SCIENCE APPLICATIONS, INC.

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ADVANCED PLANETARY STUDIES
FIFTH ANNUAL REPORT

by

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for

Planetary/Lunar Programs
Office of Space Science
NASA Headquarters
Washington, DC 20546

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FOREWORD

This report summarizes the results of advanced studies and planning support performed by Science Applications, Inc. (SAI) under Contract No. NASW-3035 for the Planetary/Lunar Programs Office, Code SL, of NASA Headquarters during the twelve-month period 1 February 1977 through 31 January 1978. A total effort of 10,098 man-hours (62 man-months) was expended on seven specific study tasks and one general support task. The total contract value was $293,045, with 87% of the work performed by the staff of the SAI/Chicago office. Inquiries regarding further information on the contract results reported herein should be directed to the study leader, Mr. John Niehoff, at 312/885-6800.

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1. INTRODUCTION

Science Applications, Inc. (SAI) participates in a program of advanced concepts studies and planning analysis for Planetary/Lunar Programs Office, Code SL, NASA Headquarters. SAI's charter is to perform preliminary analyses and assessments for Code SL planning activities. Specifically, the objective of this support is to ensure NASA with 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 sixteen SAI staff members contributed to this effort.

The purpose of this Annual Report is to summarize the significant results generated under this Advanced Studies contract during the first year, 1 February 1977 through 31 January 1978, of a two-year contract. Progress reports on the task efforts are given at scheduled quarterly 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. The intention is to direct previously uninformed, but interested readers to detailed documentation and to serve as a future reference to completed advanced studies.

Each of eight contract tasks are presented in the next section. A brief description is given of the analyses performed along with key results and conclusions. 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 24-month program of advanced studies for the Planetary/Lunar Programs Office. A schedule of eleven tasks is planned for this period; several of the tasks described here are still in progress and are to be completed in the coming year.

2. TASK SUMMARIES

A total of eleven study tasks is planned for the 24-month contract period, 1 February 1977 to 31 January 1979; these tasks are:

1. Advanced Planning Activity
2. Cost Estimation Research
3. Planetary Missions Performance Handbooks--Revisions
4. Multiple Discipline Science Assessment
5. Asteroid Workshop
6. Galilean Lander Mission Strategies
7. Asteroid Exploration Study
8. Ion Drive Transport Capabilities
9. Mars Strategy Study
10. Venus Surface Sample Return
11. Ion Drive/Solar Sail Assessment Study.

This section contains summaries of work done on eight of these tasks during the first 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 Planetary/Lunar Programs Office. The remaining seven tasks are planned efforts with specific objectives of analysis. Reported efforts on Tasks 3, 4, 10 and 11 are complete. Tasks 2, 7 and 9 are studies in progress. During the second contract year work is scheduled on all tasks except 4, 10 and 11.

A total of 10,098 man-hours (62 man-months) was expended during the first contract year on the scheduled tasks. A summary description and discussion of key results for each task are presented in the subsections which follow. The level of effort devoted to each task is given with the task title at the beginning of each subsection. Specific reports generated for each task as part of the contract are noted in the list of publications to be found in Section 3 of this report.

2.1 Advanced Planning Activity (1837 man-hours)

The purpose of this task is to provide technical assistance to the Planetary/Lunar Programs Office 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 have performed for Code SL 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 27 reportable advanced planning subtasks, performed during the first 12 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. A brief summary of each of the subtasks is presented in the subsections which follow.

In addition to these subtasks, a major task effort was undertaken with budgeted Advanced Planning Activity resources in support of Ion Drive/SAIL low-thrust propulsion system assessments performed by NASA Headquarters during the Summer of 1977. This support effort was performed at the direction of the Advanced Programs and Technology Manager of the Planetary/Lunar Programs Office. The objective of this analysis was a qualitative assessment of Ion Drive/SAIL discriminators. Results of the analysis are reported separately below in Section 2.8.

2.1.1 Performance Comparison of Mars '84 Mission Options

The purpose of this subtask was to investigate the tradeoff between level of mission definition and operational capability at Mars. Two payload levels of interest were defined: (1) orbiter and surface rover, and (2) orbiter, surface rover, and penetrators. Delivery of these

Table 1

SUMMARY OF 1977-78 ADVANCED PLANNING ACTIVITY

Subtask	Month	Subject Title	Submitted To
1	Feb 1977	Performance Comparison of Mars '84 Mission Options	Code SL/NASA
2	Mar 1977	Mission Performance Workbook: Mars Mission Options	1977 MSWG
3	Apr 1977	Planetary Five-Year Plan Planning Support	Code SL/NASA
4	May 1977	Comparison of 1981/2 and 1983 JOP Missions	Code SL/NASA
5	Jun 1977	Planetary Five-Year Plan with Delayed JOP Project Start	Code SL/NASA
6	Jun 1977	Mission Requirements for Asteroid 1977HB	Helin/CalTech
7	Jul 1977	JOP Opportunity Dependent Performance Assessment	Code SL/NASA
8	Jul 1977	Planetary Advanced Studies Workshop	Code SL/NASA
9	Aug 1977	Uranus Mission Opportunities for the Voyager PTM	Code SL/NASA
10	Aug 1977	Near-Earth Resources Workshop	Code SL/NASA
11	Aug 1977	IUS On-Orbit-Assembly Planetary Mission Capability	Code SL/NASA
12	Sep 1977	Performance Assessment of Ion Drive Encke-X Missions	Code SL/NASA
13	Oct 1977	Future Planetary Applications of Solid Propulsion	Code SL/NASA
14	Nov 1977	Low-Cost Mars Sample Return Missions Performance Analysis	Code SL/NASA

Table 1 (continued)

Subtask	Month	Subject Title	Submitted to
15	Nov 1977	Planetary Mission Performance: Ion Drive versus Centaur	Code SL/NASA
16	Nov 1977	Combined JOP/HCR Mission Performance Assessment	Code SL/NASA
17	Nov 1977	Advanced Studies Administrator Presentation	Code SL/NASA
18	Nov 1977	Future Planetary Mission Launch/Arrival Dates	Code SL/NASA
19	Nov 1977	Mars Steering Group Participation	Friedman/JPL
20	Nov 1977	Definition of the 1988 Encke Sample Return Mission	Code SL/NASA
21	Nov 1977	Review of Proposed Saturn Workshop Objectives	Code SL/NASA
22	Nov 1977	Planetary Five-Year Plan: Revised Waterfall Charts	Code SL/NASA
23	Dec 1977	Performance Comparison of Ballistic/SEP/Ion Drive Systems	Code SL/NASA
24	Jan 1978	Revised FY'77 Planetary Five-Year Planning Exercise	Code SL/NASA
25	Jan 1978	Equivalence Comparison of Ballistic and Low-Thrust Flight	Code SL/NASA
26	Jan 1978	Evaluation of IUS On-Orbit-Assembly Study Proposal	Code SL/NASA
27	Jan 1978	Preparation of Lunar/Planetary Programs Historic Waterfall Charts	Code SL/NASA

payloads was considered with either Earth-storable or space-storable propulsion, launch within either 30-day or 1-day windows. Encounter operational capability was measured in terms of final orbit size, plane change capability, maximum landing site latitude range, and residual mass margin. A total of eight combinations of payload, retropropulsion and launch window were examined. Results, presented in tabular summary form, showed final orbit sizes ranging from 5 to 10 revolutions per day. Maximum plane change capability varied from 0° to 61°. Northern latitude range varied from 0° to 30°, and residual mass margins as large as 236 kg were reported.

2.1.2 Mission Performance Workbook: Mars Mission Options

The objective of this subtask was to develop a workbook which would assist the members of the Mars Science Working Group in understanding the spectrum of options for Mars missions which were plausible within existing performance constraints. Such a workbook was constructed. The information included in it permits designs of missions comprising, individually or in combination, orbiters, impact-landers and soft-landers. Orbital mechanics data emphasize the 1984 launch opportunity (Type II transfer) but analysis of less specific mission designs in alternative opportunities is also permitted using included supporting data. Sufficient working data were provided for the 1984 opportunity to investigate mission design issues including launch window, Earth-Mars transfer, capture orbit requirements, lander site selection, plane change requirements, final orbit size selection, payload performance and propulsion system tradeoffs.

2.1.3 Planetary Five-Year Plan Planning Support

This subtask comprised a 2-month support activity related to NASA's annual Five-Year Plan planning exercise. The purpose of the exercise is to synthesize the many planning activities continually in progress at NASA into a realistic near-term plan which is consistent with anticipated funding and can serve as a useful guide for future studies. The scope of this subtask was limited to development of the Planetary/Lunar Programs' portion of the plan. Support analyses included project manpower and cost estimates, estimate revisions to accommodate both inheritance and mission scope factors, and mission integration into the Five-Year Plan. Cost estimates were worked in both fixed and real-year dollars. Programmatic results were generated and presented in waterfall chart formats to be compared against anticipated funding guidelines (planning wedges). Numerous iterations of plan project start dates required repetitive computation of project cost spreads for resource planning. Both tabular and graphical (waterfall) resource requirements were generated for the final plan options.

2.1.4 Comparison of 1981/2 and 1983 JOP Missions

This subtask provided a brief one-page comparative summary of the 1981/2 and 1983 Jupiter launch opportunities for an orbiter/probe mission. Each opportunity was summarized by the following parameter set: launch date, launch window, type transfer, ΔV budget, probe entry location, amount of pre-entry science, probe mass, orbiter and science mass, launch mass margin, and next fall-back launch option. This comparison highlights the superiority of the 1981/2 opportunity for Jupiter missions. Comments regarding programmatic impact of delaying the JOP mission until the 1983 opportunity are also included.

2.1.5 Five-Year Plan with Delayed JOP Project Start

The purpose of this task was to assess the impact of a delayed JOP project (beginning in either FY'79, FY'81 or FY'82) on the current Five-Year Plan. Fifteen different revised plans were developed and assessed in terms of revised annual funding requirements. Specific cost advantage (block buys and direct hardware inheritance) was applied where revised mission sequences provided new opportunities for common hardware development. Both tabular and graphical (waterfall) data were prepared for each of the 15 plans, along with a summary of assumptions, basic mission cost estimates and plan definitions.

2.1.6 Mission Requirements for Asteroid 1977HB

The purpose of this task was to investigate the mission requirements for a newly discovered object, 1977HB, which was thought to be comparatively accessible. Its inclination is less than 10° and its semimajor axis is 1.08 AU. Outbound and return transfer characteristics were examined over an entire synodic period of 9.33 years. A total of 32 optimum transfers were computed. Round-trip energy requirements were also compiled. The best round-trip mission found in the period 1977-86 had a total impulse requirement (assuming direct reentry upon return to Earth) of 9.18 km/sec. This is to be compared against a value of only 7.13 km/sec for sample return from Anteros, the best known case to date. The unexpectedly high round-trip ΔV requirement can be attributed to the comparatively high eccentricity of 1977HB, i.e., $e = 0.46$, which makes it a rather sharply Earth-crossing object. These results were discussed with Dr. E. Helin, who had requested the performance data, at the Asteroid Group '77 Summer Study conducted at ARC/NASA.

2.1.7 JOP Opportunity Dependent Performance Assessment

The purpose of this task was to update opportunity dependent Jupiter orbiter performance data in the Planetary Missions Performance Handbook, Vol. I: Outer Planets. In particular, the Summary Figure JO-1, showing net orbited mass capability versus launch opportunity, was redone for the 14 opportunities between 1976 and 1990. Performance was shown for the Shuttle/IUS(Twin)/Spinner launch stack, assuming 1-day windows, Earth-storable retropropulsion, payload optimized transfer times, and a 6 R_J by 30 day Jupiter capture orbit. Results reconfirmed the uniquely favorable performance of the 1981/2 launch opportunity for a Jupiter Orbiter/Probe (JOP) mission.

2.1.8 Planetary Advanced Studies Workshop

A 1-day workshop was convened by Code SL to discuss planetary advanced studies. Specific objectives of the workshop included: (1) review of the varied purposes of advanced studies; (2) definition of characteristic requirements for study of key planetary program missions; (3) examination of prerequisite mission study requirements such as computer program tasks and definitive science rationale and mission strategies; and (4) consideration of current special issues including potential uses for the Voyager PTM hardware. The workshop touched on most of these issues, but spent the majority of effort on a critical self-examination of how the advanced studies program is run. There was a general concern over the lack of mission options carried in pre-Phase A studies and a suggestion that early studies should try to avoid focusing on baseline mission profiles. A review of studies currently in progress or recently completed was also conducted with action items requested on a number of task-related issues.

2.1.9 Uranus Mission Opportunities for the Voyager PTM

The objective of this analysis was to determine the Uranus flight time performance of the Voyager PTM spacecraft. Missions to Uranus were investigated for the 1980 Jupiter/Uranus, 1981 Saturn/Uranus, and 1982 Saturn/Uranus swingby launch opportunities. The Shuttle/IUS(III) and axis-stabilized Voyager Propulsion Module (PM) were the assumed injection stages. A growth version of the PM ($\simeq$ 60% larger) was also considered. Best flight time performance was identified with the 1980 Jupiter/Uranus opportunity, but the Shuttle Orbiter will not be configured to meet the launch payload mass of the IUS(III) at this early date. The earliest possible launch would be the 1981 Saturn/Uranus opportunity which occurs several weeks before the nominal JOP window. The growth version of the PM is also needed to put the Voyager PTM on an acceptable Saturn/Uranus trajectory. Flight time to Uranus is about 8.3 years with encounter of Uranus occurring in April 1990. Swingby radius at Saturn would be within the orbit of Rhea at about 7.2 R_S, well outside the rings.

2.1.10 Near-Earth Resources Workshop

This NASA-sponsored summer study of near-Earth space resources was conducted at the UCSD campus in La Jolla, California. The purpose of the workshop was to review the potential needs and uses for near-Earth resources, and to consider what early steps should be taken in the direction of implementing those uses. Both lunar and asteroidal materials were considered as potential sources. The transportation of the materials to convenient Earth orbits was discussed, as were the alternatives for processing the available feedstocks into refined material for specific applications. It was generally agreed that the option for using space resources should be made available to our civilization within 20 to 30 years. To this end, early recommended steps of exploration included initiation of a Lunar Polar Orbiter mission, continued search for new Apollo asteroids, and serious consideration of asteroid reconnaissance missions. Also, a detailed review of space (vacuum and zero-g) beneficiation processes was recommended. SAI's participation in the workshop included a presentation on asteroid mission characteristics and opportunities, and the drafting of a chapter of the workshop report on asteroid mission characteristics.

2.1.11 IUS On-Orbit Assembly Planetary Mission Capability

The purpose of this task was a quantitative assessment of multi-Shuttle planetary mission performance. A ballistic mission capture analysis of five of the SEP/SAIL discriminator missions was performed using orbit assembly of multi-Shuttle-launched IUS stages. The purpose of the analysis was to determine just how much chemical propulsion would be needed to perform these difficult planetary missions, which are generally considered to require the prerequisite of low-thrust propulsion. The missions investigated included:

1. 1986 Mercury Orbiter w/Landers (3)
2. 1984 Comet Encke (87) Rendezvous
3. 1990 Mars Sample Return (dual launch)
4. 1983 Dual Asteroid Rendezvous w/Penetrators (2)
5. 1981 Saturn Orbiter w/Probes (2).

An IUS "triplet" configuration of three side-by-side large IUS motors (11% off-loaded) was introduced as a single stage to be incrementally added with additional Shuttle launches (one per "triplet" stage) to increase IUS(II) performance to required injected mass levels. Performance results revealed that three Shuttle launches would be required to capture each of the considered missions. Additional results included: (1) the need for very large (>6000 kg) space-storable post-injection stages on some missions; (2) reduced launch opportunity flexibility to avoid high-energy opportunities; (3) the apparent ballistic unfeasibility of rendezvous with more than two asteroids on one mission; and (4) the comparatively low energy requirement of Mars sample return using the dual launch mode.

2.1.12 Performance Assessment of Ion Drive Encke-X Missions

The objective of this analysis was to examine the extended performance potential of the 60 kw Halley Rendezvous Ion Drive low-thrust system applied to an Encke Rendezvous mission. Five multi-mission options were investigated; they are as follows:

1. Earth-Mars-Encke
2. Encke Sample Return
3. Earth-Encke-Asteroid(s)
4. Earth-Encke-Saturn Orbiter
5. Earth-Encke-Uranus Flyby.

The 1987 Encke apparition was chosen as an Encke encounter constraint. Parametric performance data, flight schedules, and trajectory profiles were prepared for each option. It was concluded that all five of the investigated options were possible with the assumed Ion Drive system.

Several offsetting factors identified, however, include long Ion Drive thrust times (up to 45,000 hrs) and additional project cost for multi-mission modules.

2.1.13 Future Planetary Applications of Solid Propulsion

A short White paper was prepared supporting the continued research and development of solid motor propulsion for planetary missions. Basic performance capability was reviewed against Earth- and space-storable propulsion alternatives. Impulse thresholds, below which solid propulsion outperforms these alternatives, were determined for several payload levels. Specific advantages of solid propulsion in four planetary mission uses were presented. Those uses included the following:

1. Kick stage Earth escape
2. Planetary orbit capture
3. Deorbit
4. Planetary ascent for sample return.

Continued research and development were recommended, with several specific areas of future study being noted.

2.1.14 Low-Cost Mars Sample Return Mission Performance Analysis

The problem addressed by this task was to determine Shuttle launch requirements for minimum cost/capability direct Mars sample return missions with specific consideration being given to short trip time capability. The sample return profile chosen consisted of direct Mars entry, direct Earth return, and direct Earth reentry. Direct Earth return opportunities from Mars included February 1986, May 1988 and July 1990. Outbound flight options investigated included conjunction ballistic, multi-rev ballistic, opposition ballistic, Venus swingby ballistic and multi-rev Ion Drive. From the performance results it was concluded that conjunction missions have the best payload margins, but require the longest stay times. Multi-rev missions are a better alternative in poorer opportunities with the slightly longer trip times. Outbound

Venus swingbys have variable opportunity-dependent characteristics. The 1986 swingby offers the shortest trip performance within a single Shuttle launch and payload assumptions. Neither Ion Drive nor direct fast ballistic options appeared competitive over the period studied against the other options. Large Mars entry masses were identified as a potential technology problem to all outbound flight options.

2.1.15 Planetary Mission Performance: Ion Drive versus Centaur

The objective of this analysis was to develop a planetary mission capture comparison between a Shuttle/Centaur launch stack and the Shuttle/IUS(Twin) augmented by a 60 kw Ion Drive low-thrust system. A set of seven missions was used to scope this analysis; the missions assumed were as follows:

1. Saturn Orbiter w/Probes (2)
2. Mercury Orbiter w/Landers (3)
3. Venus Orbit Imaging Radar
4. Mars Orbiter/Rover/Penetrators (3)
5. Mars Sample Return
6. Comet Encke Rendezvous
7. Multi-Asteroid Rendezvous.

Both nominal and growth Centaur stage designs were considered. Capture diagram results showed that the Shuttle/Centaur launch stack captured three of the seven missions (Nos. 3, 4 and 5). Using the growth Centaur stage design captures two additional missions (Nos. 1 and 2, although the landers had to be removed from No. 2). The Comet Encke Rendezvous and Multi-Asteroid Rendezvous missions were not captured ballistically by either Centaur launch stack, in the case of the asteroid mission even with only two targets. The Shuttle/IUS(Twin) augmented with Ion Drive captured all seven missions, two of them (Nos. 3 and 4) ballistically without Ion Drive.

2.1.16 Combined JOP/HCR Mission Performance Assessment

A brief review of previous Halley Comet Rendezvous (HCR) trajectory strategies and associated performance requirements was presented to illustrate the incompatibility of including a Jupiter Orbiter/Probe (JOP) on an HCR flight. It was shown that Jupiter is situated in practically the worst celestial longitude for a combined mission. It was further shown that even with much longer flight times (to better locate Jupiter), use of a Jupiter swingby and a smaller Ion Drive system, performance was still inadequate for nominal payload designs. Earlier launch dates (not later than 1980) were also completely inconsistent with development of a solar electric low-thrust system. In short, it is not possible to combine these two missions into a single low-thrust flight within reasonable development time/cost constraints.

2.1.17 Advanced Studies Administrator Presentation

A series of mission profile viewgraphs was prepared as part of a Planetary Programs presentation which was, in turn, part of the NASA Administrator's General Management Review of the Office of Space Science. Seven profile viewgraphs were prepared for the following missions:

1. 1985 Encke/Ceres Rendezvous
2. 1987 Multi-Asteroid Rendezvous
3. 1985 Encke Sample Return
4. 1992 Anteros Sample Return
5. 1991 Jupiter/Neptune Flyby
6. 1985 Fast Mars Sample Return
7. 1983 Mars Swingby/Encke Rendezvous.

Typical information included on these figures were the flight profile (either heliocentric or at encounter), key dates (launch, arrival, etc.), launch vehicle, and gross payload masses. This support activity was coordinated with JPL, which also provided a number of other mission viewgraph summaries for the presentation.

