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ATMOSPHERIC ENTRY PROBES FOR OUTER PLANET EXPLORATION

A TECHNICAL REVIEW AND SUMMARY

August 1974

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FOR OUTER PLANET EXPLORATION. OUTER
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FOREWORD

NASA Ames Research Center contracted with DYNATREND INCORPORATED to prepare this Outer Planet Entry Probe Technical Summary, which presents the results of a four-month review and analysis of prior work on scientific probes to make initial in-situ measurements of the atmospheres of Jupiter, Saturn (including Titan), Uranus, and Neptune to a pressure depth of about 10 bars. The objective of this study was to review and assess a number of other studies done in the last five years by several aerospace contractors; and to summarize and consolidate their results highlighting:

(1) the design of a common entry probe for outer planet missions
(2) the significant trades related to the development of a common probe design
(3) the impact of bus selection on probe design
(4) the impact of probe requirements on bus modifications
(5) the key technology elements recommended for advanced development

A draft of this report was made available to attendees of the Outer Planet Probe Technology Workshop at NASA Ames Research Center, Moffett Field, California on May 21-23, 1974. This document incorporates comments of the participants.

NASA and several industrial concerns under contract to NASA have performed studies of missions to the outer planets over the past several years. The major studies concerned with outer planet probe missions which provide the basis for this report were done by Martin-Marietta Corporation, McDonnell-Douglas Astronautics Company, and the Systems Division of Avco Corporation.

Study efforts have concentrated on Jupiter, Saturn and Uranus with relatively little effort addressing missions to Neptune or Titan. The studies have generally considered bus spacecraft and entry probes with total weight in the 400 to 825 kg range. The bus options considered have been limited to the spin-stabilized

Pioneer F/G and the 3-axis stabilized Mariner J-S spacecraft as modified to support an entry probe. Selected missions for the studies covered launch dates in the late 1970's and early 1980's using expendable launch vehicles in the Titan III/Centaur class and later missions using the Space Shuttle.

This summary document represents a review and coalescence of the results of studies done by NASA and a number of aerospace contractors. In addition, technical material from numerous other related documents (listed in the bibliography in Section 8.0) has been used. Some new material has been incorporated from both the Outer Planet Probe Technology Workshop* and the MJU '79 studies** performed jointly by JPL/NASA ARC during the past year. Information and data in the forms of tables, graphs, cutaway sketches, etc., have also been utilized from the reference documents. In some cases these have been used directly, and in other instances the curves or sketches were redrawn to extract only the information pertinent to the discussion contained herein. During the same time period of this study, the Martin-Marietta Corporation under contract to the NASA Ames Research Center was conducting a study to define a common Pioneer Saturn/Uranus probe using designs based on "existing" hardware from the Pioneer Venus (PV) program and to assess the effect of modifications to the PV hardware to make this equipment compatible and suitable to the Saturn and Uranus mission objectives. Information from that study (Reference 12) was made available during the preparation of this report. Subsequently, similar information has become available from Hughes Aircraft Corporation from their Pioneer Venus design (Reference 14). We have attempted to include the results of these studies in appropriate paragraphs herein. The basis of these studies, using PV hardware designs, was a departure from the other outer planet probe studies and the conclusions are felt to be noteworthy. Section 6.0 presents a summary of the PV probe adapted/designed to PS/U missions.

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* Proceedings in preparation
** Publication in process

The body of the report includes a concise summary of the conclusions reached in this study (Section 2.0). A descriptive summary of the single probe design which has been evolved from this study (Section 3.0) and the major trades which are embodied in that design (Section 4.0 - Probe, and Section 5.0 - Bus) are also included. Section 6.0 presents a summary of a Pioneer Saturn/Uranus probe based upon Pioneer Venus designs. Section 7.0 presents our recommendations for advanced development work which would greatly assist in preparing for the development of an outer planet probe.

Appendix A represents a compendium of technical data contained in a number of studies used as source material for this Technical Summary.

1.0 INTRODUCTION

The outer planets, Jupiter, Saturn, Uranus, Neptune, and Pluto with their satellites are of significant scientific interest. The question of solar system formation and evolution, and of mass and angular momentum interchanges between the expanding atmosphere of the sun and the galaxy can be addressed by exploration of the outer region of the solar system. In the context of planetary formation, knowledge at one planet or satellite can be related to that gathered at each of the other bodies in the solar system to contribute to an understanding of the origin and evolution of the solar system.

An important element of the study of the outer planets is the examination of the planetary atmospheres. Direct measurement of the composition, structure and dynamics of the planetary atmospheres will contribute greatly to an understanding of the early history of the solar system and the evolution of the Earth's atmosphere. Titan is known to have a significant atmosphere with a high concentration of methane; it is a likely place for simple organic compounds to evolve.

The outer planet probe is one link in the exploration of the atmospheres of the outer planets. In-situ measurements of the composition and structure of these atmospheres can be carried on by relatively simple experiments on the probe during atmospheric entry and descent to about the 10-bar pressure altitude.

Selection of the science experiments to be carried on the probe must consider:

- the scientific return from the measurement
- the state of development of the instrument
- the potentially severe entry environment
- the long-term deep space transit to reach the outer planets
- the limited communication capability from the outer planets to Earth

In a number of previous studies of planetary entry probes, three basic concepts of probe science, and resulting probe weight and complexity, have evolved. The lightest probes (30-50 kg) have been designed to study atmospheric structure. The science payloads of these would contain temperature, pressure and acceleration instruments. The next larger class of entry probe (100-150 kg) is designed to examine both the structure and elementary composition of the atmosphere. For the outer planets, clouds are an important aspect of composition and would come under scrutiny. The largest probe concepts which have been considered (at weights >200 kg) generally descend deeper (to the surface in the case of Viking and Pioneer Venus), include pre-entry science, detailed composition experiments, radiation balance experiments and even specific life detection instrumentation. In this study, we have selected a payload to provide structure and composition data compatible with the overall probe weights considered in the source studies.

The next section of this report contains the conclusions reached in this study and presents a summary of the baseline Outer Planet Probe Design.

2.0 CONCLUSIONS AND SUMMARY DESIGN

2.1 Conclusions

Several studies have been completed on probe design for outer planet entry missions, and despite some differences in design approach, this comparative analysis and evaluation of these prior projects has resulted in several significant conclusions which can be drawn:

(1) Entry Probe missions to the outer planets are feasible with launches planned to begin in the 1979-1980 time frame.

(2) A common probe design can be developed for entry into the atmospheres of the outer planets of Saturn (including Titan), Uranus and probably Neptune. Jupiter entry mission requirements could perhaps be included in a "common" probe design; however, the weight penalty associated with Jupiter entry would pace and direct the probe development.

(3) The basic technology for the "common" probe design exists and has been demonstrated with the possible exception of a heat shield material which can survive the outer planet entry heating. Several approaches have been recommended including the use of reflective materials. Evaluation and demonstration of these approaches awaits the development of a test facility which adequately simulates the heating and chemical environment to be experienced during outer planet entry.

(4) A common probe for missions to each of the outer planets except Jupiter (and perhaps Pluto which was not included in the study) can be designed within a weight limit of 113 kg (250 lb) including a 20% weight contingency. Inclusion of shallow entry angle (<10°) Jupiter mission requirements would increase the required probe weight to about 160 kg (352 lb). No planetary quarantine provisions are included.

(5) Both Pioneer class (spin stabilized) and Mariner class (3-axis stabilized) spacecraft have been considered as the probe bus for the outer planet missions. The Pioneer class bus with

probe can be launched to any of the outer planets by a Titan IIIE/Centaur/TE 364-4 launch vehicle. The Mariner class bus can use the same launch vehicle for Jupiter and for selected Saturn, Titan and Uranus missions using Jupiter swingby. For direct Saturn missions, or for those to Uranus via Saturn, the Mariner class bus will require uprated launch vehicle capability, solar electric propulsion, significant weight saving modification, or other launch opportunities. It may also be possible to configure a 3-axis stabilized probe-bus from other space qualified hardware. However, considering existing spacecraft and the selected launch opportunities, direct flights to Saturn in the near term are more attractive using the Pioneer spin-stabilized probe bus.

(6) From the standpoint of probe delivery, the 3-axis stabilized Mariner class bus provides some performance advantages over the spin-stabilized Pioneer class bus. Most notable among these are: (1) an improved probe-bus communication relay link design since the 3-axis stable platform permits the use of a higher gain bus receiving antenna for a higher performance communications link; (2) optical navigation capability resulting in very flexible and accurate targeting for control and knowledge of probe delivery to the entry corridor, and (3) capability to deliver the probe at zero angle of attack.

(7) Because of reliability and weight considerations, the baseline probe design is non-staged, i.e. no parachutes or other drag/stability augmentation devices and the entry heat shield is retained, during atmospheric descent. The altitude-time profile for Saturn and Uranus (and probably Neptune and Titan)* with an unstaged probe permits good reconstruction of the atmosphere with the selected instruments and their sampling rates.

The descent time from entry to the 10-bar pressure altitude may be too short in the Jovian cool dense model atmosphere to result in a good altitude profile of atmospheric constituents. Instrument operation is expected to continue for some time to lower

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* Analysis of these entry/descent profiles has not been conducted to the detail necessary to <u>insure</u> suitability of the non-staged design.

altitudes than the 10-bar design limit, but this model atmosphere would still "design" the common probe. Analysis of data from Pioneer 10 has provided information which tends to diminish the variability of the Jovian model atmospheres. The common probe design should now be reviewed for application to Jupiter missions.

(8) Several key technology items are discussed in Section 7. Each of the areas discussed is recommended for further work to advance probe development.

2.2 Summary of Baseline Spacecraft Design

The baseline probe is an 89.0 cm (35-in.) diameter 60-degree half-angle blunt cone weighing 113 kg (250 lb). The probe carries five scientific instruments: an accelerometer triad; a pressure sensor; a temperature sensor; a 0-40 AMU neutral mass spectrometer; and a nephelometer. The probe enters and descends through the planetary atmosphere without the use of drag augmentation or stabilization devices. Data taken by the probe during entry and subsequent descent to a nominal 10-bar pressure altitude are transmitted at 44 bits per second back to the flyby bus. The bus supports the probe mission from launch and in addition carries its own flyby science to examine the planet from a periapsis altitude of 50,000 to 80,000 kilometers depending upon the target planet. The spacecraft (bus and probe) is launched on a Titan IIIE/Centaur/TE 364-4 launch vehicle.

In the systems studies which form the basis for this report, probe configuration was almost universally a 60-degree half-angle blunt cone.* This entry shape has been selected for most outer planet entry missions as the best compromise between drag coefficient, aerodynamic stability, aerodynamic heating including the convective-radiative heating balance for Jovian entry, packaging, science integration, mass properties, and integration with the bus and launch vehicle. The diameter was selected as the largest size probe which integrates well onto the Pioneer bus and lies within the injected weight capability of the Titan IIIE/Centaur/TE 364-4 launch vehicle for the spectrum of outer planet missions.

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* (See Section 6.0, however)

The principal scientific objectives for early missions to make in-situ measurements within the upper reaches of the outer planet atmospheres are:

- o atmospheric structure
- o atmospheric constituents
- o cloud structure and constituents
- o planetary radiation balance
- o atmospheric dynamics

Mini-probes such as the Pioneer Venus Small Probes satisfy the first objective - atmospheric structure - through deceleration, temperature and pressure-altitude profile measurements. The probe contemplated for the early outer planet entry missions is an intermediate size probe with slightly more ambitious objectives. The addition of a 0-40AMU neutral mass spectrometer can return data on expected major constituents of the outer planet atmospheres. In addition, the nephelometer returns basic information on cloud structure, and with the neutral mass spectrometer, on cloud constituents. Further extension of the science payload car [illegible]asily be achieved without increasing the probe size and weig[illegible]ond that which can be integrated into the Pioneer bus for [illegible]h by a Titan IIIE/Centaur Class launch vehicle. The instrumen[illegible] which represent the next logical extension of the science payload also pose pacing integration problems within the probe and are not at the same stage of development as are those of the selected payload.

The decision to retain the same aerodynamic configuration throughout entry and descent rather than stage the entry configuration through the use of deceleration devices during descent significantly reduces the complexity of the probe design and flight profile and increases the probability of a successful probe entry/descent mission. The non-staged entry probe cannot tailor the descent time-altitude profile as optimally as can be accomplished by probe staging, however, adequate descent time for all scientific measurements is assured, with the possible exception of probe descent into the "cool dense" Jovian atmosphere. (See Paragraph 2.1(7)).

The spin-stabilized Pioneer spacecraft was selected as the baseline probe bus because it provides a lower cost and weight probe mission option. The 3-axis stabilized Mariner spacecraft provides a superior platform from which to support the probe. However, the Mariner probe mission improvements and superior flyby science capability are offset by the attendant increases in cost and weight.

3.0 BASELINE DESIGN

The baseline spacecraft is premised upon fulfilling all the Science Objectives of Chapter 2 for entry into the atmosphere of Saturn (including Titan), Uranus and Neptune to a depth of at least 10 bars. The design options and trades to include Jupiter entry are included in Section 4.6. The design presented has been selected as the simplest operational concept within the state-of-the-art. Where options exist which cannot be resolved entirely on technical grounds, the baseline spacecraft design represents the option which is most demanding of the probe with the alternatives discussed in Chapters 4 or 5. The baseline bus is the spin stabilized Pioneer F/G class spacecraft and the system would be launched by the Titan IIIE/Centaur/TE 364-4.

Briefly described, the baseline probe configuration consists of a spherically blunted conical forebody and a hemispherical afterbody. The forebody is a 60-degree half angle cone with a 22.9 cm (9-inch) nose radius and an 89 cm (35-inch) base diameter. The total probe weight is 113.4 kg (250 lb), including a 20% weight contingency resulting in a ballistic coefficient of 121.76 kg/m^2 (0.776 slug/ft^2). The in-board profile of the seline probe is shown in Figure 3-1 and Table 3-1 presents a weight summary. Figure 3-2 conceptually presents the probe/bus mounting.

The baseline probe will accommodate a science payload consisting of five instruments shown in Table 3-2, and this payload has been selected to provide basic information characterizing the planetary atmosphere with an instrumentation complement which provides ample measurement redundancy in the event of any single instrument failure. Table 3-3 presents the primary measurement objectives.

A pictorial sequence of events for the spacecraft, bus and probe is shown in Figures 3-3 and 3-4. During its interplanetary trajectory, the spacecraft is targeted to the probe entry point, and midcourse trajectory corrections are applied as required to

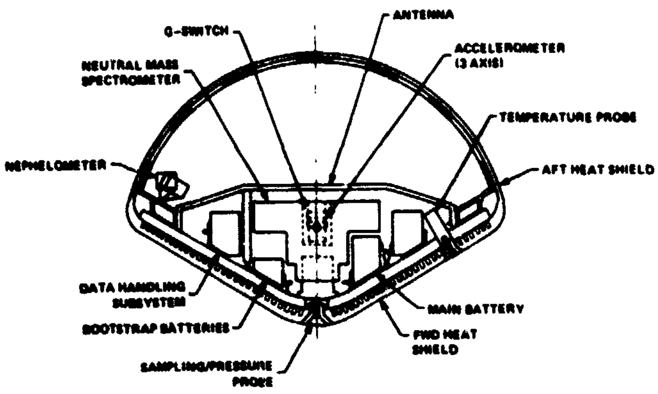


Figure 3-1. Probe in-Board Profile

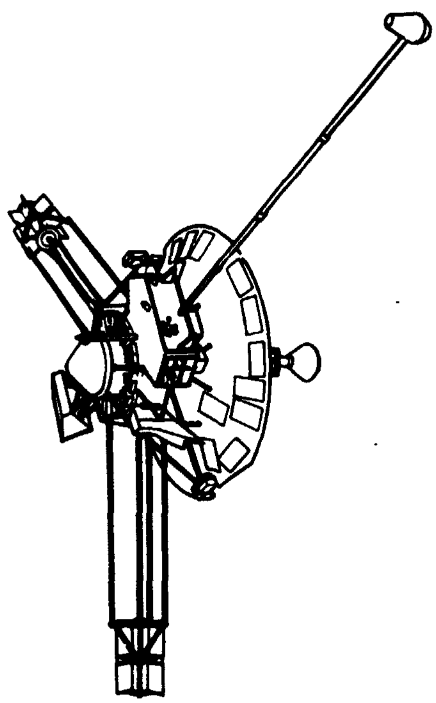


Figure 3-2. Probe on Pioneer Class Bux

Table 3-1 Probe Weight Summary

DESCRIPTION	WEIGHT (KG)	WEIGHT (LB)
STRUCTURE	13.42	29.59
HEATSHIELDS	36.63	80.75
THERMAL CONTROL	6.96	15.32
COMMUNICATIONS	9.57	21.09
ELECTRICAL POWER	9.08	20.03
PYROTECHNICS	3.74	8.24
SCIENCE PAYLOAD	10.96	24.17
INSTRUMENTATION	0.73	1.60
WEIGHT MARGIN	22.32	49.21
PROBE WEIGHT	113.40	250.00
LESS		
INTERFACE WIRING	-1.07	-2.35
EXTERNAL INSULATION	-2.66	-5.86
BEGIN ENTRY	109.67	241.79
LESS		
ABLATED MATERIAL	-8.60	-18.97
POST HEATING	101.07	222.82

Table 3-2 Baseline Science Payload

INSTRUMENT	TYPE	RANGE	WEIGHT		VOLUME		PWR (watts)
			kg	lb	cm^3	in^3	
Pressure Sensor	Capacitive	0.01 to 20 bar	0.2	0.44	181	11.1	1.2 AVG
Temperature Sensor	Resistance Wire	40 to 500°K	0.35	0.77	65	3.95	1.0 AVG
Accelerometer Triad	Force Rebalance	0.01 to 1000 G_E long. ±10 G_E lat.	0.3	0.66	101	6.2	8.2 PEAK 2.0 AVG
Neutral Mass Spectrometer	Quadropole	1 - 40 AMU	6.4	14.1	7246	442	11.0 AVG
Nephelometer	Light Backscatter	–	0.5	1.1	427	26	1.2 PEAK 1.0 AVG
TOTALS			7.75	17.1	8020	490	16.2 AVG

