.	9	7	912	GALILEO
RADIO SCIENCE	SPATIALLY ISOLATE SOURCES OF RADIO-FREQUENCY PHENOMENA USING OCCULTATION TECHNIQUES.	USES HIGH-GAIN ANTENNA AND RADIO FREQUENCY SUBSYSTEM.			GALILEO
X-RAY SPECTROMETER *	SURFACE ELEMENTAL COMPOSITION	10	10	256	LPO

* Heavy shielding and close-in orbit (to accentuate signal) required for Io and Europa. Otherwise a "telescoping" technique from outside Jupiter's magnetosphere would be required.

Table 6-1

Table 6-2

GALILEAN SATELLITE PENETRATORS - CANDIDATE SCIENCE PAYLOAD & OBJECTIVES

INSTRUMENT/EXPERIMENT	PRIMARY SCIENCE OBJECTIVE(S)	MASS (KG)	POWER (W)	DATA (BITS)	HERITAGE
SEISMOMETER	INTERNAL COMPOSITION, STRUCTURE & HOMOGENEITY. SEISMIC ACTIVITY	0.60	0.09	9.5×10^4/DAY 1.4×10^6/EVENT	PROPOSED MARS SURFACE PENETRATOR
ALPHA-PROTON BACKSCATTER/X-RAY FLUORESCENCE SPECTROMETER	SUBSURFACE ELEMENTAL COMPOSITION/ ABUNDANCE	0.40	0.10	6144/ MEASUREMENT	"
TEMPERATURE SENSORS (THERMOCOUPLES)	THERMAL CONDUCTIVITY, HEAT FLOW, SURFACE TEMPERATURE	0.07	0.02	116/SAMPLE	"
WATER DETECTOR (HEATER - P_2O_5 HYGROMETER)	SEMIQUANTITATIVE ESTIMATE OF FREE AND BOUND WATER CONTENT	0.15	5.0	3000/SAMPLE	"
ACCELEROMETER	SUBSURFACE STRATIGRAPHY THROUGH DECELERATION PROFILE	0.03	0.03	6×10^4	"
SURFACE IMAGING (SILICON PHOTOSENSOR FACSIMILE CAMERA)	SITE CHARACTERIZATION, CONDENSATES, SOIL CHARACTERIZATION	0.25	0.90	4×10^6/FULL COLOR CAMERA	"
MAGNETOMETER	MAGNETOSPHERE-SURFACE INTEGRATION, INDIGENOUS SATELLITE FIELD	0.40	0.07	30/SAMPLE	"

Table 6-3

GALILEAN SATELLITE LARGE LANDERS - CANDIDATE SCIENCE PAYLOAD & OBJECTIVES

INSTRUMENT/EXPERIMENT	PRIMARY SCIENCE OBJECTIVES	MASS (KG)	POWER (W)	DATA (BITS)	HERITAGE
MULTISPECTRAL SURFACE IMAGING (FACSIMILE CAMERAS)	SITE CHARACTERIZATION, DETAILED SURFACE STRUCTURE, EXPERIMENT SUPPORT, SYNOPTIC OBSERVATION OF JOVIAN SYSTEM	12.9	4	4×10^6/COLOR PANORAMA	VIKING LANDER
SCANNING ELECTRON MICROSCOPE/MICROPROBE*	ELEMENTAL COMPOSITION/MINERALOGIC ANALYSIS (MINOR & TRACE ELEMENTS)	25	10	TBD	AUTOMATED MOBILE LUNAR SURFACE SURVEY
X-RAY DIFFRACTOMETER	MINERALOGIC ANALYSIS, CRYSTAL STRUCTURE	5	2	TBD	AUTOMATED MOBILE LUNAR SURFACE SURVEY
α-PROTON BACKSCATTER/ X-RAY FLUORESCENCE SPECTROMETER	ELEMENTAL COMPOSITION/ABUNDANCE (MAJOR ELEMENTS)	2	1.5	4.6×10^4/ MEASUREMENT	VIKING LANDER/MARS PENETRATOR
MAGNETOMETER	MAGNETOSPHERE-SURFACE INTERACTION, INDIGENOUS SATELLITE FIELD	0.4	0.07	30/SAMPLE	MARS PENETRATOR
PETROGRAPHIC MICROSCOPE	MINERAL IDENTIFICATION	5	1	2×10^6/IMAGE	AUTOMATED MOBILE LUNAR SURFACE SURVEY
SEISMOMETER	INTERNAL COMPOSITION, STRUCTURE & HOMOGENEITY; SEISMIC ACTIVITY	2.2	4	2/SEC	VIKING LANDER
TEMPERATURE SENSOR	SURFACE TEMPERATURE	0.1	0.03	500/DAY	MARS PENETRATOR
SURFACE SAMPLER	SAMPLE HANDLING, PROCESSING & CONTROL	18	35.4		VIKING LANDER

* An Ion Microprobe Mass Analysis is a possible instrument option. Although it will require more development than the SEMM it can be more valuable in that it has the added capability of determining isotopic composition and the potential for performing automated in-situ age dating.

Payload Delivery Performance Trades

Flight mode options for payload delivery to Jupiter and its satellites consist of ballistic trajectories (either direct or indirect) and low-thrust trajectories implemented by either solar electric propulsion (SEP) or nuclear electric propulsion (NEP). In order to cull out the most promising flight mode/propulsion options for various mission concepts, parametric mass delivery performance data were generated and used as a basis for comparison and tradeoffs.

The performance trades were developed in the format of generic graphs of Jupiter approach mass requirements versus net spacecraft (orbiter) mass. An example of such a graph is given in Figure 6-1 and illustrates the trades which exist for single target orbiter missions. The Jupiter approach mass capabilities used in these curves reflect a standard Shuttle launch (65,000 lbm cargo capacity) to a 150 n mile parking orbit and injection to Earth escape by either the IUS (III) or wide body Centaur (F) upper stages. Both earth storable propellants (I_{sp} = 290 sec) and space storable propellants (I_{sp} = 370 sec) were considered as options for the spacecraft's impulsive ΔV maneuvers in the post injection phases. In computing ΔV requirements for the ballistic approach mission, it was assumed that Jupiter orbit insertion would be implemented via a powered Ganymede swingby and that subsequent close satellite encounters would be used to obtain low V_∞ approach velocities at the target satellite(s) prior to lander deployment and/or orbit capture.

Mass delivery performance trades for low thrust (NEP) approaches to Jupiter are exemplified by Figure 6-2 which plots payload mass against flight time for multi-target orbiter missions (total flight time is the summation of heliocentric and planetocentric time segments). Parameters of the NEP system adopted as a reference for this analysis are included in the figure.

Other missions for which performance trades were established include a satellite flyby tour with small landers, two target satellite orbiters, and satellite orbiters with either penetrators, small landers or large landers.

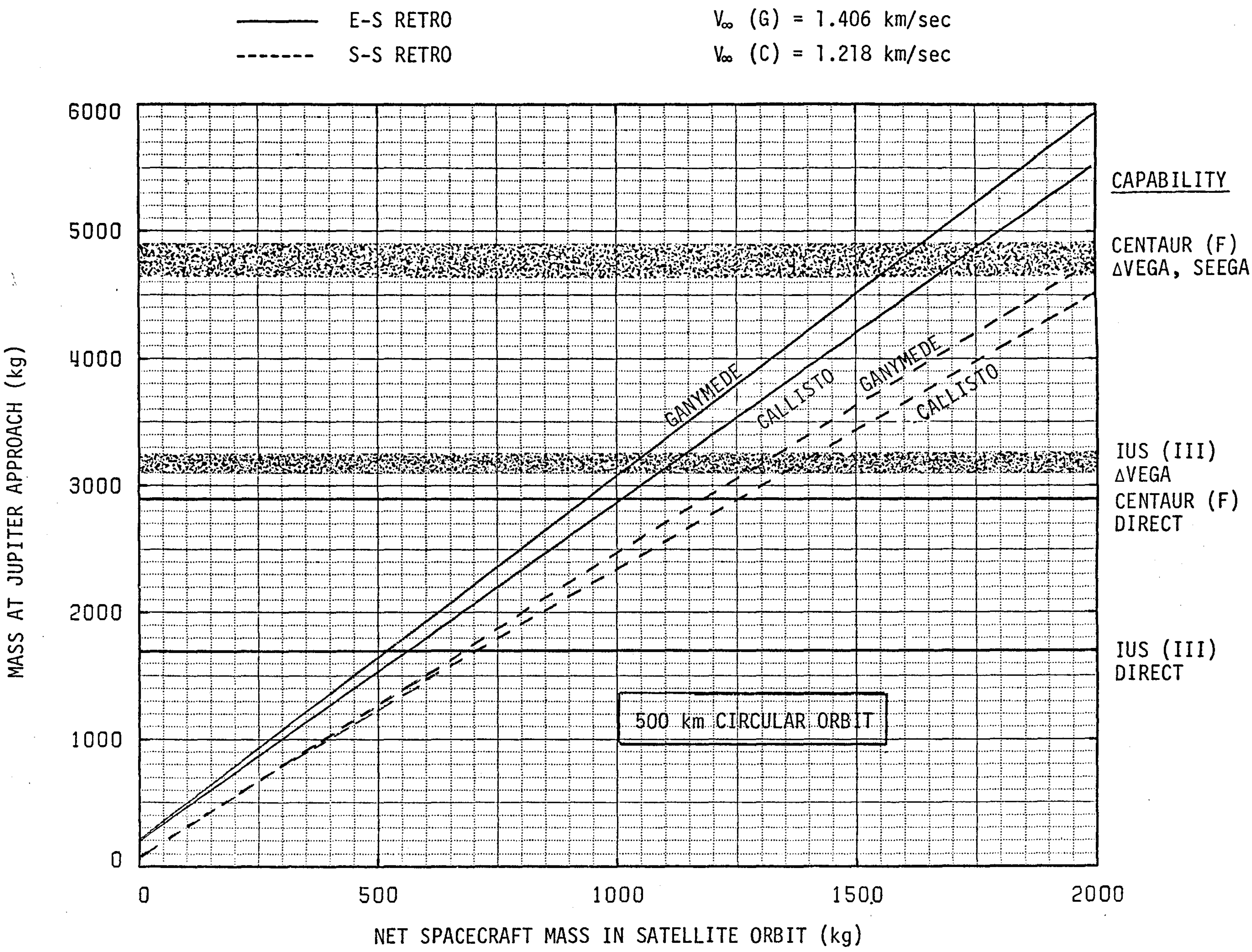


Fig. 6-1 PERFORMANCE TRADES FOR SINGLE TARGET ORBITER (CALLISTO AND GANYMEDE)