2.1.18 Future Planetary Mission Launch/Arrival Dates

The purpose of this task was to verify and document for planning purposes key dates of a variety of planetary missions actively under consideration in program planning. A total of 15 missions were reviewed, three with alternative launch opportunities. Launch, arrival and Earth return (as appropriate) dates were listed for each of the 18 cases considered. These were transmitted to NASA Headquarters and eventually used in the development of the FY'78 Five-Year Plan.

2.1.19 Mars Steering Group Participation

A Mars Steering Group was formed by JPL to coordinate and direct Mars Program studies funded as a budget line item for FY'78-'79. Because of SAI's advanced studies work in Mars program planning (Task 9 reported below in Section 2.6, as well as various Advanced Planning Activity subtasks, e.g., Sections 2.1.2, 2.1.12, and 2.1.14 above), our participation was requested by JPL and granted by NASA Headquarters. During the current year these activities (several meetings and some correspondence) are reported here as an Advanced Planning subtask. Three specific contributions were made to the steering committee between November 1977 and January 1978, in addition to meeting attendance and participation in many planning discussions. These were as follows:

1. Summary of Recent SAI Mars Advanced Studies
2. Characterization of Mars Concepts and Planning Issues
3. Development of Mars Exploration Scenarios.

The Summary of SAI Studies covered four items: (1) status of Task 9: Mars Strategies Study; (2) review of low-cost Mars sample return possibilities; (3) presentation of a combined Mars/Encke flight opportunity; and (4) a summary of possible Mars atmospheric devices. For the second item the four basic mission concept options, remote sensing, network surface science, long-range surface mobility and sample return, were summarized in terms of features and drawbacks. Related to these alternatives, five planning/strategy issues were addressed and discussed in some detail. For the third item, five Mars unmanned exploration scenarios were developed and described spanning a wide breadth of possibilities. The scenarios were titled: (1) Traditional Phase Exploration, (2) Modified Friedman Plan, (3) Sample Return First, (4) Minimum Program, and (5) Unmanned Apollo-Type Program. All of these contributions were vigorously discussed during Steering Group meetings.

2.1.20 Definition of the 1988 Encke Sample Return Mission

The purpose of this task was to develop mission profile data for a 1988 Encke Sample Return comparable to that developed in Subtask 17 (Section 2.1.17 above) for the 1985 Encke Sample Return mission. The difference between these two cases is that the 1985 launch rendezvous with Encke during its 1987 perihelion passage, while the 1988 launch encounters Encke during its 1990 passage. The overall performance between these two cases is comparable as is the total trip time. However, for the 1988 launch the outbound trip is somewhat shorter and the return trip longer to match Encke's arrival during its 1990 passage. A viewgraph mission profile with key dates and propulsion requirements was prepared comparable to the 1985 data prepared in Subtask 17.

2.1.21 Review of Proposed Saturn Workshop Objectives

A draft of 11 key questions was prepared by NASA Headquarters for consideration by a Saturn System Workshop. The purpose of this subtask was to review these questions for correctness/completeness, and to suggest appropriate changes. Revisions were made in the interest of clarity and completeness. Several questions were combined and one new question was added. The ordering of the questions was also changed to fall into four suggested categories: (1) general questions; (2) questions specific to orbiting spacecraft; (3) questions specific to entry probes; and (4) questions relevant to a Titan lander mission. The changes were telecopied back to Headquarters and incorporated into the final set of questions posed for consideration by Workshop participants.

2.1.22 Planetary Five-Year Plan: Revised Waterfall Charts

In preparation for upcoming Five-Year Planning Activities, the missions and their associated costs from the 1977 Five-Year Plan were summarized and waterfall charts of that plan prepared. The plan consisted of seven new project starts which were as follows:

		New Start
1.	Lunar Polar Orbiter	FY'79
2.	Venus Orbiting Imaging Radar	FY'80
3.	Mars-86 Orbiter/Rover/Penetrators	FY'81
4.	Comet Encke Rendezvous	FY'81
5.	Saturn Orbiter with Dual Probes	FY'82
6.	Asteroid Multi-Rendezvous	FY'83
7.	Mercury Orbiter	FY'83

Changes to these projects reflecting changes during the current study year were first made. New estimates of cost, where available, were then incorporated into an updated resources estimate. These data were then plotted as waterfall charts in both fixed FY'79 and in real year dollars. The results were sent to R. Wallace, Code SL Manager of Advanced Studies, for review. One iteration was performed following submission of these results, i.e., substitution of a cheaper Mars Polar Orbiter mission for the baseline Mars-86 concept. The revised waterfall showed that peak annual funding in FY'79 dollars was reduced from over $600M to under $500M. Total funding was also reduced over the plotted 10-year funding period (FY'78-'87) by an amount corresponding to the difference of these two missions.

2.1.23 Performance Comparison of Ballistic/SEP/Ion Drive Systems

The purpose of this subtask was to compare the performance capability of ballistic, solar electric propulsion (SEP), and Ion Drive flight modes for accomplishing four advanced planetary missions. The missions considered included:

1. Comet Encke Rendezvous
2. Saturn Orbiter-Dual Probe
3. Mercury Remote Sensing Orbiter
4. Multi-Asteroid Rendezvous Tour.

Performance assumptions based on early STS operations were used to assess the capabilities of each flight mode applied to each of these four missions. Key mission parameters, i.e., launch opportunity, flight time and reference payloads were also established for this comparative analysis. Bar charts were prepared for each mission illustrating the payload margin existing with each delivery system. A single summary bar chart illustrated two major conclusions: (1) the inadequacy of the ballistic flight mode to perform any of the four missions adequately, and (2) the superior performance of Ion Drive to SEP low-thrust systems.

2.1.24 Revised FY'77 Planetary Five-Year Planning Exercise

The purpose of this analysis was to update the FY'77 Five-Year Plan prior to initiation of FY'78 Five-Year planning. Of specific interest was the ability of the FY'77 plan to fit within a recently defined planning wedge. The Five-Year Plan mission estimates were updated along with an additional group of valuable but not "hard core" missions. The data were then plugged into the Five-Year Plan costing procedure to determine consistency with the planning wedge. Several new iterations of the plan's missions scenario were found to be necessary before the planning wedge was no longer exceeded. The resulting descoped plan and summary waterfall chart were provided to the Planetary Programs Office for presentation and discussion with OSS/NASA management. In addition, several alternative plans were developed to demonstrate increments by which the planning wedge was exceeded as the plan approached its original scope and definition.

2.1.25 Equivalence Comparison of Ballistic and Low-Thrust Flight

At the request of the Manager of Code SL's Advanced Programs and Technology, D. Herman, a technical comparison of ballistic and low-thrust flight modes was prepared to illustrate relative performance capability. For the sake of this comparison, a common performance variable, equivalent velocity increment ($\Delta\tilde{V}$) was defined. Using $\Delta\tilde{V}$ it was possible to show a comparison of payload mass fractions for each flight mode. A graphical summary of this comparison was prepared for a $\Delta\tilde{V}$ range of 0 to 14 km/sec. A space-storable retropropulsion system was chosen to represent the ballistic flight mode, while a 25 kw SEP system was used for the low-thrust flight mode. With these assumptions the ballistic flight mode offered the best performance (payload mass fraction) below a $\Delta\tilde{V}$ of 2 km/sec, while the low-thrust flight mode was superior at all $\Delta\tilde{V}$'s above 2 km/sec. A detailed summary of this analysis was documented with a specific Encke Rendezvous mission example used to verify these performance results.

2.1.26 Evaluation of IUS On-Orbit-Assembly Study Proposal

The purpose of this subtask was to evaluate a JSC/NASA study proposal for on-orbit-assembly which Planetary Programs (Code SL), was requested to assess by Shuttle Upper Stage (Code MLF). Specifically, the proposal recommended detailed consideration of an automated Mars Sample Return mission as the driver for establishment of on-orbit-assembly IUS capability. The proposal was carefully reviewed with consideration of independent Mars Sample Return performance requirements. A memo summarizing the evaluation point-by-point was written and delivered to D. Herman, Manager of Code SL's Advanced Programs and Technology. It was recommended that the JSC study be defocused from Mars Sample Return to a broader suite of advanced planetary missions to determine the general capability of on-orbit assembly and associated stage requirements, before detailed specifications were developed for specific mission applications.

2.1.27 Preparation of Lunar/Planetary Programs Historic Waterfall Charts

The objective of this brief analysis was to graphically summarize the history of Lunar/Planetary Programs funding within the perspective of constant dollars. The program resources were obtained from NASA Headquarters beginning in 1960 and taken through run-out of current approved projects, i.e., 1985. Projects were accumulated in the following seven categories:

1. Support Base
2. Lunar Missions
3. Mariner Block
4. Pioneer Block
5. Viking
6. Voyager
7. Jupiter Orbiter Probe (Galileo).

These data were then converted to FY'79 constant dollars and plotted as a waterfall chart. The chart shows that current and run-out funding in Lunar/Planetary Programs is below all funding levels since FY'61, i.e., $200M (in FY'79 dollars).

2.2 Cost Estimation Research (809 man-hours)

Cost estimation analysis has been an on-going Advanced Studies support task for five years. Its objective is to develop and implement a methodology for predicting costs of future lunar and planetary flight projects. Its purpose is to provide reasonably accurate cost estimates based on pre-Phase A study definitions to key advanced planning activities within the Planetary/Lunar Programs Office. A flight project cost estimation model has been in existence at SAI for the past four years as a result of this task effort, and has been regularly improved and expanded in scope of application as a result of this on-going research. The nature of the work falls into one of three general subtasks:

1. Flight Project Data Collection
2. Modeling Analysis
3. Cost Prediction.

Work is done in all three subtask areas each year. The level of effort expended on data collection has stabilized during the past several years with three to four flight projects being tracked at any given time. There has been a shift in emphasis, however, within the other tasks with increasingly more effort now expended on applications and less on modeling. This occasionally changes as new features are added to the cost model, but emphasis generally continues on applications. Each of the subtasks is briefly summarized in the following subsections.

2.2.1 Flight Project Data Collection

Since data collection began more than five years ago, every effort has been made to incorporate all relevant lunar and planetary flight project data into the model. Direct labor, burden, materials and miscellaneous costs are tracked on every element of each project. These data are then reduced into new categories consistent with modeling algorithms used in the cost model.

During the 1977-78 contract period new data were collected on three flight projects: Viking Orbiter, Voyager (Mariner Jupiter/Saturn) and Pioneer Venus. The Viking Orbiter project, per se, is essentially complete; the new data represent extended mission costs. With the successful launch of the two Voyager spacecraft in mid-1977, the emphasis for this task now shifts from hardware associated costs to mission operation and data analysis costs. The Pioneer Venus project is important in two basic respects: it is the first flight project providing "real" data for atmospheric entry probes; and it is a complex project, involving the design, construction and operation of four differing types of spacecraft. This second aspect poses challenging questions in terms of cost allocation and modeling; e.g., how to correctly prorate support category costs such as program management among the various spacecraft, or how to correctly account for hardware commonality.

2.2.2 Cost Modeling

The initial objective of the cost modeling subtask was the development of a flight project cost prediction analog whose input requirements could be restricted to pre-Phase A level mission definitions. Such a cost model, using direct labor hours as the working cost parameters, has been developed at SAI and is actively in use. The on-going purpose of this subtask is to refine and expand the model's scope of application as permitted by the expanding base of flight project data resulting from the effort expended in the previous subtask.

Development of the cost model was initiated with the redistribution of flight project cost data into a minimum set of categories, each of which was to be modeled as a function of some pre-Phase A mission parameter(s). The categories found to be most acceptable for this purpose fell naturally into two classes: (1) subsystem hardware costs which have both nonrecurring and recurring elements, and (2) project support costs which are recurring elements scaled (in part) to the magnitude of total hardware costs. The specific categories used are as follows:

1. <u>Hardware Categories</u>

   Structure
   Propulsion
   Guidance and Control
   Communications
   Power
   Science

2. <u>Support Categories</u>

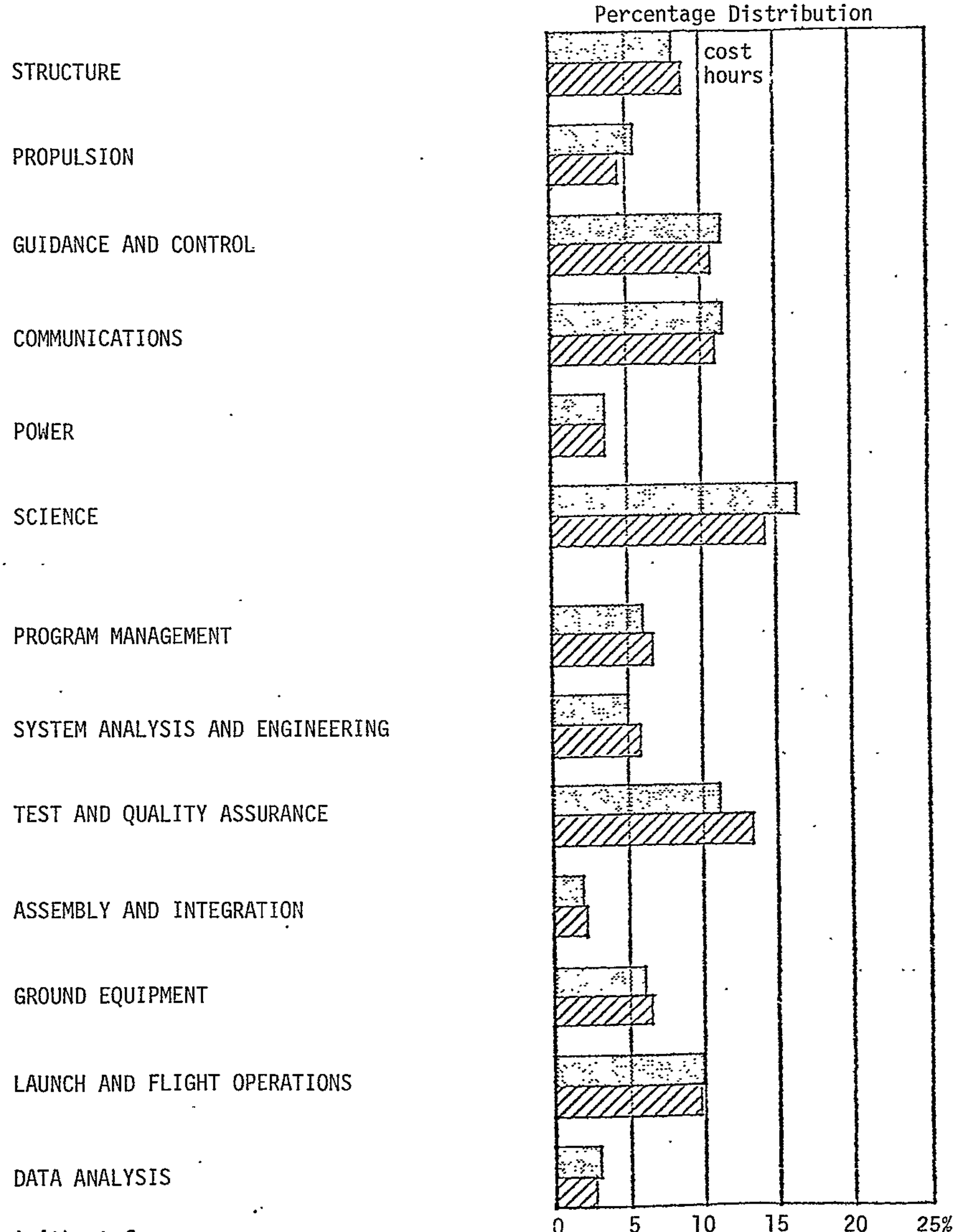
   Program Management
   Systems Analysis and Engineering
   Test and Quality Assurance
   Assembly and Integration
   Ground Equipment
   Launch and Flight Operations
   Data Analysis

An obvious dependent parameter choice for modeling the costs of these categories is dollars. However, the use of dollars often obscures the real cost because of wage inflation factors, overhead rates, fees, etc. Planetary missions are typically characterized by very low production volume and high development costs, i.e., they are labor-intensive endeavors. Hence, the use of direct labor hours was considered as a possible alternative to dollars. Productivity rather than wage rate (and hence inflation factors) becomes a key measure of cost when using direct labor hours. Also, direct labor is a common denominator of NASA cost reporting requirements from which overhead, G&A and fee are computed. Of concern in the use of direct labor hours was the omission of project materials costs. To examine how well direct labor alone could track total project cost, comparisons are continually made between cost per category and direct labor per category. For both parameters, percentage comparisons averaged over the entire ten-project data base are shown in Figure 1 for each category defined above. The comparison validates the credibility of direct labor hours to adequately track total project cost. Further analysis of the data base also revealed that direct labor hours represent 30% of total flight project cost with only a few percent variation over the entire data base. It was concluded that the labor hours are indeed a very good parameter of cost, and further that modeling project direct labor is essentially equivalent to modeling total planetary flight project costs.

The choice of direct labor hours to model cost opened the way for the actual modeling analysis. Labor estimating relationships (LERs)

Figure 1

PERCENT COMPARISON OF DOLLARS* AND LABOR HOURS--ALL MAJOR PROJECTS (AVG)



*without fee.

were developed for each cost category. The nonrecurring direct labor hours (NRDLH) of the hardware categories were modeled first since they were most readily associated with pre-Phase A mission parameters, particularly weight. Recurring direct labor hours (RHLD) were modeled next as a function of the NRDLH and number of flight articles. Prelaunch support category direct labor hours were modeled as a function of the accumulated total hardware direct labor hours. Launch and post-launch functions were modeled from pre-Phase A mission parameters, particularly event times, as well as accumulated direct labor hours.

A flow chart depicting the total estimation procedure is presented in Figure 2. The heavy arrows indicate the primary flow of the estimation process using the various LERs outlined above. Both hardware and support category direct labor hours (DLH) are converted to dollars using modeled category wage rates and inflation factors consistent with the anticipated flight project period. These costs are accumulated to a total direct labor (DL) project cost which is then ratioed up (÷ by 30%) to finally determine total project cost. Note that inheritance (cost saving) factors can be added to the input stream at the hardware cost level to reduce required NRDLH levels for subsystem development. Inheritance is considered as a percentage of each category which qualifies for cost savings with actual savings accrued at as many as three levels of inheritance. Reductions in hardware NRDLH are allowed to ripple through the estimation procedure so that additional savings are also realized in associated support categories. The inheritance method is sufficiently general to permit eventual inclusion of standardized hardware cost benefits when such data become available from flight project experience.

Both the LERs and their synthesis into an estimation procedure are the subjects of the continued analysis of this subtask. As a result of this on-going effort the cost model is now applicable to a wide scope of mission concepts including flybys, orbiters, entry probes, landers, and sample returns. Subtask analysis is currently focused on improving

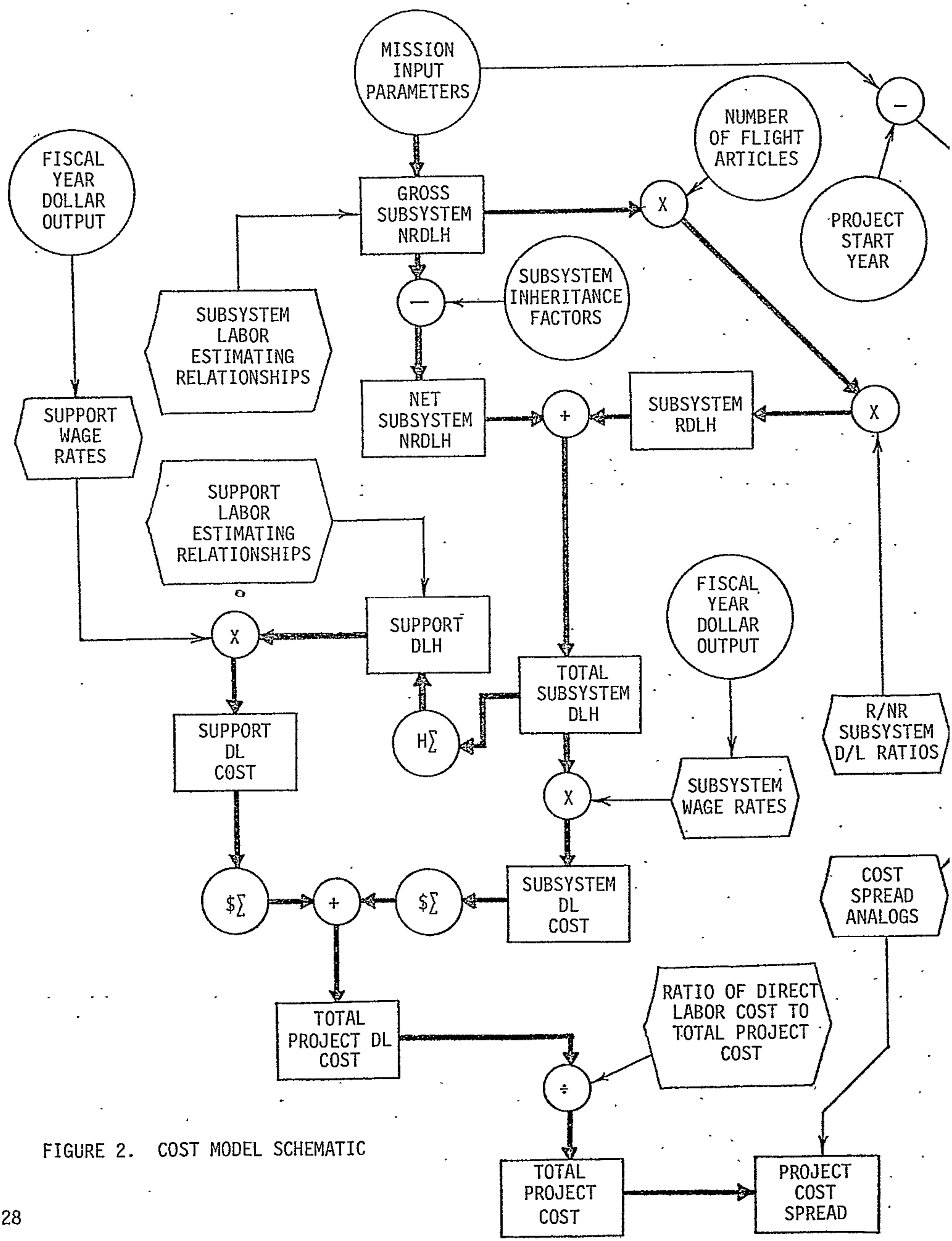


FIGURE 2. COST MODEL SCHEMATIC

entry probe cost estimates with results not yet complete as Pioneer Venus flight project data are still being collected. As the model has been expanded and improved, so also have the input requirements increased. The current list of possible input parameters is presented in Table 2. This list will undoubtedly continue to grow with further model improvements, but will be diligently constrained to a pre-Phase A study information level.