Table 3-3 Scientific Measurement Objectives

SCIENTIFIC OBJECTIVE	PRESS.	TEMP	ACC.	NMS	NEPH.
Atmospheric Density	X	X	△	X	
Atmospheric Temperature	X	△	X	X	
Atmospheric Pressure	△	X	X	X	
Atmospheric Constituents	X	X		△	X
Cloud Location/Structure	X	X	X	X	△
Cloud Composition	X	X	X	△	X
Atmospheric Turbulence	X	X	X		X

△ Direct Measurement
X Related Measurement

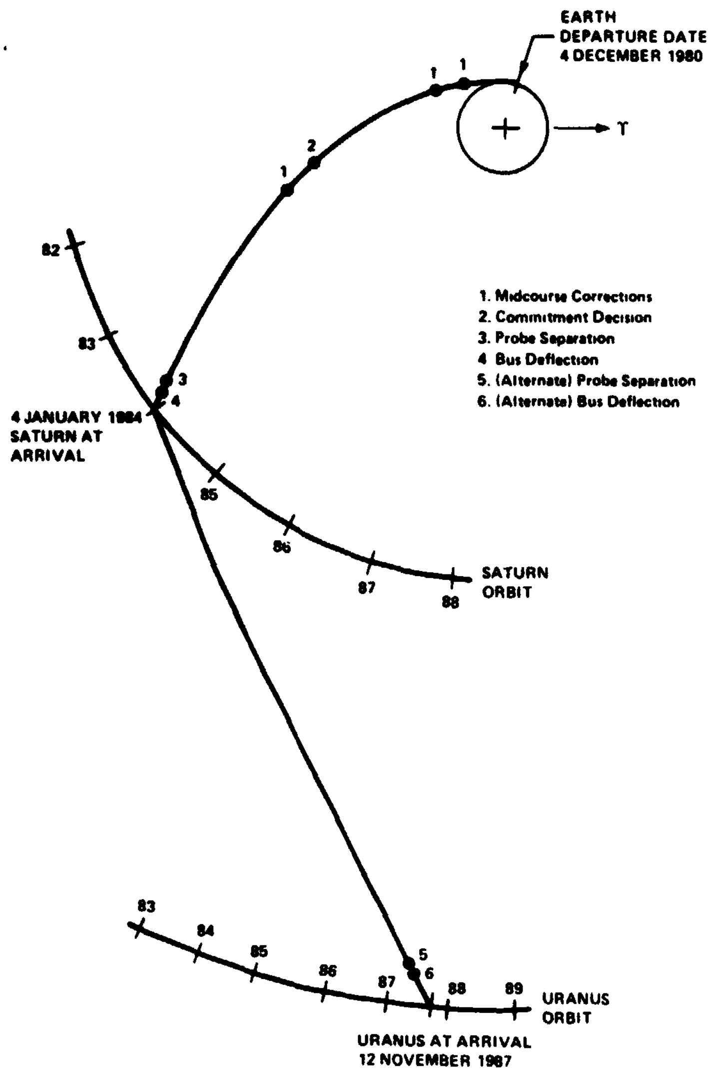


Figure 3-3a. Saturn/Uranus Mission Profile

MJU 79 REFERENCE TRAJECTORY

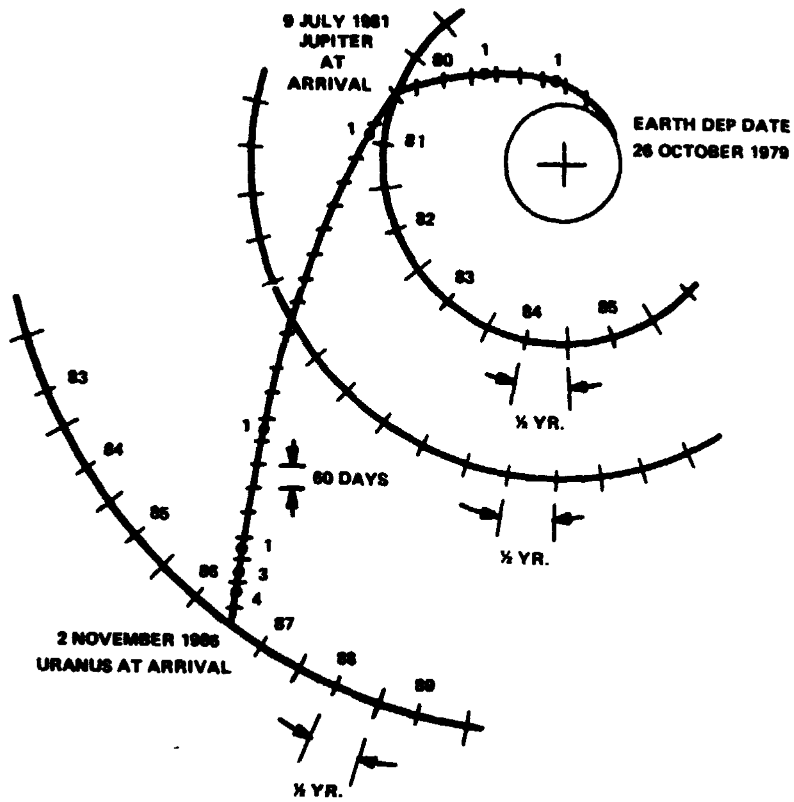


Figure 3-3b. Jupiter/Uranus Mission Profile

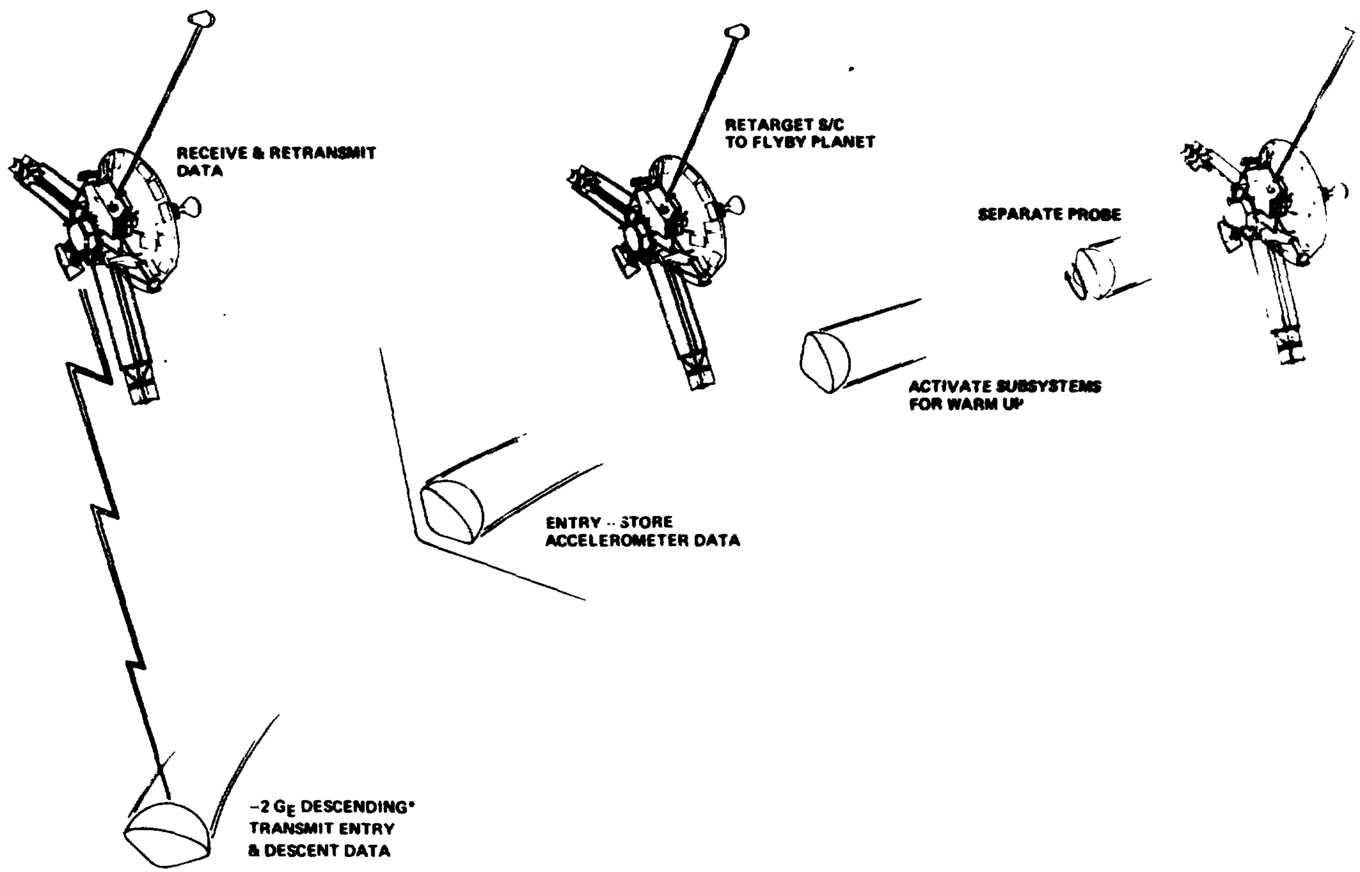


Figure 3-4. Probe Deployment Sequence

maintain the entry coordinates and entry time within design limits. The probe is released from the spinning spacecraft approximately 25 to 30 million kilometers from the planet and the bus is then retargeted to overfly the probe during probe entry. (The best communication geometry is generally obtained when the bus is directly over the probe at the end of the probe mission.) The Pioneer bus retargeting maneuver must be accomplished without losing spacecraft communications lock with Earth. This requires the deflection maneuver to be made in two separate thrusting operations, one along the Earth line (spacecraft axis) and one normal to the Earth line.

During the post separation coast phase, the probe is essentially dormant except for the entry timer which is initiated prior to separation. During this long coast the battery remains very cold and thus essentially remains at its full charge. The entry timer activates probe subsystems nominally 40 minutes ($\pm$ 5 minutes) before entry. At this time a pyrotechnic gas generator activates the battery by forcing the electrolyte from a reservoir into the cells. The battery heater, programmer, accelerometers, engineering instruments, and transmitter oscillators are also activated. After 30 minutes, the battery heater is deactivated; the battery temperature having increased enough to allow the battery to support the remaining load.

During entry, from first indication of deceleration ($-0.0004G_E$) until $-2G_E$ descending, 3-axis acceleration data are stored in a solid state memory for subsequent playback during descent. The other science instruments are activated at $-2G_E$ (descending) and remain activated throughout the mission. Typical sample intervals, quantization levels and the resulting data rates during entry and descent are shown in Table 3-4.

Concepts for the inlet design for the pressure sensor and neutral mass spectrometer, the temperature sensor deployment mechanism, and the nephelometer installation are shown in Figures 3-5 thru 3-7. Detail design of the integration of these instruments should

Table 3-4 Data Rate Requirements

DATA TYPE	SAMPLE INTERVAL (sec) Entry	SAMPLE INTERVAL (sec) Descent	Word Length (bits)	Sample Length (words)	DATA RATE (bits/sec) Entry	DATA RATE (bits/sec) Descent
Pressure	--	50	10	1	--	0.2
Temperature	--	50	10	1	--	0.2
Acceleration						
Longitudinal	0.2	50	10	1	50	0.2
Lateral (Each Axis)	0.2	50	7,10*	1	70	0.2
Neutral Mass Spectrometer	--	405	9	634	--	14
Nephelometer	--	20	10 6	4 3	--	2
Engineering and Calibration	0.2	Various	6	1	30	2
Housekeeping	--	--	--	--	30	3.12
Total Entry Data Rate					$\boxed{180}$	
Entry Data Playback						22
Total Descent Data Rate						$\boxed{44}$

*7 during entry, 10 during descent

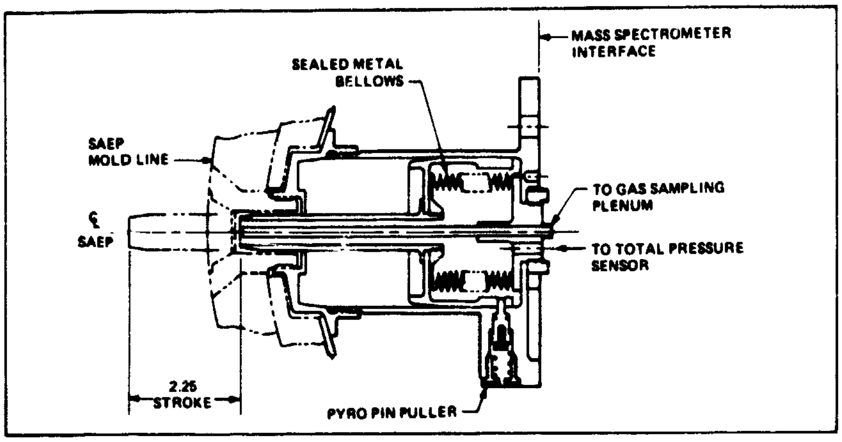


Figure 3-5. Gas Sampling/Pressure Probe

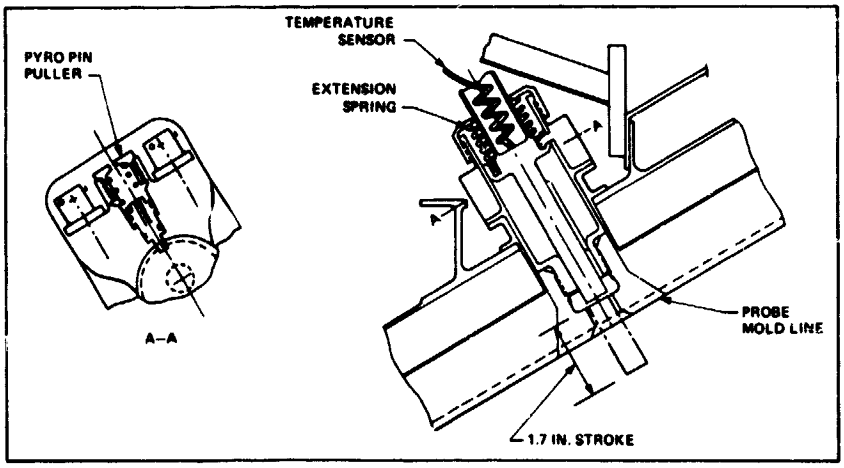


Figure 3-6. Temperature Probe

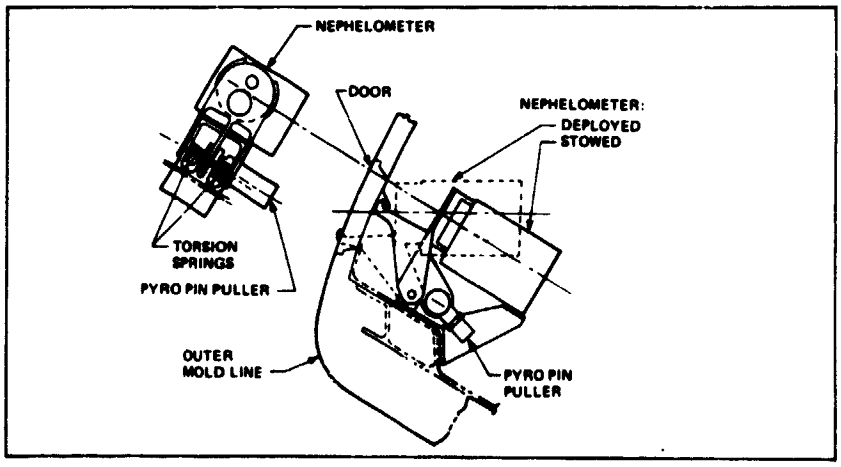


Figure 3-7. Nephelometer Installation

be based upon development projects as recommended in Section 7 of this report.

The communications sub-system on the probe is required to transmit science and engineering data to the spacecraft bus for relay to Earth continuously from entry throughout the remainder of the mission. The subsystem operates at a carrier frequency of 400 MHz to reduce atmospheric losses and to gain best advantage from the low gain receiving antenna on the spinning spacecraft. The characteristics of the communications subsystem are shown in Table 3-5.

Table 3-5 Communications Subsystem Characteristics

Transmitter Power	40 watts
Probe Antenna Gain	7.0 dB on axis
Probe Antenna beamwidth	66° axisymmetric
Bus Antenna Gain	3.8 dB peak
Bus Antenna beamwidth	50° Torroidal (65° off roll axis)
Communications range	120,000 km
Bit rate	44 bits/second

The transmitter crystal operates for 40 minutes prior to entry to stabilize the carrier frequency and at the termination of entry ($-2G_E$ descending) the power amplifier is activated and data transmission begins. Science and engineering data, interleaved with the playback of stored entry data, are transmitted continuously until the probe subsystems cease operation at some altitude below the 10-bar pressure level, or until the bus passes from view.

The probe uses a microstrip antenna with a peak on-axis gain of 7.0 dB. The measured pattern is broad with a 66-degree 3 dB beamwidth. The pattern falls off gradually beyond 66 degrees and the 100-degree beamwidth gain is still above isotropic (+0.6 dB).

The receiving antenna on the Pioneer bus is a loop vee antenna with a preset beam center (110° off roll axis) and a 50-degree 3 dB beamwidth and a peak gain of 3.8 dB.

A 44 bit per second data rate is transmitted at 40 watts using FSK modulation with a rate 1/2 Viterbi code.

The major probe subsystems are powered by a 28-volt 239 watt-hour remotely activated silver zinc battery packaged with its auto-activation mechanism in a toroidal shape to allow packaging within the probe for the most favorable forward c.g. location. The probe timer, main battery activation, and ordnance functions are powered by two 6-volt 12-watthour nickel cadmium bootstrap batteries, activated during manufacture, and with a storage capacity far exceeding that required for the timer and ordnance functions (0.12 watt hours). The batteries will support the ordnance firing current requirement.