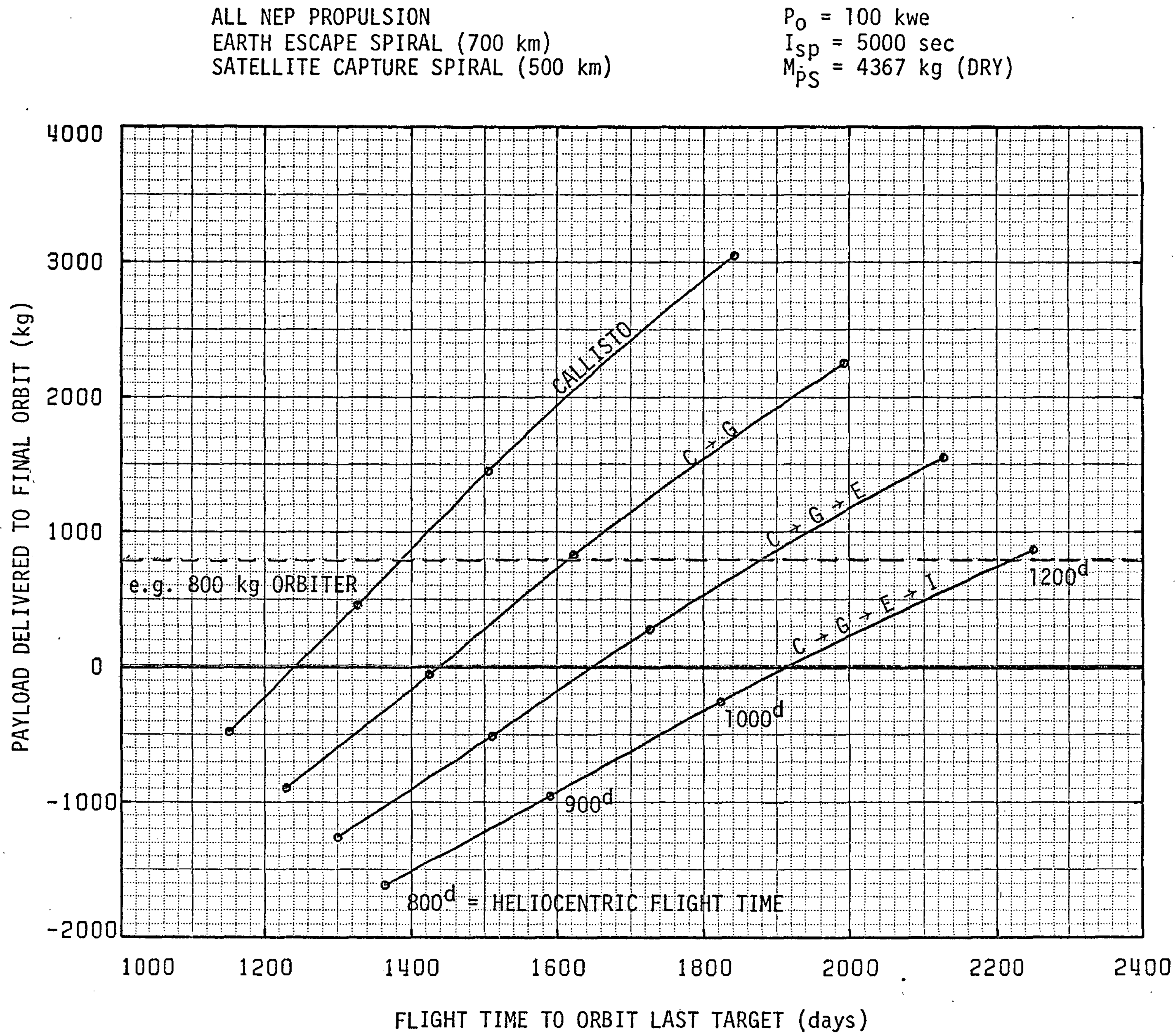


Fig. 6-2 NEP PERFORMANCE TRADES FOR MULTI-TARGET ORBITER

Orbital Operations

Supplemental information needed to compute the propulsion and shielding requirements of applicable mission scenarios was obtained by analyzing various operations at Jupiter including: (1) satellite gravity assisted touring; (2) penetrator/lander deployment techniques; (3) trim maneuvers to overcome stability problems in satellite orbit; and (4) science directed stay-times at each satellite. For ballistic approach missions, post-insertion maneuvers are comprised of satellite targeting and navigation during the tour, bus deflection after lander deployments, and satellite orbit capture/maintenance as the case may be. These maneuver requirements range from 680 to 3917 m/sec depending on the mission mode and target. Total spacecraft ΔV requirements (exclusive of lander retro ΔV) were found to range from 1943 to 5180 m/sec for direct ballistic and SEEGA flight modes, or from 2548 to 5785 m/sec for ΔVEGA flights.

Solid rocket ΔV requirements for lander deployments are treated as part of the lander system sizing. In the case of small landers deployed on satellite approach (i.e. flyby tour), the retro ΔV is of the order 3500 m/sec. Out-of-orbit deployment ΔV lies in the range 1500 to 2000 m/sec depending on the satellite target. For the large Viking-class soft lander, the ΔV range is 1150 to 1650 m/sec (solid rocket) with an additional 320 to 420 m/sec required for terminal maneuvers (monopropellant vernier).

Due to the large perturbing effect of Jupiter on spacecraft orbiting its satellites, the matter of orbit stability and lifetime was one of particular concern. Starting from near circular (e = 0.05) 500 km polar orbits, a perturbation history was calculated for each satellite and the orbital decays were noted. In general, it was found that the orbital lifetimes (especially for Io and Europa) were less than the minimum stay-times desired for useful imaging coverage. A ΔV budget for orbit was therefore computed for each satellite. The values shown in Table 6-4 are for orbiter lifetimes corresponding to the stay-times required for 25% and 100% coverage at 70 m resolution.

From the many science objectives considered for these missions, orbital

TABLE 6-4

TOTAL ΔV REQUIRED TO RETURN SPACECRAFT TO INITIAL ORBIT AFTER PERIAPSE ALTITUDE HAS BEEN PERTURBED $\pm$ 50 km

SATELLITE	ORBIT* INSERTION ERROR (m/sec)	REINITIALIZED ORBIT (e)	REINITIALIZED ORBIT (h_p x h_a)	25% COVERAGE MINIMUM STAY-TIME (days)	25% COVERAGE MINIMUM STAY-TIME (orbits)	25% COVERAGE ΔV (m/sec)	100% COVERAGE MINIMUM STAY-TIME (days)	100% COVERAGE MINIMUM STAY-TIME (orbits)	100% COVERAGE ΔV (m/sec)
CALLISTO	39	.002	500 x 512	20.7	152	39	82.8	611	39
GANYMEDE	44	.002	500 x 513	24.5	190	44	98.0	763	44
EUROPA	31	.002	500 x 508	7.0	58	31	42.5	350	48
IO	39	.002	500 x 509	11.6	110	59	46.8	442	139

* INITIAL ORBIT ECCENTRICITY = 0.05 AFTER ORBIT INSERION MANEUVER

imaging coverage was selected as the satellite stay time criterion because it represents a clear cut measure of science performance that is uniquely time related. To be specific, this measure refers to the fraction or percentage of a satellite mapped at a given resolution. By assuming a polar orbit, the potential exists for 100% global coverage, and this has been used to define the upper bound for satellite stay-times. The lower bound or minimum stay-time at each satellite was selected to be the satellite orbit period. Figure 6-3, which plots the percentage of Ganymede coverage as a function of stay-times and image resolution, is an example of the analyses performed for all the satellites. The results are summarized in Table 6-5 and take into account the information rate consequences of excessive image overlap and limited orbiter to Earth communications capacity (67.2 kBPS).

Radiation Shielding Analysis

The vulnerability of many spacecraft components to the high flux levels of energetic particles trapped in Jupiter's magnetosphere, make radiation shielding a major determinant of mission performance. Using fluence spectra obtained from the Neil Divine/JPL model of Jupiter's radiation belts, extremely simplified (spherical) shielding configurations and radiation design criteria established for Galileo (and projections for future missions), spacecraft shielding requirements were derived for the entire range of mission options considered in this study.

A determination of shielding effectiveness in reducing the internal radiation environment of a spacecraft generally requires a detailed radiation transport analysis based on a specified structural configuration. However, even if specific configurations were known at this time, such an analysis was well beyond the scope of this task. Instead, published electron, electron-bremsstrahlung and proton depth dose data were used to parametrically relate the ionizing dose and displacement fluence of shielded spacecraft components to the shielding thickness (approximated as a spherical shell) and the external fluence spectra experienced during a mission.

Basically, shielding thicknesses capable of limiting the electron

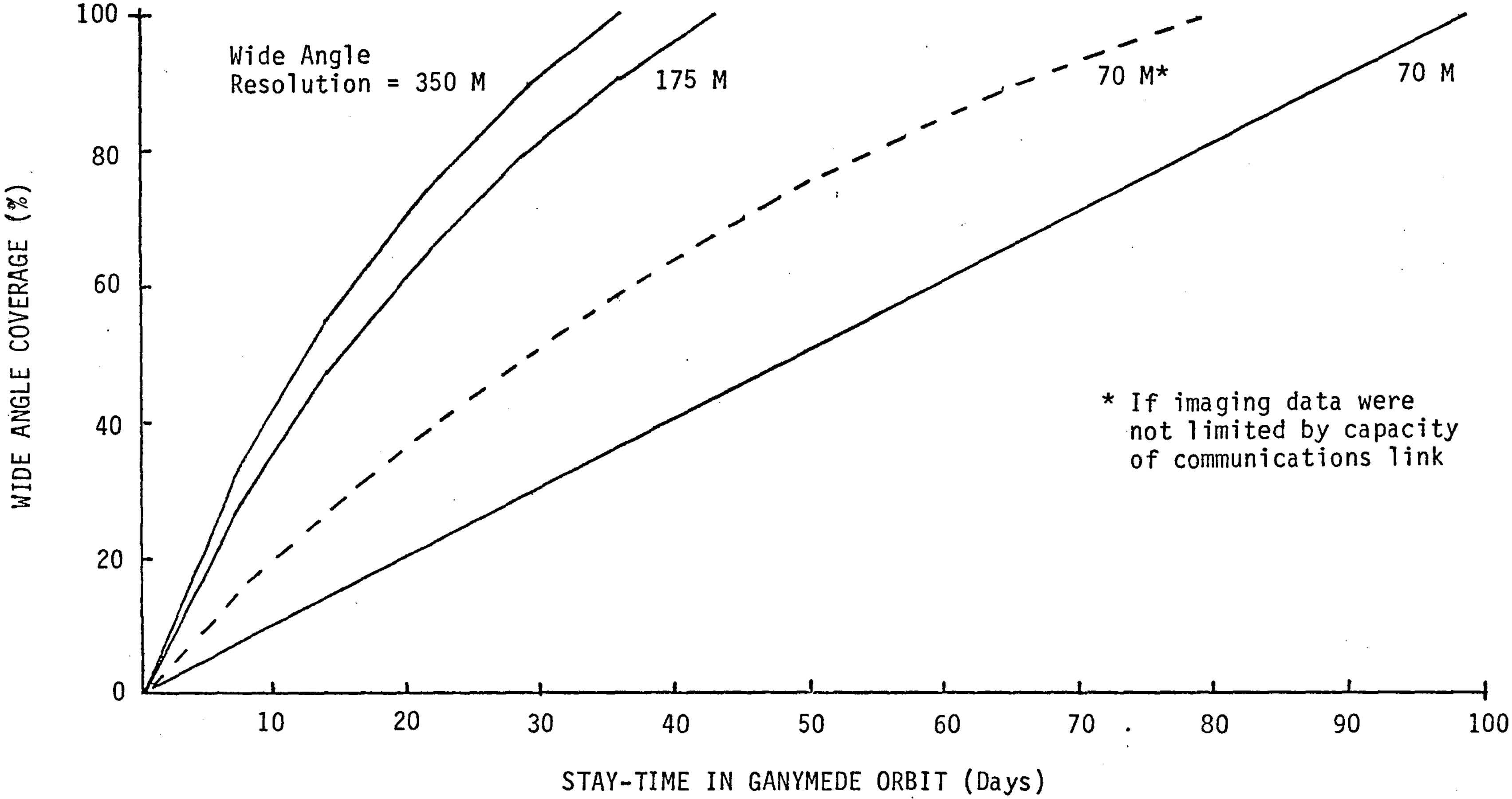


FIG. 6-3 ORBITAL IMAGING COVERAGE OF GANYMEDE

TABLE 6-5

ORBITER IMAGING - STAY TIME REQUIREMENTS

SATELLITE	ROTATION PERIOD MIN. STAY TIME (DAYS)	ORBITER ALTITUDE (km)	WIDE ANGLE RESOLUTION (m)	COVERAGE FOR MIN. STAY TIME (%)	STAY TIMES REQUIRED FOR INDICATED COVERAGE (DAYS)			
					25%	50%	75%	100%
CALLISTO	16.69	500	70	20.5	20.7	41.4	62.1	82.8
		1250	175	52.0	8.0	16.0	29.0	50.0
		2500	350	77.0	5.3	10.6	15.9	33.2
GANYMEDE	7.16	500	70	7.5	24.5	49.0	73.5	98.0
		1250	175	26.2	6.9	15.1	26.5	43.0
		2500	350	31.4	5.6	12.5	21.9	35.7
EUROPA	3.55	500	70	13.1	7.0	15.7	27.0	42.5
		1250	175	19.6	4.7	10.3	17.7	28.5
		2500	350	22.4	4.0	9.0	15.5	24.9
IO	1.77	500	70	4.2	11.6	23.3	35.0	46.8
		1250	175	10.5	4.7	9.9	16.9	26.5
		2500	350	11.2	4.2	9.2	15.7	24.9

ionization dose to 75 krads (Si) and the proton displacement fluence (20 MeV equivalent) to 5 x 10^9 p/cm^2 were specified in order to satisfy Galileo criteria; but to account for the future possibility of additional radiation hardening of electronic components, alternate shielding thicknesses were derived on the basis of a 625 krad (Si) dose limit. Intrinsic spacecraft shielding equivalent to 100 mil aluminum (0.686 g/cm^2) was assumed as the reference point for computing the additional shielding needed to meet these mission radiation limits. Then, using stay-time/image coverage as a parameter, shielding requirements analyses were performed for orbiters, landers and penetrators at all four satellites. The results are summarized in Tables 6-6 and 6-7 according to the payload delivery mode.