Cost model accuracy objectives are twofold: (1) estimates of total costs for projects included in the data base should not differ from actual by more than 10%; (2) new project estimates should not be in error by more than 20% with mission scope held constant. Error analysis of the model against the data base presently shows a mean error of -6.4% in cost (i.e., underestimating) with a mean absolute error of 12.9%. Applications to date against existing programs not in the data base indicate that errors for new flight projects are probably not greater than 25%.

2.2.3 Applications

Applications of the cost model have continued to increase with its refinement and expanding scope. During the past contract period, the model was used extensively in support of advanced planning activities by the Planetary/Lunar Programs Office.

Table 2

COST MODEL INPUT PARAMETERS

- Mission Factors
  - Fiscal Wage Date
  - Date of First Launch
  - Number of Flight Articles
  - Mission Duration
  - Encounter Time
  - Launch Windows
- Structure
  - Total Weight of Structure Subsystem
  - Weight of Mechanisms and Landing Gear
  - Weight of Thermal Control, Pyrotechnics and Cabling
- Propulsion
  - Dry Weight of Propulsion System
  - Liquid Vernier Dry Weight
  - Aerodeceleration Subsystem Weight
- Guidance and Control
  - Total Weight of Guidance and Control Subsystem
  - Weight of Radar in Guidance and Control Subsystem
- Communications
  - Weight of Radio Frequency Subsystem
  - Weight of Data Handling Subsystem
  - Diameter of Antennas
- Power
  - Weight of Power Subsystem Excluding RTGs
  - Number of RTG Units per Spacecraft
  - RTG Fuel Loading (thermal watts)
- Science
  - Total Weight of Science Experiments
  - Weight of Lander Surface Experiments
  - Pixels per Line of TV

2.3 Planetary Missions Performance Handbook, Volume I Revisions (2016 man-hours)

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 planning. In the past, two types of NASA handbooks have been prepared, each presenting a particular type of fundamental mission data: (1) raw trajectory data handbooks such as the NASA SP-35 series, and (2) propulsion system performance handbooks such as the NASA Launch Vehicle Estimating Factors Document. To make use of these data in performance analyses, 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 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. The basic format for presentation of outer planet payload results is net payload versus trip time. Additional data are included to investigate performance sensitivity to changes in orbit size. Further, a series of working graphs, presented as appendices, allow the analyst direct access to the underlying trajectory and propulsion data, and facilitate generation of additional performance results from perturbed mission assumptions. Volume I (Revised) incorporates the most recently obtainable propulsion system definitions, timely interplanetary transfer techniques, and currently prevailing mission 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.

2.3.1 Scope

Volume I of the PMP Handbook deals with missions to the five outermost planets. With the issuance of the first revision, the scope has been expanded to include new mission modes and more opportunities to the furthest targets. The full revised scope of missions and opportunities is shown in Table 3. The missions in this table are organized by final target, reflecting the overall organization of the Handbook itself. Thus, Jupiter/Neptune and Saturn/Uranus/Neptune flyby missions are both to be found under Neptune Flybys.

Orbiter missions are presented for Jupiter, Saturn and Uranus. In each case, orbiter mass constraints on the direct ballistic transfers suggest consideration of alternate flight modes. For Jupiter, six opportunities for a Venus-Earth Gravity Assist (VEGA) have been explored in some detail, and found to produce some payloads which double the corresponding direct ballistic payloads. Saturn and Uranus missions are better served by a variation on this technique: the Deep-Space Impulse-Earth Gravity Assist (ΔVEGA) mode. Solar Electric Propulsion, reevaluated in recent work (Ion Drive), finds application in missions to trans-jovian targets.

Flyby missions are presented for all five targets. Although a direct ballistic flyby will suffice for the nearer targets, direct flight times to the outermost targets are prohibitively long. Gravity-assisted swingby of one or more intermediate targets is used to reduce flight time (or increase payload) to Uranus, Neptune and Pluto. Note that opportunities for such a swingby mission are grouped in three or five consecutive launch years. These groupings recur periodically with best performance typically obtained in the middle opportunity. Early launch opportunities in the group are sometimes fictitious because the relative positions of the bodies involved require a swingby below the surface of the gravity-assist planet. At the same time, the latest opportunities in the group have the greatest swingby distances--so great, in some cases,

Table 3

SCOPE OF VOLUME I, PMP HANDBOOK

MISSION		LAUNCH OPPORTUNITY (19xx)															
		80	81	82	83	84	85	86	87	88	89	90	91	92	93	94	95
JUPITER	BALLISTIC FLYBY	X	X		X	X	X	X	X	X	X	X					
	BALLISTIC ORBITER	X	X		X	X	X	X	X	X	X	X					
	VEGA ORBITER		X		X	X		X		X		X					
SATURN	BALLISTIC FLYBY	X	X	X	X		X	X	X	X	X	X					
	BALLISTIC ORBITER	X	X	X	X		X	X	X	X	X	X					
	ION DRIVE ORBITER	X	X	X	X		X	X	X	X	X	X					
	ΔVEGA ORBITER	X	X	X	X		X	X	X	X	X	X					
URANUS	BALLISTIC FLYBY						X					X					X
	J/U SWINGBY													X		X	X
	BALLISTIC ORBITER						X					X					X
	ION DRIVE ORBITER						X					X					X
	ΔVEGA ORBITER						X					X					X
NEPTUNE	J/N SWINGBY													X		X	X
	S/U/N SWINGBY*	X	X	X	X		X										
	U/N SWINGBY*						X	X	X	X	X						
PLUTO	J/P SWINGBY*											X	X	X			

*both ballistic and Ion Drive

that the gravity-assist impulse to the spacecraft is of little or no value in shaping the final leg of the transfer. These two extremes are particularly apparent in missions which utilize a Jupiter swingby: because Jupiter's heliocentric distance is much less than that of any of the trans-saturnian planets, the advantageous Earth-Jupiter-target geometry phasing is of short duration. The consequence, as seen in Table 3, is that some opportunities from each group are culled out. Thus, each of the Jupiter swingby groups shows only three real launch opportunities.

Calendar year gaps appear in the table for several reasons. Launch opportunities to all of the outer planets are spaced at intervals of slightly more than 1 year, with Jupiter the longest at about 13 months. Therefore, an occasional calendar year will not contain a launch opportunity. For example, there is no Saturn opportunity in calendar 1984. Note that this produces a corresponding gap in the Saturn/Uranus/Neptune missions as well. The occurrence of VEGA launch windows is a much more involved matter. Previous work has searched the 1980's decade and produced the six viable opportunities shown. (In contrast, note that a ΔVEGA opportunity exists for every direct launch opportunity, the difference being a matter of flight time.) Uranus missions are shown every fifth year, although opportunities exist every year. The annual performance changes for Uranus opportunities are small because of the planet's relatively slow motion about the Sun. It suffices to show data from every fifth opportunity to adequately represent launch year dependent performance variations.

2.3.2 Revisions

The revised issue of PMP Handbook, Volume I, has changed both the scope of missions covered and some of the assumptions underlying the performance analysis. To summarize, the major revisions are:

1. Change reference launch window extent from 21 days to 10 days.

2. Add current IUS candidate performance based upon the Boeing Burner II vehicle (replacing the earlier Transtage concepts).

3. Include satellite-assisted capture performance for Jupiter and Saturn Orbiter missions (Galilean satellites and Titan, respectively).

4. Add new swingby mission opportunities to the outermost planets.

5. Add VEGA/ΔVEGA option for orbiter missions to Jupiter, Saturn and Uranus.

6. Add Ion Drive Solar Electric Propulsion option (replacing previous SEP concepts).

7. Add working graphs for the mission analyst.

Necessary descriptions of these items appear below.

Propulsion System Application. The propulsion systems used to define payload performance fall into three classes: (1) launch vehicles, (2) interplanetary Ion Drive (low-thrust) systems, and (3) retropropulsion stages for midcourse maneuvers and capture into orbit. All systems used here are presumed to be available for the entire period of application, except for the Space Tug, which is not expected to be available until 1985 or later.

Launch Vehicles. Two base launch vehicles are used for outer planet applications in the PMP Handbook series. They are the expendable Titan IIIE and the reusable Space Shuttle. There are a number of existing and conceptual chemical upper stages and kick stages which may be used in combination with either or both of the basic vehicles. To show mission performance in terms of deliverable mass, a subset of these stages has been chosen and mated to the base launch vehicles. Performance curves which suffice to define these combinations are given in Figures 3-5. The lone expendable vehicle, the Titan/Centaur with MJS propulsion module attached, is shown throughout this volume as a benchmark for comparison with previous work. It is expected that all NASA interplanetary missions in the next decade will be carried as Shuttle payloads. Current studies have centered upon two Interim Upper Stage (IUS) candidates: the IUS(Twin) and the IUS(Twin)/Spinner. Both of these (as well as all other recent IUS candidates) are based upon the Boeing Burner II IUS concept. The former consists of twin "large" stages, each of which has a gross ignition mass of slightly more than 10,000 kg. The latter also includes a third Spinning Solid Upper Stage to capture higher energy missions. The low energy IUS(Twin) sees application in Volume I for VEGA/ΔVEGA and Ion Drive missions, all of which require launch energies of less than $\sim 50\ km^2/sec^2$.

The Tug represents the ultimate upper stage design for the Shuttle. Two versions are considered here. The first is a recoverable Tug(R) which uses an upper stage motor (EE-Kick) for return propulsion. As an expendable vehicle, with or without the Earth-Escape-Kick stage, the Tug(E) is the most capable ballistic performer. The kick stage, a large attitude-stabilized solid motor suggested by MSFC, is appended for very high-energy missions.

All launch vehicle performance shown here assumes trans-target injection from the standard 160 n.mi. circular parking orbit (90 n.mi. for Titan IIIE). Payload adapters are excluded at this point. However, payload performance data in subsequent sections take into account a 35 kg adapter for ballistic missions and a 115 kg adapter for Ion Drive missions. All launches are assumed to be due east from KSC. Non-easterly launches are accounted for in the following way: for the expendable Titan IIIE, a non-easterly launch penalty is imposed when the declination of the launch asymptote (DLA) is in the range 28.5° to 52.4°. For DLAs greater than 52.4°, an additional dog-leg maneuver

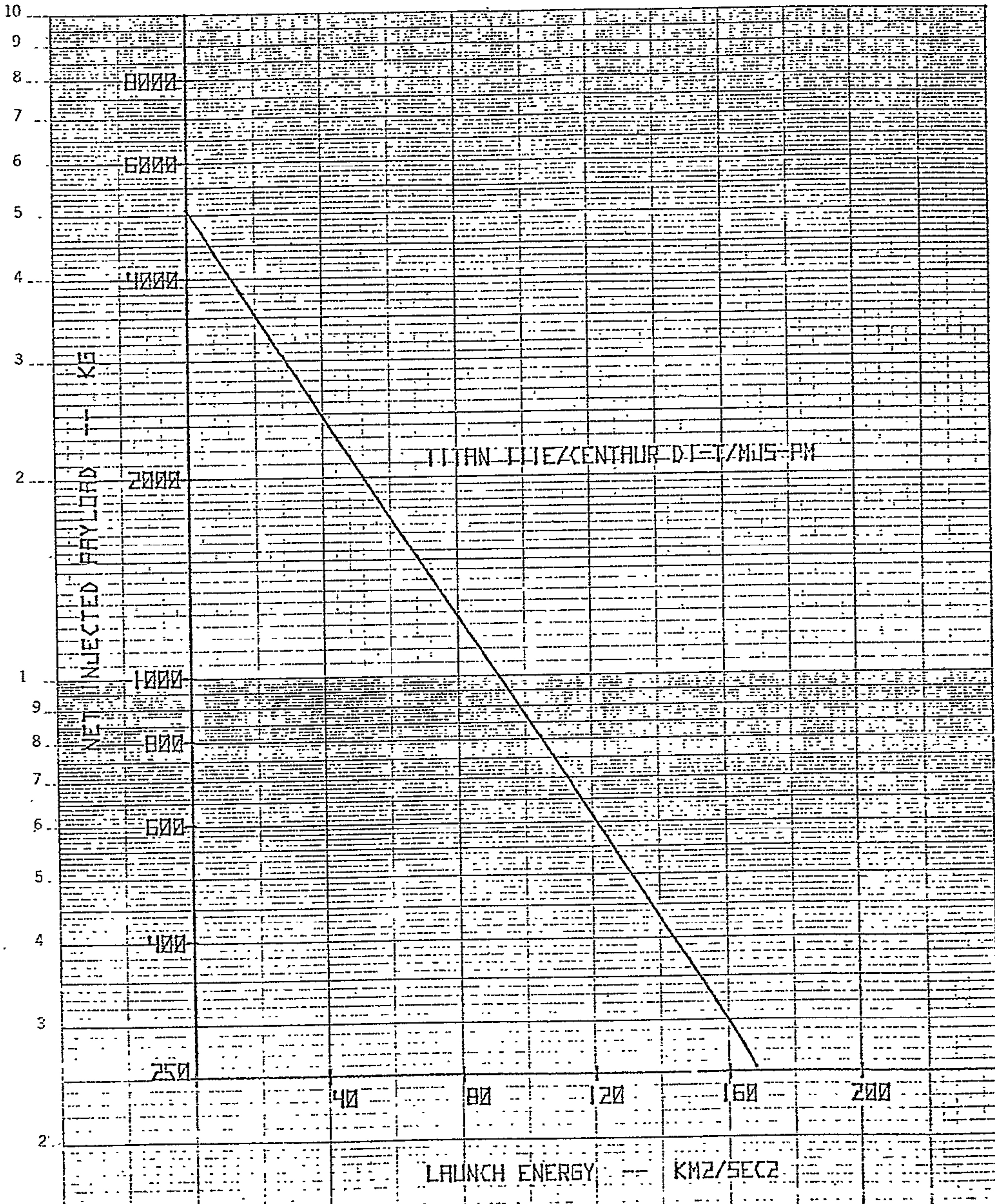


FIGURE 3. EXPENDABLE LAUNCH VEHICLE PERFORMANCE

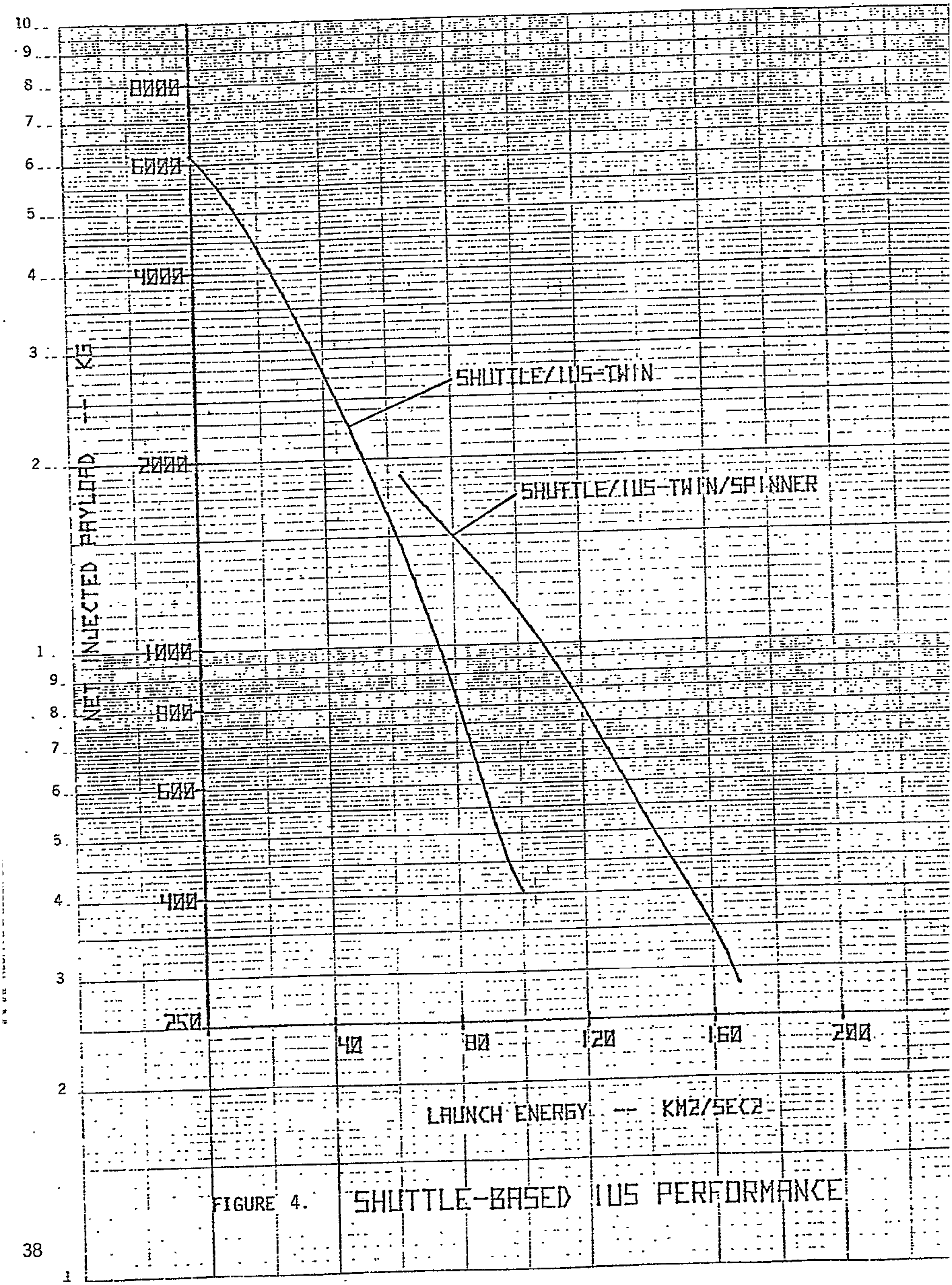


FIGURE 4. SHUTTLE-BASED IUS PERFORMANCE

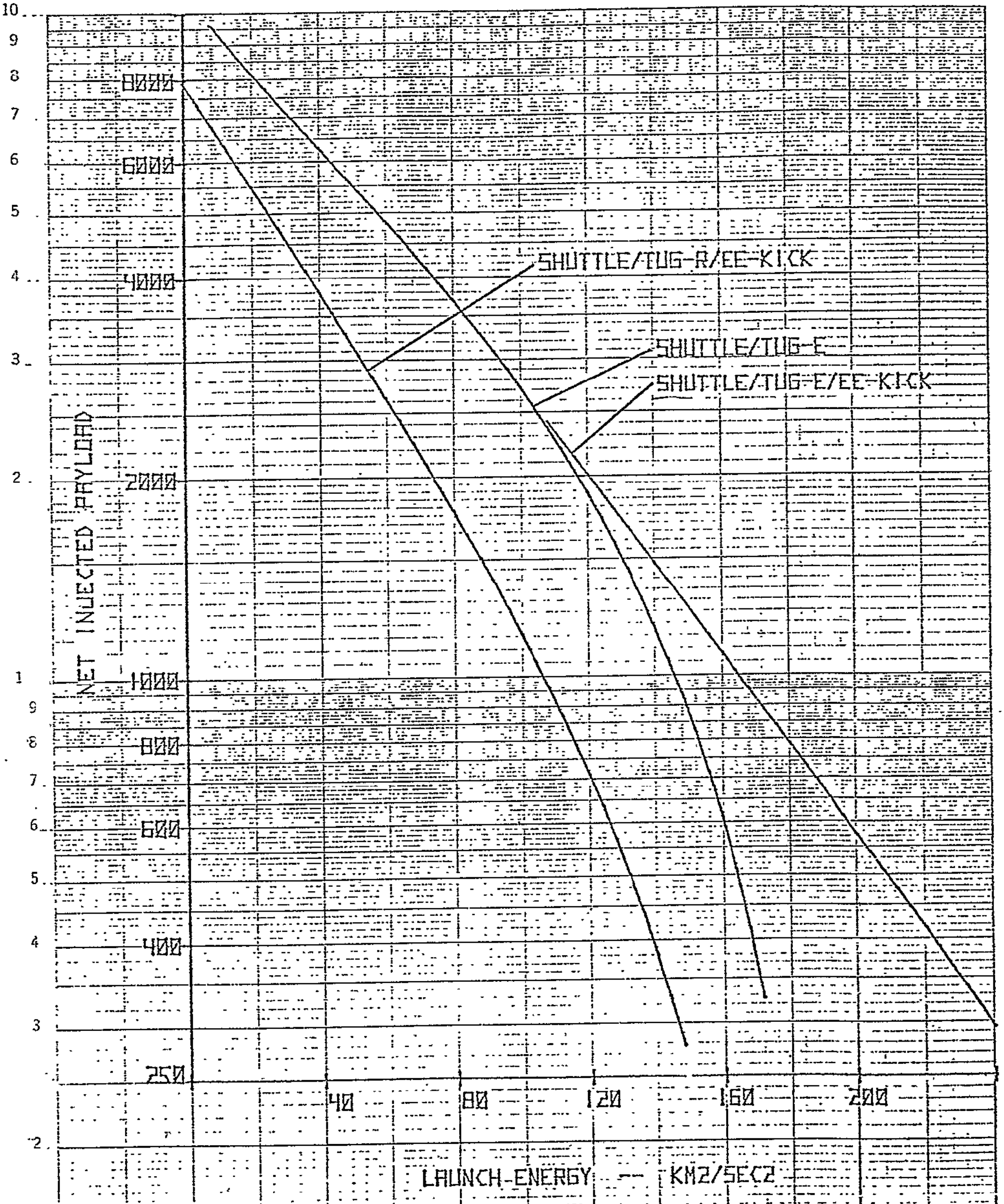


FIGURE 5. SHUTTLE-BASED TUG PERFORMANCE

penalty is imposed. For Shuttle-based launches, a dog-leg maneuver penalty is imposed only if DLA exceeds 43.5° and Shuttle cargo bay mass is exceeded. (The latter, for a given vehicle, is dependent upon DLA.) The upper stage must pay the entire dog-leg penalty for DLA > 57°. This limit is imposed by range safety constraints. Performance penalties for a 10-day launch window are imposed on all launch vehicle options by increasing the optimum launch energy.