The energy requirements of the probe subsystems are listed in Table 3-6.

Table 3-6 Equipment Power/Energy Requirements

EQUIPMENT	UNIT POWER (WATTS)	TIME (MIN.)	ENERGY (WATT-HOURS)
ENTRY DETECTION:			
X-DAY CLOCKS (2)	140×10^{-6}	51,420	0.24
G-SWITCH	0.2	45	0.15
DATA HANDLING SUBSYSTEM	10.0	145	24.0
TRANSMITTER - OSC/MOD	1.0	145	2.4
POWER AMPLIFIER	89	100	148.0
SCIENCE:			
MASS SPECTROMETER	11.0	110	20.0
GETTER PUMP HEATER	30.0	10	5.0
ORDNANCE RELAYS (16)	3.0	0.001	0.05
ACCELEROMETER	2.0	145	4.8
PRESSURE GAGE	1.2	100	2.0
TEMPERATURE GAGE	1.0	100	1.7
NEPHELOMETER	2.0	100	3.3
HEATER	1.0	10	0.17
ORDNANCE RELAYS (14)	3.0	0.001	0.04
BATTERY HEATER	30	30	15.0
EQUIPMENT ENERGY			227.0
DISTRIBUTION LOSSES (5%)			11.0
TOTAL ENERGY REQUIRED			238.0

The data handling subsystem provides programming, and power switching for electrical power control and distribution. The data handling system contains the probe timers and the initiation relays. Commands from the bus, the probe timers and/or inertia switches activate the initiation relays. Battery power is distributed directly to the user equipment and any regulation or conversion is done within the user equipment.

The resulting power profile is shown in Figure 3-8.

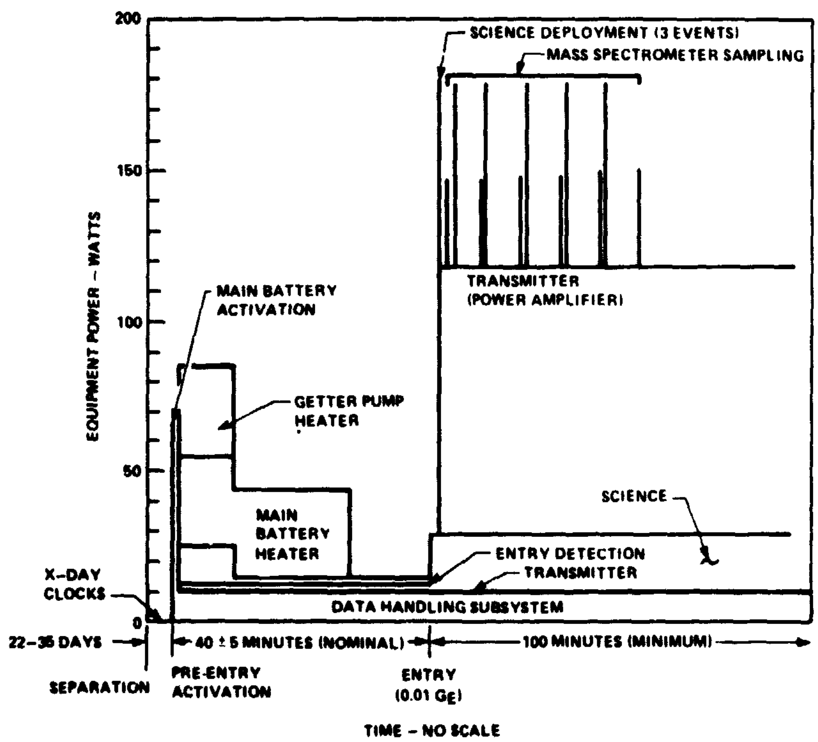


Figure 3-8. Power Profile

Pyrotechnic devices are used on the probe to initiate operational sequences. These devices are considered Category B ordnance in terms of the ETR safety requirements. The bridgewires are of 1-ampere, 1-watt, no fire design. Two positive actions are necessary to initiate the pyrotechnic. The ordnance functions which must be safed and initiated are listed in Table 3-7.

Table 3-7 Ordnance Functions

DEVICE	POWER SOURCE	INITIATOR QTY
URANUS DECISION (PYRO RELAY)	SPACECRAFT	2
INITIATE PROBE (PYRO RELAY)	SPACECRAFT	2
SEVER UMBILICAL CABLE	SPACECRAFT	2
SEPARATION PINS (3)	SPACECRAFT	6
INITIATE BATTERY	PROBE	2
OPEN NEPHELOMETER PORT HOLE	PROBE	2
DEPLOY TEMPERATURE PROBE	PROBE	2
DEPLOY MASS SPECTROMETER TUBE	PROBE	2
MASS SPECTROMETER:		
OPEN SAMPLING TUBE 1	PROBE	1
CLOSE SAMPLING TUBE 1	PROBE	2
OPEN SAMPLING TUBE 2	PROBE	1
CLOSE SAMPLING TUBE 2	PROBE	2
OPEN SAMPLING TUBE 3	PROBE	1
CLOSE SAMPLING TUBE 3	PROBE	2
OPEN SAMPLING TUBE 4	PROBE	1
CLOSE SAMPLING TUBE 4	PROBE	2
OPEN SAMPLING TUBE 5	PROBE	1
CLOSE SAMPLING TUBE 5	PROBE	2
OPEN SAMPLING TUBE 6	PROBE	1

All are initiated by Single Bridgewire Apollo Standard Initiators with the power to activate supplied directly from the bootstrap batteries.

The entry environment encountered at the outer planets varies with mission design atmospheric model. The design requirements necessary to accommodate the planets considered are:

	Excluding Jupiter	Jupiter
entry loads	$750G_E$	$400G_E$
entry peak dynamic pressure	1.0 MN/m^2	0.5 MN/m^2
entry peak heating rate	295 MW/m^2	193 MW/m^2
entry maximum integrated heating	613 $MW\text{-}sec/m^2$	1250 $MW\text{-}sec/m^2$

The probe forebody structure is a honeycomb sandwich aeroshell consisting of a fiberglass outer face sheet, a fiberglass honeycomb core and an aluminum inner face sheet with four integrally machined rings. The four internal stiffening rings help react the circumferential loads, distribute concentrated inertial loads, and provide equipment mounting surfaces. The afterbody structure is an all-fiberglass honeycomb dome transparent to radio transmission.

The heat shield is dense carbon phenolic-1448 kg/m^3 (90 lb/ft^3), bonded directly to the forebody fiberglass face sheet. The inner section of the structural heat shield is hollowed out to reduce density -579 kg/m^3, (36 lb/ft^3) and thermal conductivity. The afterbody heat shield is rf transparent elastomeric -322 kg/m^3, (20 lb/ft^3).

The probe is subjected to extremes in the external environment from near absolute zero during interplanetary flight to the atmospheric heating during the short entry phase. The thermal control system must withstand these extremes and maintain an acceptable thermal environment for the probe subsystems. The non-operating and operating temperature requirements of the equipment are shown in Table 3-8.

Table 3-8 Equipment Temperature Requirements

ITEM	NON-OPERATING TEMPERATURE LIMITS (°F)	OPERATING TEMPERATURE LIMITS (°F)
DATA HANDLING	-40 to 160	-40 to 160
TRANSMITTER	-40 to 160	-40 to 160
MASS SPECTROMETER	-65 to 250	-20 to 160
TEMPERATURE GAGE	-65 to 250	-65 to 200
PRESSURE GAGE	-65 to 250	-65 to 200
ACCELEROMETER	-65 to 250	-40 to 160
NEPHELOMETER	-45 to 160	-20 to 160
MAIN AgZn BATTERY	-40 to 80	40 to 145
NiCd BATTERY	-40 to 32	-20 to 100

The thermal control subsystem consists of a multi-layer insulation blanket surrounding the probe until it burns off during entry, 4-one watt radioisotope heater units within the equipment compartment, surface coatings on the probe-spacecraft adapter section and electrical heaters powered by the spacecraft while the probe remains attached during interplanetary flight. The thermal control subsystem is shown schematically in Figure 3-9.

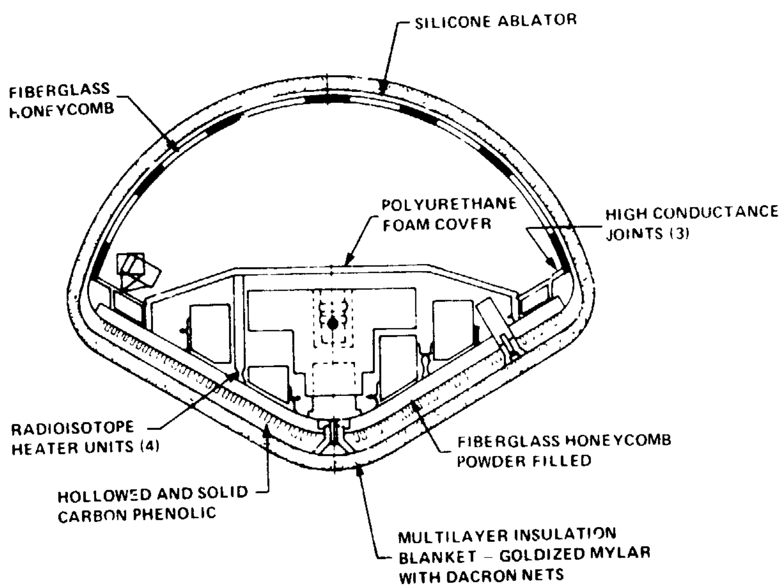


Figure 3-9. Thermal Control System

4.0 PROBE SYSTEM AND SUBSYSTEM TRADES

This section discusses the major probe system and subsystem trades and the basis for selecting the baseline design of Section 3. The reference system studies which provide the source material for this document each present self-consistent probe system and subsystem design concepts. There are, however, both major and minor differences between these concepts. It is not the intent of this summary to present the trades which have led to design choices made in common in all of the reference studies unless they are of major importance to the probe design. These trades are well documented in the references. It is rather the intent of this Section 4 summary to assemble the key data on major probe trades between different approaches taken in the reference studies. As stated in the Foreword, information from the Reference 12 and 14 studies became available subsequent to this one. Most of the results of those studies do not impact the selected baseline probe configuration; however, their basis (use of Pioneer Venus equipment for a Pioneer Saturn/Uranus mission) established designs from different requirements. The configuration resulting from these is summarized in Section 6 herein.

4.1 Mission Parameters

The scientific objectives of the Outer Planet Entry Probe mission are to explore the atmospheres of Jupiter, Saturn, Titan, Uranus and Neptune to:

(1) Determine the structure and composition of the atmospheres to a depth of at least 10 bars, and

(2) Determine the altitude and composition of any clouds which may be present

Several mission opportunities have been examined, however, the Saturn 1979, Saturn-Uranus 1980, and the Jupiter-Uranus-Neptune 1979 opportunities are considered typical. The Saturn Mission opportunities can also provide for Titan encounter. The multiple planet missions are designed to allow the option of a probe mission at any of the intermediate planets or by using the swingby mode, extending the probe mission to the most distant planet. The baseline probe

(Chapter 3) is designed to accommodate entry into the atmosphere of any of the five planets of interest (except Jupiter*) without change in the probe hardware.

The matrix of missions examined is drawn from the literature. These missions and the principal characteristics of each are shown in Table 4-1.

4.2 Probe System Configuration Trades

The major requirement which dictates the probe configuration is the acquisition and transmission of data about the planetary atmospheres. The probe must complete the entry phase above the altitude at which measurements begin and must transmit the data taken down to a pressure altitude of at least 10 bars before the bus has passed from view. The major configuration parameters which must be selected include: entry shape, ballistic coefficient, entry/descent staging (if required), and multi-planet configurations.

The model atmospheres used for this investigation are given in the literature** and shown graphically in Figure 4-1. Except for the Titan models, they have been used extensively to study outer planet entry. The principal constituents of the model atmospheres are molecular hydrogen, helium, water, methane, neon, ammonia, and possibly nitrogen (Titan). Three models (cool dense, nominal and warm extended) are given for each planet. For Titan, two models (dry adiabatic lapse rate, and wet adiabatic lapse rate) are given.

Most outer planet probe studies have selected the 60° blunt cone as the most effective forebody configuration for the probe. This shape provides a high drag coefficient, good aerodynamic

* The probe design can be used for entry at Jupiter with a change in aeroshell heat shield and structure. All internal components remain as shown, however, science data may be degraded. See Section 4.2 and 4.6.

**References (31), (51), (52), (53)

Table 4-1 Mission Parameters

	MISSION MODE	LAUNCH DATE	ARRIVAL DATE	C_3 (km^2/sec^2)	TRIP TIME (DAYS)	LAUNCH WINDOW (DAYS)	TARGET PLANET FLYBY RADIUS	PROBE RELEASE RADIUS (10^6 km)	DEFLECTION MODE	DEFLECTION ΔV (m/s)	COMMUNICATION LEAD TIME (MIN)	MAX. COMMUN. RANGE (10^3 km)	ENTRY ANGLE γ_E (DEG)	ENTRY VELOCITY V_E (km/sec)	ENTRY ANGLE OF ATTACK α_E (DEG)
Jupiter 78	–	10/10/78	8/10/80	122	670	20	2 R_J	10	Probe	221	–	–	–20	59.7	0
	–	9/26/78	11/4/81	99	1,136	20	2 R_J	10	Probe	241	–	155	–30	49	0
	–	10/10/78	3/27/80	112	534	30	1.6 R_J	30	Probe	44	27	68	–10	48	0
▶ Jupiter 79	–	11/7/79	9/17/81	105	680	20	2 R_J	10	Probe	221	–	–	–20	59.7	0 ◀
▶	–	11/7/79	9/17/81	105	680	20	2 R_J	30	Probe	71	–	–	–15	59.7	0 ◀
	–	11/7/79	9/17/81	105	680	20	6 R_J	30	Probe	256	–	–	–15	59.7	0
	–	11/6/79	4/30/81	112	541	30	6.8 R_J	30	Probe	337	114	437	–15	48	0
	–	11/6/79	4/30/81	112	541	30	6.8 R_J	30	Probe	312	114	437	–15	48	0
Saturn 77	JS	9/5/77	12/8/80	111	1,190	20	2 R_S	10	Probe	221	–	–	–20	–	0
	JST	9/4/77	2/16/81	107	1,261	20	2.3 R_S	10	Probe	170	–	–	–25	–	0
▶ Saturn 79	–	11/21/79	3/2/83	140	1,197	15	2.3 R_S	30	S/C	52	89	108	–30	–	0 ± 1.8 ◀
▶	–	11/22/79	4/8/83	135	1,233	15	2.25 R_S	28.9	S/C	69	60	101	–30	36.8	11 ± 2.2 ◀
▶ Saturn 80	SU	12/3/80	5/9/84	140	1,258	15	3.8 R_S	30	S/C	96	126	197	–30	–	0 ± 19 ◀
▶	SU	12/4/80	1/4/84	140	1,126	11	2.3 R_S	28.9	S/C	73	60	104	–30	37.1	10.4 ± 2.6 ◀
▶	SU	12/1/80	1/4/84	141	1,129	10	2.3 R_S	30	S/C	70	82	107	–30	37	13.5 ± 1.2 ◀
▶ Uranus 79	JUN	11/6/79	5/19/86	113	2,386	20	2.4 R_U	9.8	Probe	170	–	–	–60		0 ◀
▶	JUN	10/26/79	11/2/36	107	2,564	21	3.0 R_U	17.5	S/C	80	–	–	–40	–	0 ◀
▶ Uranus 80	SU	12/3/80	8/24/88	140	2,826	15	2 R_U	10	S/C	85	111	87	–60	–	0 ± 3.2 ◀
▶	SU	12/4/80	11/12/87	140	2,533	11	3.5 R_U	27	S/C	68	48	107	–40	25	15.4 ± 2.6 ◀
▶	SU	12/1/80	11/5/87	141	2,529	10	3 R_U	25	S/C	61	92	94	–35	25	7.3 ± 4.2 ◀
▶ Neptune 79	JUN	11/6/79	3/12/90	113	3,779	15	3 R_N	10	Probe	95	–	–	–20	–	0 ◀
	JUN	10/26/79	12/21/90	107	4,074	21	3.0 R_N	20	S/C	80	–	–	–20	–	0
▶ Titan 78	JST	11/9/78	3/7/83	96	1,610	20	4.2 R_T	2.2	S/C	200	120	59	–60	5.8	40 ◀

▶ Reference Missions ◀

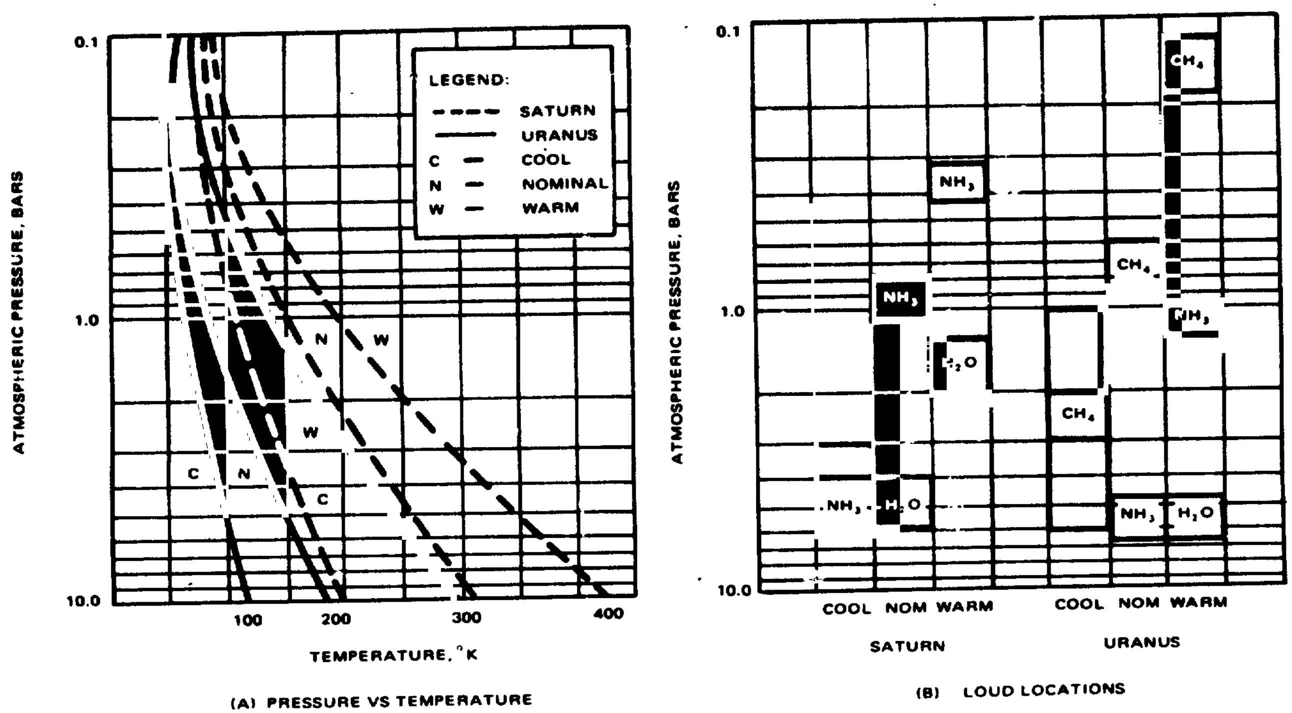


Figure 4-1. Model Atmospheres in the Probe Terminal Descent Region

stability throughout the flight regime, efficient packaging volume, and minimizes the required heat shield weight. Figure 4-2 shows the heat shield mass fraction (relative to total probe entry weight) as a function of cone half-angle for different entry angles (the curves on the figure happen to be for the planet Jupiter, but the shapes are believed typical for entry into Saturn and Uranus). Increasing the half angle significantly above 60° increases the radiative heating component over much of the probe forebody without significantly increasing the drag coefficient (and thus not reducing total heating input).