These results show that from a shielding standpoint, NEP spiral trajectories are better than ballistic tours for missions to Callisto and Ganymede; however, because they require larger transit times through high flux regions, they demand substantially greater shielding for missions to Europa and Io. It is also evident that prohibitively large shielding masses all but eliminate consideration of Galileo-class electronics for missions to these two inner satellites (short stay-time missions to Europa appear acceptable), and even with additional radiation hardening, the shielding requirements for any Io missions seem to be impractically high. For brevity, only shielding masses corresponding to the minimum and 100% coverage stay-times are included in these summary tables. However, requirements based on other imaging coverage resolution criteria were also determined for use in the selection and assessment of the reference mission set.

Reference Mission Profiles and Cost Estimates

Table 6-8 lists the various spacecraft mass elements used in the analysis of mission capability. Not included are the mass estimates for radiation shielding and retropropulsion (unless indicated) since they must be uniquely derived for each mission. Mission capability is summarized in terms of mass margins by the matrix format of Table 6-9. The margins are stated at Jupiter approach for the ballistic delivery modes, whereas for

Table 6-6

SHIELDING MASS ESTIMATES - BALLISTIC DELIVERY, SATELLITE ORBITER MISSIONS

	ORBITER		LARGE LANDER		HARD LANDER		PENETRATOR	
	MIN STAY* TIME	100% ORB** COVERAGE	MIN STAY TIME	100% ORB COVERAGE	MIN STAY TIME	100% ORB COVERAGE	MIN STAY TIME	100% ORB COVERAGE
CALLISTO	21 KG	21 KG	18 KG	18 KG	5 KG	5 KG	2 KG	2 KG
GANYMEDE	22	33	18	23	5	7	2	2
EUROPA	93 (33)	456 (90)	52 (17)	204 (51)	16 (5)	61 (15)	6 (2)	22 (5)
IO	(216)	(467)	(169)	(261)	(50)	(78)	(18)	(28)

* ONE SATELLITE PERIOD
** 70M RESOLUTION WIDE ANGLE IMAGING
( ) ASSUMES COMPONENT RADIATION HARDENING 10 x GALILEO

Table 6-7

SHIELDING MASS ESTIMATES - NEP DELIVERY, SATELLITE ORBITER MISSIONS

	ORBITER		LARGE LANDER		HARD LANDER		PENETRATOR	
	MIN STAY* TIME	100% ORB** COVERAGE	MIN STAY TIME	100% ORB COVERAGE	MIN STAY TIME	100% ORB COVERAGE	MIN STAY TIME	100% ORB COVERAGE
CALLISTO	0 KG	0 KG	0 KG	0 KG	0 KG	0 KG	0 KG	0 KG
GANYMEDE	0	0	0	0	0	0	0	0
EUROPA	(98)	(135)	(73)	(84)	(22)	(25)	(8)	(9)
IO	X	X	X	X	X	X	X	X

* ONE SATELLITE PERIOD
** 70M RESOLUTION WIDE ANGLE IMAGING
( ) ASSUMES COMPONENT RADIATION HARDENING 10 x GALILEO
X SHIELDING THICKNESS PROHIBITIVELY LARGE

TABLE 6-8

SPACECRAFT MASS ELEMENTS* (GALILEAN SATELLITE MISSIONS)

ORBITER	780 KG + SHIELDING
ORBITER W/LARGE LANDER	845 KG + SHIELDING
PENETRATOR (1) W/RETRO	
FLYBY DEPLOYMENT	200 KG
ORBIT DEPLOYMENT	85-100 KG
HARD LANDER (1) W/RETRO	
FLYBY DEPLOYMENT	340 KG
ORBIT DEPLOYMENT	140-175 KG
LARGE LANDER	515 KG + SHIELDING
ORBITER PROPULSION (BALLISTIC)	EARTH STORABLE <u>OR</u> SPACE STORABLE
LARGE LANDER PROPULSION	SOLID/MONO
NEP SYSTEM (100. KW)	4365 KG + PROPELLANT

*INCLUDES SCIENCE AND 10% CONTINGENCY

Table 6-9

GALILEAN SATELLITE MISSION CAPABILITY MATRIX (MASS MARGINS IN KG)

DELIVERY MODE / MISSION MODE	DIRECT BALLISTIC					ΔVEGA BALLISTIC						LOW THRUST	
	IUS(III)/E-S	IUS(III)/S-S	CENT(F)/E-S	CENT(F)/S-S		IUS(III)/E-S	IUS(III)/S-S	CENT(F)/E-S	CENT(F)/S-S			NEP (SPIRALS)	
SATELLITE FLYBY TOUR													
PENETRATORS C→G→E		-	-	230			580	1520	2230				
HARD LANDERS C→G→E		-	-	-			-	800	1580				
SINGLE TARGET ORBITERS													
CALLISTO		-	555	1005		745	1355	2300	3000				
GANYMEDE		-	380	890		570	1240	2125	2890				
EUROPA		-	-	265		-	615	1340	2265				
IO		-	-	160		-	510	1160	2160				
ORBITER W/PENETRATORS (3)													
CALLISTO				-		-	340	950	1990				
GANYMEDE				-		-	65	500	1715				
EUROPA				-		-	-	300	1530			360	
ORBITER W/HARD LANDERS (3)													
CALLISTO							-	345	1530				
GANYMEDE							-	-	1155				
EUROPA							-	-	1105			135	
ORBITER W/LARGE LANDER													
CALLISTO									-			1020	
GANYMEDE									-			145	
EUROPA									-			-	
MULTIPLE TARGET ORBITER													
CALLISTO/GANYMEDE							-	-	715			1465	
GANYMEDE/EUROPA									-			635	
C/G/E									-			635	

NEP low-thrust delivery the margins apply directly to the orbiter (net spacecraft) mass. Dark shading is used to indicate mass margins in excess of 500 kg, light shading is used for positive margins less than 500 kg, and negative margins are indicated either by a negative sign or a blank area. Jupiter radiation shielding requirements have been accounted for in the margin calculations. Data for Callisto and Ganymede orbiters reflect 100% coverage orbit lifetimes and Galileo design electronics shielding. In the case of Europa and Io orbiters, the margin data reflects reduced stay-times (less than 100% coverage) and/or hardened electronics -- particularly for Io.

Several conclusions may be drawn from this analysis: (1) the IUS(III) capability is limited to the ΔVEGA flight mode and mission modes including only satellite flyby/penetrators and single target orbiters; space-storable propulsion may also be required in some cases. (2) Centaur(F) capability allows direct ballistic flights for the above mission modes but requires space-storable propulsion for all but the Callisto and Ganymede orbiters. (3) Centaur capability with ΔVEGA (or SEEGA) extends the mission capture range through orbiters with multiple small lander deployments and, with space-storables, the multiple Callisto/Ganymede orbiter mission can also be captured. (4) NEP capability appears to be most apropos for the most demanding missions, namely, multiple target orbiters and large lander deployment (Callisto or Ganymede only).

Based on identified performance capabilities and potential science return, five reference missions were developed as examples suitable for further consideration including cost estimation. Profiles of these reference missions are presented in Tables 6-10 through 6-14. Each profile consists of a mission description, pertinent trajectory data, and a summary of mass performance, including the margin at the beginning-of-mission. Penetrators are listed as complete systems in the mission profiles, i.e. including retro propellant and inerts. A launch vehicle adapter of 3.4% of the injected mass was assumed for the ballistic missions. NEP airborne support equipment (ASE) was assumed to be 3000 kg.

It will be noted that all satellite targets except Io are included in

Table 6-10

GALILEAN SATELLITES MISSION PROFILE

- **MISSION CONCEPT**.......... Satellite Flyby Tour with Penetrators (3)

- **DESCRIPTION**

Flight Mode...........................	Ballistic Direct
Target Approach Mode..................	Ganymede assisted JOI, Low V_∞ Tour
Launch Vehicle........................	Shuttle/Centaur(F) (C_3 = 77 km^2/sec^2)
Spacecraft Retropulsion...............	Earth Storables (ΔV = 1.943 km/sec)
No. of satellite encounters...........	26 (C-6, G-14, E-5, I-1)
No. of satellites orbited.............	0
No. of lander deployments.............	3 (one each to C, G, E)

- **TRAJECTORY**

Launch Date.............	7 Jan 1994	End of Mission.............	26 Dec 1997
Jupiter Approach Date...	4 Jul 1996	Earth-Jupiter Flight Time...	908 days
Initial Capture Orbit................	122 days		
Satellite Tour Time..................	422 days to Io		
Spiral Orbit Capture Time............	—		
Satellite Orbit Stay-Time............	—		

- **MASS PERFORMANCE (kg)**

GALILEO DESIGN ELECTRONICS

Orbiter including shielding..........	873
Orbiter retro inerts.................	242
Orbiter retro propellant............	1266
Penetrator System..........201 (x3) =	603
Launch Vehicle Adapter..............	101
Required Injected Mass.........	3085
Injected Mass Capability........... (Shuttle/Centaur(F)* @C_3 = 77 km^2/sec^2)	3350
	265 Margin

*Uprated Shuttle 70K

Table 6-11

GALILEAN SATELLITES MISSION PROFILE

- **MISSION CONCEPT**.......... Ganymede Orbiter

- **DESCRIPTION**

Flight Mode...........................	Ballistic Direct
Target Approach Mode..................	Ganymede assisted JOI, Low V_∞ Tour
Launch Vehicle........................	Shuttle/Centaur(F) (C_3 = 77 km^2/sec^2)
Spacecraft Retropulsion...............	Earth Storables (ΔV = 2.626 km/sec)
No. of satellite encounters...........	21 (C-8, G-13)
No. of satellites orbited.............	1
No. of lander deployments.............	0

- **TRAJECTORY**

Launch Date.............	7 Jan 1994	End of Mission.............	29 Apr 1998
Jupiter Approach Date...	4 Jul 1996	Earth-Jupiter Flight Time...	908 days
Initial Capture Orbit................	122 days		
Satellite Tour Time..................	445 days		
Spiral Orbit Capture Time............	—		
Satellite Orbit Stay-Time............	98 days		

- **MASS PERFORMANCE (kg)**

GALILEO DESIGN ELECTRONICS

Orbiter including shielding..........	813
Orbiter retro inerts.................	279
Orbiter retro propellant............	1532
Launch Vehicle Adapter..............	89
Required Injected Mass.........	2713
Injected Mass Capability........... (Shuttle/Centaur(F) @C_3 = 77 km^2/sec^2)	2880
	167 Margin

Table 6-12

GALILEAN SATELLITES MISSION PROFILE

- MISSION CONCEPT.......... Europa Orbiter with Penetrators (3)

- DESCRIPTION

Flight Mode........................... Ballistic ΔVEGA
Target Approach Mode.................. Ganymede assisted JOI, Low V_∞ Tour
Launch Vehicle........................ Shuttle/Centaur(F) (C_3 = 28 km^2/sec^2)
Spacecraft Retropulsion............... Earth Storables (ΔV = 3.572 km/sec)
No. of satellite encounters........... 21 (C-6, G-14, E-1)
No. of satellites orbited............. 1
No. of lander deployments............. 3 (one each to C, G, E)

- TRAJECTORY

Launch Date............ 9 Dec 1991 — End of Mission............. 3 Feb 1998
Jupiter Approach Date... 3 Sep 1996 — Earth-Jupiter Flight Time... 1730 days
Initial Capture Orbit............... 122 days
Satellite Tour Time................. 370 days
Spiral Orbit Capture Time........... —
Satellite Orbit Stay-Time........... 27 days

- MASS PERFORMANCE (kg)

HARDENED ELECTRONICS = 10 x GALILEO

Orbiter including shielding	858
Orbiter retro inerts	495
Orbiter retro propellant	3072
Penetrator System..........201 (x3) =	603
Launch Vehicle Adapter	173
Required Injected Mass	5200
Injected Mass Capability (Shuttle/Centaur(F) @C_3 = 28 km^2/sec^2)	6000
Margin	800