2.3.3 Ion Drive (Low-Thrust) Propulsion

Throughout the PMP Handbook series, low-thrust Solar Electric Propulsion is treated in modular form, rather than as a spacecraft-integrated system, in order to facilitate wide-ranging cross-comparisons with ballistic data. The Ion Drive propulsion module consists of a concentrated solar array power source, coupled through conventional power processing units to drive an array of electron bombardment thrusters. Installed solar cell power at beginning of life is fixed at one of 25, 40 or 60 kw (see Table 4). Degradation of this power level is estimated to be 12% over the duration of the mission. For purposes of performance evaluation, the entire 12% is subtracted at beginning of life to yield the effective cell powers shown. The parabolic concentrator has a maximum power concentration factor of 3.2. Thus, at a distance of 4.5 AU, the available power is 20.8% of P_{max} compared with a conventional SEP power of only 5.8%. Thruster specific impulse is fixed at 4000 sec, an average value representative of current design points. There results a propulsion efficiency of about 68%. Propellant mass and thrust direction for each mission/opportunity combination is determined by CHEBYTOP.

2.3.4 Retro Stages

Retro stages are the third class of propulsion used in PMP Handbook payload performance computations. Specifically, they are used for orbiter missions, all of which are presumed to require a chemical retro stage

Table 4

ION DRIVE (SEP) OPTIONS FOR PMP HANDBOOK, VOL. I

Thrust Module:	120 kg bimod unit (two thrusters each unit, one always on standby)
Interface Module:	260 kg
Concentration Factor:	3.2
Power Degradation:	$P_o = 0.88\ P_{BOL}$ - effective cell power (P_o) is degraded 12% from installed beginning of life power (P_{BOL})
Maximum Input Power (55°C limit):	$P_{max} = 1.20\ P_o$ for $R \leq 1.35$ AU
Specific Impulse:	4000 sec
Propulsion Efficiency:	$\eta = 0.6796 = \dfrac{b}{1 + (d/c)^2}$, where $\begin{cases} b = 0.775 \\ d = 14.7 \\ c = I_{sp} \times 9.80665 \times 10^{-3} \text{ km/sec} \end{cases}$
Auxiliary Power:	Included in above

P_{BOL} (kw)	No. Units	P_o (kw)	M_{PS} (kg)	α (kg/kw)	Propulsion Time (days)*
60	5	52.8	1700	32.197	800
40	4	35.2	1300	36.932	800
25	4	22.0	1090	49.545	800

*Shortened for J/P swingby missions.

for impulsive orbit capture. (Satellite-assisted captures, discussed below, still require retropropulsion.) Orbiter performance data are restricted to single stage applications, since multi-stage retro systems are considered unnecessary for the ranges of target approach mass and capture impulse presented. Two retro options are considered (Table 5): a flight-proven bipropellant Earth-storable system with I_{sp} = 300 sec, and a present technology space-storable system with I_{sp} = 370 sec. Both of these are rubber stages: the propellant tanks are sized to the specific conditions of planet approach mass and approach velocity.

2.3.5 Satellite-Assisted Orbit Insertion

Satellite-assisted capture has been shown to be capable of effecting orbit capture impulse savings of several hundred meters/second at Jupiter. This savings is a function of both the required orbit insertion impulse and the desired periapse. The technique is applied here to the four Galilean satellites and to Titan. The satellite-assisted maneuver is a powered swingby which produces a final (post-encounter) orbit whose periapse lies inside that of the satellite. Although an impulse is required at swingby, the gravity-assist effect of close satellite encounter produces a savings over the unassisted impulse requirement. Performance results throughout the Handbook are shown for fixed orbit sizes, specified by periapse radius and period. Orbits achieved with satellite-assisted capture are also fixed, but the values shown for orbit period are nominal only. In fact, the period in every such instance has been adjusted to produce an orbit which is resonant with that particular satellite. Actual values for orbit parameters are shown in Table 6.

Note that the periapse radius selected for each of the Galilean satellites is about nine-tenths of the satellite's own periapse. In each case, the effect of the satellite-assist maneuver is maximized for insertion into a final orbit whose perijove lies just inside the

Table 5

RETRO STAGE SCALING LAWS FOR PMP HANDBOOK, VOL. I

- Stage Sizing Equation:

$$M_s = M_o(1 + f)\left[1 - e^{-(\Delta V/c)}\right] + M_e$$

where:

M_s = retro stage mass (kg)
M_o = planet approach mass (kg)
M_e = retro engine mass (kg)
f = tankage structure factor (fraction of required propellant)
ΔV = retro velocity impulse for capture (km/sec)
c = exhaust velocity (km/sec).

- Stage Option Characteristics:

	Earth-Storable	Space-Storable
Retro Engine Mass, M_e	65	85
Tankage Factor, f	0.13	0.14
Specific Impulse, I_{sp}	300	370
Exhaust Velocity, c	2.942	3.629

Table 6

ORBIT SIZES ACHIEVED BY SATELLITE-ASSISTED CAPTURE

	Io	Europa	Ganymede	Callisto	Titan
• Body Constants					
Gravitational Parameter (km^3/sec^2)	5934	3196	9885	7172	9140
Radius (km)	1829	1500	2635	2500	2916
Orbit Period (days)	1.77	3.55	7.15	16.69	15.95
• Orbit Periods (Days)					
Nominal Period 90	90.27	92.30	92.95	83.45	95.70
Nominal Period 120	120.36	120.70	121.55	116.83	127.60
Nominal Period 150	150.45	149.10	150.15	150.21	143.55
• Orbit Periapse (Planet Radii)	5.3	8.5	13.5	23.7	20.2

satellite's own orbit. For Titan-assists at Saturn, the optimum capture impulse savings seems to occur arbitrarily close to the satellite: the performance shown here is for a post-swingby periapse of 20.2 Saturn radii, or about 0.99 of Titan's periapse. Actual performance--capture impulse as a function of approach velocity and final orbit period--is shown for all five satellites as working graphs in the appendices.

VEGA/ΔVEGA Missions. Recent studies have taken detailed looks at the application of two types of Earth gravity-assisted swingbys to increase net payload deliverable to outer planet targets. In general, payload availability on outer planet missions is constrained by high launch energy requirements for the direct ballistic transfer. These two alternatives offer significantly lower launch energies at the expense of added midcourse impulses and longer trip times. VEGA (Venus-Earth Gravity Assist) and ΔVEGA (midcourse ΔV-Earth Gravity Assist) trajectories require the Earth reencounter to increase heliocentric energy and shape the final outer planet transfer leg. The resulting performance proves to be quite attractive: one can generally expect payloads which are two or three times as large as that for the corresponding direct transfer. This volume incorporates the trajectory work done on VEGA/Jupiter missions. Extensive additional parametric data were generated for ΔVEGA opportunities to Saturn and Uranus. Although both modes are applicable to all three targets, it turns out that VEGA missions are better, although more inconsistent performers than ΔVEGA missions to Jupiter. For Saturn and Uranus the situation is reversed. ΔVEGA missions are able to deliver larger payloads to target orbits. Therefore, PMP Handbook Volume I incorporates VEGA/Jupiter missions and ΔVEGA/Saturn and Uranus missions.

2.3.6 Handbook Organization

Payload performance results and basic transfer characteristics are organized by final target and mission type in the sections which follow. There are eight of these sections:

Jupiter Flybys
Jupiter Orbiters
Saturn Flybys
Saturn Orbiters
Uranus Flybys
Uranus Orbiters
Neptune Flybys
Pluto Flybys.

Note that organization by final target means, for example, that Jupiter/Neptune, Saturn/Uranus/Neptune and Uranus/Neptune missions are all to be found in the section titled "Neptune Flybys." Each of these sections is tabbed, and has its own pagination for referencing convenience. Within these sections, a consistent pattern of organization is followed. It begins with an introductory subsection which briefly describes the mission alternatives presented, lists the launch opportunities, presents a summary of payload performance sensitivity to launch opportunity and defines the propulsion options considered. The remainder of the section contains payload performance data organized by launch opportunity and mission type. Each new opportunity and mission type is set off by a colored page. The launch year is cited in the upper corners of each page of performance results as a quick reference aid.

The specific format and amount of data presented vary with the type of mission considered. For flyby missions just one graph is presented for each launch opportunity. It presents the tradeoff of net swingby payload for trip time to the target planet. For gravity-assisted swingby missions (outer planet swingbys), three graphs are shown for each opportunity: (1) net payload versus flight time to final target; (2) trip time to intermediate target(s) as a function of trip time to final target; and (3) swingby miss distances versus trip time. More extensive data are presented for orbiter missions in order to examine payload trades over various orbit sizes. Each opportunity is first characterized by a graph of net orbited payload versus trip time. (A reference orbit and retro stage are chosen for this purpose.) Then, several subsidiary tables show payloads for variation in launch vehicle, retro stage and orbit size.

An example of the tabular format appears as Table 7. Here, the 1985 Saturn Orbiter is shown, using the IUS(Twin)/Spinner over a 10-day window with space-storable retropropulsion at Saturn. An impulse budget of 100 m/sec is allowed for midcourse retargeting and orbit trims. Note that the flight mode is direct--as opposed to VEGA or ΔVEGA. The top half of the table shows net orbited payload as a function of trip time for a variety of orbits, specified by period and periapse radius (in Saturn radii). For example, in order to orbit a payload of 500 kg safely above the ring system (i.e., at 4 Saturn radii), it is necessary to allow for a flight time of 1800 days--nearly 5 years. The orbit period has little effect on payload here, so any of the three may be chosen, consistent with other mission design requirements. The bottom half of the table shows the required size of the chosen retro system needed to deliver the corresponding payload from the top half of the table. The sum of these two masses equals the injected mass at Earth departure. The asterisk below the last column in Table 7 indicates that the figures in that column represent satellite-assisted capture performance.

All of the performance graphs of net payload versus trip time show several curves, each identified by a number which may be found in the fold-out Launch Vehicle Glossary at the rear of the volume. A set of appendices presents extensive working graphs for the mission analyst. Instructions for use of these data are contained in the appendices.

SATURN ORBITER MASS PERFORMANCE

FLIGHT MODE	DIRECT
LAUNCH VEHICLE	SHUTTLE/IUS(TWIN/SPINNER)
LAUNCH WINDOW	10 DAYS
RETRO SYSTEM	SPACE-STORABLE: ISP = 370 SEC
EXCESS DV	100 M/SEC

*** NET USEFUL PAYLOAD (KG) ***

PERIOD (DAYS)	TRIP TIME (DAYS)	(YRS)	ORBIT PERIAPSE RADII (PLANET RADII) 1.1	2.0	3.0	4.0	6.0	8.0	20.2
90.0	1000	2.74	77	31	2	0	0	0	0
90.0	1200	3.29	266	204	159	127	83	52	10
90.0	1400	3.83	444	380	330	293	239	200	155
90.0	1600	4.38	569	510	463	426	371	329	300
90.0	1800	4.93	649	595	551	517	464	423	410
120.0	1000	2.74	78	33	4	0	0	0	0
120.0	1200	3.29	269	207	163	131	87	57	14
120.0	1400	3.83	449	385	337	300	246	207	163
120.0	1600	4.38	575	517	471	435	380	340	312
120.0	1800	4.93	655	602	560	527	475	435	424
150.0	1000	2.74	79	34	5	0	0	0	0
150.0	1200	3.29	271	210	166	134	90	60	15
150.0	1400	3.83	452	389	341	304	251	213	166
150.0	1600	4.38	578	521	476	440	387	347	316
150.0	1800	4.93	659	608	566	533	483	444	429

*** RETRO STAGE MASS (KG) ***

PERIOD (DAYS)	TRIP TIME (DAYS)	(YRS)	ORBIT PERIAPSE RADII (PLANET RADII) 1.1	2.0	3.0	4.0	6.0	8.0	20.2
90.0	1000	2.74	335	380	410	0	0	0	0
90.0	1200	3.29	365	427	471	503	548	578	620
90.0	1400	3.83	348	412	462	499	553	592	637
90.0	1600	4.38	317	377	424	461	516	557	586
90.0	1800	4.93	293	347	390	425	478	519	532
120.0	1000	2.74	334	379	408	0	0	0	0
120.0	1200	3.29	361	423	467	499	544	574	616
120.0	1400	3.83	343	407	455	492	546	585	629
120.0	1600	4.38	312	370	416	452	506	547	575
120.0	1800	4.93	287	339	381	415	467	506	517
150.0	1000	2.74	333	377	407	0	0	0	0
150.0	1200	3.29	359	421	465	497	541	571	615
150.0	1400	3.83	340	403	451	488	541	579	626
150.0	1600	4.38	308	366	411	446	500	540	571
150.0	1800	4.93	283	334	376	408	459	498	512

SATELLITE-ASSISTED ORBIT INSERTION: *

2.3.7 Analysis Summary

Initial interplanetary transfer analysis was performed largely with two computer codes--MULIMP for ballistic trajectories, and CHEBYTOP II for low-thrust. Values shown for all trajectory parameters are taken from the optimum Type I transfer, except for launch energy (C3), which is penalized according to the extent of the chosen launch window. The performance data based upon these trajectories are generally conservative estimates of net payload. In many cases, detailed trajectory and propulsion analysis at the Phase-A mission study level will yield improved performance.

2.4 Multiple Discipline Science Assessment (984 man-hours)

The planning of most planetary missions is based only upon objectives for planetary science. This report takes a general look at other science disciplines to determine where and when it is appropriate to include them in the planning of planetary missions. Some specific examples of multiple discipline opportunities are then selected and for each a brief description of the mission characteristics is given.

2.4.1 Identification of Science Objectives

There are many science disciplines that can profit from observations made in space. The search for those science opportunities which can be accomplished together with the study of planets considered the general disciplines of astronomy, physics, the geosciences and the applied sciences. In astronomy, there is interest in both solar observations and in views of stars and galaxies. The areas of physics which deserve special mention are solar physics, cosmology and gravitational physics. The geosciences and the applied sciences are included because they may benefit from the technological developments needed for future planetary missions.

There are three types of commonality of interest that can unite planetary objectives with the other sciences. The first and simplest is the case of using a planetary mission spacecraft to carry out an experiment for the other discipline. Particle and field observations illustrate this type of commonality well. The second type is the use of a single vehicle to deliver two (or more) spacecraft to their targets. It is not unusual for a single launch vehicle to place several satellites into Earth orbit. However, no attempt has yet been made to send two spacecraft to Earth escape with a single delivery system. All such opportunities would require one standard ballistic launch and may also involve common use of low-thrust propulsion during an interplanetary trajectory. A search method based upon science objectives is applied

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to the cases where a single spacecraft can be used and to opportunities using a single delivery vehicle. The last type of commonality is based upon joint use of a system. The interest here is upon complete systems such as atmospheric probes, rovers, etc., and not on a subsystem such as attitude control, propulsion or a science instrument.

2.4.2 Characteristics of Promising Opportunities

The most promising opportunities identified during this study are listed in Table 8 and described briefly below.

Mercury Orbiter. A spacecraft in orbit about Mercury can acquire unique data on the Sun and on relativistic gravitational effects. The advantages for solar observations are a five to ten times greater solar flux, a longer time for observing individual features and the possibility of using Mercury as an occulting disk for coronal studies. From a much more accurate determination of Mercury's orbit, information on the internal structure of the Sun can be derived and tests can be made on relativistic gravity theories.

To accomplish any or all of the above objectives there must be changes in the set of instruments, in the spacecraft systems or in the spacecraft operations. For example, any useful solar observations will require several instruments with high spatial and/or spectral resolution to investigate the disk and the corona at visible and ultraviolet wavelengths. Another desirable instrument is a neutron detector, since the flux of solar neutrons is greatly attenuated at the Earth by radioactive decay. To determine Mercury's orbit accurately either the spacecraft must be "quiet," i.e., at least for several orbits on a regular basis, there must be no unknown forces acting upon it. Some improvements in spacecraft tracking procedures and equipment may be needed.

The additional mass for science instruments is estimated to be between 30 and 90 kg and that for spacecraft subsystems other than

Table 8

PROMISING MULTIPLE DISCIPLINE SCIENCE OPPORTUNITIES

Type of Commonality	Planetary Use	Additional Disciplines	Relevant Observations
Single spacecraft	Mercury Orbiter	Solar astronomy; Gravity physics	Solar images; Relativistic effects
	Mars Sample Return	Solar physics; Applied science	Collection of samples exposed to solar particles
	Neptune or Pluto Flyby	Solar physics; Stellar astronomy	Interstellar neutral H and He; Magnetic field, cosmic rays
	Any Mission	Solar physics; Stellar astronomy	Fields, particles; Gamma ray bursts
Single launch vehicle	Mercury Orbiter	Solar astronomy; Solar physics	Solar images from 0.2 AU synchronous orbit
	Mars Orbiter	Solar astronomy; Solar physics	Solar images and particles from 90° orbit
	Neptune or Pluto Flyby	Solar astronomy; Solar physics	Solar data down to 0.02 AU or from 90° orbit
Single system	Atmospheric Probe	Geoscience	Upper atmosphere structure, composition
	Remotely Piloted Vehicles	Geoscience	Atmosphere structure, composition

propulsion is taken as 15 to 45 kg. If the orbiter is in a circular polar orbit, 25 to 75 kg of additional propellant is required to put the extra mass into orbit. For an elliptical orbit, the added propellant is only 5 to 15 kg. The resulting spacecraft has relatively complex pointing requirements since it must point instruments at the Sun and Mercury, communicate with the Earth and control its temperature.

Mercury orbiter missions are delivered by low-thrust propulsion systems for which an increase in the required net mass on approach can be achieved with a longer flight time. Typically this sensitivity is 0.3 days per kilogram although a specific case could be up to a factor of two different. Thus, the additional flight time for 30 to 90 kg of additional multiple discipline science is 15 to 45 days in the elliptical orbit case and 21 to 63 days if the spacecraft is in a circular orbit.