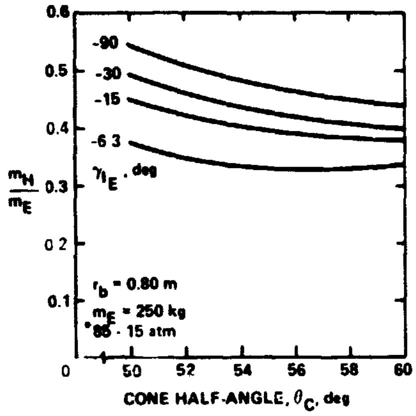


Figure 4-2. Variation of Heat Shield Mass Fraction with Cone Half Angle

A comparison of the cold wall entry conditions as a function of the planet considered, probe shape, atmospheric model, entry angle and ballistic coefficient is shown in Tables 4-2, 4-3, & 4-4. The reference shapes considered are shown in Figure 4-3.

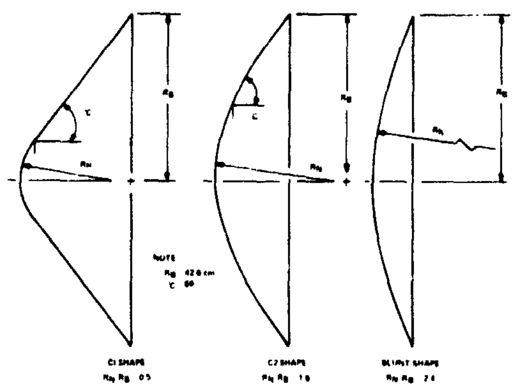


Figure 4-3. Entry Probe Forebody Configurations Evaluated in Study

*85% H_2 - 15% He

Table 4-2 Cold Wall Entry Conditions for the C1 Shape in the Saturn Entry Environment

Case	Atmos.	Entry Angle γ (Deg)	Ballistic Coef. (g/cm²)	Entry Velocity V_E (km/sec)	Maximum Pressure P (atm)	Maximum Heat Rate q (kw/cm²)	Heating Pulse (kw-sec/cm²)		
							Total q_t	Conv. q_c	Rad. q_r
				Stagnation Point					
1	Cold-Dense	−50.6	14.9	30.4	19.6	26.75	64.08	37.7	26.3
2	Cold-Dense	−19.6	14.9	28.2	7.36	7.11	56.2	48.6	7.57
3	Warm	−19.6	14.9	28.2	2.6	2.57	66.7	65.6	0.08
4	Warm	−50.6	14.9	30.4	6.94	5.32	54.6	53.4	1.2
5	Nominal	−50.6	12.1	32.0	9.53	9.42	52.7	45.8	6.97
6	Nominal	−19.6	12.1	32.0	3.85	5.16	75.7	70.6	5.05
13	Cold-Dense	−50.6	17.7	30.4	23.2	33.49	78.4	40.9	37.5
14	Cold-Dense	−50.6	12.1	32.0	14.82	29.2	76.4	41.4	34.9
18	Cold-Dense	−19.6	12.1	32.0	6.0	11.73	87.4	62.7	24.7
19	Nominal	−15.0	14.0	32.0	4.40	5.72	88.6	31.9	6.7
20	Nominal	−25.0	14.0	32.0	7.16	7.86	72.7	64.7	7.95
21	Nominal	−40.0	14.0	32.0	10.87	10.58	62.4	52.9	9.48
22	Cold-Dense	−15.0	14.0	32.0	6.85	12.61	101.4	73.1	28.3
24	Cold-Dense	−40.0	14.0	32.0	16.94	33.06	90.1	46.3	43.7
				Body Position R = 0.634 R_B					
1	Cold-Dense	−50.6	14.9	30.4	17.0	33.4	91.4	67.04	24.3
2	Cold-Dense	−19.6	14.9	28.2	6.37	9.74	73.1	67.6	5.47
3	Warm	−19.6	14.9	28.2	2.23	3.19	72.86	72.8	0.03
4	Warm	−50.6	14.9	30.4	5.97	8.45	76.1	75.6	0.47
5	Nominal	−50.6	12.1	32.0	8.21	13.44	74.8	68.7	6.14
6	Nominal	−19.6	12.1	32.0	3.31	6.57	85.7	81.4	4.26
13	Cold-Dense	−50.6	17.7	30.4	20.22	41.5	113.3	77.8	35.4
14	Cold-Dense	−50.6	12.1	32.0	12.97	35.7	100.4	64.3	36.1
18	Cold-Dense	−19.6	12.1	32.0	5.23	13.82	103.1	77.0	26.1
19	Nominal	−15.0	14.0	32.0	3.78	7.30	98.3	93.7	4.9
20	Nominal	−25.0	14.0	32.0	6.16	10.72	91.5	85.5	5.95
21	Nominal	−40.0	14.0	32.0	9.37	14.9	86.3	79.1	7.21
22	Cold-Dense	−15.0	14.0	32.0	5.97	14.65	118.2	89.3	28.9
24	Cold-Dense	−40.0	14.0	32.0	14.8	42.16	119.0	74.0	45.0

Table 4-3 Cold Wall Entry Conditions for the C1 Shape in the Uranus Entry Environment

Case	Atmos.	Entry Angle γ (Deg)	Ballistic Coef. (g/cm²)	Entry Velocity V_E (km/sec)	Maximum Pressure P (atm)	Maximum Heat Rate q (kw/cm²)	Heating Pulse (kw-sec/cm²)		
							Total q_t	Conv. q_c	Rad. q_r
				Stagnation Point					
46	Cold-Dense	−49.4	14.9	26.4	20.6	63.2	108.3	36.0	72.3
47	Warm	−39.4	14.9	26.4	3.3	2.44	48.0	48.6	0.01
49	Cold-Dense	−24.4	14.9	26.4	11.4	34.2	113.4	48.6	64.9
51	Nominal	−25.0	14.0	25.2	4.3	3.0	41.0	40.9	0.08
52	Nominal	−40.0	14.0	25.2	6.53	3.75	33.9	33.9	0.09
53	Nominal	−50.0	14.0	25.2	7.77	4.12	31.3	31.2	0.11
54	Cold-Dense	−25.0	14.0	25.2	9.97	22.67	81.8	41.2	40.6
55	Cold-Dense	−40.0	14.0	25.2	15.15	34.89	80.3	33.7	46.6
56	Cold-Dense	−50.0	14.0	25.2	18.1	42.6	79.6	31.0	48.8
				Body Position R = 0.634 R_B					
46	Cold-Dense	−49.4	14.9	26.4	18.36	74.47	131.8	53.4	78.5
47	Warm	−39.4	14.9	26.4	2.84	3.34	57.9	57.9	0.07
49	Cold-Dense	−24.4	14.9	26.4	10.16	40.3	130.5	60.5	70.0
51	Nominal	−25.0	14.0	25.2	3.70	4.49	53.5	53.5	0.02
52	Nominal	−40.0	14.0	25.2	5.63	6.34	49.5	49.5	0.03
53	Nominal	−50.0	14.0	25.2	6.71	7.31	47.8	47.8	0.04
54	Cold-Dense	−25.0	14.0	25.2	8.85	26.9	94.7	52.5	42.2
55	Cold-Dense	−40.0	14.0	25.2	13.45	41.8	96.9	48.5	48.4
56	Cold-Dense	−50.0	14.0	25.2	16.02	50.4	97.5	46.9	50.5

Table 4-4 Effect of Shape on the Cold Wall Entry Environment

Case	Shape	Atmos.	Entry Angle γ (Deg)	Ballistic Coef. (g/cm²)	Entry Velocity V_E (km/sec)	Maximum Pressure P (atm)	Maximum Heat Rate q (kw/cm²)	Heating Pulse (kw-sec/cm²)		
								Total q_t	Conv. q_c	Rad. q_r
					Stagnation Point, Saturn Entries					
25	C2	Cold-Dense	−15	14.0	32.0	6.855	14.9	104.9	61.5	43.3
26	C2	Cold-Dense	−40	14.0	32.0	16.94	39.9	98.4	39.3	59.1
27	Blunt	Nominal	−15	15.8	32.0	4.96	6.33	78.5	57.2	21.3
29	Blunt	Nominal	−40	15.8	32.0	12.25	14.75	66.9	38.6	28.3
30	Blunt	Cold-Dense	−15	15.8	32.0	7.73	21.8	138.1	52.9	85.1
31	Blunt	Cold-Dense	−40	15.8	32.0	19.09	60.2	141.7	33.7	108.1
32	C2	Nominal	−15	14.0	32.0	4.39	5.4	79.0	69.5	9.53
34	C2	Nominal	−40	14.0	32.0	10.87	10.82	58.5	44.8	13.6
					Stagnation Point, Uranus Entries					
64	Blunt	Nominal	−40	15.8	25.2	7.37	2.78	25.1	24.6	0.4
67	Blunt	Cold-Dense	−40	15.8	25.2	17.1	64.2	131.0	24.7	106.3
71	C2	Nominal	−25	14.0	25.2	4.302	2.54	34.8	34.6	0.13
72	C2	Nominal	−50	14.0	25.2	7.777	3.50	26.5	26.3	0.18
					Body Position R = 0.634 R_B, Saturn Entries					
26	C2	Cold-Dense	−40	14.0	32.0	15.05	44.9	126.6	78.8	47.7
27	Blunt	Nominal	−15	15.8	32.0	4.69	7.47	102.0	87.6	14.5
31	Blunt	Cold-Dense	−40	15.8	32.0	18.14	61.4	160.3	73.1	87.2
32	C2	Nominal	−15	14.0	32.0	3.85	7.79	105.8	100.2	5.61
					Body Position R = 0.634 R_B, Uranus Entries					
64	Blunt	Nominal	−40	15.8	25.2	6.97	6.34	49.0	48.9	0.16
67	Blunt	Cold-Dense	−40	15.8	25.2	16.3	63.7	138.9	50.5	88.3
71	C2	Nominal	−25	14.0	25.2	3.77	4.79	56.9	56.8	0.03
72	C2	Nominal	−50	14.0	25.2	6.83	7.78	[illegible]	50.7	0.04

Figure 4-4 indicates a few of the afterbody configurations which have been studied. The most effective configuration appears to be the full diameter spherical section afterbody with its center of curvature at the vehicle center of gravity, however, additional ballistic range tests and wind tunnel validations are needed to verify characteristics. The forces resulting from fluctuating afterbody pressure all act through the centroid producing no net destabilizing moment. This configuration has a hypersonic zero

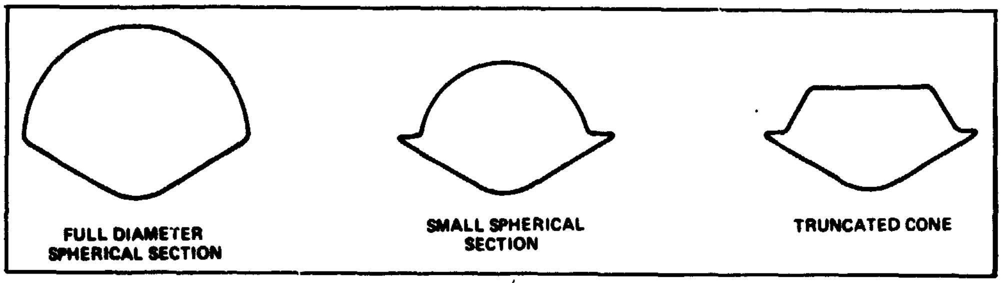


Figure 4-4. Afterbody Configuration

angle of attack drag coefficient of 1.34 reducing to 0.92 at subsonic velocities. Ballistic range testing has been conducted for this shape over the transonic and subsonic flight regimes. The aerodynamic coefficients derived from this test program were used to evaluate the stability of the probe in a six-degree of freedom dynamic analysis. A typical entry into the Saturn nominal atmosphere with an initial angle of attack of 12.6 degrees and a spin rate of 5 revolutions per minute is shown in Figure 4-5 and 4-6.

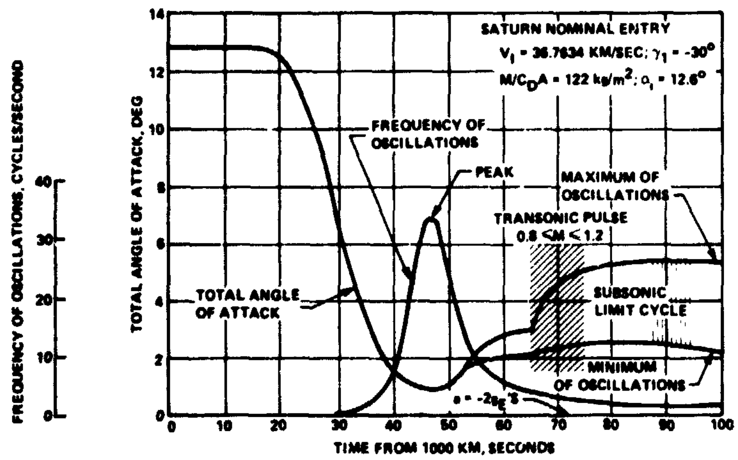


Figure 4-5. Motion Analysis

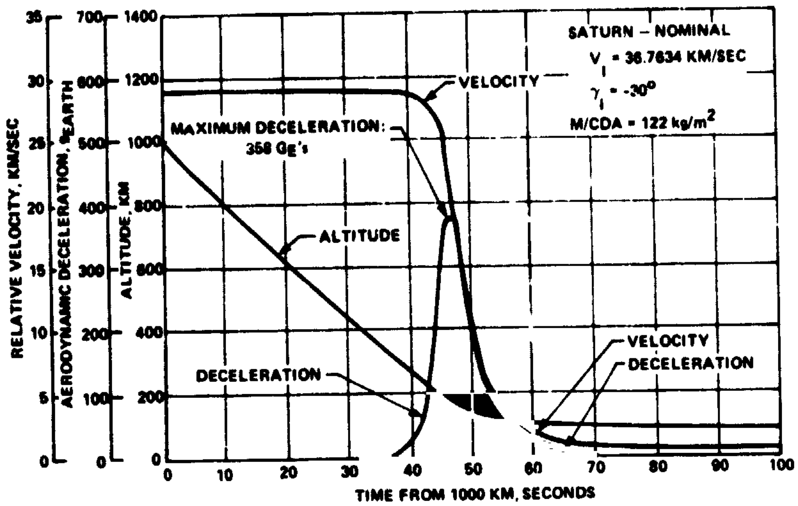


Figure 4-6. Motion Analysis

The probe ballistic coefficient must be selected to satisfy a set of potentially conflicting requirements. The ballistic coefficient must be low enough to complete the entry phase above the highest altitude of interest so that post entry measurements can begin, and it must be high enough to descend through the atmosphere and transmit all data taken to at least the 10-bar pressure altitude level while the flyby bus is still overhead. In most studies, these conflicting requirements are satisfied by staging the entry probe, thereby providing a different ballistic coefficient for the entry, post entry, and sometimes, the descent mission phases as shown in Figure 4-7. The design conditions which constrain the ballistic coefficient have been selected somewhat differently in various studies. For staged entry probes, the criterion for parachute deployment is generally taken as Mach 0.7 above 100 millibar pressure altitude. For non-staged entry probes, the criterion for initiating measurements has typically been taken as $-2G_E$ descending above the tropopause (-0.7 to $-3G_E$ was used). These criteria are most severe when applied to entry through the cool dense atmosphere models and are listed in Table 4-5. The entry into the Jovian cool dense atmosphere presents the most severe constraint on the ballistic coefficient as shown in Figure 4-8.