Table 6-13

GALILEAN SATELLITES MISSION PROFILE

- MISSION CONCEPT.......... Callisto/Ganymede Orbiter with Ganymede Large Lander

- DESCRIPTION

Flight Mode........................... NEP
Target Approach Mode.................. Low Thrust Spirals
Launch Vehicle........................ Shuttle (Escape Spiral from 700 km)
Spacecraft Retropulsion............... —
No. of satellite encounters........... 2
No. of satellites orbited............. 2
No. of lander deployments............. 1

- TRAJECTORY

Launch Date................ Any year — End of Mission............. —
Jupiter Approach Date...... — — Earth-Jupiter Flight Time... 420 +1200 = 1620 days
Satellite Tour Time.................. —
Spiral Orbit Capture Time............ 373 days (Jupiter approach→end of mission)
Satellite Orbit Stay-Time............ $83^C + 98^G$ = 181 days
Total Mission Duration............... 2174 days = 5.95 years

- MASS PERFORMANCE (kg)

GALILEO DESIGN ELECTRONICS

Orbiter including shielding	852
NEP Dry Mass	4367
NEP Propellant	11007
Lander including shielding	516
Lander retro inerts.......Solid 50, Mono 49	99
Lander retro propellant...Solid 570, Mono 92	662
ASE	3000
Required Shuttle Cargo Mass in 700 km orbit	20503
Shuttle Cargo Capability at 700 km	21000
Margin	497

TABLE 6-14

GALILEAN SATELLITES MISSION PROFILE

- MISSION CONCEPT .... Multiple Target Orbiter

- DESCRIPTION

Flight Mode	NEP
Target Approach Mode	Low Thrust Spirals
Launch Vehicle	Shuttle (Escape Sprial from 700 km)
Spacecraft Retropulsion	—
No. of satellite encounters	3
No. of satellites orbited	3
No. of lander deployments	0

- TRAJECTORY

Launch Date	Any year	End of Mission	—
Jupiter Approach Date	—	Earth-Jupiter Flight Time	378 + 1100 = 1478 days

Satellite Tour Time	—
Spiral Orbit Capture Time	449 days (Jupiter approach → end of mission)
Satellite Orbit Stay-Time	$83^{C} + 98^{G} + 42^{E}$ = 223 days
Total Mission Duration	2150 days = 5.89 years

- MASS PERFORMANCE (kg)

Orbiter including shielding	915
NEP Dry Mass	4367
NEP Propellant	10580
ASE	3000
Required Shuttle Cargo Mass in 700 km orbit	18862
Shuttle Cargo Capability at 700 km	21000
	2138 Margin

HARDENED ELECTRONICS = 10 x GALILEO

these examples. Notwithstanding the great scientific interest in Io, an orbiter mission there is extrememly difficult due to the radiation environment, and it is not clear that the added shielding mass indicated could even be implemented in any realistic design. One might consider Io an end-of-mission objective (without added shielding) on the satellite flyby tour mission. Another point to note is the penetrator deployment policy on the Europa orbiter mission. Here we have selected the option of deploying one penetrator each to Callisto and Ganymede on those satellite flybys enroute to Europa, with the last penetrator deployed on approach to Europa prior to orbit insertion.

Table 6-15 presents detailed cost estimates for the five reference missions. These estimates were generated by use of the SAI Planetary Program Cost Model with the following general guidelines and assumptions. All costs are presented in fiscal year 1982 constant collars. Reference missions are independent of each other; thus, there is no basis for costing common hardware buys among the five missions although cost savings from hardware design heritage are included. The flyby or orbiter spacecraft and the large lander are costed as one flight qualified unit; the penetrators are costed as three flight qualified units plus one flight qualified spare. Twenty percent contingency is applied to each estimate in the form of APA/Management Reserve.

Cost estimates for each of the five reference missions range from $260M to $672M. Coincidentally perhaps, the above limits correspond to NEP payload delivery for the multiple target orbiter and orbiter/large lander missions. The real project costs are understated in these cases since the Shuttle launch cost, the unit or recurring cost of the NEP system, and NEP operational cost are not included in these numbers. One might expect the add-on transportation cost to be of the order $200M to $250M. The cost range of the ballistic reference missions is $370M to $488M; the add-on transportation cost for the Shuttle/Centaur launch would be about $100M in FY'82 dollars.

TABLE 6-15

MISSION COST ESTIMATES
(FY 1982 Million Dollars)

	Satellite Flyby Tour with Penetrators	Ganymede Orbiter	Europa Orbiter with Penetrators	Callisto/Ganymede Orbiter with Ganymede Large Lander*	Multiple Target Orbiter*
Program Management	$ 19M	$ 14M	$ 19M	$ 32M	$ 11M
Spacecraft Science	42	42	45	42	45
Penetrator Science	26	—	26	—	—
Lander Science	—	—	—	106	—
Spacecraft System	151	147	156	116	105
Penetrator System	51	—	51	—	—
Lander System	—	—	—	163	—
RTG's	20	19	20	9	—
Launch + 30 Days Operations	18	14	18	32	11
Development Subtotal	327	236	335	500	172
Mission Operations	48	54	58	54	40
Data Analysis	15	18	14	6	5
Flight Project Subtotal	63	72	72	60	45
Total	390	308	407	560	217
APA/Reserve	78	62	81	112	43
Grand Total	$468M	$370M	$488M	$672M	$260M

* Exclusive of costs of NEP stage and NEP flight operations

2.7 Advanced Propulsion Data Base

Propulsion system performance, availability, and cost are among the key issues that will determine the direction and effectiveness of NASA's future solar system exploration program. Launch vehicle/upper stage injection systems as well as in-transit spacecraft propulsion are of concern in this regard. The objectives of this task were to appraise the long-term (25-year) propulsion demands, to analyze the characteristics and effectiveness of candidate advanced systems, and to provide a comparative performance assessment of these systems. The scope of this study evolved as part of the contractual effort in response to NASA Headquarters' direction; it was focused on particular aspects of the problem rather than being all-encompassing. Specifically, the major thrust of this work concentrated on two areas of current interest: (1) the implications of launching planetary missions from a space station base in low Earth orbit; and (2) the propulsion system requirements apropos to carrying out a fast trip time program of far outer planet orbiters with atmospheric probes. The following is a brief summary of the work accomplished in these two areas.

Performance Assessment of Space Station Launches

The principal role of a permanent manned space station would be the construction, maintenance, and launching of space systems. Since the station would be serviced by Shuttle (or advanced vehicle) launches from Earth, the selection of SOC orbit altitude and inclination would be constrained by considerations of performance accessibility and station keeping (against atmospheric drag). For purposes of this analysis, a circular orbit at 370 km (200 n. mile) altitude and 28.3° inclination seems a reasonable choice. Higher values of inclination are, of course, possible, but significant differences from the launch site latitude (Cape Canaveral assumed) could incur Shuttle cargo mass reduction.

Injection of a planetary spacecraft onto an Earth-escape hyperbolic asymptote is governed by the laws of orbital mechanics and proper timing with the objective of minimizing the mission velocity-change (ΔV) requirements. A specified injection asymptote within an allowable launch date window is achieved by proper selection of the orientation of the injection plane and the point of injection within this plane. For the usual operation of Earth launch to parking orbit and then injection, the proper conditions are achieved by daily launch timing, selection of parking orbit inclination, and short duration coast periods. Additional plane change adjustment made in combination with the injection burn (dog-leg maneuver) may be necessary to achieve asymptote declinations above the maximum permissable inclination of the parking orbit. Most planetary mission opportunities do not require this additional correction.

The situation of injection from a space station orbit is quite different since the station orbit plane will not generally have the proper nodal orientation at the desired launch date. This orbit plane does precess due to Earth oblateness perturbation, so the proper orientation will eventually occur on a periodic basis. At the nominal altitude and inclination, the nodal regression is 7.2°/day. Injection from the station orbit will generally incur some performance penalty due to two conditions: (1) incorrect position of orbit node on desired launch date, and (2) possible requirement of the injection declination being greater than the station orbit inclination. Of the two, the first is probably more important.

Figure 7-1 delineates the possible injection strategies. Plane changes in Earth orbit generally require large ΔV maneuvers unless the nodal misalignment is small (e.g. see Figure 7-2). It is usually much more efficient to adopt a passive "launch date timing" strategy as a baseline solution to the injection problem with plane changes reserved only for fine tuning purposes.

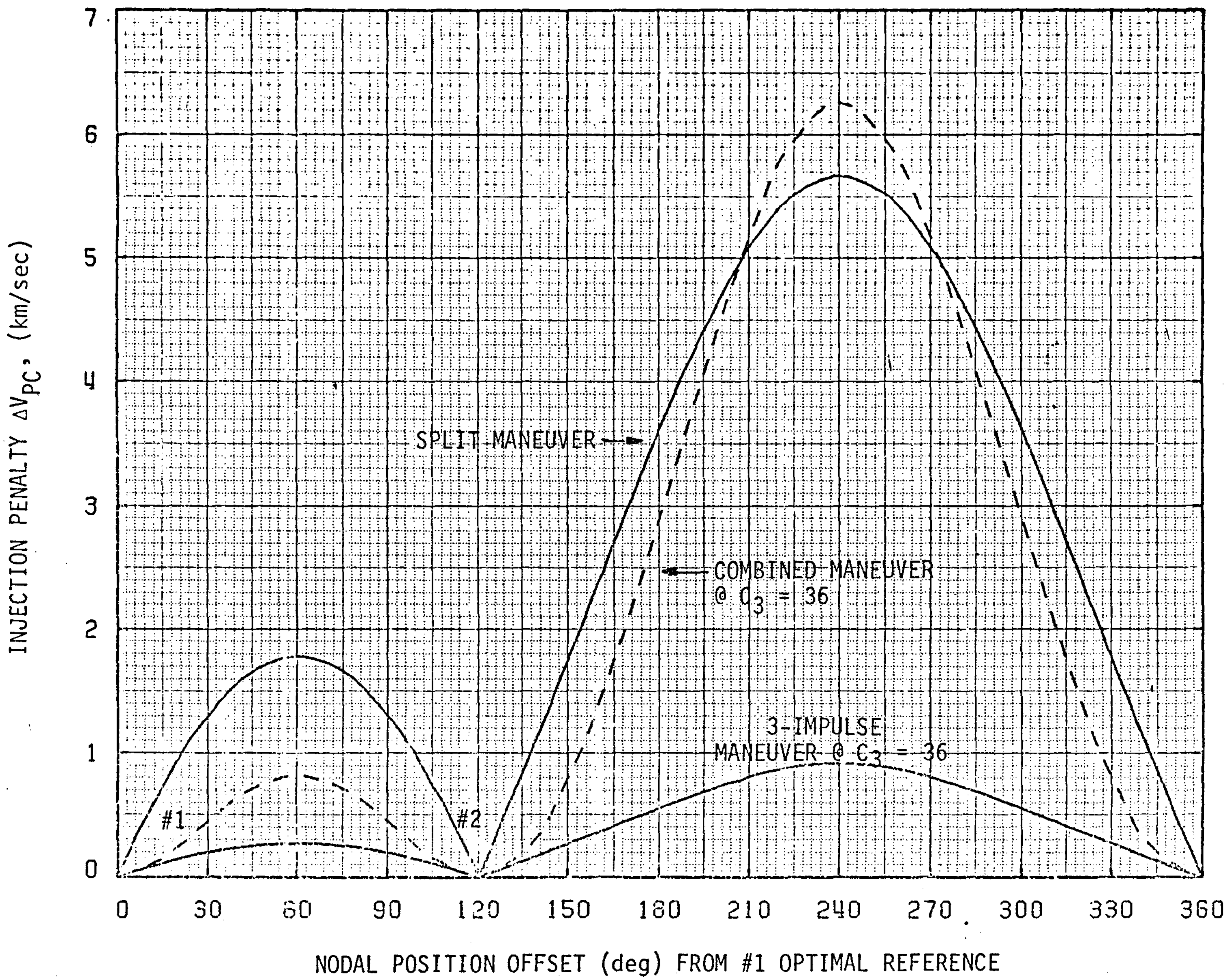
Figure 7-3 illustrates the launch timing strategy for the outbound leg of a Mars Sample Return mission. The required position of the injection node is double-valued when |DLA| is less then the station orbit inclination; otherwise it is single-valued. Station orbit precession is indicated