Mars Surface Sample Return. The multiple discipline opportunity that can be easily combined with a Mars Surface Sample Return (MSSR) mission is the return of samples exposed to the deep space environment. These samples would be carefully selected and prepared so that they could be used for studies of solar wind ions, solar flare particles and micrometeoroids, and for investigations of the effects of deep space environments on materials. Analyses would be done using the many powerful techniques available in Earth-based laboratories. The samples would be deployed while in deep space and retrieved prior to capture in Earth orbit. This experiment should be part of an Earth return vehicle (ERV) that stays in orbit about Mars.

The nominal experiment returns 5 kg of samples and adds about 20 kg to the net mass of the ERV at launch. It is recommended that an additional 20 kg be allocated to a package of particles and field instruments to measure the interplanetary environment to which the samples are exposed. As an example, the overall increase in the injected mass of the ERV for a 1988 MSSR mission is about 325 kg for the nominal sample.

Each additional kilogram of sample requires about 20 kg of injected mass. This opportunity is generally easily accommodated within dual launch concepts for MSSR for which the ERV usually has more than a 325 kg margin.

Neptune or Pluto Flyby. A spacecraft on a Neptune or Pluto Flyby mission offers an excellent opportunity for study of the interstellar medium and its interaction with the solar wind. These objectives require observations during the cruise phases of the mission, particularly after the planet encounter. It is presumed that most of the necessary instrumentation is already included for the purpose of measuring the interplanetary particles and fields. Some increases may be needed in the sensitivity and/or the energy ranges of these instruments. One obvious additional experiment is a detector for neutral atoms and molecules. Its impact on the spacecraft is negligible. The inclusion of this objective implies that the mission duration should be as long as possible--limited only by spacecraft reliability or communication capability. Some subsystem modifications may be advised.

Jupiter swingby trajectories are the best choice--the relevant opportunities are 1990, 1991, 1992 and 1994 for Neptune and 1989, 1990 and 1991 for Pluto. Flight time can also be decreased by using a larger launch vehicle, e.g., a Tug instead of an IUS, but this is not always advantageous because the faster trajectory has an unallowable swingby distance below Jupiter's surface. In addition, some restrictions on the planetary encounter may be necessary to put the spacecraft on a post-encounter trajectory that escapes the solar system at a rapid rate. Obviously, the fastest trajectory to Neptune or Pluto is desired.

Mercury Orbiter and Synchronous Solar Observatory Missions. A single low-thrust propulsion system can deliver an orbiter to Mercury and a solar observatory to a 0.20 AU circular orbit. A spacecraft in a circular orbit at 0.20 AU has an orbit period of about 30 days which is also the rotation period of the solar photosphere. Thus, this spacecraft can observe continuously any feature on the solar disk or in the solar corona. To accomplish this objective requires using instruments with high spatial and/or spectral resolution and wide spectral range. There are also opportunities as the Earth-Sun-spacecraft angle constantly changes for stereo observations of the Sun. This orbit puts severe stress on the thermal control subsystem, but there is no reason to believe this problem cannot be solved. It is expected that the science instrument package would be 100 to 200 kg and that the spacecraft would be 600 to 800 kg. The latter may be reduced some if the low-thrust system can be retained to provide power and some attitude control functions

These combined missions can be performed by the 60 kw Ion Drive low-thrust propulsion system studied for Comet Halley Rendezvous but without a concentrator. While the flight times are longer than for either mission alone, it is possible to do many combinations. For example, the single circular orbiter for Mercury (1200 kg) and a 600 kg solar observatory required flight times of 500 and 775 days, respectively, when an IUS(Twin) is used, and only 420 and 690 days when the Tug(R)/EE-Kick is used. Alternatively, the Tug(R)/EE-Kick allows the 1200 kg and 500 day Mercury mission to be performed in conjunction with a 950 kg and 830 day solar observatory. Useful payloads can be delivered using a somewhat less powerful and less advanced low-thrust system, but the flight times are longer.

Missions to Planets and a Solar Polar Observatory. The out-of-the-ecliptic missions considered here result in highly inclined (>50°) circular orbits. The purpose of such a mission is to study the structure of the Sun and of interplanetary space as a function of solar latitude. There are also some possibilities for stereoscopic solar imagery and for low background astronomical observations. This solar polar observatory would have high spatial and/or spectral resolution instruments covering most, if not all, spectral regions. Total spacecraft mass is expected to be about 800 kg, including 200 kg of science. The planetary missions considered are a Mars Orbiter and Jupiter Swingby missions to Neptune or Pluto. The Mars Orbiter could be a geophysical orbiter, perhaps including penetrators or supporting some surface system. The Mars approach mass for these options would be 1200 to 1600 kg. The Neptune mission is assumed to include an atmospheric probe for a total mass of 800 kg while the Pluto case employs only the 600 kg flyby spacecraft. Swingby opportunities to Neptune and Pluto begin in 1990 and 1989 respectively, as cited above.

After injection to Earth escape by an IUS(Twin) and delivery of a 1600 kg Mars Orbiter, Ion Drive (60 kw) can then take an 800 kg payload to an inclination of 53° (the orbit period is 1.88 years). The overall flight time is about 1000 days. Reducing the Mars Orbiter to 1200 kg results in an inclination of 62° at about 1250 days. The Jupiter swingby mode can easily give a 90° inclination. Using a Tug(R)/EE-Kick for injection to Earth escape gives a solar observatory mass of 640 and 760 kg for the Neptune and Pluto missions respectively. These payloads could be increased by increasing the flight time beyond the 1280 days considered here or by going to a circular orbit larger than 1.0 AU. It may be possible to do these missions with a smaller and less advanced low-thrust system, but a longer flight time is needed to increase inclination after Mars encounter or to provide adequate payload at Jupiter via a SEEGA trajectory.

Missions to Planets and a Solar Probe Mission. Another interesting mission for investigations of the Sun is the Solar Probe mission which goes to a perihelion of 0.02 AU. This can only be done using a Jupiter swingby where there is again the opportunity to send a second spacecraft on to Neptune or Pluto. The choice of perihelion allows *in situ* study of the solar corona at 4 solar radii and offers reasonable hope for a technical solution of the thermal control problem. Significant information is also obtained on the solar gravitational potential and effects predicted by relativistic gravitational theories. Both remote sensing optical instruments and in situ particle and field instruments are desired. The range of science payloads is assumed to be 50 to 100 kg resulting in a spacecraft mass of approximately 600 to 800 kg. About 25% of this is the mass of the heat shield used for thermal control.

Both ballistic and low-thrust trajectories can be considered for these opportunities. The nominal ballistic missions are easily done by the Tug(E)/EE-Kick with flight times of about 2.4 and 7.2 years to the Sun and either Neptune or Pluto, respectively. This Tug vehicle can do both missions and deliver more payload in less time than can be done for any single target by a single IUS vehicle. The orbit period of the solar probe is 5 years, typically, so the mission may be limited to only one solar encounter. Ion Drive (60 kw) can be used after the Jupiter swingby to reduce the period of this orbit to between 1.2 and 2.0 years depending on the spacecraft mass and the launch vehicle.

Atmospheric Probe. Atmospheric probes have been used or are planned for planetary studies at Venus, Jupiter, Saturn and Titan. In all cases the major objective is to obtain a vertical profile of basic in situ data on the structure and composition of the atmospheres. The average properties of the Earth's atmosphere are well-known. Variations are studied using aircraft, balloons and sounding rockets. In the future, this detailed vertical structure information for both the Earth and the planets could be obtained with atmospheric probes. The proposed concept for using probes at Earth is based upon delivery of many probes to orbit as a partial Shuttle payload and recovery of all systems for subsequent reuse. This is necessary to make this approach cost competitive with other ways to obtain similar data. If such a concept can be developed, then a new technology base is established which reduces the design and construction expenses for subsequent, more sophisticated planetary probes.

Remotely Piloted Vehicles. A remotely piloted vehicle (RPV) could be used in the exploration of a planet with an atmosphere, particularly

Venus and Mars. The airbreathing RPV is now a well-developed concept for both military and civilian applications on the Earth. The preferred system would be designed to operate at a pressure of 5 mbar and can be used at an altitude of about 40 km on the Earth or near the surface of Mars. At Mars, the RPV could be used to study the atmosphere, obtaining horizontal profiles of its properties, to look at the surface with high spatial resolution remote sensing instruments or to transport small payloads, such as surface samples or small experiment packages. Conceptual designs of airplanes to operate at thin atmospheres for long durations are characterized by high life-to-drag ratios and large dimensions--the same characteristics found in gliders. For common applications, the source of power must be able to operate in a CO_2 atmosphere (e.g., a hydrazene engine, a primary battery or a nuclear thermal generator).

Fields, Particles and Gamma Ray Bursts. Particles and field observations have frequently been included in the scientific payloads of planetary missions. There is a continuing need for particle and field data at heliocentric positions other than that occupied by the Earth. Particle data are desired for the solar wind ions and electrons, the solar flare particles and the low-energy cosmic rays. Field data consist of the magnetic field, the electric field and the electromagnetic waves generated by local plasma phenomena and by remote sources, especially the Sun. Various instruments are available to perform these measurements. There are missions like Pioneer Venus '78 with limited capabilities using three instruments weighing only 5 kg and also missions like Voyager capable of measuring all the above properties with six instruments weighing almost 40 kg. Thus, when planning future planetary missions, 10 to 25 kg of the science payload should, if possible, be allocated for particle and field instruments.

The locations of the recently discovered gamma ray bursts are determined by triangulation using time of arrival data. Two (or more) detectors on planetary spacecraft can be used to determine accurate source locations for identification with known astronomical objects. Such an instrument need not be large; the Pioneer Venus device is only 2.4 kg.

2.4.3 Conclusions and Recommendations

The most favorable opportunity appears to be the Mercury Orbiter mission with solar/gravity science. It and the other single spacecraft opportunities do not require significant advances in spacecraft or propulsion technology. In general, the single launch vehicle opportunities require advanced propulsion systems and/or SEEGA trajectories. Additional study is recommended to determine feasibility of the Solar Probe mission, the atmospheric probes for the Earth and RPVs for Mars.

2.5 Asteroid Exploration Study (781 man-hours)

A major objective of this investigation is to assess asteroid mission concepts with emphasis on the multiple-target rendezvous scenario. The lack of such an evaluation was identified by the Terrestrial Bodies Science Working Group (TBSWG) as a current deficiency which prevents meaningful comparisons with other proposed planetary missions. It is expected that this study will help to remove this deficiency by a definitive analysis of exploration goals and strategies coupled with mission requirements and performance capabilities.

This multi-faceted study is currently in progress and is scheduled across the second year of the present contract period. Completion is anticipated for early summer 1978 so that study results may be presented to this year's COMPLEX meeting. To aid this progress summary, Table 9 shows a preliminary outline of the final report which serves also as a study task outline. A checkmark indicates subtasks completed and written up in draft form, an asterisk indicates subtasks in progress, and unmarked items denote work not yet initiated. Approximately 35% of the total scheduled effort is finished at this time. The following paragraphs summarize some of the key elements of the asteroid exploration study.

2.5.1 Exploration Strategies

The asteroids are distinct bodies with many differences among them. The main questions of scientific interest, as listed in Table 10, relate to comparisons among the different types of bodies and to their distributional characteristics and physical properties. To the extent feasible, space missions should be sent to as wide a variety of bodies as possible and make the widest variety of measurements. Variables of importance to target selection include diameter, composition (e.g., spectral class), and semimajor axis. Since no set of missions can hope to visit representatives of all types of asteroids, it is clear that maximum use must

Table 9

ASTEROID EXPLORATION STUDY REPORT OUTLINE (PRELIMINARY)

SUMMARY

VOLUME I

✓ 1. INTRODUCTION

✓ 2. EXPLORATION STRATEGIES

- 2.1 Current Knowledge of Asteroids
- 2.2 Goals of Exploration
- 2.3 Target Selection Criteria
- 2.4 Definition of Mission Modes
- 2.5 Selection of Study Strategies

✓ 3. SCIENCE OBJECTIVES

- 3.1 Physical Properties
- 3.2 Geochemical Composition
- 3.3 Thermal Characteristics
- 3.4 Surface Properties
- 3.5 Relation of Mission Science Objectives to Asteroid Knowledge and Solar System Exploration Goals

4. MEASUREMENTS AND INSTRUMENTATION

- * 4.1 Measurement Definition and Analysis
- * 4.2 Candidate Remote Sensing Instrumentation
- * 4.3 Candidate In-Situ Instrumentation

5. TARGET ENCOUNTER ANALYSIS

- 5.1 Optical Recovery/Approach
- * 5.2 Remote Sensing Position Requirements
- * 5.3 Station-Keeping/Circumnavigation and Orbit Maneuvers
- 5.4 Penetrator (Hard Lander) Deployment
- * 5.5 Mass and Gravity Field Determination

6. PAYLOADS

- 6.1 Candidate Science Payloads
- 6.2 Science Payload Characteristics: Mass, Power Requirements, Data Requirements, Volume

Table 9

REPORT OUTLINE (continued)

VOLUME II

7. TRAJECTORY REQUIREMENTS AND PERFORMANCE

* 7.1 Candidate Propulsion Systems
* 7.2 Requirements Inferred from Asteroid Orbit Distributions
  7.3 Requirements for Selected Targets

8. EXAMPLE MISSION DESIGN SUMMARIES

8.1 Multi-Asteroid Survey Missions
8.2 Sample Return Missions
8.3 Program Cost Estimates

9. CONCLUSIONS/RECOMMENDATIONS

Table 10

ASTEROID SCIENCE OBJECTIVES

1. Testing and verification of inferences from ground-based studies
2. What are the genetic relationships among the various types of asteroids and other small bodies?
3. How did the nature of solar nebula condensates vary as a function of position in the early solar system?
4. What do asteroids reveal about early solar system environments?
5. Do the asteroids portray a tableau of the early processes of planetesimal accretion?
6. Why has the thermal evolution of various asteroids differed?
7. Nature of collisional evolution
8. Nature of orbital dynamic evolution

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9. Assessment of the potential for future economic utilization

be made of ground-based and Earth-orbital techniques for extrapolating the close-up measurements to the large number of remaining objects.

The kinds of space missions that might be accomplished can be broken down into four classes: (1) flyby, (2) rendezvous (orbiter), (3) docking (lander), and (4) sample return. Various combinations are also possible; for example, an orbiter which releases a small soft-lander or surface penetrator. Of these classes, flyby at speeds greater than 1 km/sec is deemed to be of little scientific merit because of the short stay time, inadequate spatial resolution, and insensitivity to many important remote sensing measurements. That is not to say that multiple fast flybys would be worthless compared to not going to an asteroid at all, but only that it yields limited science return relative to the rendezvous mode. Given the expectation of advanced propulsion technology (Solar Electric Propulsion) being available by the mid-1980's, the rendezvous mode strategy is both feasible and much preferable.

Multiple-target rendezvous has been selected as the baseline exploration strategy for this study. The primary candidate targets are taken to be those in the main belt (2.1 to 3.5 AU), although the closer Apollo/Amor bodies and the more distant bodies near Jupiter are not necessarily excluded from consideration. The outliers may be included as targets of opportunity when feasible. Target selection criteria is not at all straightforward, but certain guidelines based on current knowledge may be established.

1. A variety of compositional types should be visited, especially representatives of the apparently primitive asteroids (the C-type) and of thermally differential classes (those believed to have metallic or chondritic compositions, such as the M-, S- and E-types).

2. A distribution of sizes is also important: the largest asteroids contain most of the mass in the system, are most likely to be original accretions as distinct from fragments, and are likely to have experienced the most complex planetary processes. Small asteroids, on the other hand, will most likely provide pristine rocks relatively unaffected by processes of thermal modification of regolith burial. Carefully selected examples

of small asteroids, in a Hirayama family for instance, may reveal interior properties of the precursor body.

3. Variation in orbital distance is important for sampling the changes which occurred in solar nebula condensates. One possible mission scenario is to start at a target in the inner fringe of the main belt and progress gradually toward the outer fringe. This policy is also compatible with minimizing trajectory energy requirements.

4. One of the larger asteroids, such as Ceres or Vesta, might be included in the multi-target mission, although not necessarily as the first target.

5. The study should provide data which show the requirement tradeoffs between deployment of small landers at each target versus the addition of more rendezvous targets in the mission sequence.

Analysis of TRIAD file data has led to a comparative classification (Priority 1, 2 and 3) of C-, M-, U- and S-type targets. The priority grouping reflects the quality of current observational knowledge more so than any intrinsic weighting of scientific interest. However, this information serves in lieu of any other culling out process and will be utilized for purposes of specific target searches.

Sample return missions permit the full array of sophisticated laboratory techniques developed for analysis of Moon rocks to be applied to asteroid material. Provided that samples are selected with sufficient care to ensure high scientific potential and, in particular, that they are not equivalent to an already existing meteorite sample, this mission mode could greatly extend our insight of the nature of the bodies from which the samples were taken. It is hard to judge from our present vantage point whether the scientific merit of a single target asteroid sample return justifies the expected high cost of such a mission. There is one possible exception worth noting: that is, if the nation embarks on a program of exploitation of extraterrestrial resources, a precursor sample return mission is likely to be essential. The most probable targets of exploitation are the nearby Apollo/Amor asteroids. Hence, this study will investigate the sample return requirements but only for one or two examples of near asteroid targets.

2.5.2 Science Objectives and Measurements

The goals of exploration outlined in Table 10 are broad, far-reaching objectives in contrast to measurement data obtained from specific experiments chosen to maximize the scientific return of the particular mission. We address the relation between science objectives and experiment/mission modes to provide an assessment of how well mission science meets the goals of asteroid exploration. The eight science questions fall into four broad areas and may be related to specific experiments as shown in Table 11. That a specific experiment is relevant to a particular question does not mean that the measurement will "answer" the question but only that the interpretation of experiment results are relevant to that particular question. A table such as this is most useful for identifying gaps and redundancies in the mission science relationship to mission objectives.

These measurement techniques have been evaluated for relative effectiveness on the various mission classes. For the remote sensing techniques (i.e., imaging, reflectance spectroscopy, etc.) a rendezvous (orbiter) mission is the clear choice over a flyby mission. When a penetrator or hard lander is incorporated into a rendezvous mission many in situ techniques (α-scattering, seismometry, etc.) can be supported and this also increases the scientific potential of the asteroid mission. Candidate instruments will be described based upon detailed requirements for these measurement techniques*. When possible, the candidate instruments will be based on existing design concepts.

Table 11

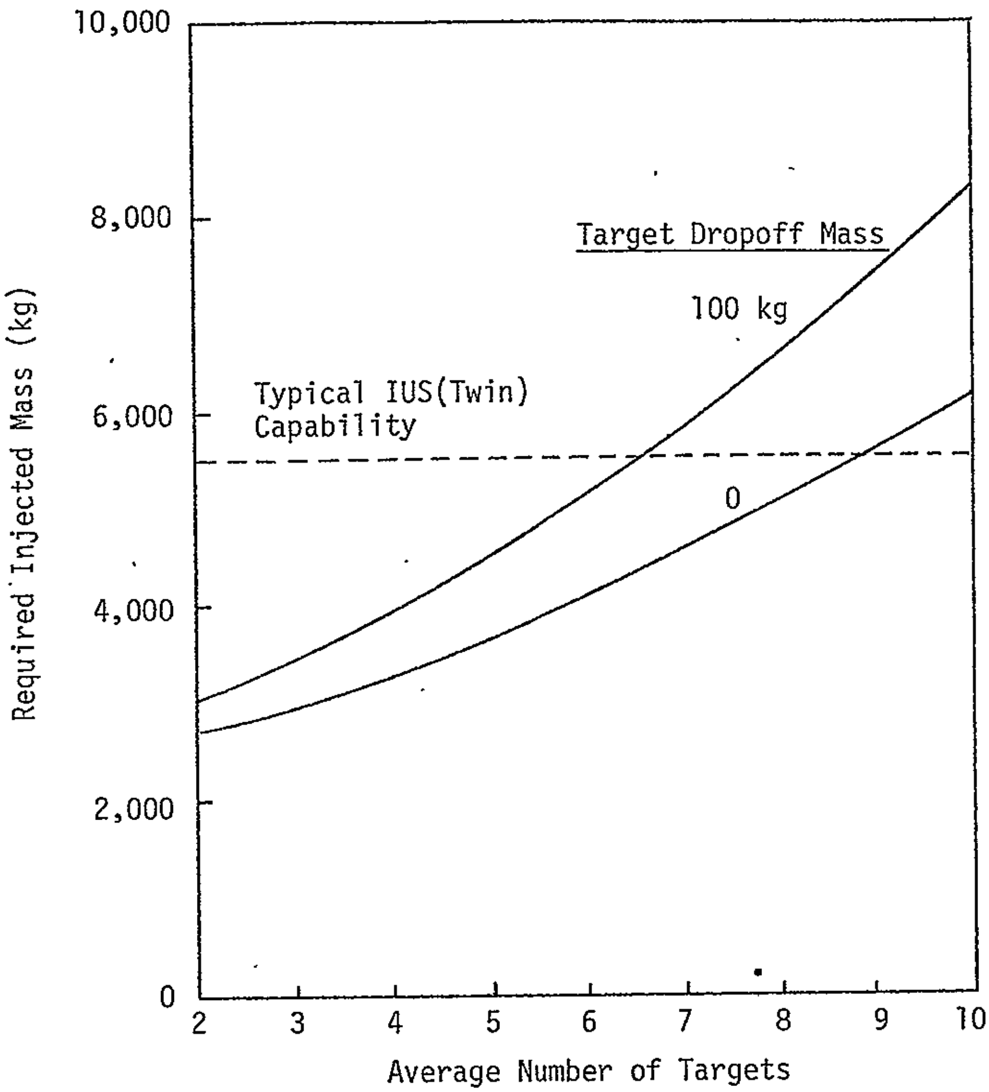
RELATION OF SCIENCE EXPERIMENTS TO ASTEROID SCIENCE GOALS

Experiment/Measurement Technique	Measured Parameter	Science Objective Defined in Table 10							
				Asteroids on Probes of Early Solar System			Origin and Evolution of Asteroids		
		1	2	3	4	5	6	7	8
Radio Tracking	Mass Mass Distribution	x	x					x	
Imaging	Surface Morphology Surface Properties Size and Shape (vol) Phase Function	x	x		x	x	x	x	
IR-Visible Reflectance Spectroscopy	Surface Mineralogy	x	x	x	x		x	x	x
UV Spectroscopy	Surface Mineralogy	x	x	x	x		x	x	x
Gamma Ray Spectroscopy	Near Surface Elemental Composition	x	x	x	x		x		
X-Ray Fluorescence	Surface Elemental Composition	x	x	x	x				
Magnetometer	Magnetic Field	x	x		x		x		
Heat Flow	Heat Flux at Surface	x	x	x	x		x		
Seismometer	Internal Structure	x	x	x	x		x	x	
Thermal IR	Soil Parameters	x						x	
α-Scattering	Surface Elemental Composition	x							
Surface Imaging	Detailed Surface Structure	x							

2.5.3 Trajectory Analysis

It is understood from the start that the conventional ballistic flight mode using chemical propulsion is very ill-suited to asteroid rendezvous. Transfer from Earth to a typical target in the main belt requires a launch energy C3 of about 40 $(km/sec)^2$ and a rendezvous ΔV of about 5 km/sec. Average ΔV requirements for subsequent asteroid-to-asteroid rendezvous are of the order of 3 km/sec. With launch capability limited to the Shuttle/IUS(Twin) vehicle, it is not impractical to accomplish single target rendezvous (at the expense of a very large spacecraft retro system), but multi-target rendezvous is virtually impossible.