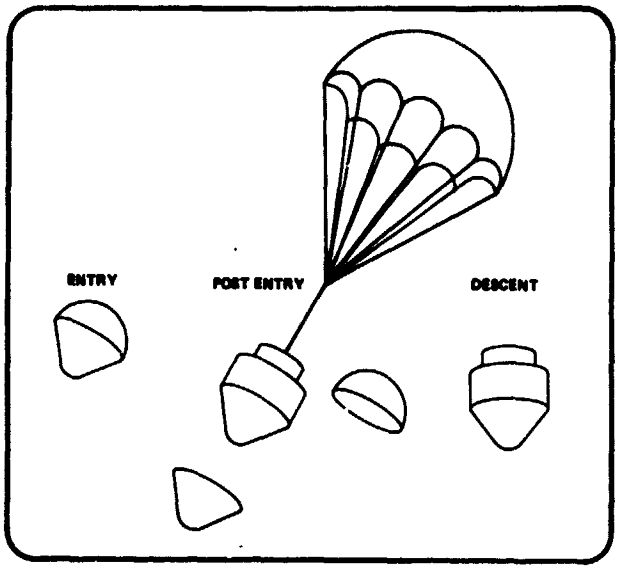


Figure 4-7. Stage Entry Probe

Table 4-5 Design Conditions for Exit from Entry

	Tropopause Cool dense atmos.		100 mbar Cool dense atmos.
	Alt (km)	Pressure (mb)	Alt (km)
Jupiter	19.4	259	31.1
Saturn	35.3	204	48.2
Uranus	35.8	330	50.4
Neptune	15.2	660	30.6

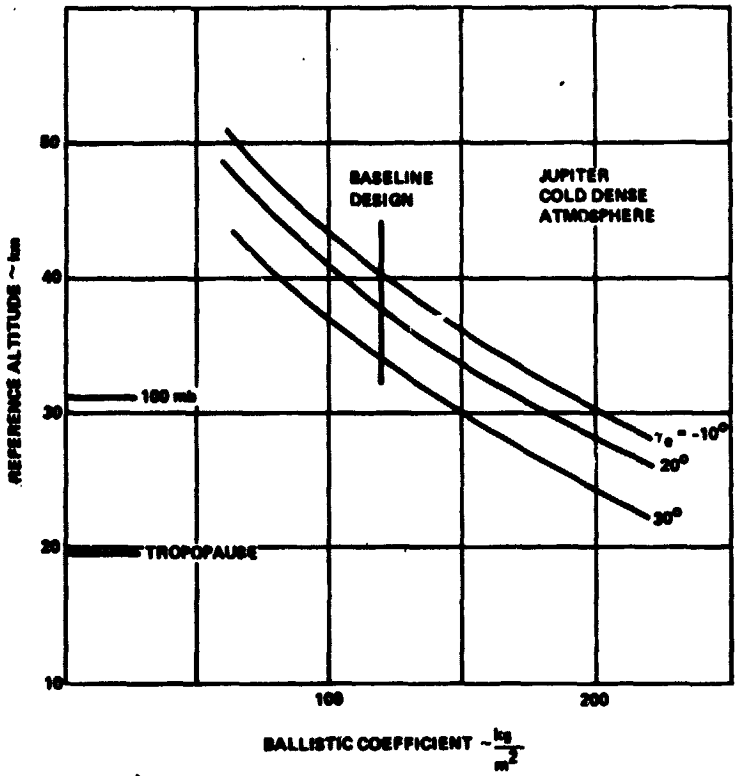


Figure 4-8. Entry into the Jovian Cool Dense Atmosphere

The criterion for ballistic coefficient during the atmospheric descent phase is established by the communications geometry for relaying data from the probe to Earth via the flyby bus. The communications geometry requires the flyby bus to be directly over the entry probe at the end of the probe mission (transmission of data obtained down to at least the 10 bar pressure altitude). For unstaged probes, this descent time is determined by the configuration which more nearly satisfies both the entry and descent criteria for data gathering. Staged entry probes on the other hand accept the different ballistic coefficient criteria at the expense of weight and complexity.

Another major factor influencing the selection of ballistic coefficient results from packaging constraints. The 60° half angle blunt cone aeroshell shape is limited to an 89 cm (35 in) diameter by the Pioneer bus and within this diameter, the probe equipment must be packaged to produce a resultant probe center of gravity as far forward as possible for good aerodynamic stability during entry and descent. In addition, the inertial moments ratio $I_{roll}/I_P I_Y$ must be ≥ 1.2 for good dynamic stability during the long autonomous probe coast from separation to entry. These packaging constraints tend to limit the ballistic coefficient to less than 150 kg/m^2 (0.96 $slug/ft^2$).

The trade between staged and non-staged probes is summarized in Table 4-6. The staged probe provides two, or in some cases three, different values of the ballistic coefficient during entry and descent to better tailor the time-altitude profile.

Table 4-6 Major Staging Trades

UNSTAGED ENTRY	STAGED ENTRY
• Staging complicates design parachute deployment & heat shield jettison ~ questionable reliability • Lighter weight ~16 kg • Aeroshell protects equipment during descent	• Better accommodates conflicting ballistic coefficient requirements • Exposes sampling inlets after entry • Uncovers communications antenna • Slower descent rate for more science data

The lighter weight and significantly improved reliability of the non-staged entry probe are sufficiently attractive to prompt design efforts to overcome the mitigating factors. The descent time for non-staged entry into each of the planets is adequate for science data collection (with the possible exception of the Jupiter cool dense model atmosphere).

Figure 4-9 presents unstaged entry and descent profiles into Saturn and Uranus model atmospheres. For comparison purposes, typical three-stage Jovian entry and descent profiles are shown in Figure 4-10 and also on Figure 4-10 is the profile for the baseline un-staged probe (β= 122 kg/m^2). This curve indicates the probe reaches the 10-bar pressure altitude approximately eleven minutes after Jupiter entry.

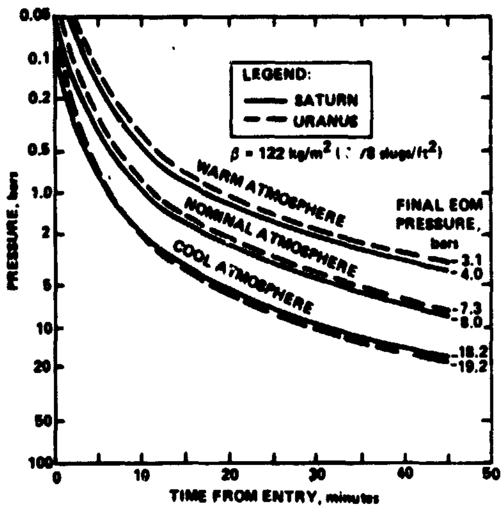


Figure 4-9. Pressure Descent Profiles

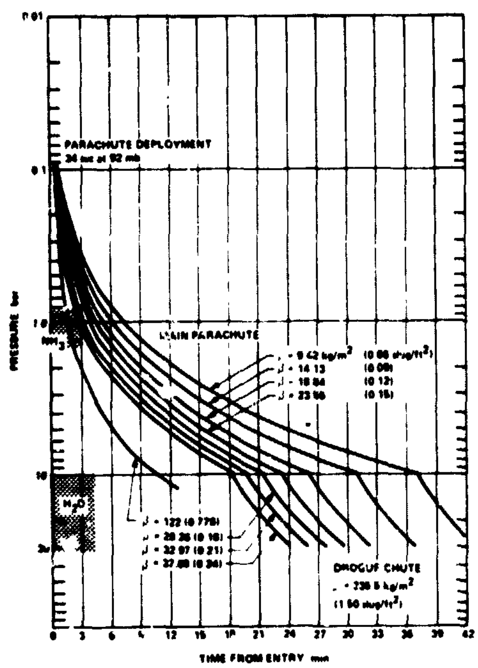


Figure 4-10. Pressure Descent Profile Parametrics -- Jupiter Cool/Dense Atmosphere

Subsystem design considerations in the un-staged configuration must also be concerned with the atmospheric sampling and data return. Designs have been proposed for the science sampling inlets and more work is recommended as discussed in Section 7 of this report. The communications system degradation which results from the use of a radome over the antenna is slight at 400 MHz. Detail design efforts should be continued with the objective of the use of the non-staged entry probe as represented by the baseline design.

The study reported in Reference 12 resulted in a probe configuration which may be a reasonable compromise of weight/reliability/instrument design factors; namely a deployable probe nose cap which exposes the necessary atmospheric instrument inlets after the entry phase is completed.

In summary, the baseline probe design has been selected for its relative simplicity, commonality of subsystems, and nearer term availability for missions to the majority of the outer planets of interest. Inclusion of the constraining Jupiter entry/descent requirements causes such significant design, development and spacecraft bus integration complexities that a "common" probe becomes a Jovian probe to be used for other planetary missions. On the other hand, with the selected baseline probe design, first order science investigations can be accomplished at most of the outer planets -- Saturn (and Titan), Uranus, and Neptune. Second order investigations of these planets would dictate the next level of probe evolution (size, payload, bus, etc.). At that juncture, in both complexity and development status, the "common" probe design could conceivably include the Jovian requirements.

4.3 Experiment Selection and Integration

As stated in the Introduction to this report, the selection of the science experiments to be carried on the first generation outer planet atmospheric probe must consider:

- o scientific return from the measurement
- o state of development of the instrument
- o severity of the entry environment
- o long-term deep space transit
- o limited communication capability from the outer planets to Earth

Many candidate instruments (Table 4-7) were examined in selecting the payload for the probe. The selected instruments are

Table 4-7 Candidate Science Instruments

PRE-ENTRY	ENTRY	DESCENT
Langmuir Probe Ion Retarding Potential Analyzer Neutral Particle Retarding Potential Analyzer Ion Mass Spectrometer	* Accelerometer Triad	* Accelerometer Triad * Temperature Gauge * Pressure Gauge * Neutral Mass Spectrometer * Nephelometer Solar Radiometer (UV) Cloud Particle Size Analyzer Beta Scatter UV Photometer Thermal Radiometer (R) Gas Chromatograph

* Selected Instruments

similar or identical to those being developed for the Pioneer Venus probe mission, they satisfy the instrument selection criteria stated above, and provide answers to the first order questions about the structure and composition of the atmospheres of the outer planets. Although desirable, no pre-entry instruments were included in the selected payload since the integration and operational difficulties presented by pre-entry measurements exceeds their relative scientific value on small entry probes. The extension of probe science capability to include radiometric measurements (through the use of the solar radiometer) has been examined in some detail. This instrument was not selected for the probe science payload largely because it implies a mission requirement for dayside entry, a difficult requirement to meet for most of the missions under consideration.

Special integration requirements for the selected payload are relatively minimal. One of the instruments, the temperature gage, must be deployed into the flow stream after entry and three of the instruments require access through the probe aeroshell after entry:

the nephelometer must have access through a port or window to monitor atmospheric aerosols; and the total pressure sensor and the neutral mass spectrometer must be ported to the ambient atmosphere. The accelerometer triad must be mounted at the probe center of gravity and accurately aligned with the probe axes.

4.4 Communications Subsystem Trades

The design of the communications subsystem is dominated by the communications link geometry, and the choice of bus (i.e. 3-axis stabilized or spin-stabilized). The communications link geometry can be modified by mission design to fall within the requirements of Table 4-8, for each of the missions contemplated.

Table 4-8 Communications Geometry

	PIONEER	MARINER
Max. Communications Range at Entry	120,000 km	125,000 km
Max. Communications Range at End of Mission	105,000 km	80,000 km
Max. Probe Look Angle	60 deg	90 deg
Max. Bus Look Angle Excursion	45 deg	40 deg
Max/Min Range Rate	$25/-20$ km/sec	$25/-20$ km/sec
Max/Min Range Acceleration	$8/-1$ m/sec^2	$8/-1$ m/sec
Date Rate	44 bits/sec	44 bits/sec

Some mission designs in the reference literature exceed these requirements, however, each of these missions had some peculiar requirement dictating its selection.

The communications geometry is established by the series of probe and bus maneuvers which take place at the time of probe separation from the bus; and for most missions these are programmed to occur between 10 and 30 million kilometers before planetary encounter. The choice between probe maneuvers, bus maneuvers, or a combination of both is predicated upon planetary quarantine philosophy, dispersion sensitivity, ease of implementation, and operational constraints.

In the baseline mission, the spacecraft is targeted for the probe entry point; and after probe release, the bus is retargeted to fly by the planet at the desired periapsis altitude. In addition, the bus approach velocity is reduced slightly to place the bus directly over the probe at the end of the probe mission (nominally 10 bars atmospheric pressure).

Bus selection governs the type of relay link receiving antenna which can be used. The 3-axis stabilized bus could use a fixed antenna (which could be a dish at L-band) with a gain of 10-12dB, without requiring it to be moveable. Alternatives to this include a shaped beam, with the antenna still fixed, to take advantage of the narrow range of look angles to be covered and a somewhat higher gain antenna that could be programmed to follow the expected probe track. The spinning bus, however, must use either a despun antenna of similar gain and beamwidth or a fairly low gain antenna with an axisymmetric pattern. It is not feasible to integrate a despun antenna into the Pioneer class bus without significant size, weight and cost impact. An axisymmetric antenna with a peak gain of about 3.8 dB (50° half power beamwidth aimed at a look angle of 65° referenced to the forward roll axis) will accommodate the range of look angles anticipated for all the missions.

Regardless of the choice of receiving antenna, lowering the carrier frequency increases the received signal power if the antenna beamwidth is limited only by geometrical uncertainties and not by aperture size.

The three axis stabilized bus can use either UHF ($\sim$400MHz) or higher frequencies. However, there are performance advantages for higher frequencies, and a moderate gain antenna; the frequency is then limited to about L-band ($\sim$860 MHz) by antenna size limitations. For an axisymmetric antenna on the spin stabilized bus, a 400 MHz frequency was chosen because of the size limitation on the probe antenna. The other factors influencing the choice of relay link operating frequency such as atmospheric attenuation, ionospheric loss depending on peak electron density (e/cc), and power amplifier efficiency, result in a total effect of less than 1.5 dB over the range of frequencies 400 MHz to 1.0 GHz, as shown in Figures 4-11, 4-12, and 4-13. The use of a toroidal antenna also results in a lower effective antenna noise temperature. The planet noise, which is large at low frequencies, has little effect on the antenna noise temperature of a toroidal antenna; but it completely determines the antenna noise temperature of a narrow beam antenna directed at the planet. The effective antenna noise temperature at Saturn of each antenna type over the frequency range of interest is shown in Figure 4-14.

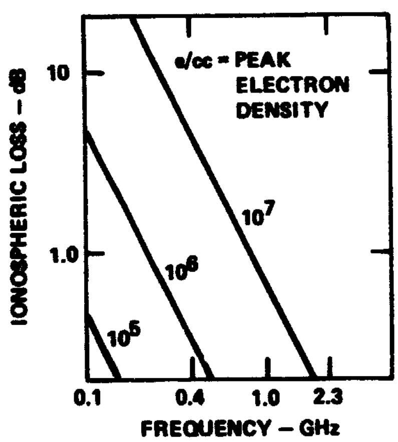


Figure 4-11. Ionospheric Loss on Saturn and Uranus

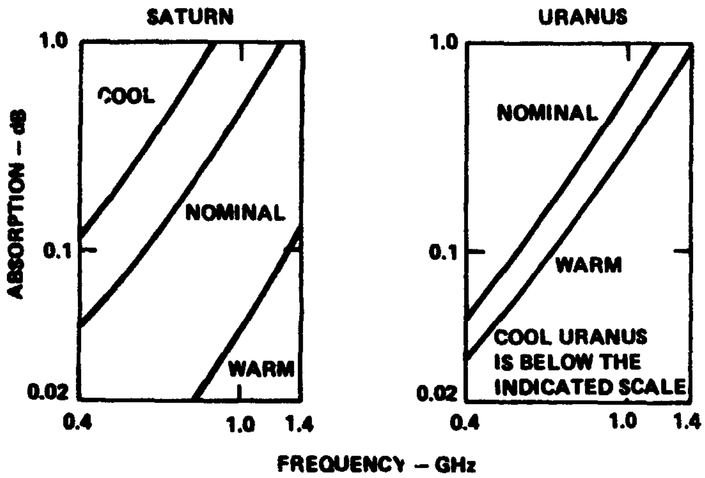


Figure 4-12. Atmospheric Absorption at Ten Bars Pressure Levels

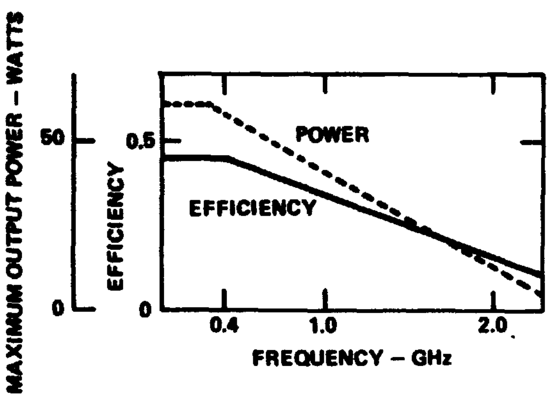


Figure 4-13. Generic Transmitter Characteristics

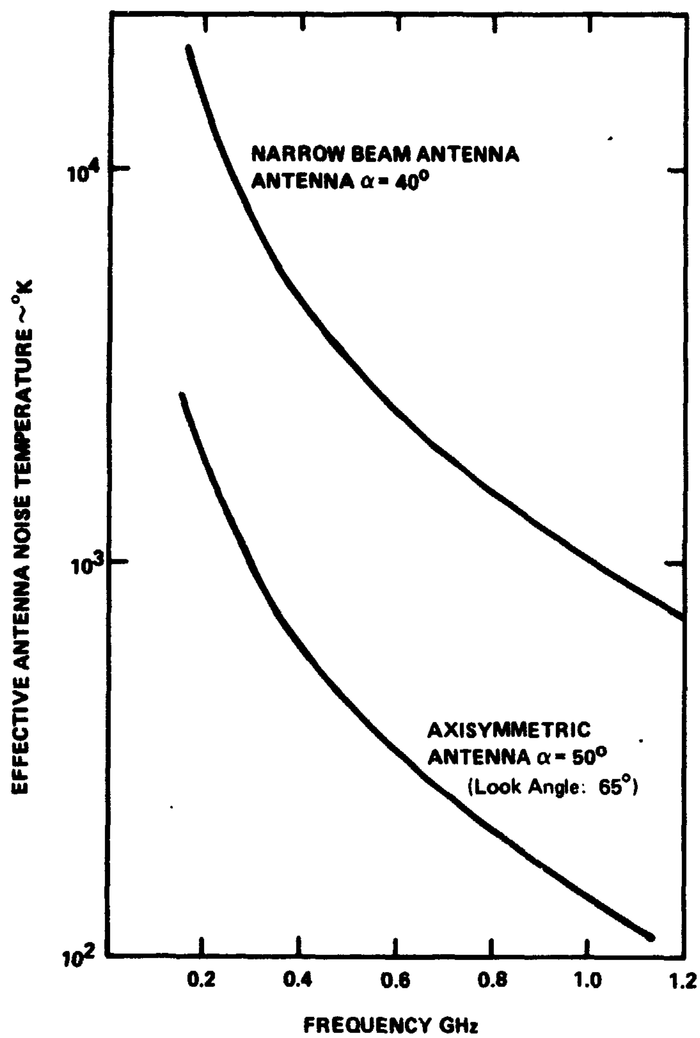


Figure 4-14. Effective Antenna Noise Temperature vs Frequency for Outer Planet Bus Antennas at Saturn

The two primary modulation techniques considered are PSK-PM and FSK. A comparison was made of the performance of a phase tracking receiver for PSK-PM modulation with equal carrier and data power against an AFC receiver with a BT = 2 tracking FSK modulation. The PSK-PM modulation was shown to have a 2.8 dB advantage under Gaussian noise channel conditions. However, an analytical comparison cannot be made under the assumption of a fading channel and, if an AFC receiver for FSK can be built with a BT = 1, the performance of FSK system under the idealized Gaussian noise conditions would match that for PSK-PM. The fading effects for the Saturn/Uranus atmospheric turbulence model are not expected to cause loss of lock in a PSK-PM receiver; and the fading effects should be even less on an AFC receiver for FSK. Since the turbulence model for Saturn/Uranus is of necessity very tenuous, the FSK system has been selected for the baseline mission. A detailed FSK receiver design was developed (Reference 4) and is currently being studied under fading channel conditions via computer simulation.