Fig. 7-1

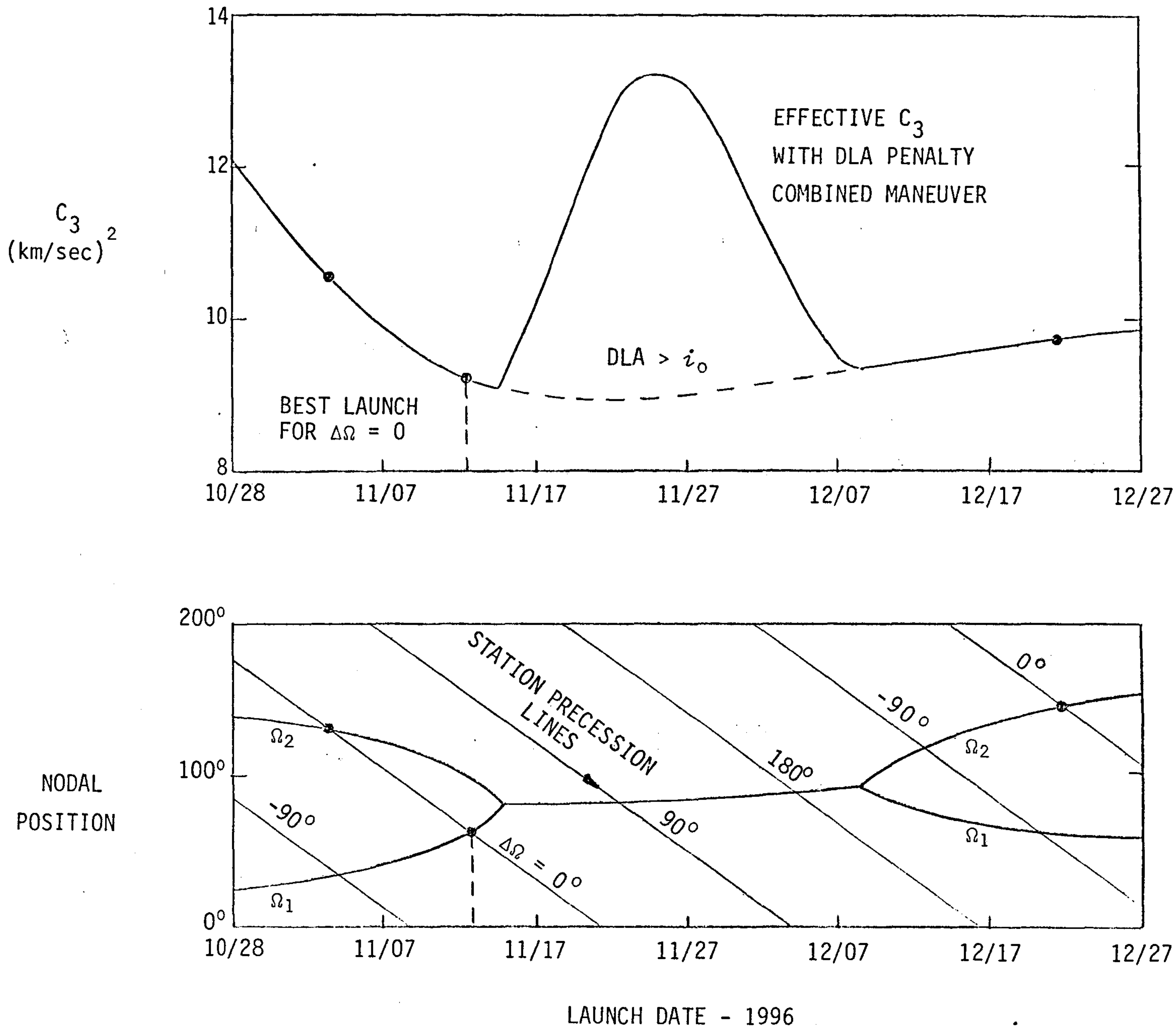
SPACE STATION/PLANETARY INJECTION STRATEGIES

	UTILIZATION PRIORITIES
• ACTIVE (PROPULSIVE PLANE CHANGES)	
o EARTH-ORBITAL	AS NEEDED FOR DLA TARGETING & LAUNCH DELAYS
– SPLIT MANEUVER	– STATION ORBIT REALIGNMENT: EXPENSIVE
– COMBINED MANEUVER	– NON-PLANAR ESCAPE: LESS EXPENSIVE
– THREE-IMPULSE MANEUVER	– APOAPSE PLANE CHANGE: LEAST EXPENSIVE (BUT REQUIRES 24^h INTERMEDIATE ORBIT)
o INTERPLANETARY	
– BROKEN PLANE TRANSFERS	VERY EFFECTIVE ON SOME MISSIONS IN REDUCING DLA PENALTIES AND IN IMPROVING OFF-OPTIMAL ESCAPE REQUIREMENTS FOR PASSIVE STRATEGY (SEE BELOW)
• PASSIVE (LAUNCH DATE TIMING)	BASELINE SOLUTION
– STATION ORBIT PRECESSION	WAITS FOR ORBIT REALIGNMENT, ACCEPTING SOME PERFORMANCE LOSS FROM RESULTING OFF-OPTIMAL LAUNCH DATE

Fig. 7-2



STATION ORBIT INJECTION TO |DLA| = 15° WITH CORRECTION OF NODAL POSITION OFFSET



SELECTION OF LAUNCH DATE VIA PRECESION - 1996 MARS SAMPLE RETURN

Fig. 7-3

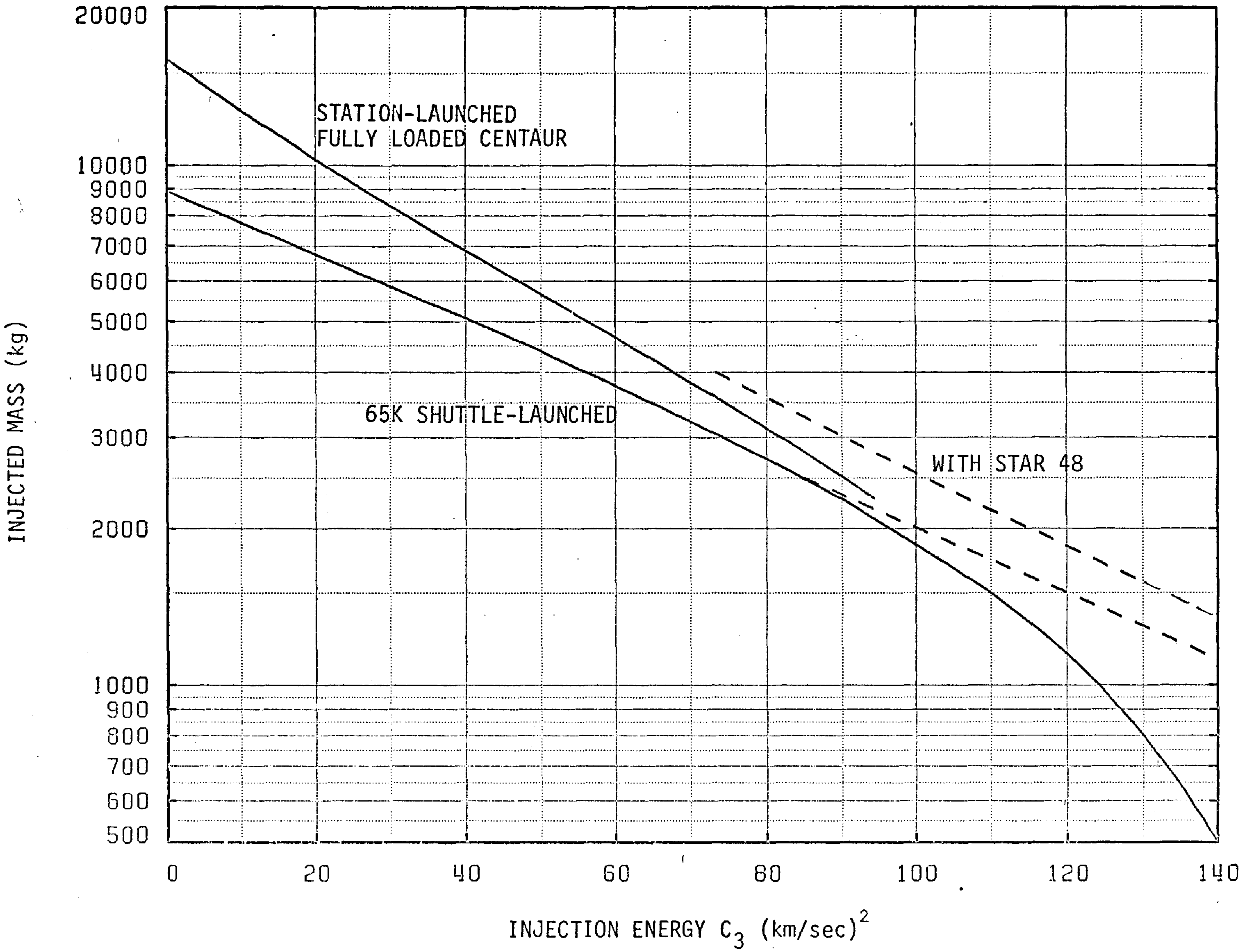
by the set of parallel lines for various nodal position offsets at some specific reference epoch; note that the designation $\Delta\Omega = 0^{\circ}$ is an arbitrary reference over the full 360° range of potential station nodes, but is usually chosen to represent near-optimum injection performance. The intersection of any given precession line with the required nodal position curve thus determines several possible launch dates over the extended launch window. In the example illustrated for $\Delta\Omega = 0$ there are three possible launch dates between 10/28/96 and 12/27/96; the best launch date selection is 11/12 on the basis of minimum injection energy C_3. The best launch dates for other values of $\Delta\Omega$ in the range 0° to 360° are chosen in a similar manner.

The performance penalty for station launches on off-optimal launch dates may be offset by the advantage of being able to fully load propellant in the upper stage unconstrained by Shuttle cargo weight limitations. This advantage is shown in Figure 7-4 for the Wide-Body Centaur; note that the performance gain is greatest at low values of injection energy which is the case for missions to Mars.

Results of the Shuttle vs. Station-launched performance for the 1996 Mars Sample Return mission are described in Tables 7-1 and 7-2 and Figures 7-5 and 7-6. Note that both Mars Orbit Rendezvous (MOR) and Direct Return (DR) flight modes are considered. Performance comparisons are given in two categories: (1) injected mass margin for nominal values of system mass elements, and (2) sample capsule mass capability for maximum injected mass and all other mass elements at nominal values. Assuming the Centaur upper stage, station launch performance is far better than Shuttle-launched missions, and, for the DR mode in particular, is mission enabling.