Solar Electric Propulsion, particularly the recently proposed systems which utilize array concentrators, does offer the needed performance for multi-asteroid rendezvous, as well as for sample return missions. To illustrate this point, consider Figure 6 which shows the injected mass required to rendezvous, on average, with N main belt targets. (These results are only tentative and will be refined as part of the trajectory analysis task described below.) The typical launch capability of the IUS(Twin) is 5500 kg at C3 = 7 $(km/sec)^2$. An average of six targets are accessible assuming a 100 kg dropoff (lander) at each target. Two additional targets may be picked up if landers are not deployed at any asteroid. These results assume a 40 kw propulsion system (with concentrators) weighing 1300 kg exclusive of propellant. Such a design does represent advanced rather than current SEP technology. For a more conservative design weighing 2000 kg, the average number of targets decreases to four or five. A further decrease to only two or three targets would be associated with a SEP system without concentrators. An important fact to note is that the time interval between encounters is about 1.5 years in order to obtain reasonable mass performance. Hence, a six-target sequence would take about 9 years to complete. This lifetime "burden" must be borne by the science investigation team as well as the subsystem reliability design.



Propulsion System	1300 kg (40 kw, CF = 3.2)
Mission Module	500 kg
1st Target M_F/M_O	0.75
Subsequent M_F/M_O	0.90

FIGURE 6. CAPABILITY OF ADVANCED SOLAR ELECTRIC PROPULSION FOR MULTIPLE ASTEROID RENDEZVOUS MISSIONS

The trajectory analysis task in progress is comprised of two parts, both of which are expected to yield useful results in characterizing performance capability. The first subtask involves a new approach based on statistical analyses of asteroid orbit elements and the requirements imposed on orbit-to-orbit transfers. An asteroid survey computer program is being developed which will generate the statistical distributions of main belt asteroid orbits under a variety of conditions relating to spectral classification and diameter limitations. This information will then be used in a Monte Carlo analysis to obtain the distribution of transfer propellant requirements. The final desired result is a set of probability curves associated with specified conditions on launch vehicle, propulsion system, required payload, flight time, etc. These curves will state, under the given conditions, the probability of achieving rendezvous with N targets over the total spectrum of launch opportunities, i.e., assuming a random launch date. The importance of this kind of information to mission planners is that it provides a measure of what is possible to achieve both in terms of average performance and extreme bounds. Such results will also be useful as a global reference when assessing the performance of specific targeted missions.

The second subtask in the trajectory analysis involves search and generation of representative examples of multi-target mission scenarios. This could be a very time-consuming effort compared to the statistical approach, because the combination and permutation of candidate target sequences is virtually unlimited. It is therefore necessary to place some practical constraints on this task. Some relevant guidelines have already been mentioned in the discussion of exploration strategies. We have set as a goal the generation of 10 specific targeted sequences covering the launch opportunity period 1985-1994. Generally, one example for each launch year will be sought; however, in some instances it may be of interest to examine targeting variations (branching options) about a nominal case. The example missions obtained in this study, taken together with other examples expected from concurrent JPL studies, should provide mission planners and science investigators with sufficient data for, at least, tentative decision-making purposes.

2.6 Mars Strategy Study (1107 man-hours)

The primary objective of this study is to devise a framework for evaluating the relative merits of approaches to future Mars exploration. This framework has two basic aspects, one essentially scientific and one essentially technical. The scientific aspect is the assembly of a set of fundamental but achievable goals for new knowledge to be acquired, taking into consideration the known complexity of Mars, and the power of analytical techniques available today. The more technical aspect of the framework is an analysis of the possible interplay of separable exploration modes; orbital science, network science, mobile laboratory science, and returned sample science are considered here. We ask how execution of one mode can aid the design and execution of a succeeding type, how its results can clarify interpretation of other data and how missions using multiple modes simultaneously can be more scientifically productive. Mission strategies are identified that include sample return as the primary goal and that have the highest scientific potential. It is recommended that detailed performance and cost data be generated for these strategies.

Originally a three phase study was contemplated. This summary describes the analyses performed during the first phase. A draft report of this work has been exposed to critical review by the following scientists and engineers active in planetary mission planning: A. A. Albee (Cal. Tech.), L. Friedman (JPL), H. Masursky (USGS), J. Minear (NASA/JSC), T. A. Mutch (Brown U.), and J. M. Papike (SUNY/Stony Brook). In the second phase, the comments and recommendations of the reviewers are to be incorporated into a revised report. The revision is expected to include more detail in the analysis of basic science objectives for a Mars surface analysis mission and in the assessment of various technical approaches to meeting these objectives. The planned third phase included analysis of mission performance and cost estimates for recommended missions and strategies. This phase is now decoupled from the other phases. One aspect of this work is discussed in Section 2.1.19.

2.6.1 Scientific Objectives

Our framework for discussing scientific objectives for Mars exploration (Table 12) includes investigations of planetary formation, endogenic processes, atmosphere-lithosphere interactions and exogenic processes, and is based upon previous work on the application of planetary missions to the problems of planetary origin, evolution and solar system history. However, emphasis is given here to studies for which surface samples contain a record of the processes of formation and evolution. Stress is also placed on processes of global significance rather than on local geological problems.

2.6.2 Analysis of Surface Science Missions

The choice of mission modes and the sequence in which they are launched can have important implications for the scientific return of a Mars exploration program. The most significant interactions are: (1) orbital science is needed to select sites and to support surface science (both mobile laboratory and sample return), (2) mobile laboratory surface science enhances the value of sample return science. An orbital science mission should precede surface science if it is to be used to achieve complete flexibility in landing site selection and to make changes in the science payload and methods of data analysis and interpretation. Orbital science should be done in conjunction with surface science to determine whether conditions at potential landing sites are appropriate for a landing and for making simultaneous orbital and surface observations of dynamic phenomena. The value of network science appears to be independent of the types of surface science on the same or earlier missions, but can be enhanced by simultaneous orbital observations of atmospheric dynamics.

Mobility is an important factor in the design of a surface science mission. From a scientific point-of-view, a mobility range of at least 100 m (e.g., a tethered rover) is required for a surface science mission to any landing site. This assures that the samples and rocks analyzed

Table 12

SCIENTIFIC FRAMEWORK FOR FUTURE MARS EXPLORATION IN POST-VIKING ERA

INVESTIGATIONS OF PLANETARY FORMATION AND ENDOGENIC PROCESSES

Planet Formation

1. To determine the physical properties and chemical composition of the hypothesized solar nebula when the material of Mars condensed and accreted.
2. To assess the state of the planet during later stages of accretion.

Planetary Structure

1. To assess the state of planetary differentiation involving major separations into core/mantle/crust/atmosphere.
2. To determine when this differentiation occurred.
3. To examine the nature of differentiation processes in the crust and upper mantle and their time scale.
4. To study the present state of the planet's interior.

Atmospheric Evolution

1. To determine the degassing history of the martian interior.
2. To measure the present abundances and distribution of volatiles in the atmosphere and lithosphere.
3. To assess the influence of atmospheric escape processes on atmospheric evolution.

Igneous Petrogenesis

1. To define the processes involved in igneous rock formation.
2. To characterize geochemically the different types of volcanic activity.
3. To establish a time scale for volcanic activity on Mars.

INVESTIGATIONS OF ATMOSPHERE-LITHOSPHERE INTERACTIONS AND EXOGENIC PROCESSES

Weathering and Soil Chemistry

1. To characterize the processes that formed the martian regolith (soil).
2. To assess the role of water in regolith formation.
3. To characterize the chemical activity of the regolith.
4. To determine why organics are absent from the regolith.

Erosion and Transport

1. To characterize the processes that formed the martian regolith.
2. To assess the role of water in regolith formation.
3. To characterize eolian processes and features.
4. To determine the age and origin of channels.

Sedimentary and Cryospheric Processes

1. To determine the composition and stratigraphy of layered sediments and ices in the polar regions.
2. To determine the composition and stratigraphy of layered sediments in equatorial regions.
3. To study the interplay between recent climate and recently accumulated sediment.

Biology

1. To determine whether life exists on Mars
2. To determine whether life has ever existed on Mars.

and/or collected are not contaminated by the landing and are representative of the local area. However, only a limited number of science objectives can be met by landing at a homogeneous site with a rover having a 100 m range. This rover could be dependent on the primary lander vehicle for power, communications and science instrument support. Many more objectives can be met by targeting to a specific site such as areas where two or three units are in contact or where a layered sequence of rocks is exposed along an accessible slope. This type of site requires a mobility sufficient to remove the landing error and/or offset and to perform the desired traverse. With available technology the landing error is likely to be tens of kilometers so that this factor dominates the mobility requirement. If it were possible to reduce the landing error so that it is small compared to the traverse requirement of about 10 km, then the design and operation of the rover could be simplified. A long-range rover should be self-sufficient with its own power source, relay communications link and supporting science instruments. A rover that has a range of up to 10 km can be simpler because of the reduced need for traverse speed. Consequently, the rover can be teleoperator controlled and less rugged.

The relationship of mobile laboratory surface science to sample return science is primarily one of documenting samples using imagery and detecting significant changes in surface characteristics (e.g., soil and rock types, composition). Although suggested by some previous studies, the following roles for mobile laboratories are <u>not</u> found to be very useful: (1) a substitute for sample return; (2) an essential precursor to sample return; (3) a means for selecting return samples; and (4) a follow-on to sample return.

An orbiting (or atmospheric) vehicle can be used to acquire scientific data that are pertinent to the selection and qualification of landing sites, the planning of detailed traverses within landing site areas and the understanding of data obtained by the surface science in a broader planetary context. Both imaging and nonimaging (e.g., gamma ray spectroscopy and reflectance spectroscopy) experiments are needed, but only the former offers new design challenges. A surface resolution of approximately 0.5 m is required to locate the landed vehicle on the Mars surface and to detect obstacles in the path of a rover. This is possible for an advanced orbital imaging system; image motion compensation is necessary and the estimated instrument mass is about 100 kg. However, variable contrast attenuation due to dust or fog is expected to be a problem for an orbital imaging system. With an active transponder on the surface vehicle, it can be located by an orbital radar imaging system with a surface resolution of approximately 40 m. The radar also provides information about surface slopes and roughness which can be used to identify areas that are difficult to traverse. Imaging from an airplane and a tethered balloon require smaller, less sophisticated sensors; however, these systems are new development efforts.

2.6.3 Conclusions and Recommendations

Our specific conclusions and recommendations are:

1. Raising the level of martian science to allow detailed comparisons with the Earth and Moon requires that sample analysis be implemented by returning samples to Earth or Earth orbit.

2. The sample return mission should include mobile laboratory surface science, in particular, imaging and sophisticated methods for sample acquisition; precursor mobile laboratory science is not necessary.

3. Surface mobility makes an essential contribution to the value of the sample collected during a sample return mission.

4. Orbital science and network science must be justified on their own merits except that imaging and possibly geochemical measurements can be used in landing site selection and for putting the surface science data into a global context.

5. A rationale does exist for conducting a program of sample return science at Mars in which operations extend over several launch and Earth return opportunities. With this type of strategy the earliest samples returned may be gathered by a tethered rover from a homogeneous site, followed by larger volumes of samples derived from a greater variety of terrains and units using a vehicle with greater mobility.

2.7 Venus Surface Sample Return (1428 man-hours)

2.7.1 Introduction

The objective of this task was to examine the feasibility and technological implications of Venus surface exploration via the sample return mission concept. In attempting to design such a mission for Venus it is soon obvious that the major factor to be dealt with is the hostile environment which this planet presents to systems entering its lower atmosphere, particularly near its surface where temperatures are estimated at approximately 768°K, pressure at 94 atm and density at 65 kg/m^3.

The high venusian temperature, in addition to its obvious impact on material used for lander and ascent system components, has a substantial influence on the entire mission design. Since the temperature exceeds the limits to which present day electronics and rocket propulsion systems can be subjected without deterioration, thermal protection of the entire rocket ascent vehicle is needed. In addition to requiring the use of a rather large pressure vessel to enclose and protect the vehicle, this also implies the use of an alternate method of ascent from the surface, i.e., a hydrogen-filled balloon, thus requiring delivery of either H_2 or H_2 generation equipment to the surface of Venus. Although future advancements in electronics or propellant technology may alleviate these requirements, it has been the policy in this study to avoid such speculation whenever possible. Other environment related design considerations are described below.

The high atmospheric pressure of Venus most significantly affects the mass requirements of structural components; especially pressure vessels which must maintain their integrity in the presence of both high pressure differentials and intense heat. This requirement leads to unusually massive structures, whose weight, in turn, adversely affects the design requirements of practically all other subsystems. Again, in keeping with the policy stated above, the study did not consider possible mass reductions which might be achieved through advances in materials technology or by complex system designs to balance internal and external pressure.

The high density of the venusian atmosphere has both positive and negative effects on the sample return mission. By drastically reducing velocity through atmospheric drag and thereby eliminating the need for descent propulsion or elaborate deceleration devices, the high atmospheric density definitely proves to be an asset for the descent/landing phases of the mission. However, during ascent, this high density produces drag losses which sharply reduce the performance of the rocket propulsion system. This makes it necessary to both delay rocket ignition until the ascent balloon reaches higher and less dense altitudes, and also to utilize a low drag, i.e., slender body ascent vehicle configuration.

By adapting the stated approach which minimizes dependence on technology advancements it has been possible to measure mission feasibility specifically in terms of mass performance margins which exist over the repeated 8-year cycle of round-trip trajectories between Earth and Venus. For this purpose, the study ground rules presumed the collection and return via direct Earth entry of a 1 kg sample, the availability of projected STS launch vehicles and retropropulsion systems, and the capability of automated rendezvous and sample transfer in Venus orbit. The study scope included examination of both direct and out-of-orbit Venus entry options and both single and dual launch options at Earth. However, the single launch option was dropped early in the study when it became clear that there was no way it could meet the injected mass requirements.

Due to the very preliminary nature of this investigation several important mission elements were felt to be outside the scope of this study and were not given consideration at this time. Among these were landing site selection, communications requirements, guidance/rendezvous requirements, Venus surface operations and science objectives other than those associated with the sample return.

2.7.2 Baseline Mission Scenario

A baseline Venus Sample Return mission was formulated early in the study as a reference point on which subsystem design requirements and mass estimates could be based. Although other plans could be devised, and may even possess advantages over the suggested baseline, it is believed that the scenario described below represents a valid, "workable" solution to the many problems associated with this mission.

The baseline mission assumes a dual Earth launch. One payload will contain an orbiter and an Earth Return Vehicle (ERV) to carry the sample back from Venus. The second launch payload will contain the Venus entry module which includes the lander and ascent systems. Since the stay time within the Venus atmosphere is, by necessity, limited to only a few hours, the launches are phased so that the orbiter will arrive at least a day or two earlier than the lander. This will allow it sufficient time to maneuver into a low altitude (300 km) rendezvous orbit in preparation for the sample ascent. In so doing, the time between the ascent and the eventual sample transfer to the orbiter is minimized, thereby relieving the lifetime requirement of batteries used in the ascent payload.

The baseline entry module is presumed to enter the Venus atmosphere directly. Aeroshell and heat shield design provisions will allow entry angles ranging from 20° to 45°. After the maximum G-loading and heating pulse is completed, which normally occurs above 65 km, the module will descend unstaged (i.e., with aeroshell and remaining heat shield intact) to an altitude of approximately 2 km. With an estimated ballistic coefficient of 400 kg/m^2, this descent phase will take about 45 minutes and the velocity at the 2 km altitude will be nearly 12 m/sec. The aeroshell/heat shield will then be jettisoned and a parachute deployed to reduce the terminal velocity to 5 m/sec. At this point approximately 6 minutes remain until touchdown. During this time the landing gear is extended to absorb the impact and provide a stable base for the landed vehicle.

Inflation of the ascent balloon and sample collection begin almost immediately after touchdown. These surface operations are scheduled to last 1 hour after which the balloon should be completely inflated and the 1 kg surface sample stored in a protective cannister. Since the rocket propellant and electronic systems cannot be exposed to the atmosphere at Venus' surface, the sample cannister must temporarily be stowed outside the sealed pressure vessel enclosing these components.

Upon separation from the lander, the balloon and pressure vessel ascend slowly through the atmosphere until 3-1/3 hours later they reach the balloon equilibrium altitude of 60 km. At that time the pressure vessel seal will be broken and the sample cannister will be stored in the payload compartment of the rocket ascent vehicle. The remaining portion of the pressure vessel can then be jettisoned. This is followed by ignition of the first stage rockets and separation from the balloon.

First stage cutoff occurs at an altitude of approximately 100 km. The ascent vehicle then coasts to an altitude of 300 km where the second stage rocket will be ignited to achieve an elliptical orbit. A third stage burn will be used for orbit circularization and trim. With the ascent payload in a circular orbit, the orbiter/ERV will be maneuvered for rendezvous and the subsequent transfer of the sample cannister to the ERV. After a stay time of from 1.2 to 1.4 years in orbit (depending upon the opportunity), the ERV will be inserted into its Earth return trajectory. Total mission time could range from 2.1 to 2.5 years, again depending upon the opportunity.

It should be noted that the parameter values used to describe the mission scenario were derived in performing the study and were not specified *a priori*.

2.7.3 Analysis and Results

Several separate but related analyses were performed during this study. Principal among these were the efforts directed to determining mass/performance requirements for the ascent systems (rocket propulsion and balloon) thermal control, lander/sampling system, entry/descent systems (parachute, heat shield and aeroshell), and interplanetary vehicles (orbiter/ERV, lander bus and launch vehicles). In the course of this task it was found that the most effective manner of performing these analyses was in the sequence stated above, i.e., in essentially inverse chronological order starting with the payload ascent to Venus orbit.

Rocket propulsion requirements for the final ascent were determined by computing the payload mass fraction delivered to rendezvous orbit as a function of the launch altitude (altitude at first stage ignition) and initial kick angle. Launch altitudes from 50 to 60 km were examined as they appeared to represent a range where improved performance resulting from higher altitude launches might be balanced by increased balloon mass and ascent time requirements. Furthermore, expansion of the range did not seem useful since drag losses at 60 km are sufficiently small that additional performance improvement resulting from launches above this altitude could be expected to be minor; and atmospheric temperatures below 50 km (>350°K) are higher than the operating design points for many of the electronic components.