The choice of coding scheme has been narrowed to a choice between long- and short-constraint-length convolutional codes. Only convolutional codes have been considered since the code generation on the probe is the easiest to implement and theoretical and experimental data to date show them to yield the best performance. The decision between long- or short-constraint-length codes is not easily made. The long convolutional codes (25-40 bits) require sequential decoding, whereas the short convolutional codes (less than or equal to 10 bits) can be decoded via a maximum liklihood Viterbi decoder. Under Gaussian noise statistics, the sequential decoding yields higher performance than the Viterbi decoding; however, since it uses a much longer tree length, it has a larger data loss if a block of data has a burst of errors. Thus, it is postulated that Viterbi decoding is less sensitive to slow fades than sequential decoding. The baseline design uses hard decision Viterbi decoding, but the selection of code type should be made by comparing performance of each under simulated fading conditions.

Decoding performance can be enhanced by 1.2 to 1.4 dB if a soft (8 level) decision variable rather than a hard (2 level) decision variable is used. Using soft decisions, a larger number of encoded data symbols must be handled -- three times as many for 8-level soft decisions. Thus the data to be transmitted from the bus to the Earth must be tripled and if a real time relay of the data cannot be relied upon, the bus storage required would be tripled. The baseline mission of about 1-1/2 hours (shorter at Jupiter and Titan) at 44 bits required 475 kilobits of storage for a rate 1/2 hard decision decoder and 1.4 megabits of storage for a soft decision decoder. If the larger storage requirement can be accommodated, the performance advantage will improve the link margin.

Table 4-9 shows a communications link design for a spin-stabilized bus at 400 MHz and a 3-axis stabilized bus at 860 MHz, each using BT = 2 FSK modulation with a rate 1/2 hard decision (K = 10) Viterbi code.

Table 4-9 Communications Link

	PIONEER S/C 400 MHz		MARINER S/C 860 MHz	
	Nominal	Adverse Tolerance	Nominal	Adverse Tolerance
Transmitter Power (40W)	16.0 dBW	–	16.0 dBW	–
Transmit Line Loss	–0.5 dB	0.1 dB	–0.5 dB	0.1 dB
Transmit Antenna Gain	6.6 d3	0.6 dB	6.5 dB	1.8 dB
Free Space Loss (120 x 10^3 km)	–186.1 dB	0.7 dB	–192.7 dB	0.3 dB
Planetary Atmosphere Loss (10 bar)*	–0.4 dB	1.0 dB	–1.9 dB	1.0 dB
Polarization Loss	–0.2 dB	0.1 dB	–0.3 dB	0
Receiving Antenna Gain	1.5 dB	0.6 dB	12.0 dB	1.0 dB
Receiving Line Loss	–0.5 dB	0.1 dB	–0.5 dB	0.1 dB
Net Circuit Loss	–179.6 dB	3.2 dB	–177.4 dB	4.3 dB
Total Received Power	–163.6 dBW	3.7 dB	–161.4 dBW	4.3 dB
Effective System Noise Temperature	800°K		1650°K	
Receiver Noise Spectral Density	–199.6 dBW/Hz	0.2 dB	–196.4 dBW/Hz	0.4 dB
Processing Loss	–1.0 dB	0	–1.0 dB	0
Fading Loss	–1.0 dB	0	–1.0 dB	0
Received Data Power	–165.6 dBW	3.2 dB	–163.4 dBW	4.3 dB
Data Bit Rate (44 bps)	16.4 dB	0	16.4 dB	0
Threshold $^{Eb}/No$	10.6 dB	1.0 dB	10.6 dB	1.0 dB
Threshold Data Power	–172.6 dBW	0.2 dB	–169.4 dBW	0.4 dB
Perf Margin	7.0 dB	4.4 dB	6.0 dB	4.7 dB
Nominal less Adverse Tolerance	2.6 dB		0.3 dB	

*Includes ionospheric loss.

4.5 Probe Power Subsystem Trades

Several electrical power and energy sources have been extensively studied for use in probe missions to the outer planets. The principal requirements which the system must meet are:

- o up to 11-year deep space storage (Neptune mission)
- o launch and entry environment ($750G_E$)
- o unattended operation
- o up to 35-day operation at 280 microwatts
- o total energy - 239 watthours
- o peak power - 120 watts

Batteries remain the most attractive power source for outer planet missions compared to the cost and weight of Radio Isotope Thermoelectric Generators (RTG's), and the inadequate solar flux at the outer planets for consideration of solar arrays.

In the battery selection both nickel-cadmium and silver-zinc were evaluated; the former offering many charge-discharge cycles over a long life in exchange for weight penalties (22 watt hours per kg); and the latter providing savings in volume and weight (97 watt hours per kg) in exchange for a single discharge cycle on the probe mission. Because of the long cruise life requirement, the silver zinc batteries are not activated until forty minutes (± 5 minutes) prior to entry and this activation is accomplished by signal from the data handling subsystem's entry timer. The lifetime of current silver zinc battery cells is limited by the organic separator material. Although long-life cells using inorganic materials are under development, batteries using these cells will not have completed life tests at the time the probe design is committed for 1979 - 1980 launches. Auto-activated batteries are manufactured, assembled and flown in the dry charged state with the electrolyte stored in a reservoir separated from the cell by a frangible diaphragm. Prior to entry the battery is activated by forcing the electrolyte from the reservoir into the cells under pressure from a pyrotechnic gas generator. Battery lifetime of 5 to 10 hours can be realized before the battery self discharges along the electrolyte wetted surface of the fill manifold. The 5 to 10-hour life is more than adequate for high power operation during entry and descent.

After separation from the bus, the probe is in an autonomous cruise, low power phase for up to 35 days during which the entry timer (280 microwatts, 0.24 watthour total) is powered by the wet charge nickel cadmium battery. This secondary (bootstrap) battery is small in weight and volume and its size is determined by its requirement to also provide actuation current (5 amperes) to the main battery pyrotechnic auto-activation device described above.

4.6 Aeroshell Structure and Heat Shield Trades

This section briefly describes the structure and heat protection system considerations and trade-offs associated with outer planet entry probe missions. The discussion is divided into two parts. First, the considerations associated with a common Saturn/Uranus/Neptune aeroshell design are presented. Second, the changes to the aeroshell design required for Jupiter entry are discussed. It should be reiterated that the high entry velocities associated with outer planet probe missions result in large heating loads (especially for Jupiter entry), dynamic pressure loads and deceleration loads. Consequently, the probe aeroshell represents a substantial fraction of the total probe entry weight and care must be taken to optimize its design. This is a particularly difficult task for the heat shield design, since relatively little is known about the behavior of materials subjected to the combined effects of high convective and radiative heating and shear forces. A strong need exists for the development of a test facility to adequately simulate the heating environment to be experienced during outer planet entry.

For comparative purposes, Table 4-10 presents several important entry conditions associated with entry into Jupiter, Saturn, (and Titan), Uranus and Neptune. The results of Table 4-10 apply to a 60° half angle blunted cone configuration with a ballistic coefficient of about 125 kg/m^2. Maximum dynamic pressures, G-loadings and heat transfer rates are encountered during steep entry into the cold Uranus model atmosphere; whereas maximum integrated heating is encountered during shallow entry into the nominal Jupiter model atmosphere, (the Jupiter cold model atmosphere appears very improbable after the Pioneer 10 data has been evaluated). These values are very sensitive to the atmospheric models used which should be further refined as outer planet exploration progresses.

Table 4-10 Outer Planet Entry Conditions

	Jupiter*	Saturn	Titan	Uranus	Neptune
Entry Velocity (km/sec)	59 to 61	36 to 38	5 to 12	22 to 25	25 to 28
Entry Angle (deg)	−5 to −10	−15 to −40	−60	−30 to −50	−20 to −30
3σ Entry Angle Dispersion (deg)**	0.5	9 to 1	15	15 to 7	–
Max. Entry Inertial Loads (G_E)	400	750	36	750	300
Max. Peak Dynamic Pressure (MN/m^2)	0.50	0.91	0.17	1.0	0.5
Max. Peak Heating Rate (MW/m^2)	193	120	11	295	68 est.
Max. Integrated Heating ($MW\text{-}sec/m^2$)	1250	613	216	390	375

*Post Pioneer 10 Mission
**Upper bounds – could be improved by use of optical navigation (e.g., Uranus - ½°)

4.6.1 Saturn/Uranus/Neptune Entry

For this set of entry missions the aeroshell structure is designed to accommodate Uranus entry which results in the most severe dynamic pressure and deceleration loading environment. The heat shield design is governed by the warm Saturn and cool Uranus entry environments which result in the most severe heating conditions.

<u>Aeroshell</u> <u>Structure</u> - Maximum dynamic pressure and deceleration loads occur for the steep entries into the cool Uranus atmospheric model (see Table 4-10). Peak dynamic pressure is about 1,000,000 N/m^2 (125 psi) and maximum deceleration loads are about

750 G_E. The aeroshell is designed to resist the external aerodynamic pressure against buckling. Structural design concepts which have been considered for the aeroshell include semi-monocoque and ring stiffened honeycomb construction. Although it is possible to consider advanced materials (e.g. boron or graphite reinforced composites) the studies have mainly focussed on conventional materials such as aluminum, titanium and fiberglass core honeycomb with fiberglass and aluminum face sheets. Figure 4-15 shows parametrically, aluminum and titanium aeroshell structure weights as a function of design pressure for different aeroshell diameters. The information is presented for semi-monocoque construction, but comparable weights can be achieved with honeycomb construction using fiberglass core and fiberglass and aluminum face sheets. Maximum design pressures for Saturn/Uranus entries are of the order of 1.4 MN/m^2 (200 psi) including load safety factors resulting in aeroshell structural weights of about 8.5 to 11.5 kg (20-25 lbs) for reasonable aeroshell diameters.

As can be seen from Figure 4-15, for moderate design pressures and aeroshell diameters, aluminum has a small weight advantage over titanium. However, titanium aeroshell structure weights are less sensitive to increased design pressures (particularly for the larger aeroshell diameters). Consequently, the use of titanium eases the problems associated with designing a common aeroshell structure for both Jupiter entry and Saturn/Uranus/Neptune entry. In addition, titanium has better structural properties at elevated temperatures and its use frequently results in lower total heat shield/structure weight since the aeroshell can sustain higher heat shield back face temperatures. In those cases where detailed temperature-time histories indicate that an aluminum aeroshell structure is feasible, weight savings of around a kilogram can be achieved compared to titanium.

Ring stiffened honeycomb-type construction is an attractive alternative to semi-monocoque construction. The use of a fiberglass

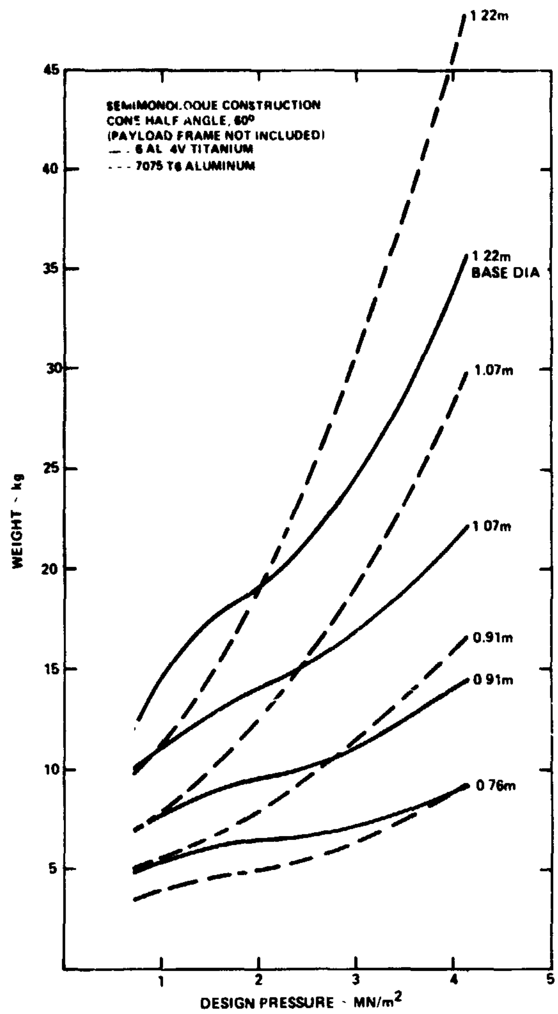


Figure 4-15. Aeroshell Structure Weights

core inhibits heat flow into the interior of the probe. The outboard face sheet can also be fiberglass to allow operation at elevated heat shield back face temperatures while the inboard face sheet can be aluminum. This also allows the inboard face sheet to be machined integrally with the stiffener rings.

Aeroshell Heat Shield - The bulk of the heat shield analyses performed in the various studies are derived from the pioneering work of M.E. Tauber at NASA/ARC (refs 17 and 18). For Saturn/Uranus/Neptune entry this work has focussed on deriving heat shield mass fractions for probes in the weight class of 100-250 kg with both blunt body ($R_N/R_B = 2$) and conical configurations (60° half angle).

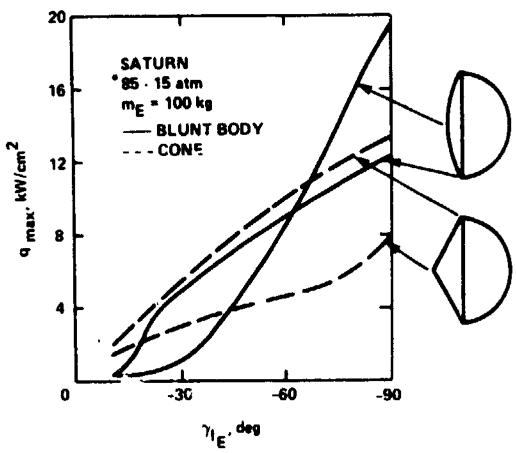


Figure 4-16. Peak Heating Rates During Saturn Entry

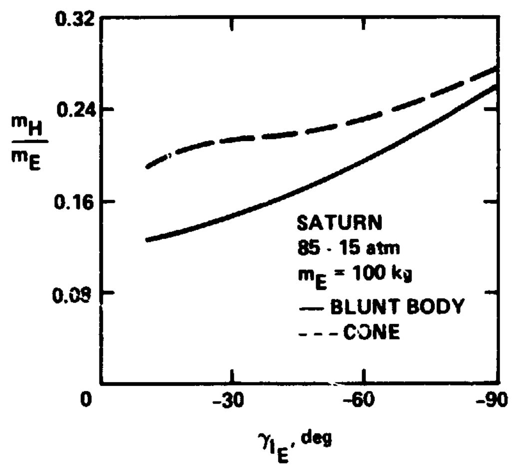


Figure 4-17. Saturn Entry Heat Shield Mass Fractions

Figure 4-16 shows the peak heating rates for both configurations for Saturn entry, as a function of entry angle, at the stagnation point as well as the shoulder of the aeroshell. The thin shock layer associated with the conical shape results in much lower radiative heating (and hence lower total heating) at steep entry angles. The blunt body experiences significantly lower heating only for shallow entry angles. A comparison of graphitic heat shield mass fractions for Saturn entry is shown in Figure 4-17. Although somewhat lower mass fractions are obtained for the blunt body, the difference at steep entry angles is small. Furthermore, as discussed below the conical configuration is much more compatible with Jupiter entry conditions (where

* 85% H_2 - 15% He

theblunt bodies encounter enormous radiative heat loads). In addition, descent aerodynamic stability and packaging considerations favor blunted conical configurations, which, consequently, are generally selected for outer planet entry.