Other mission applications and upper stage selections can lead to different results. An example of a contrary result favoring Shuttle launches is illustrated by Figure 7-7. This mission is a rendezvous with the near-Earth asteroid Anteros launched in the unfavorable 1999 opportunity (high DLA) and utilizing the IUS(II)/Star 48 upper stage. In this

Fig. 7-4



WIDE BODY CENTAUR LAUNCH PERFORMANCE

Table 7-1

1996 MARS SAMPLE RETURN MISSION - TRAJECTORY PARAMETERS

	SHUTTLE-LAUNCHED*	STATION-LAUNCHED**
LAUNCH DATE	11/17/96 - 11/27/96	11/10/96 - 12/15/96
MARS ARRIVAL DATE	9/17/97 - 10/19/97	9/05/97 - 12/06/97
MARS DEPARTURE DATE	8/20/98	7/07/98 - 9/25/98
EARTH RETURN DATE	8/09/99	7/16/99 - 9/01/99
INJECTION ENERGY C_3, $(KM/SEC)^2$	8.9 - 9.0	9.1 - 13.2
INJECTION |DLA|, (DEG.)	29.8 - 31.8	24.9 - 31.8
MARS ARRIVAL V_{HP}, (KM/SEC)	2.9 - 3.3	2.9 - 4.6
MARS DEPARTURE V_{HP}, (KM/SEC)	2.6	2.4 - 3.1
EARTH RETURN V_{HP}, (KM/SEC)	4.0	3.8 - 4.9
EARTH RETURN DAA, (DEG.)	33.1	28.3
TOTAL MISSION ΔV - DR MODE, (KM/SEC)	14.0	13.8 - 14.8

* RANGE OVER 10^d SHUTTLE-LAUNCHED WINDOW

**RANGE OVER 360^o OF POSSIBLE STATION NODAL LOCATIONS

Table 7-2

1996 MARS SAMPLE RETURN MISSION - MASS DEFINITION

SYSTEM MASS ELEMENT	- - - NOMINAL VALUE (KG) - - - DIRECT RETURN MODE	MOR MODE
AEROCAPTURE/ENTRY	1500	1500
ORBITER	-	550
LANDER (W/ROVER)	650	650
ASCENT VEHICLE SUBSYSTEMS	-	95
EARTH RETURN VEHICLE	120	-
SAMPLE CAPSULE	30	30
SAMPLE	5	5
SUBTOTAL W/O PROPULSION	2305	2830
INJECTED MASS REQUIREMENT		
SHUTTLE-LAUNCHED	8320	5555
STATION-LAUNCHED*	8345 - 9085	5515 - 5910

*RANGE OVER ALL POSSIBLE NODAL POSITIONS OF SPACE STATION

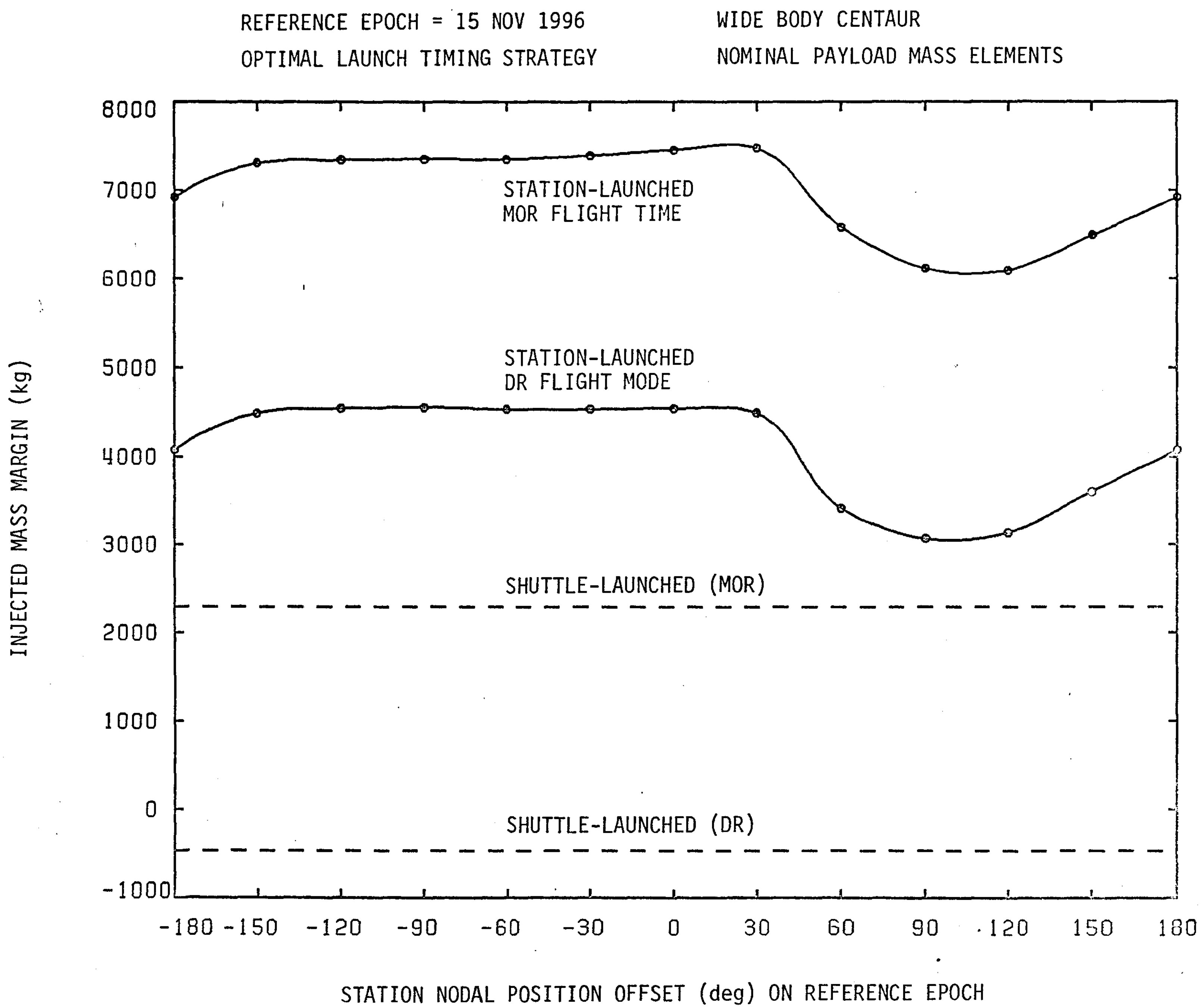


FIG. 7-5 INJECTED MASS MARGIN COMPARISON FOR 1996 MARS SAMPLE RETURN

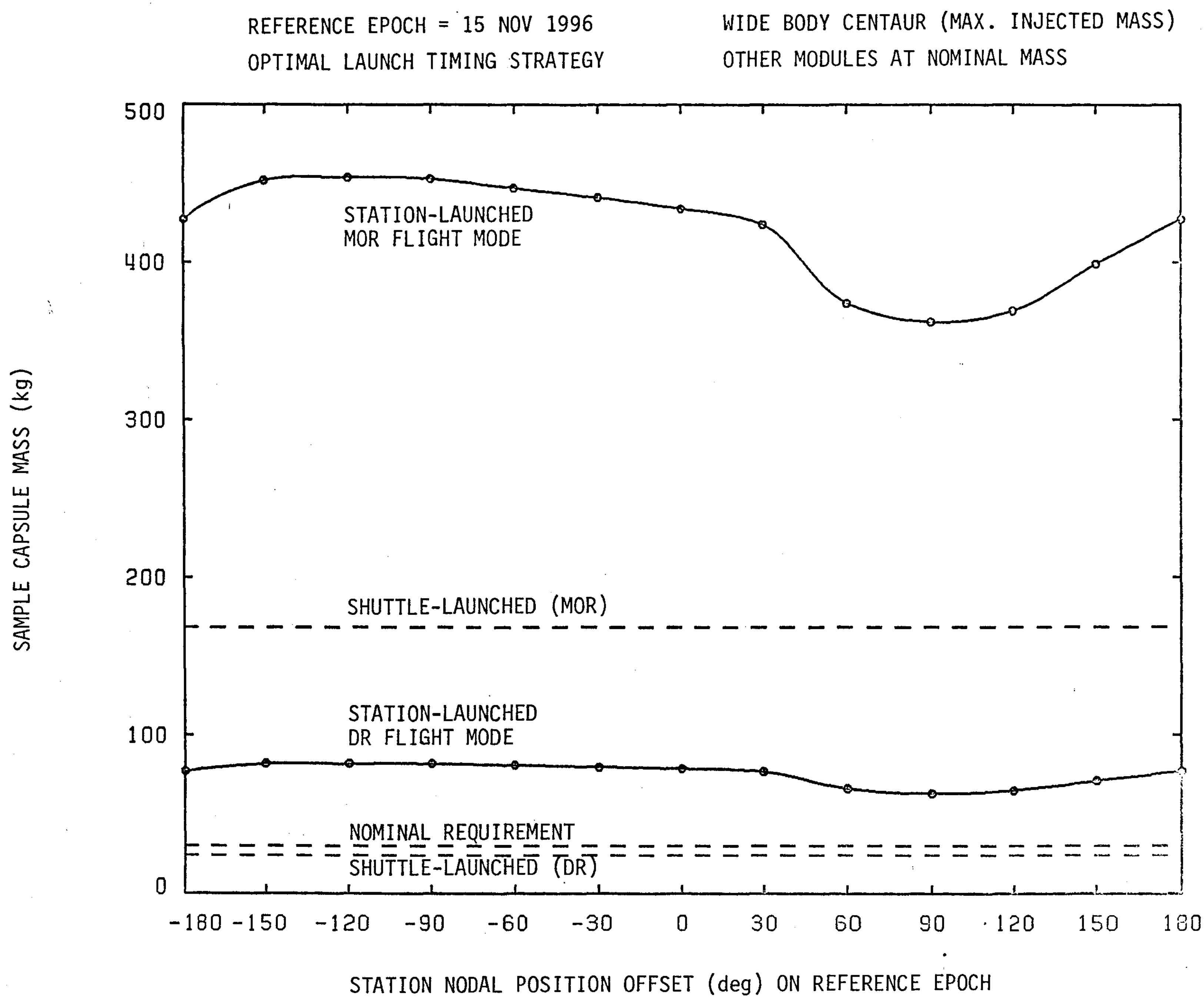
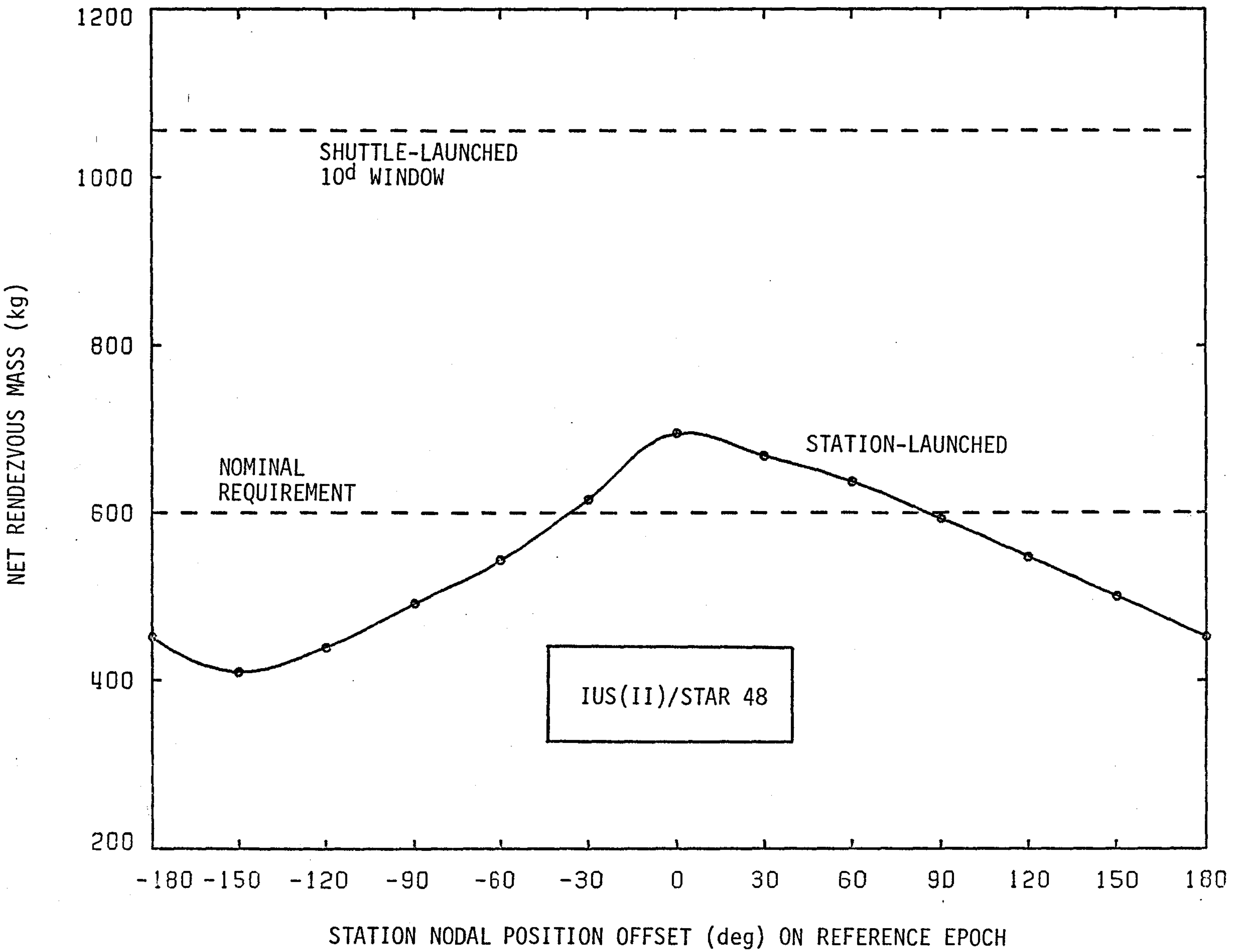


FIG. 7-6 SAMPLE CAPSULE MASS CAPABILITY COMPARISON FOR 1996 MARS SAMPLE RETURN

Fig. 7-7

REFERENCE EPOCH = 30 JUN 1999
OPTIMAL LAUNCH TIMING & BROKEN PLANE STRATEGY



PERFORMANCE COMPARISON FOR 1999 ANTEROS RENDEZVOUS case the Shuttle launch capability can deliver a net rendezvous payload of 1056 kg whereas the station launch performance mostly falls below the nominal requirement of 600 kg.