Results of the analysis indicate that launches at the high end of this range, i.e., 60 km, provide the best performance by a margin which could not be overcome by the balloon ascent considerations mentioned previously. Assuming a 60 km launch at an optimum kick angle of 87°, it was determined that a 1284 kg initial launch mass was required to insert a 35 kg payload* into the 300 km circular orbit desired for rendezvous. This can be compared with 1850 kg and 3627 kg, the intial masses required at launch altitudes of 55 km and 50 km respectively.

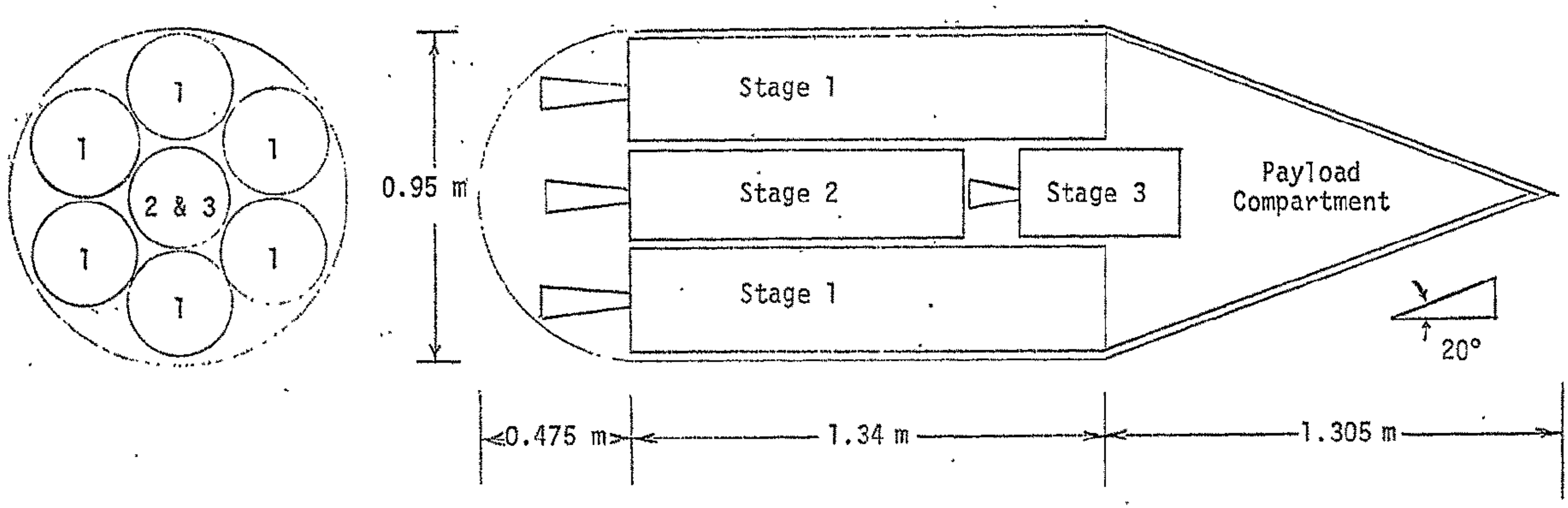
*The selected payload mass of 35 kg is small compared to values being suggested for Mars sample return payloads. Its selection was based on the possibility of capturing the mission with a single IUS(Twin) stage launch. Use of multiple launches or more advanced STS vehicles would permit a larger payload.

The requirement for a low drag coefficient ascent vehicle led to the rocket propulsion configuration shown in Figure 7 which in turn dictated the dimensional requirements of the protective pressure vessel. Structural requirements of the pressure vessel were conservatively based on an unreinforced monocoque structure of uniform thickness.* Both beryllium and titanium were considered as potential candidates for this application and their required thicknesses were determined on the basis of buckling resistance and compressive yield strength under external pressures up to 100 atmospheres. The required material thicknesses were found to be approximately 3.5 cm for beryllium and 2 cm for titanium; however, the beryllium structure at 495 kg possesses a mass advantage of nearly 200 kg over its titanium counterpart.

Two thermal control concepts for protecting the contents of the pressure vessel throughout the descent, surface operations and ascent phases of the mission were considered during the study. Both concepts employ the use of a phase change material (PCM), e.g., lithium nitrate-trihydrate, to absorb the incoming heat; however, they differ in placement of insulation internal or external to the pressure vessel. The mass advantage of the former approach was found to be overwhelming since a near vacuum could be created in the pressure vessel allowing the use of very effective superinsulation (multiple reflective layer) blankets. The total thermal control mass required through this approach is only 30 kg including both insulation and PCM material. On the other hand, the high atmospheric pressures at Venus tend to degrade the effectiveness of Min-k, the external insulation material, and the required thermal control mass approaches 740 kg.

The thermal control components, pressure vessel and rocket ascent vehicle (including payload and conical shroud) constitute an 1815 kg payload for the ascent balloon. This payload mass and the equilibrium altitude of 60 km specifies a balloon radius of 9.5 m. Further analysis

*Set equal to the maximum required thickness of any one section of the pressure vessel.



DIMENSIONAL REQUIREMENTS			
Rocket Stage	1	2	3
Diameter	0.2965	0.2715	0.2643
Case Length (m)	1.376	0.964	0.442
Nozzle Length (m)	0.2593	0.2295	0.1502
Internal Insulation Thickness (cm) 2			
PCM Volume (m^3) 0.016			
Available Payload Volume (m^3) 0.281			

STRUCTURAL REQUIREMENTS				
Material	Beryllium		Titanium	
Failure Criteria	Buckling	Yield	Buckling	Yield
Cone Thickness (cm)	1.252	3.251	1.843	1.317
Cylinder Thickness (cm)	1.372	3.459	2.018	1.402
Hemisphere Thickness (cm)	0.522	1.730	0.845	0.701
Maximum Anticipated Pressure 100 Atm				
Safety Factor 1.25				

Fig. 7 PRESSURE VESSEL CONFIGURATION AND REQUIREMENTS

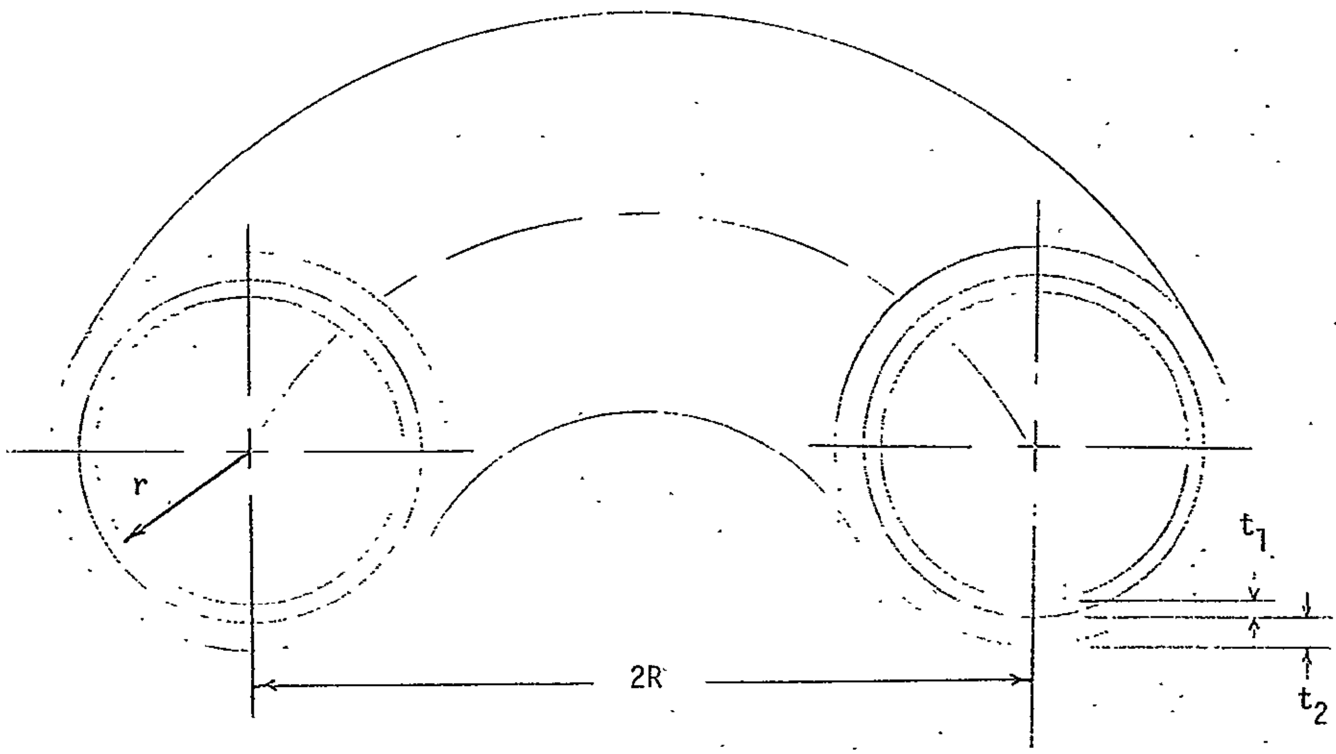
of the requirements for a partially filled non-extensible balloon present possible tradeoffs involving gas volume and ascent time. Because gas transport mass increases dramatically with increased gas loading and thermal control mass goes up more slowly with increased ascent time, the tradeoff is always biased towards less gas and longer ascents. An H_2 gas charge of 134 kg was selected. This provides an initial acceleration of approximately 0.2 g_V (Venus gravity) and an ascent time of 195 minutes.

The choice of hydrogen over helium for the balloon gas was due to the permeability of most balloon materials to helium at Venus temperatures. Hydrogen, furthermore, possesses a mass advantage but this was a secondary consideration. For convenience in packaging, a toroidal shaped pressure vessel, as shown in Figure 8, was specified as a container for the hydrogen which was to be cryogenically stored and transported in its liquid state. Thermal protection during transit from Earth to Venus was assumed to be provided by a thick blanket of superinsulation. This was augmented by a layer of Min-k insulation which could perform more effectively under the atmospheric pressure encountered during descent. The total mass of the gas transport system including gas vented during transit was computed to be 1178 kg. The table in Figure 8 also compared mass requirements for a case where the initial acceleration is 0.5 g_V, and clearly illustrates the tremendous mass penalty (489 kg) paid for reducing ascent time less than 1 hour.

Landing system mass requirements were based on previous mass estimates for Mars sample return missions. A total allowance of 386 kg was provided for the lander which includes the following subsystems: telecommunications power, data handling and control, pyrotechnics, cabling, devices, antenna, TDR, sample acquisition and structure.

The total landed mass based on summation of all the major systems discussed to this point is 3383 kg. At the time of parachute deployment, this mass will be descending at a rate of approximately 12 m/sec with an expected impact velocity of over 10 m/sec. Through the use of parachute

Fig. 8 LIQUID HYDROGEN GAS TRANSPORT SYSTEM



DESIGN COMPARISON FOR TWO ACCELERATION PARAMETERS*	λ = 0.5	λ = 0.2
Ascent Time to 60 km	145	195
Torus Radius R (m)	1.0735	0.9636
Pressure Vessel Radius r (m)	0.4485	0.3986
Internal Volume (m^3)	4.263	3.0226
MIN-K Thickness t_1 (cm)	1.6	1.5
Superinsulation Thickness t_2 (cm)	10.4	9.1
Initial Gas Mass (kg)	284	201
Landed Gas Mass (kg)	210	134
Pressure Vessel Mass (kg)	822	581
Insulation Mass (kg)	561	396
Initial Mass Total (kg)	1667	1178

*Based on descent β = 500 kg/m^2

sizing equations it was determined that an 8 m (constructed) diameter parachute would be adequate to reduce the impact speed to 5 m/sec. Allowing for the use of a higher than normal density fabric to withstand the Venus temperatures, a parachute mass of 60 kg was specified.

At this point the analyses had proceeded backwards in the mission to Venus entry where the major issues were the required masses for the heat shield and aeroshell. Scaling relationships for the heat shield were obtained for laminar and turbulent convective heating during entry, for the radiative heating component, and for the relationship between these heating components and the heat shield masses for the Pioneer/Venus probes. The scaling reference point was the heat shield requirement for the P/V small probe. A scaling relationship was also developed for the aeroshell based on structural buckling of a conical shell. The reference point in this case was the aeroshell mass requirement for the P/V large probe. Using these derived scaling laws, it was found that the heat shield and aeroshell masses required for the baseline sample return mission were 327 kg and 1647 kg respectively. These values presume a 1991 launch which is the best opportunity. For the worst opportunity (1988) these values increase to 459 kg and 1718 kg.

The mass requirements of all the Venus entry systems are summarized in Tables 13 and 14. Table 13 applies to the 1991 launch opportunity and assumes direct Venus entry, while Table 14 is for out-of-orbit entry. The major difference in the two is the smaller heat shield and aeroshell mass required when entering from orbit. This is a result of both lower entry velocity and greater control of entry angle.

The launch requirements over the opportunity cycle for both direct and orbit entry options are graphically depicted in Figures 9 and 10; supporting data are found in Table 15. The injected mass requirements for the orbiter/ERV and the lander/ascent vehicle payloads are represented by the shaded bars in the diagram. Capabilities of potential launch vehicles are superimposed over these requirements to clearly indicate positive and negative mass margins where they exist.

Table 13

VENUS ENTRY SYSTEMS MASS SUMMARY

1991 DIRECT ENTRY CASE

SYSTEM		MASS (kg)
Payload (includes 1 kg sample)		35
Rocket Ascent Stages		
Stage 1	1095	
Stage 2	108	
Stage 3	46	1249
Protective Pressure Vessel (beryllium)		495
Thermal Control		
Insulation	14	
PCM	16	30
Conical Ascent Shroud		6
Balloon		
Gas (H_2)	134	
Fabric	54	188
VENUS LAUNCH MASS		2003
Gas Transport System		994
Lander Systems		386
TOTAL LANDED MASS		3383
Descent Parachute		60
Aeroshell		
Cone	941	
Aft Cover	368	
Auxiliary Structure	338	1647
Heat Shielding		
Cone	281	
Aft Cover	46	327
VENUS ENTRY MASS		5417

Table 14

VENUS ENTRY SYSTEMS MASS SUMMARY

SYSTEM		OUT-OF-ORBIT ENTRY CASE MASS (kg)
Payload (includes 1 kg sample)		35
Rocket Ascent Stages		
Stage 1	1095	
Stage 2	108	
Stage 3	46	1249
Protective Pressure Vessel (beryllium)		495
Thermal Control		
Insulation	14	
PCM	16	30
Conical Ascent Shroud		6
Balloon		
Gas (H_2)	134	
Fabric	54	188
VENUS LAUNCH MASS		2003
Gas Transport System		994
Lander Systems		386
TOTAL LANDED MASS		3383
Descent Parachute		50
Aeroshell		
Cone	628	
Aft Cover	368	
Auxiliary Structure	338	1334
Heat Shielding		
Cone	177	
Aft Cover	46	223
VENUS ENTRY MASS		4990

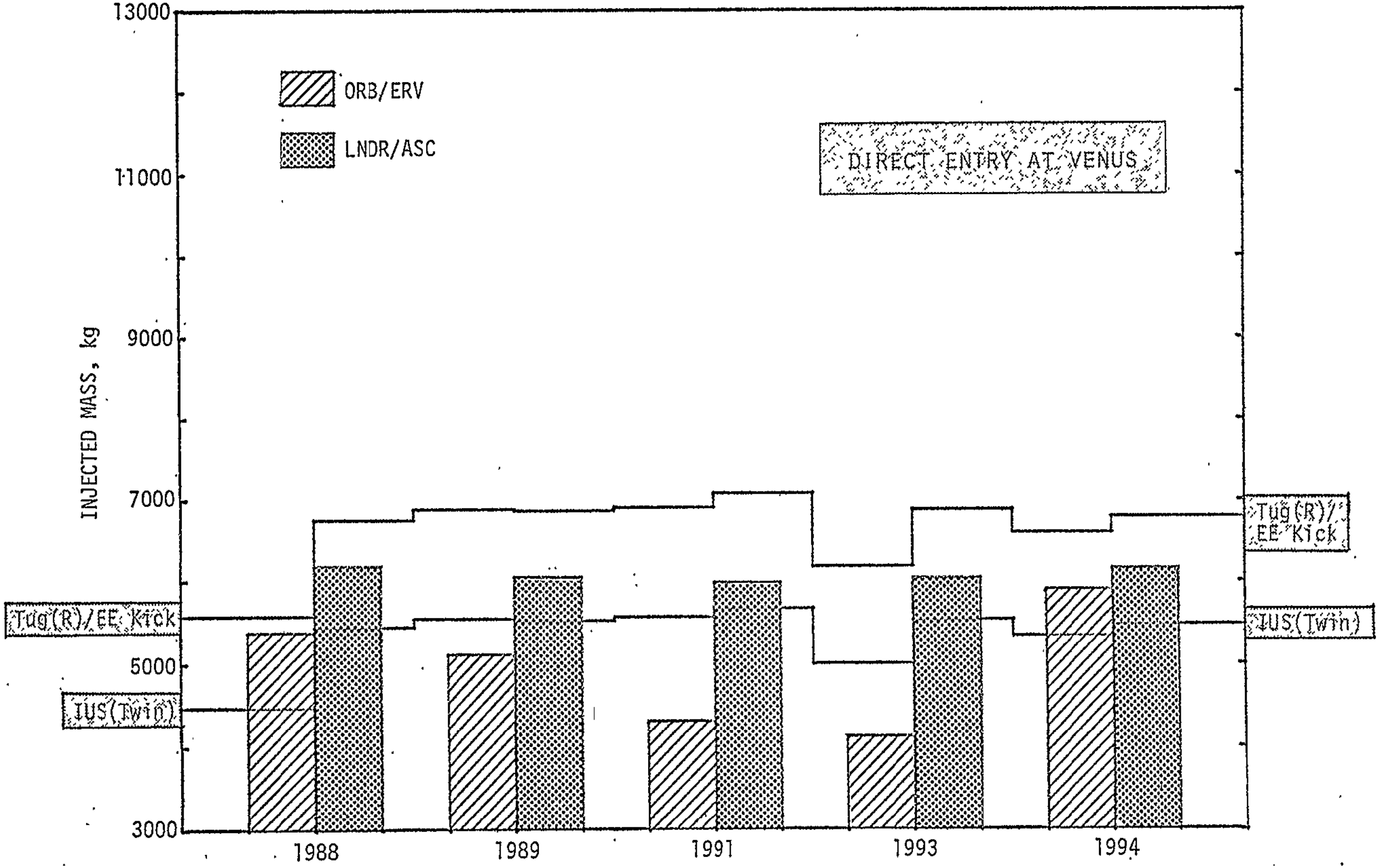


FIGURE 9. VENUS SAMPLE RETURN LAUNCH REQUIREMENTS

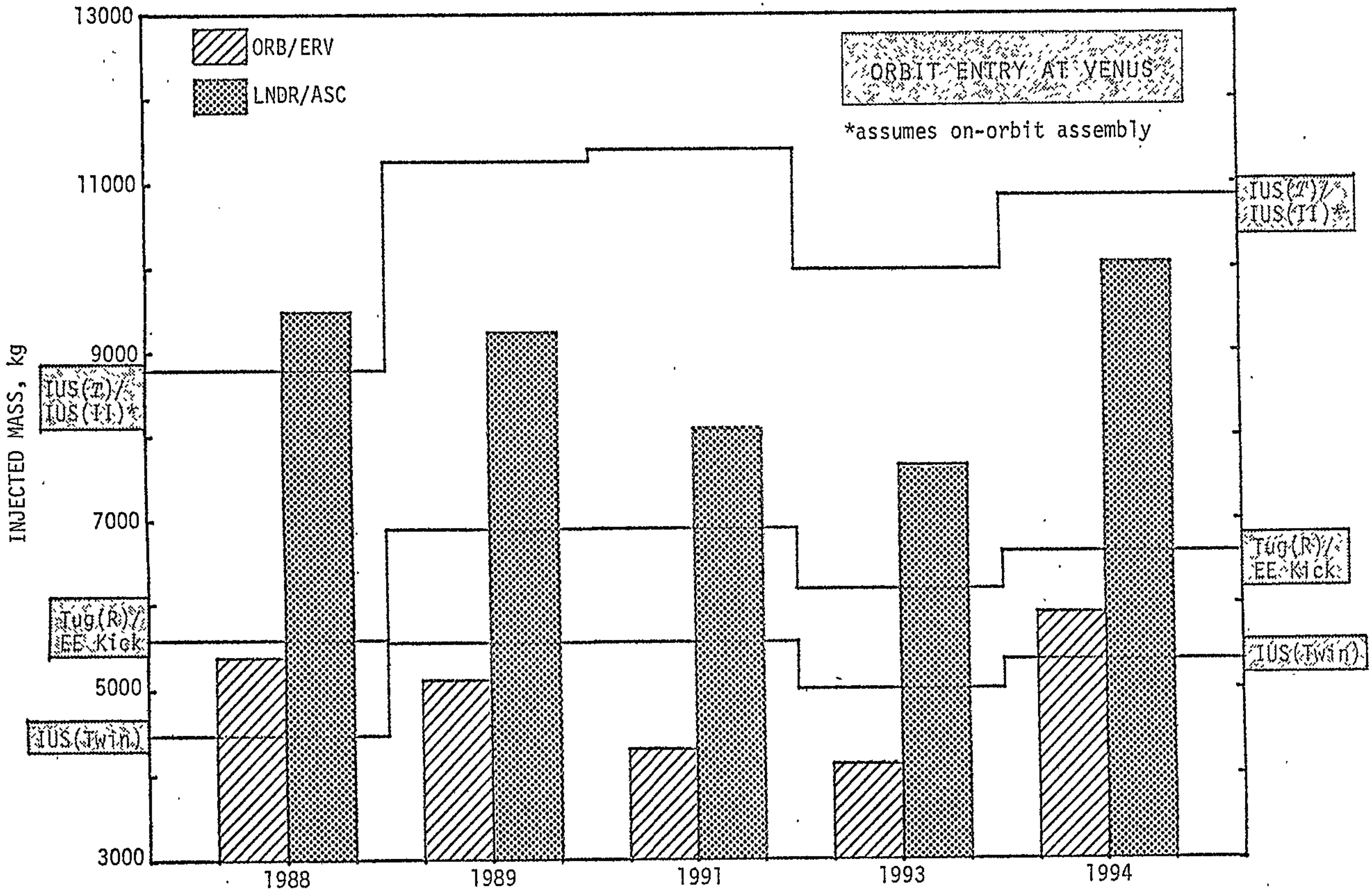


Fig. 10 VENUS SAMPLE RETURN LAUNCH REQUIREMENTS

Table 15

VENUS SURFACE SAMPLE RETURN OPPORTUNITIES

Opportunity	Vehicle	Launch Date	Arrival Date	Departure Date	Return Date	Stay Time (days)	Total Trip Time (days)	Performance Rank
1988	LNDR/ASC	18 Mar 1988	8 Sep 1988					Worst
	ORB/ERV	4 Apr 1988	28 Jul 1988	4 Dec 1989	4 Jun 1990	494	808	
1989	LNDR/ASC	30 Oct 1989	15 Apr 1990					Intermediate
	ORB/ERV	2 Nov 1989	16 Apr 1990	27 Jun 1991	8 Dec 1991	437	769	
1991	LNDR/ASC	21 May 1991	2 Nov 1991					Best
	ORB/ERV	29 May 1991	15 Nov 1991	1 Feb 1993	9 Jun 1993	444	750	
1993	LNDR/ASC	31 Dec 1992	22 May 1993					Intermediate
	ORB/ERV	2 Dec 1992	21 May 1993	17 Oct 1994	25 May 1995	514	904	
1994	LNDR/ASC	4 Aug 1994	6 Dec 1994					Intermediate
	ORB/ERV	14 Aug 1994	12 Dec 1994	7 May 1996	2 Dec 1996	512	851	