The heating environment associated with nominal atmosphere Neptune and Uranus entry is milder than for Saturn, resulting in lower heat shield mass fractions (See Figure 4-18). Consequently, the probe heat shield is usually designed by the Saturn entry environment.*

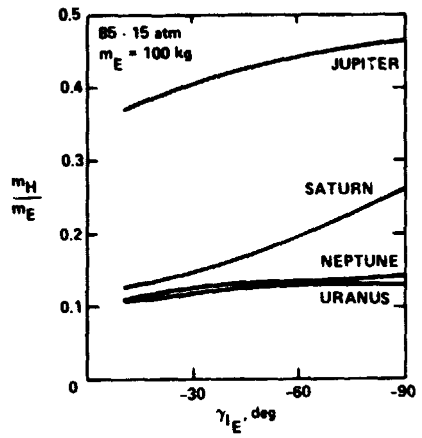
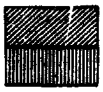
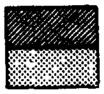
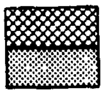
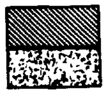


Figure 4-18. Comparison of Heat Shild Mass Fractions for Entry into the Outer Planets

Several different ablator/insulation heat shields have been considered in the studies, namely, ATJ Graphite/low density carbon felt insulator, carbon phenolic/hollowed carbon phenolic, quartz/hard compacted fiber insulation (HCF), carbon phenolic/HCF and Teflon/HCF. The quartz and Teflon approaches represent reflective heat shields designed to reflect the shock layer radiation and thus minimize heat input to the heat shield. Table 4-11 presents

* Maximum heating rates are encountered during entry into the cool Uranus atmosphere, but maximum integrated heating occurs during entry into the warm Saturn atmosphere.

Table 4-11 Heat Shield Candidates

ABLATOR/ REFLECTOR / INSULATION						
ABLATOR	CARBON PHENOLIC	CARBON PHENOLIC	CARBON PHENOLIC			ATJ GRAPHITE
REFLECTOR	–	–		TEFLON	QUARTZ	
INSULATION	HOLLOWED CARBON PHENOLIC	HONEYCOMB REINFORCED LOW CONDUCTIVITY ABLATOR	HARDENED COMPACTED FIBERS (HCF)	HCF	HCF	LOW DENSITY CARRON FELT
STRUCTURAL PERFORMANCE	ADEQUATE	ADEQUATE	ADEQUATE FOR ρ_{HCF} - 35 PCF	ADEQUATE ρ_{HCF} = 15 PCF	ADEQUATE ρ_{HCF} 15 PCF	ADEQUATE
AVAILABLE MATERIAL PROPERTIES	YES	YES	EXTRAPOLATED HCF	EXTRAPOLATED TEFLON	EXTRAPOLATED QUARTZ	EXTRAPOLATED GRAPHITE

the various heat shield/insulation candidates. All heat shields appear to be feasible concepts but the carbon and graphite based ablator designs have, at present, less design uncertainties associated with them than the reflective heat shield designs. In addition, for the non-Jupiter entries, the radiative heating component is less of a problem.

Fabrication confidence favors the use of the first two heat shield candidates shown in Table 4-11. The first candidate consists of carbon phenolic where ablation performance is provided by the dense, homogenous carbon phenolic (outer portion of the heat shield) while insulation performance is provided by mechanically hollowing out the inner portion of the carbon phenolic and filling it with carbon fiber insulation. The amount of hollowing out is determined by the heat shield strength required to withstand the high entry pressure and G-loading. Effective hollowed out densities of about 577 kg/m^3 (36 lb/ft^3) can be used (compared to a density of 1442 kg/m^3 (90 lb/ft^3) for the solid carbon phenolic). Shallow entry into the warm Saturn atmosphere represents the worst case heat shield condition and results in a heat shield where the dense carbon phenolic is about 1.27 cm (0.5 inches) thick and the hollowed-out carbon phenolic thickness is about 1.65 cm(0.65 inch). This limits the bond line temperature to 427°C (800°F) for a non-jettisoned heat shield concept. The second candidate consists of solid carbon phenolic backed up by a honeycomb reinforced low conductivity ablative material to provide insulation performance. For this heat shield concept, a carbon phenolic thickness of about 1.27 cm (0.5 inches) is required together with an insulation thickness of about 0.8 cm (0.33 inches).

Table 4-12 presents results of heat shield sizing calculations for variations in probe shape, atmospheric model, entry angle, ballistic coefficient, and planet for candidate heat shield concepts. The probe shapes are shown in Figure 4-3.

4.6.2 Jupiter Entry

The Pioneer 10 flyby at Jupiter has provided an improved estimate of the ephemeris and the atmosphere of Jupiter. Both of these factors have contributed to a reduction in the entry environment for the outer planet probe.

Prior to the Pioneer 10 flyby the uncertainty in the position of Jupiter resulted in an entry angle that ranged from -15 to -30 deg. With the improvement in the ephemeris, it will now be possible to target the probe near the skip-out boundary, about -5 to -7 deg with a 3σ dispersion of 0.5 deg. The shallow entry angle reduces both the loads and heating rate. A reduction in loads results in a smaller weight fraction for the aeroshell and internal support structure and also eases the qualification and acceptance testing of the probe and its equipment. Furthermore, a shallower entry angle reduces the heating rates and narrows the gap between the actual environment and the environment that can be simulated in a ground test facility. Probe surface heating rates are now comparable to Saturn/Uranus environments and bounded by Earth entry R/V experience. The penalty for shallow angle entry is the longer heat pulse and resultant thicker heat shield requirements. This is a small price considering that now a state-of-the-art heat shield material like carbon-phenolic can be used with an adequate safety margin.

Table 4-12 Summary of Sizing Predictions

Case	Planet	Atmos.	γ (Deg)	Shape	Ballistic Coef. (g/cm^2)	Entry Velocity V_E (km/cm)	Heat Shield Concept	Ablative Material	Bondline Temp. (°R)	Ablator Thickness (Inches)
					Stagnation Point					
1	Saturn	Cold	−50.6	C1	14.9	30.4	Sandwich	C-P	4746	0.30
									1880.	0.40
2	Saturn	Cold	−19.6	C1	14.9	28.2	Sandwich	C-P	3811.	0.30
									1603.	0.50
3	Saturn	Warm	−19.6	C1	14.9	28.2	Sandwich	C-P	2843.	0.45
									2308.	0.50
4	Saturn	Warm	−50.6	C1	14.9	30.4	Sandwich	C-P	3464.	0.30
									1997.	0.40
5	Saturn	Nom.	−50.6	C1*	12.1	32.0	Sandwich	C-P	2670.	0.30
									1711.	0.40
6	Saturn	Nom.	−19.6	C1*	12.1	32.0	Sandwich	C-P	3161.	0.40
									2050.	0.50
11	Saturn	Cold		C1	14.9	30.4	Sandwich	Carbon	2644.	0.70
14	Saturn	Cold	−50.6	C1*	12.1	28.2	Sandwich	C-P	1600.	0.50
18	Saturn	Cold	−19.6	C1*	12.1	32.0	Sandwich	C-P	3521.	0.40
19	Saturn	Nom.	−15.0	C1	14.0	32.0	Semi-Inf.	C-P	2460.	0.374
21	Saturn	Nom.	−40.0	C1	14.0	32.0	Semi-Inf.	C-P	2460.	0.273
24	Saturn	Cold	−40.0	C1	14.0	32.0	Semi-Inf.	C-P	2460.	0.453
25	Saturn	Cold	−15.0	C2	14.0	32.0	Semi-Inf.	C-P	2460.	0.354
31	Saturn	Cold	−40.0	B	15.8	32.0	Semi-Inf.	C-P	2460.	0.724
33	Saturn	Nom.	−15.0	B	15.8	32.0	Semi-Inf.	Silica	2460.	0.453
35	Saturn	Nom.	−40.0	B	15.8	32.0	Semi-Inf.	Silica	2460	0.308
37	Saturn	Nom.	−15.0	C1	14.0	32.0	Semi-Inf.	Silica	2460.	0.447
41	Saturn	Nom.	−40.0	C1	14.0	32.0	Semi-Inf.	Silica	2460.	0.321
43	Saturn	Nom.	−15.0	C1	14.0	32.0	Semi-Inf.	Silica	2460.	0.473
45	Saturn	Nom.	−40.0	C1	14.0	32.0	Semi-Inf.	Silica	2460.	0.358
46	Uranus	Cold	−49.4	C1	14.9	26.3	Sandwich	C-P	3737.	0.60
									1990.	0.65
47	Uranus	Warm	−39.4	C1	14.9	26 4	Sandwich	C-P	2541.	0.40
									1666.	0.50
49	Uranus	Cold	−24.4	C1	14.9	2[illegible].4	Sandwich	C-P	1933.	0.70
51	Uranus	Nom.	−25.0	C1	14.0	32.0	Semi-Inf.	C-P	2460.	0.272
53	Uranus	Nom.	−50.0	C1	14.0	32.0	Semi-Inf.	C-P	2460.	0.207
54	Uranus	Cold	−25.0	C1	14.0	32.0	Semi-Inf.	C-P	2460.	0.430
56	Uranus	Cold	−50.0	C1	14.0	32.0	Semi-Inf.	C-P	2460.	0.438
63	Uranus	Nom.	−25.0	B	15.8	32.0	Semi-Inf.	Silica	2460.	0.338
65	Uranus	Nom.	−50.0	B	15.8	32.0	Semi-Inf.	Silica	2460.	0.257
66	Uranus	Cold	−25.0	B	15 8	32.0	Semi-Inf.	Silica	2460.	0.365
68	Uranus	Cold	−50.0	B	15.8	32.0	Semi-Inf.	Silica	2460.	0.333
73	Uranus	Nom.	−25.0	C2	14.0	32.0	Semi-Inf.	Silica	24[illegible]	0.346
74	Uranus	Nom.	−50.0	C2	14.0	32.0	Semi-Inf.	Silica	24[illegible]	0.265
					Body Position $R = 0.634\ R_B$					
32	Saturn	Nom.	−15.0	C2	14.0	32.0	Semi-Inf.	Teflon	–	0.582
42	Saturn	Cold	−15.0	B	15.8	32.0	Semi-inf.	Teflon	–	0.926

*Probe radius is 17.5 inches

Data reduction following the Pioneer 10 flyby has permitted a narrowing in the range of model atmospheres for probe design. The pre-Pioneer 10 models were based on a nominal model atmosphere with a hydrogen to helium ratio of 6.5, a cold-dense model which yielded the peak load and peak heating rates had a ratio of 2.2, and a warm-expanded model atmosphere which resulted in the greatest integrated heating had a ratio of 15.2. Based on the data returned, the expected Jupiter atmosphere ranges from the nominal model to the warm-expanded model.

Both the improved ephemeris which permits a shallower entry angle and the elimination of the cool-dense model atmosphere contribute to the reduction in loads and heating rate. Thus, the severe pre-Pioneer 10 loads and heating rates that characterize Jupiter entry and provide the criteria for the design of a "common" outer planet probe were eliminated. It is shown in Table 4-10 that Saturn and Uranus entry now yield the maximum entry loads and that Uranus entry results in the maximum heating rates. It is the warm-expanded model atmosphere that yield the greatest integrated heating, and it can also be seen in Table 4-10 that Jupiter entry sets this design requirement.

Therefore, it is possible to design a common Jupiter/Saturn/Uranus/Neptune from the viewpoint of loads without incurring significant penalty. For a 113 kg class probe, the heat shield penalty for maintaining a common outer planet probe design is about 45 kg for a Jupiter probe. If, however, it is desirable to increase the scientific payload, or extra launch weight is not available, then it may be more efficient to have a uniquely designed probe just for Jupiter.

Aeroshell Structure

The parametric information shown in Figure 4-15 is also applicable to Jupiter entry. It can be seen that the aeroshell weight necessary to satisfy entry into Jupiter/Saturn/Uranus/Neptune mission does not exceed about 7 kg.

Aeroshell Heat Shield

Tauber, et al (Reference 17) have examined the entry environment and heat shield requirements for Jupiter entry over a range of atmospheres and probe sizes and weights (100, 250 and 500 kg). The low end of the weight scale essentially represents the smallest feasible instrumented vehicle, the mid-range represents the entry vehicle payload limit for the Pioneer-type spacecraft and the high weight was the maximum entry vehicle payload capacity projected for the JPL thermoelectric outer planetary spacecraft (TOPS).

Heating rates experienced during entry into Jupiter exceed present day experience by more than an order of magnitude and potential heat shield problems such as particulate removal, spalling, thermal stress failure are very difficult to assess. Such effects are neglected in the analyses which generally assume orderly thermochemical ablation characteristics.

Typical peak-heating rates for two locations on a 60° half-angle cone are shown in Figure 4-19 as a function of entry angle. The maximum rates are entirely due to radiation and are lower at the stagnation point because the shock layer is thinner (small initial nose blunting is assumed). As expected, the heating rates increase rapidly with steepening entry angle leading to a desire to limit entry angles to values below -30°.

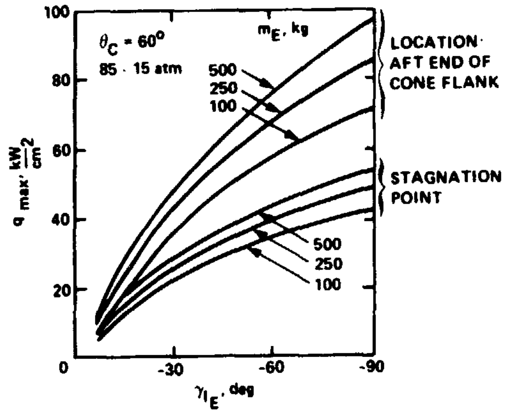


Figure 4-19. Maximum Heating Rates – Jupiter Entry

The degree to which existing heating simulation facilities fail to meet the required conditions for Jupiter entry is illustrated in Figure 4-20. The typical entry trajectories shown result in combined radiative and convective rates which are frequently an order of magnitude higher than can be presently simulated. To proceed without such tests would require acceptance of higher mission risk.

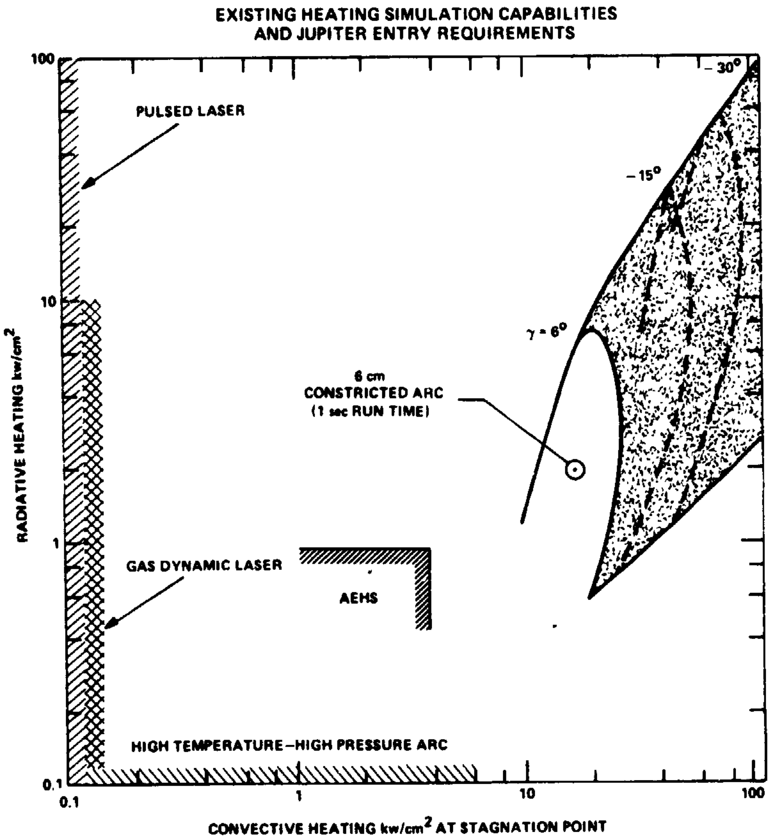


Figure 4-20. Existing Heating Simulation Capabilities and Jupiter Entry Requirements

The choice of 60° for the cone half-angle minimizes the required heat shield mass fraction. This can be seen from Figure 4-21 where mass fractions are presented as a function of cone half-angle for different entry angles. A significant increase in cone half-angle beyond 60° can increase the shock layer thickness resulting in increased radiative heating and increased heat shield weight requirements.

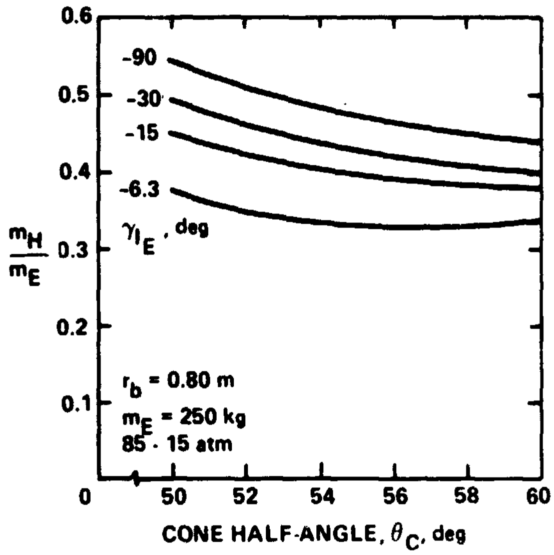


Figure 4-21. Variation of Heat Shield Mass Fraction with Cone Half-Angle (Jupiter Entry)

The probe ballistic coefficient (m_E/C_DA) affects the heat shield mass fraction as well. This is illustrated in Figure 4-22 which shows that the heat shield mass fractions increase significantly as the ballistic coefficient drops below 100 kg/m^2.