On the basis of these examples and other results obtained from the study one may draw several generalized conclusions. A fundamental tradeoff exists between Shuttle-launched and station-launched planetary missions which is manifested in terms of off-optimal launch date penalties vs. the potential advantages of fully loaded upper stages. Over a broad range of planetary missions, these tradeoffs tend to favor: (1) the Shuttle for small payload mission implemented with smaller upper stages, e.g. the IUS(II); or (2) the space station for larger payload missions implemented with larger upper stages, e.g. the Wide-Body Centaur or equivalent OTV.

Propulsion Options for Fast Trips to the Outer Planets

The desire here is to deliver orbiter and probe payloads to the far outer planets (Uranus, Neptune and Pluto) in trip times under 10 years, and preferably under 7 years. The capability for accomplishing this very difficult goal may be defined and measured in terms of various options regarding launch/upper stage injection, interplanetary flight mode, and type of propulsion or vehicle system used for the post-launch mission phases.

The first step in the capability analysis is to define the upper stage choices to be examined along with their relevant parameters. Injected mass vs. injection energy performance may then be calculated. These choices and parameters are presented in Table 7-3. The first three chemical stages are considered for use with an uprated 85K Shuttle on a standard launch to a 150 n.mi. parking orbit. The airborne support equipment (ASE) must be subtracted from the 85,000 lbm cargo load; this imposes an upper limit to the initial stack mass (stages plus payload) at the time of ignition and could require propellant offloading of a stage having a large maximum propellant capability -- particularly in the low C_3 energy range. Of the two general OTV's employed on standard

Table 7-3

CHEMICAL UPPER STAGE PARAMETERS

UPPER STAGE	THRUST LEVEL(lbf)	SPECIFIC IMPULSE(sec)	MAXIMUM PROPELLANT(lbm)	BURNOUT[(1)] WEIGHT(lbm)	ASE[(2)] (lbm)
OTVS	16,500	442.4	38,000	6000(6500)	5600
CENTAUR(WB)	32,800	448.4	46,000	6467(6920)	9184
OTVL	30,000	464.6	62,000[(3)]	8640(9150)	9000
OTVLL	30,000	464.6	128,000	- (17,000)	-
CRY01	10,000	460.0	14,000	2470	-
CRY02	15,000	464.6	27,300	4020	-

(1) Parenthetical value applies when used as 1st stage

(2) ASE applies only to Shuttle(85K) launches; may require propellant offloading

(3) For OTVL(R) injection to 24-hour orbit, 7750 lbm reserved for return ΔV

launches, the first is herein designated OTVS (18 ft length) with the suffix "S" meaning small or short. This stage is based on a recent conceptual design by NASA-MSFC and proposed as a possible alternative to the Wide-Body Centaur development. The second and larger OTVL (35 ft length) is taken from a Boeing study of systems for the disposal of nuclear wastes in space. This stage used in a reusable mode and designated OTVL(R) is also considered. The Wide-Body Centaur (29 ft length) has previously been defined by NASA-LeRC.

The very large OTVLL stage (58 ft length) which is also based upon the Boeing study is considered only for the on-orbit launch option.* Two kick stages designated CRYO1 and CRYO2 are considered for both launch options; the first of these was conceptualized by SAI while the second was again based on the Boeing study. Assembled on-orbit combination of upper stages considered for the sample problem are OTVS/OTVS, OTVL(R)/OTVS, OTVL/CRYO2 and OTVLL/CRYO2.

Injected mass performance of Shuttle-launched and on orbit-launched chemical stages is shown in Figures 7-8 and 7-9. These data were generated using the defined parameters by SAI's STAGE program which accounts for finite thrust gravity losses. Also indicated on these performance graphs are the injected mass requirements of several ballistic flight mode options which are discussed next.

<u>ΔVEGA and SEEGA</u>. Consider first the indirect flight mode options, ΔVEGA and SEEGA, and the Uranus mission which is by far the least difficult. Direct ΔVEGA flights to Uranus require a 3+-year Earth-to-Earth transfer leaving less than 4 years for the Earth-to-Uranus leg. At such fast flight times the Uranus approach speed exceeds 20 km/sec making propulsive orbit insertion impossible for all practical purposes. Direct SEEGA flights offer slight relief because the first transfer leg is 2+ years. The 7 year flight approach speed reduces to about 16 km/sec but this is still much too high for propulsive orbit insertion considering the net approach mass delivery capability of the reference 31.6 kw

* On-orbit launch may be synonomous with operations from a space station base in low Earth orbit.

800 KG ORBITER, 300 KG PROBE

● ALL PROPULSION, 2-STAGE SPACE STORABLE

▲ AEROCAPTURE/EARTH-STORABLE, $T_F = 7^y$

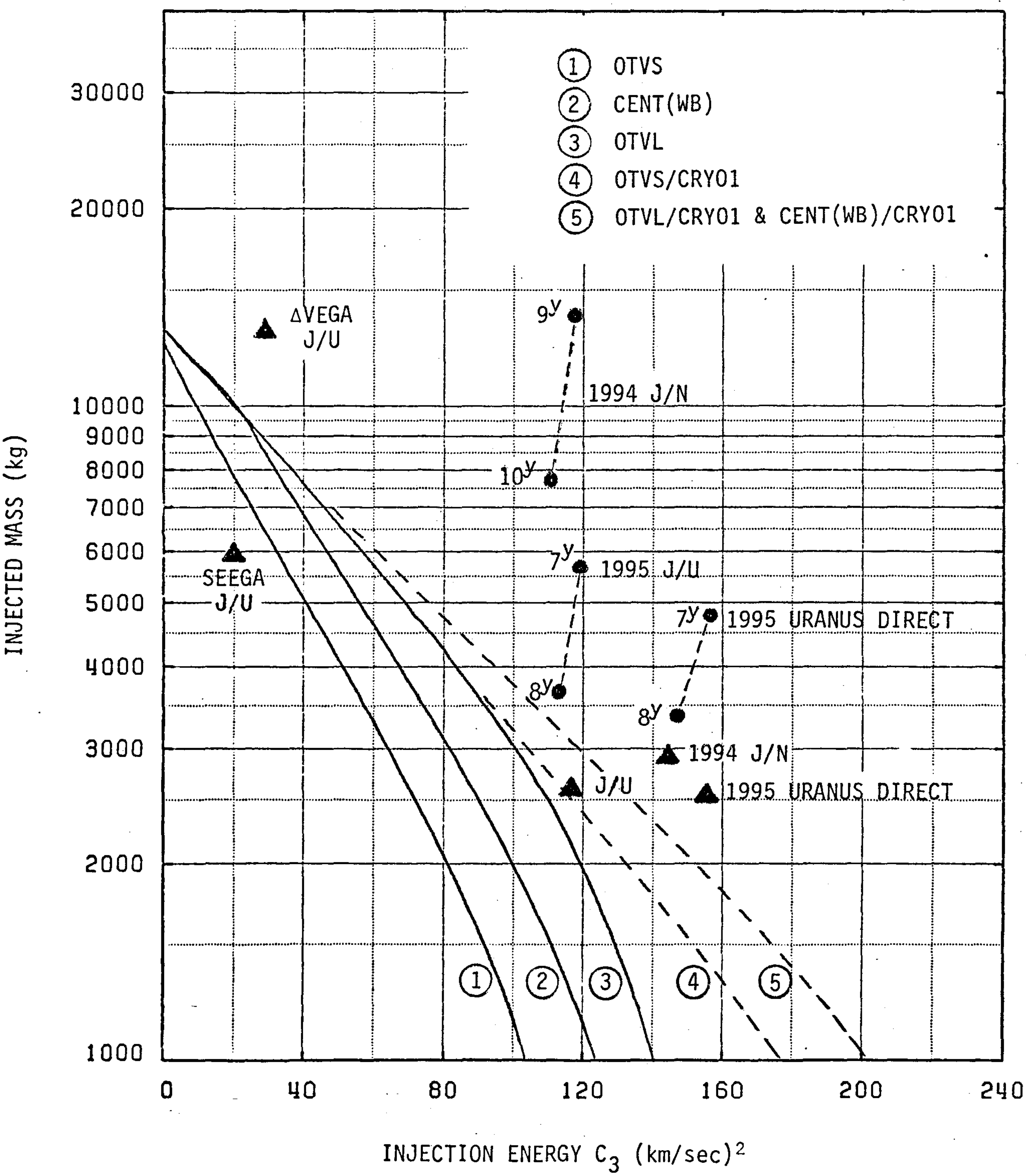


FIG. 7-8 UPPER STAGE PERFORMANCE FOR SHUTTLE (85K) LAUNCH - 150 NM

800 KG ORBITER, 300 KG PROBE

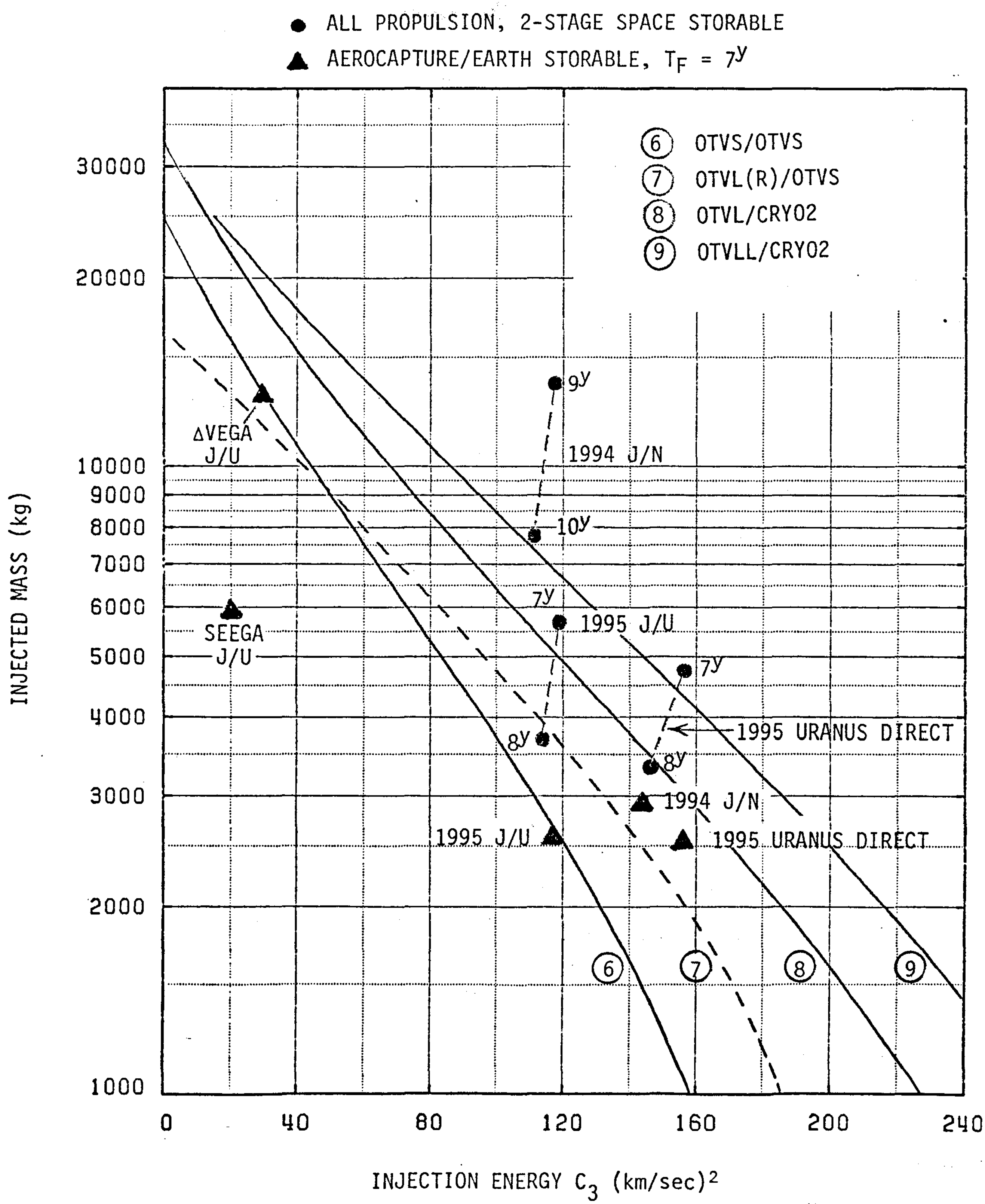


FIG. 7-9 UPPER STAGE PERFORMANCE FOR ON-ORBIT (SOC) LAUNCH - 200 NM

SEPS (less than 3500 kg) injected to Earth escape by the Shuttle (85K)/ Centaur (WB); a point of reference is the Shuttle (65K)/Centaur (WB) for which the Uranus net approach mass is about 2000 kg.