From Figure 9 it is seen that the IUS(Twin) has negative mass margin for every lander launch and is therefore unable to capture any mission opportunity by itself. However, it is capable of injecting the orbiter/ ERV in the 1989, 1991 and 1993 opportunities and could therefore perform one of the two launches required. Assuming direct entry at Venus, the recoverable Tug with an EE-Kick stage is shown to be capable of capturing all the mission opportunities; however, if orbit entry is desired, Figure 10 indicates that it too is deficient with regard to the lander launch. Performance of missions having the orbit entry option requires either multiple IUS launches with on-orbit-assembly or use of an expendable Tug.* But even in the multi-launch example shown, an assembled IUS(Triplet) and IUS(II) lack the performance required to launch the lander in the 1988 opportunity.

2.7.4 Conclusions

The most important conclusion one can draw from this study is that a Venus Sample Return mission can be performed. However, it has been shown to be a difficult mission, requiring the performance capability of projected STS launch vehicles in addition to a dual launch and possibly on-orbit assembly to inject the necessary mass.

In interpreting this conclusion it should be understood that it is based on the premise that severe mass penalties in the form of a heavy pressure vessel and gas transport systems would be accepted in order to avoid major technology issues, i.e., development of electronics and propellants capable of reliable operation in the hostile Venus environment. Such developments could of course relieve the total mass requirements of the mission, but would not by themselves be sufficient to alter conclusions regarding mission difficulty, since several other technology and engineering issues remain as matters for future concern. These include questions pertaining to balloon and parachute materials, PCM emplacement and containment, and the automated operations of balloon deployment, rendezvous and sample transfer to both the ascent payload and the ERV.

*Although not shown in the figure, the capability of the expendable Tug would lie somewhere between the recoverable Tug and the IUS(T)/IUS(II) assembly.

2.8 SEP/Sail Discriminator Assessment (756 man-hours)

This task was part of a comprehensive analysis of two promising advanced low-thrust systems, Ion Drive (SEP) and Solar Sail. This analysis supported the work of an Assessment Committee (Mr. F. Demeritte, Chairman) which was organized to evaluate the technology readiness and development risk of building either system in time for a Halley Comet Rendezvous launch in early 1982. This appraisal also considered five other future high-energy missions which can be accomplished using these advanced low-thrust systems. SAI also assisted the Assessment Committee in two other tasks (Mission Performance Verification and Development Cost/Time Risk Assessment) which were supported by the Office of Aeronautic and Space Technology.

The objective of the appraisal was to compare and assess qualitatively the impact of SEP and Sail on mission performance, science, spacecraft and navigation. (Cost risk factors are excluded from this subtask.) The approach was as follows:

1. Consider all six baseline missions.
2. Define 28 discriminators.*
3. Determine the importance of individual discriminators for each mission.
4. Assess discriminator impact using baseline mission data.
5. Determine overall impacts.

JPL provided data covering a specified set of mission parameters for the six missions. For each mission the relevant discriminators were selected (independent of the SEP/Sail data) and assigned high, medium or low rankings according to the level of importance that particular discriminator had in a mission feasibility study. The impact assessment of a discriminator resulted in a qualitative rating of favors SEP, favors Sail or the impact is about the same for either low-thrust system. With respect to some parameters such as payload, flight time, power, etc., a quantitative assessment was performed. However, other factors such as

*This step is described in Advanced Planetary Studies, Fourth Annual Report, Science Applications, Inc., Report No. SAI 1-120-580-A4, July 1977.

science interference, target approach constraints, viewing constraints, etc., could only be evaluated qualitatively. Consequently, the overall impact assessment is a qualitative one.

This analysis indicates that both systems can do all six missions. Figure 11 shows that the two low-thrust propulsion systems are equally suited (within 10%) for the following missions:

Comet Halley Rendezvous
Comet Encke Rendezvous
Mercury Orbiter with Rough Landers.

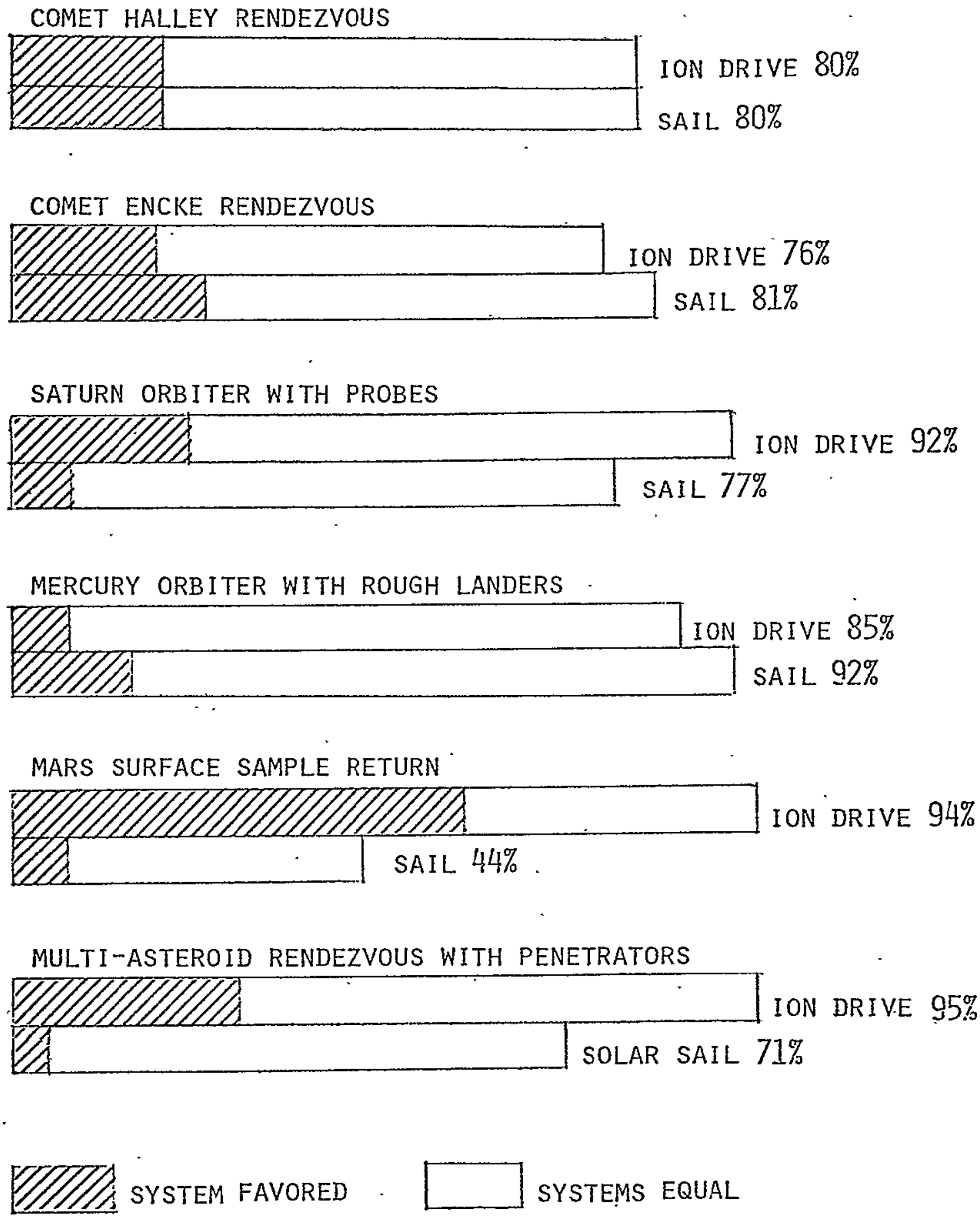
The Ion Drive system (SEP) has more favorable impacts for the remaining three missions, namely:

Saturn Orbiter with Probes
Mars Surface Sample Return
Multi-Asteroid Rendezvous with Penetrators.

The general features of the impact assessment are described below. These results suggest that the systems have similar overall impacts for inner planet and comet rendezvous missions. This situation probably also applies to nontargeted missions, such as a solar probe or out-of-the-ecliptic mission. The Ion Drive system appears to have the advantage for outbound missions and for multi-leg missions with near-target operations.

For all discriminators rated as high importance, Figure 12 shows that Ion Drive is favored much more often than Sail, although just over 50% of these discriminators are rated equal. This ability to do better on very important mission considerations strengthens the overall preference for Ion Drive shown in Figure 11. This also accounts for the higher rating earned by Ion Drive when the assessment which weights the discriminators according to their importance is compared to the assessment using an unweighted average.

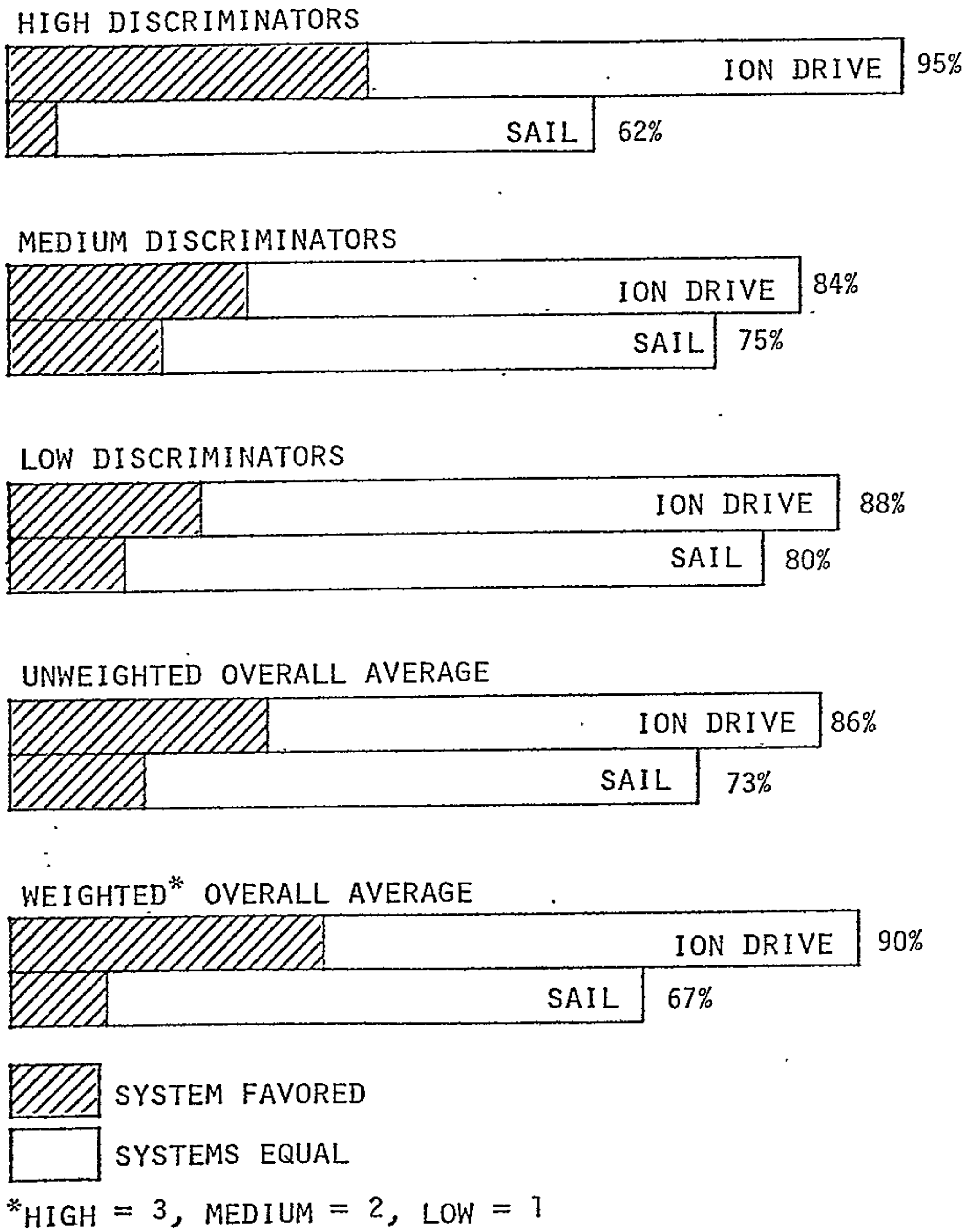
With respect to the performance discriminators, the Shuttle/IUS(Twin) is the <u>launch vehicle</u> in all cases. The minimum <u>launch window</u> is always



*UNWEIGHTED AVERAGES OF ALL RELEVANT DISCRIMINATORS

IMPACT ASSESSMENT FOR EACH MISSION*

Figure 11



IMPACT ASSESSMENT SUMMARY FOR ALL MISSIONS

Figure 12

C-2

met, but in some cases a significantly longer launch window is available at little cost in performance. Ion Drive uses less flight time to complete four discriminator missions, but Sail offers shorter flight time for Encke and Mercury. Ion Drive has a significant advantage for comet missions because its arrival time is pre-perihelion while Sail is always post-perihelion. For the Mars and Multi-Asteroid missions, both have adequate stay time. In general, Ion Drive has less sensitivity to increased payloads where the increase is provided by a longer flight time and not via a reduction in contingencies. Both systems have no in-flight sensitivity to transport system degradation. In the case of Ion Drive a worst case estimate of in-flight degradation is made and then the rated system thrust is reduced by this factor for the whole mission. The worst case estimate for thrust loss by Sail is negligible and is, therefore, ignored.

In all cases the two low-thrust systems deliver identical science instrument packages to the targets. Therefore, the discriminators in the science area are related to operations. (See also the spacecraft discriminators.) There is a fundamental difference between Ion Drive and Sail with respect to cruise science interference. Ion Drive has significant problems with static magnetic fields from the ion thrusters and with electromagnetic interference during thruster operations. These problems limit the opportunities for high quality interplanetary particles and fields measurements to coast periods. However, long intervals when the ion thrusters are idled are rarely included in these minimum flight time missions. Coast periods do occur in missions which are designed for reduced initial mass (equivalent to reduced fuel). This problem also applies to encounter science interference when Ion Drive uses the nonjettisoned option, although now coast periods are the normal mode of operation. Viewing constraints do not inhibit the acquisition of science data with either low-thrust system. The Ion Drive solar array or its thrust vestor can be briefly relocated to provide a view that nominally is blocked by structure. The Sail blades do block a large area, but as they rotate a gap between blades appears once each 30 sec which can be used to acquire the science data. Attitude stability is adequate with both systems for the acquisition of science data. Pointing accuracy, nominally 1.0 mrad, and stability of $<10^{-5}$ rad/sec are independent of the way by which the attitude is controlled.

Among the spacecraft discriminators, the major differences are associated with the ability of Ion Drive to make effective use of the nonjettisoned option. To support the low-thrust systems, the required power/command support from the spacecraft is not large for either. However, in the nonjettisoned mode Ion Drive supplies about 600 watts of provided power support, all that is needed for spacecraft operation.

Communication constraints are not significant for either system. Ion Drive uses one high gain antenna and has nearly 4π coverage while Sail needs two such antennas for the same coverage. There are no problems with viewing constraints or attitude stability for either Ion Drive or Sail which would affect spacecraft operations (e.g., pointing a high gain antenna, acquiring a stellar reference, etc.). The thermal control impact is larger for Sail because more time is spent inside the Earth's orbit. Ion Drive is able to use its low-thrust system to perform all near-target maneuvers for several missions whereas Sail uses a separate supporting chemical propulsion system. The mission descriptions all assumed a single Shuttle/IUS(Twin) launch and had (with one exception) no obvious assembly/departure constraints. Target approach constraints are most significant for comet missions and there is a definite preference for the approach conditions of the Ion Drive mission. During cruise Ion Drive is more maneuverable than Sail, but after the Sail is jettisoned, the independent spacecraft is more maneuverable than the nonjettisoned Ion Drive system. The assessment of the maneuverability constraint discriminator is, therefore, dependent upon the relative importance of cruise versus encounter operational maneuvers. Docking load constraints are important for the sample return mission and for the Sail version of the multi-asteroid mission. The mission definitions were not sufficiently complete to determine the impacts of docking constraints. Ion Drive has a beam neutralizer which controls its <u>electrical charging</u>; Sail does not need one.

Navigation data are available only for the Comet Halley Rendezvous mission. These data indicate that the two systems have similar capabilities and/or requirements for <u>viewing constraints</u>, <u>attitude stability</u>, <u>operational procedures</u> and <u>accuracy</u>. There is no reason to believe that the assessment would have been different had data been available for other missions. The estimated accuracy is directly related to the assumed error model for the low-thrust noise. The model for Ion Drive is based upon previous system studies while there has been less experience in modeling errors for Sail. Neither mission definition includes a calculation of the attitude stability required for on-board target acquisition. It is possible that the stability needed for acquisition is more demanding than that for science data in which case the nominal acquisition time must be delayed or additional effort expended designing a more stable platform.

3. REPORTS AND PUBLICATIONS

Science Applications, Inc. is required, as part of its advanced studies contract with the Planetary/Lunar Programs Office, 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 during the contract period 1 February 1977 through 31 January 1978 is presented below. Unless otherwise indicated, these documents are available to interested readers upon request.

3.1 Task Reports for NASA Contract NASW-3035

1. "Mission Performance Workbook - Mars Mission Options (Emphasis 1984)," Report No. SAI 1-120-839-T9, March 1977.

2. "Ion Drive/Solar Sail Assessment Study," Report No. SAI 1-120-839-S3, August 1977.

3. "Planetary Missions Performance Handbook - Volume I (Revision A), Outer Planets," Report No. SAI 1-120-839-S2A, February 1978.

4. "Multiple Discipline Science Assessment," Report No. SAI 1-120-839-S4, December 1978.

5. "Advanced Planetary Studies Fifth Annual Report," Report No. SAI 1-120-839-A5, December 1978.

6. "Advanced Planning Activities, February 1977 - January 1978," Report No. SAI 1-120-839-M9, December 1978.

3.2 Related Publications

1. "Asteroid Return Trajectories," J. C. Niehoff, at Eighth Lunar Science Conference, NASA/JSC, March 1977.

2. "Round-Trip Mission Requirements for Asteroids 1976AA and 1973EC," J. C. Niehoff, Icarus 31, 430-438, August 1977.

3. "Asteroid Mission Alternatives," J. C. Niehoff, at Asteroid Workshop, University of Chicago, January 1978.

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