Jupiter swingby (gravity-assist) opportunities to Uranus and Neptune exist during the early to mid-1990's. The best launch year for fast Uranus orbiter missions is 1995 because the Uranus approach speed is lower. For both ΔVEGA and SEEGA flight modes the actual launch would occur in 1993 because of the 2+-year Earth-to-Earth transfer requirement. A Jupiter swingby appended to ΔVEGA provides slight but insufficient relief for the 7-year mission to Uranus employing all-propulsive maneuvers. The Uranus approach speed is 17.9 km/sec and the total post-injection ΔV requirement for the spacecraft is 13.9 km/sec. Delivery of a 300 kg probe and 800 kg orbiter employing a 4-stage space-storable retropropulsion system (I_{sp} = 370 sec) would require an injected mass of about 140,000 kg! -- well off the scale of the upper stage capability diagram. The SEEGA/ Jupiter swingby flight is also not a viable option.

Aerocapture at Uranus has been studied in sufficient detail to conclude that it is basically feasible, at least from an aerodynamic viewpoint, at entry speeds up to 30 km/sec; Neptune aerocapture should likewise be feasible. The scaling relationship used for estimating the mass of an aerocapture vehicle of 5 meters length is:

$$M_{ac} = 39.27\, V_e + 86.4 \text{ kg}$$

where V_e is the entry speed in km/sec. This mass includes the aeroshell structure, thermal protection system (carbon-phenolic), and reaction control system. Using this relationship we have calculated the total injected mass requirements for various flight mode options. The post-aerocapture orbit assumed is approximately 1.0 planet radius periapse to 60^d orbit period, with a 550 m/sec ΔV budget allowed for the orbiter to implement probe deployment (out-of-orbit) and other plane change/ orbit adjustment maneuvers.

In the case of the ΔVEGA/Jupiter swingby/aerocapture option, the 7-year flight requires a post-injection ΔV requirement during transit of 3.63 km/sec for ΔVEGA midcourse and powered Earth swingby maneuvers. This requirement is met by a separate retropropulsion system attached to the aerocapture vehicle and jettisoned well before Uranus encounter. The injected mass requirement, assuming a 2-stage Earth-storable system (I_{sp} = 295 sec), is about 13,300 kg at a C_3 = 28. This point is shown on Figures 7-8 and 7-9 to indicate upper stage capture capability. A 2-stage space-storable retro would reduce the injected mass requirement to about 9950 kg. Note that this mission would be captured only by the on-orbit launch options. The same type of analysis indicated that the SEEGA/Jupiter swingby/aerocapture mission to Uranus in 7 years or less could be captured by the Shuttle-launched injection stages. This capability is enabled by the fact that the large ΔV requirement on the Earth-to-Earth leg is supplied by the more efficient low-thrust propulsion system.

Aerocapture combined with Jupiter swingby was also examined for the Neptune Orbiter/Probe mission with launch in 1992 (1994 Earth return). The ΔVEGA impulse requirements for a 7-year flight are so large as to make this flight mode virtually impossible even with on-orbit injection stages. With SEP-SEEGA, it is again indicated that sufficient mass could be delivered to Neptune in 7 years flight time; however, the swingby distance at Jupiter is very close to the surface (less than $3R_J$). Also, Neptune entry speeds about 35 km/sec raise important aerocapture technology questions -- further verification analysis would be needed for this option. In the case of a Pluto orbiter, none of the techniques discussed so far would be enabling since Pluto's tentative atmosphere would probably not admit the aerocapture technique, and the high approach speeds taken together with its small gravitation field would result in prohibitively massive chemical propulsion systems.

Direct Flight. Injected mass requirements for direct ballistic flights to Uranus and Neptune are indicated in Figures 7-8 and 7-9. As before, an 800 kg orbiter and 300 kg probe mass are assumed for purposes of this performance trade analysis. The all-propulsion mode assumes a 2-stage space-storable retro for orbit capture. Neither Uranus nor Neptune missions of 7 years flight time duration are captured, even with on-orbit launches, unless a Jupiter swingby is allowed. With Jupiter swingby, only the Uranus mission is captured, and then only for the largest of the on-orbit injection stages.

Aerocapture improves this situation significantly. The 7-year flight to Uranus is possible with the Shuttle(85)/Centaur(WB)/CRY01 if Jupiter swingby is allowed; direct flights without Jupiter swingby require the increased capability of on-orbit assembly. Capture of the Neptune mission requires both Jupiter swingby and on-orbit assembly. An important observation to note is that aerocapture allows relatively low values of injected mass.

NEP Flight. The great potential of NEP application to outer planet missions lies in the fact that the nuclear reactor power source operates independently of distance from the sun. This characteristic of useful thrust acceleration at a large distance allows the vehicle system to slow down near the target planet ($V_{HP} \simeq 0$) and achieve orbit capture with relatively small propellant expenditure. NEP is the only mass delivery system considered here that will accomplish the Pluto orbiter mission.

Another feature of NEP is that it may be employed during planetocentric operations, i.e., to spiral out from Earth orbit to escape conditions and to spiral in to planet capture orbit without any intervening phase of chemical propulsion. The nominal Earth escape spiral begins from a 700 km orbit (nuclear safe altitude) and terminates when $C_3 = 0$ energy conditions are attained. Typical initial acceleration values are 2 to 3 × 10^{-4} m/sec^2 with corresponding Earth escape spiral times of 250 to 380 days and a mass fraction of about 0.878. Another possible escape mode is initial departure orbit at 12.2 Earth radii for the option of injecting the NEP/payload system to an intermediate 24-hour orbit above the Van Allen radiation belt. In this case a reuseable OTV could perform the injection maneuver from low orbit and a kick stage could be used to circularize the orbit at the higher altitude.

Planet capture spiral calculations assume initial capture orbits having semi-major axes of 59 Uranus radii, 75 Neptune radii and 3 Pluto radii. Typical initial acceleration values at planet approach are 4 to 6 $\times 10^{-4}$ m/sec^2 with corresponding spiral times less than 30 days and mass fraction above 0.98. The NEP could continue to spiral down to lower energy orbits. For example, at Uranus with $a_o = 5 \times 10^{-4}$, an additional 30 days of thrusting would reduce the orbit from 59 R_U to 20 R_U with an additional 2.5% of propellant expenditure.

The analysis* showed that the nominal escape spiral mode of operation will not capture any of the three missions with a 7-year maximum flight time constraint. However, if the NEP system is injected beyond Earth escape conditions by the OTVL/CRYO2 upper stage (on-orbit launch), then Uranus orbit is achieved in about 5 years and Neptune and Pluto orbits just barely at the 7-year limit. An intermediate case of escape spiral from a 24-hour Earth orbit was also examined; this option barely captured the Uranus mission but left Neptune and Pluto unattainable. Jupiter swingby was examined briefly for the Neptune mission but offered no apparent relief.

The 7-year trip time missions can be captured by eliminating the Earth escape spiral phase and, instead, injecting to escape ($C_3 > 0$) using high-energy chemical stages assembled and/or fueled on-orbit. Table 7-4 summarizes the launch/upper stage capture of the far outer planet missions assuming the reference 100 kw NEP system and mission

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* NEP assumptions are a 100 kw system operating at 5500 sec specific impulse and weighing 4367 kg dry. Mission payload modules are 1250 kg (Uranus and Neptune) and 950 kg (Pluto).

UPPER STAGE MISSION(1) CAPTURE MATRIX -- REFERENCE NEP(2) INJECTED TO $C_3 \geq 0$

DATA ENTRIES ARE INJECTED MASS (kg) AND C_3 $(km/sec)^2$

MISSION	SHUTTLE(85K)	ON-ORBIT LAUNCH			
	OTVL OR CENTAUR(WB)	OTVL(R)/OTVS	OTVS/OTVS	OTVL/CRYO2	OTVLL/CRYO2
URANUS $T_F = 5.53^y$			11,050 39	9,800 69	9,150 93
URANUS $T_F = 6.08^y$		9,550 46	9,550 46	8,800 76	8,400 100
URANUS $T_F = 6.63^y$	10,200 18	8,800 53	8,850 51	8,250 81	7,950 105
NEPTUNE $T_F = 7.18^y$				12,300 54	11,300 76
PLUTO $T_F = 7.14^y$				11,200 60	10,400 83

(1) MISSION PAYLOAD MODULES ARE 1250 kg (URANUS & NEPTUNE), 950 kg(PLUTO)

(2) P_o = 100 kw, I_{sp} = 5500 sec, M_{ps} = 4367 kg, 5% LV ADAPTER

Table 7-4

payload modules. Although the Uranus mission is captured by the Shuttle(85K)/Centaur(WB), the "smallest" common capability for all three missions is the OTVL/CRYO2 stage launched on-orbit.

In summary then, two enabling technologies (post-launch) emerge from this analysis: aerocapture and NEP. Aerocapture can be employed only at Uranus and Neptune with Pluto still requiring NEP. Aerocapture needs to be combined with a Jupiter swingby which occurs only at specific launch opportunities in the 1990's. Two potential options exist with aerocapture: SEP delivery using SEEGA trajectories and probably Shuttle-based launches, or ballistic trajectories definitely requiring on-orbit assembly. NEP delivery also requires on-orbit launches with upper stage injection to escape.

3. REPORTS AND PUBLICATIONS

Science Applications, Inc. is required, as part of its Advanced Studies contract with the Earth and Planetary Exploration Division, to document the results of its analyses. This documentation traditionally has been in one of two forms. First, reports are prepared for each scheduled contract task. Second, publications are prepared by individual staff members on subjects within the contract tasks which are considered of general interest to the aerospace community. A bibliography of the reports and publications completed (or in preparation) concerning work performed during the contract period 1 February 1981 through 1 May 1982 is presented below. These documents are available to interested readers upon request.

3.1 Task Reports for NASA Contract NASW-3035

1. "Advanced Planning Activities, February 1980 - January 1981", Report No. SAI-1-120-526-M16, April 1981.
2. "Cost Estimation Model for Advanced Planetary Programs - Fourth Edition", Report No. SAI-1-120-768-C9, November 1981.
3. "Advanced Planetary Studies Ninth Annual Report and Five-Year Contract Summary", Report No. SAI-1-120-768-A9, August 1982.
4. "Advanced Planning Activities, February 1981 - January 1982", Report No. SAI-1-120-768-M18 (in preparation).

3.2 Related Publications

1. "Mars Missions of Opportunity", The Case for Mars Conference, University of Colorado, Boulder, Colorado, May 1981.
2. "Scientific Activities on the Martian Surface", The Case for Mars Conference, University of Colorado, Boulder, Colorado, May 1981.
3. "Should Man on Mars be the Next Major Goal of the Space Program", The Case for Mars Conference, University of Colorado, Boulder, Colorado, May 1981.
4. "Mission Concepts for Venus Surface Investigation", Paper No. 81-184, AAS/AIAA Astrodynamics Specialist Conference, Lake Tahoe, Nevada, August 1981.
5. "Near Earth Asteroids: A Survey of Ballistic Rendezvous and Sample Return Missions", Paper No. 81-185, AAS/AIAA Astrodynamics Specialist Conference, Lake Tahoe, Nevada, August 1981.
6. "A Systematic Method of Generating Galilean Satellite-to-Satellite Transfers for Orbiter/Lander Missions", Paper No. 81-118, AAS/AIAA Astrodynamics Specialist Conference, Lake Tahoe, Nevada, August 1981.
7. "Galilean Satellite Mission Concepts", Paper No. AIAA-82-1460, AIAA/AAS Astrodynamics Conference, August 1982.