A comparison between the heat shield mass fractions required for Jupiter entry and those for the other outer planets is shown in Figure 4-18. At low entry angles, Jupiter mass fractions are about three times higher while at the steeper entry angles the Jupiter mass fractions are about double those for Saturn. This implies that the heat shield weight required for the Jupiter entry probe is about three times that required for the baseline Saturn/Uranus/Neptune probe design. Consequently, the heat

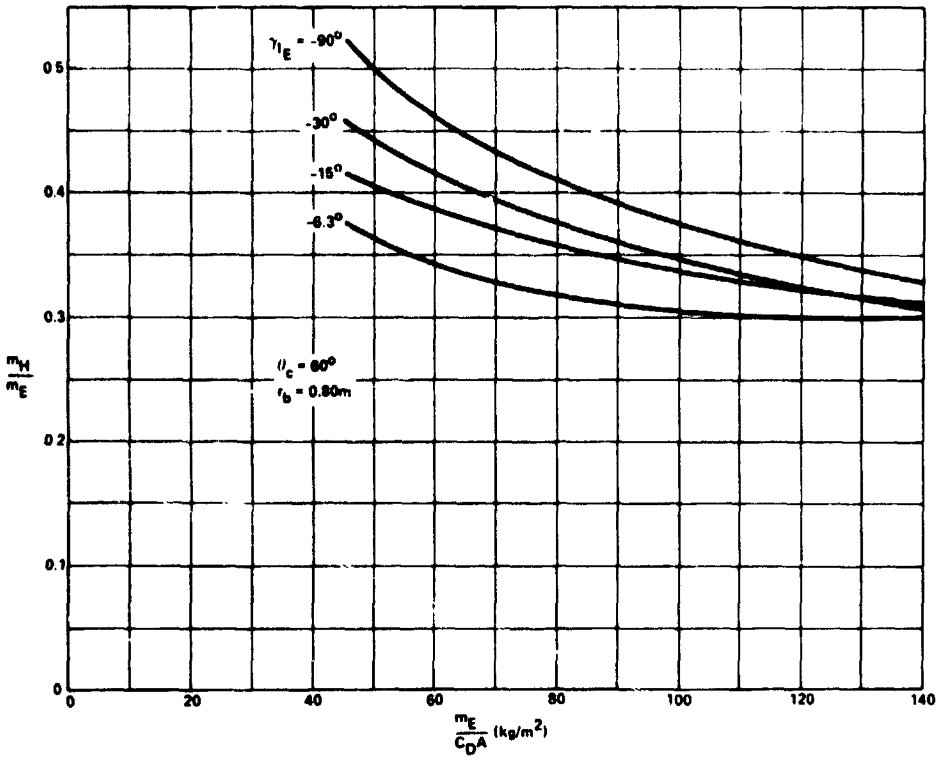


Figure 4-22. Variation of Heat Shield Mass Fraction with Ballistic Coefficient (Jupiter Entry)

shield weight penalty for Jupiter is approximately 45 kg (65 lbs) using a graphitic ablator design concept.

Reflective heat shields offer the possibility of reducing the required heat shield mass fraction below that associated with graphitic ablators. Materials such as Teflon, Boron Nitride, Quartz and others are under active consideration. However, reflective heat shields represent an advance in the present state-of-the-art for heat protection systems and will require substantially more development and testing before their effectiveness can be fully assessed. The gains possible for Saturn, Uranus and Neptune are modest; for Jupiter, a major improvement would be of great value.

4.7 Thermal Control Trades

The design of the thermal control system over the entire mission requires a distinct approach to each separate mission phase. During the pre-launch phase, conventional shroud cooling using circulated N_2 gas will be employed. Near Earth, during ascent and parking orbit, the entire spacecraft is protected from the brief ascent heating pulse by the shroud; and the remainder of the ascent and Earth orbit phase are of very short duration and are similar to the interplanetary phase.

The interplanetary transit is characterized by ever decreasing solar flux as the spacecraft recedes from the sun as shown in Figure 4-23. The probe is mounted in the shadow of the large Earth communications dish, where it receives no direct thermal input from the sun, and thermal control is achieved by covering the probe with a high performance multilayer insulation blanket to reduce heat losses and by making up these small losses by internal heating. Both electrical and radioisotope heaters have been considered for this function. Probe thermal control can also be augmented by providing electrical heaters on the probe attachment points powered from the spacecraft RTG and by providing high thermal conductivity paths into the probe at these points.

During the 20-35 day autonomous probe flight phase, the probe must provide its own source of thermal energy to make up heat lost through the multilayer insulation blanket. The solar flux input, now able to reach the probe, is very low at any of the target planets (See Figure 4-23) and is largely rejected by the multilayer insulation blanket. The size and weight of a battery required to support electrical heaters during this phase (20-35 days) is prohibitive but radioisotope heater units (RHU's) can provide the required heat at a reasonable weight.

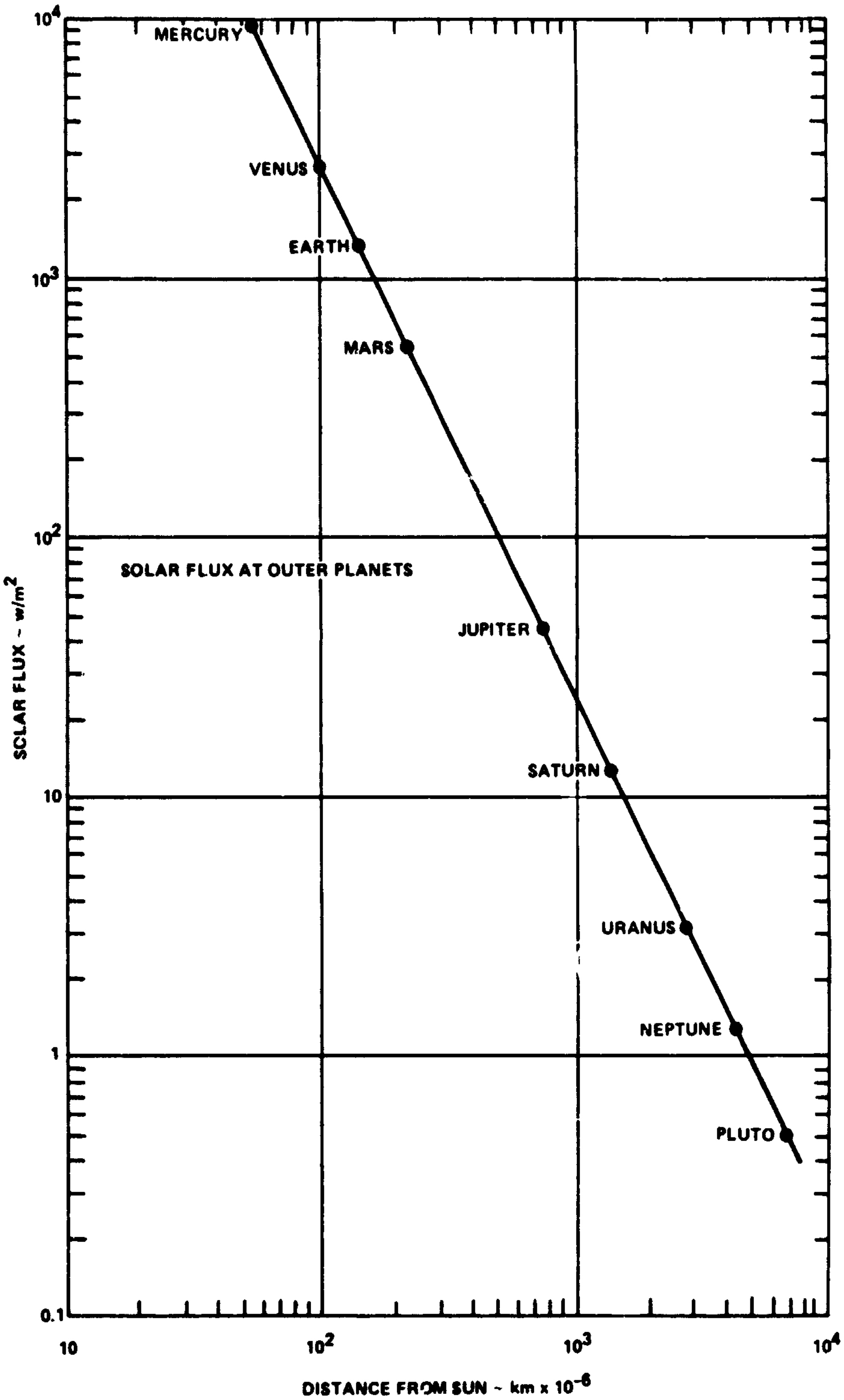


Figure 4-23. Solar Flux at Outer Planets

As the thermal performance of the multi-layer insulation blanket is improved, the RHU heat input requirements are reduced. The thermal performance of the blanket is a function of its design, material selected, thickness and fabrication technique. The information utilized in conducting the probe thermal control trades is shown in Figure 4-24. As indicated on this figure,

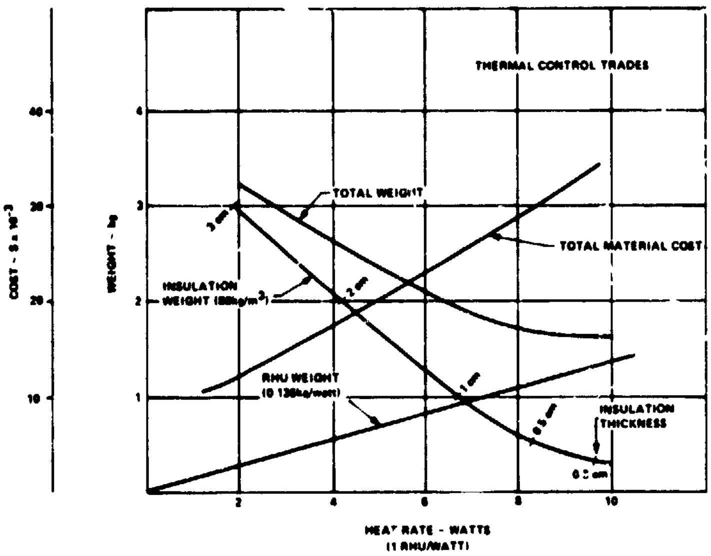


Figure 4-24. Thermal Control Trades

the minimum weight combination would require 9 RHU's and a 0.4 cm thick insulation blanket; however, the cost trade suggests a smaller number of RHU's. The baseline design was selected as 4 RHU's and a 2 cm thick insulation blanket.

About 40 minutes before entry, the 30-watt main battery heater is activated for 30 minutes to raise the battery and, by conduction, other internal equipment to acceptable operating temperature levels. Pyrotechnic devices for accomplishing this heating were not evaluated in the reference studies, but may present an attractive alternative from standpoints of weight and required power.

During the atmospheric entry phase, the multi-layer insulation blanket quickly burns off and the probe heat shield provides protection against the severe entry heating pulse. Communication testing of this is required. Thermal energy stored in the aeroshell structure and heat shield during the entry phase tends to offset the cooling effect of the planetary atmosphere during the 30-minute to 1 hour descent and the heat shield also provides insulation against the cool planetary atmosphere. Internal thermal dissipation of the electronic payload, primarily from the power amplifier, yields about 80 watts during the descent phase.

As a result, from entry thermal energy, component generated heat, the cooling by the atmosphere, and the insulating properties of the heat shield, very little temperature change occurs within the probe during its data gathering descent through the atmosphere.

5.0 BUS SYSTEM AND SUBSYSTEM DESIGN TRADES

Many of the probe design features and trades (Section 4) have an impact upon the choice of bus and upon the bus design modifications which must be made to accommodate the probe. There are also significant differences in probe design which result from the selection of the supporting bus. The probe mission can be supported by either a spinning bus (Pioneer class) or a 3-axis stabilized bus (Mariner class). Each of these has significant advantages and disadvantages. The major trades are shown in Table 5-1.

Table 5-1 Major Bus/Probe Trades

	PIONEER CLASS	MARINER CLASS
Launch Vehicle	TITAN IIIE/CENTAUR/ TE364-4	TITAN IIIE/CENTAUR/MJS PROP. MOD. SHUTTLE/CENTAIR/BII or SHUTTLE/45,000 lb. AGENA/BII
Relay Link Communications	Lo-Gain Bus Antenna 400 MHz Frequency	Low to Moderate Gain Antenna UHF or L Band
Separation	Spinning Probe No., Zero but Small Angle of Attack	Probe Spin-Up Required Zero Angle of Attack
Mounting	Center-Line	Parallel to Centerline

5.1 System Configuration Trades

The Pioneer class spinning bus for outer planet entry missions has been studied by TRW (Reference 8). The modifications to the Pioneer F and G spacecraft required for these missions will be discussed in subsequent sections. The overall Pioneer bus launch weight (exclusive of bus science and the probe itself but including the modifications to support the probe) is estimated at 323 kg (710 lb). The Mariner class 3-axis stabilized bus has been considered by Martin-Marietta (Reference 3) and Jet Propulsion Laboratory (Reference 2). The modifications to the Mariner J-S spacecraft will also be discussed in subsequent sections. The overall Mariner bus launch weight (exclusive of bus science and probe) estimated at 639 kg (1405 lb).

Estimated performance data for four launch vehicle configurations are shown in Figure 5-1. These launch vehicle options are:

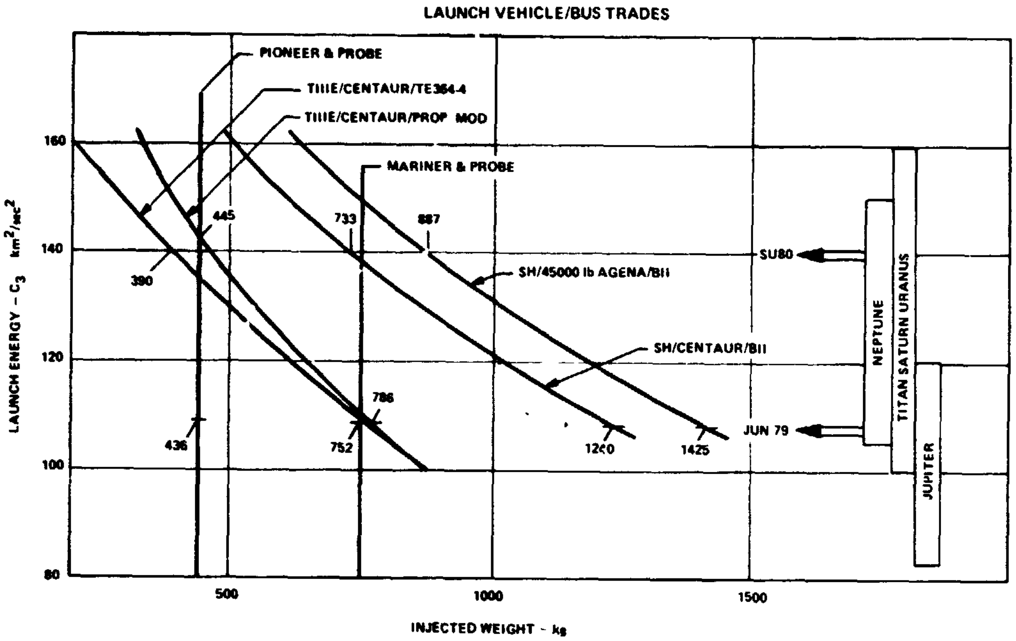


Figure 5-1. Launch Vehicle/Bus Trades

(1) Titan IIIE/Centaur/MJS Propulsion Module; (2) Titan IIIE/Centaur/TE 364-4 (2300); (3) Shuttle/Centaur/Burner II (2300); and (4) Shuttle/45000 lb Agena/Burner II (2300). The performance data shown assumes a 343 km (185 nm) parking orbit. The injected weight includes the spacecraft, probe and adapter. The weight of the Pioneer class bus with a 114 kg (250 lb) probe and the weight of the Mariner class bus with a similar probe are as indicated on Figure 5-1. The difference between the injected weight capability of a particular launch vehicle combination and the bus/probe weight is available as growth margin or for the addition of bus science for a flyby mission.

The launch energy (C_3) for minimum energy trajectories to each of the five planets for a variety of different missions are listed in Table 4-1, Section 4.1, and the range of values is shown in Figure 5-1. A C_3 of 107 km^2/sec^2 is representative of the 1979 Jupiter-Uranus-Neptune opportunity and a C_3 of 140 km^2/sec^2 is representative of the 1980 Saturn-Uranus opportunity. The injected weight capability for each launch vehicle combination for each of these opportunities is shown on Figure 5-1.

For probe targeting and planetary encounter, the choice between probe maneuvers, bus maneuvers, or a combination of each is predicted upon communications geometry, planetary quarantine philosophy, dispersion sensitivity, ease of implementation, and operational constraints.

The planetary quarantine requirements that the planet under investigation be protected against inadvertent contamination by Earth organisms is usually taken as a restriction against targeting the spacecraft to impact the planet or its atmosphere at any time during the mission. This approach requires that the entry probe make at least some of the required maneuvers at separation; that is, the bus must always be targeted for a safe flyby distance from the planet and the probe trajectory must be deflected to intercept the planetary atmosphere. The additional complexity of adding the necessary propulsion and thrust vector control (usually a moderately high probe spin rate which must be reduced before entry) capabilities to the probe when these features are already incorporated in the bus has been overriding. In addition, probe mission success is much more sensitive to the targeting dispersions resulting from maneuver implementation errors than is bus mission success.

The magnitude and direction of the velocity increment (ΔV) necessary to accomplish the bus retargeting maneuvers varies with specific mission design but generally lies between 65 and

100 m/sec. The timing and bus attitude maneuver sequence necessary to proper ΔV application depends largely on bus operational constraints; i.e. whether a spin-stabilized or 3-axis stabilized bus is utilized. In order to minimize the propellant necessary to provide the required ΔV, the loss of the Earth lock must be an acceptable condition during the maneuver. For the Pioneer, the bus retargeting maneuver must be made without losing spacecraft communications lock with Earth; requiring the maneuver to be made less efficiently. For Mariner, there are no constraints on thrust vector orientation. Consequently, the spacecraft can be positioned to provide ideal entry conditions for the probe at zero angle of attack. The probe is then spun up, released, and the bus retargeted to conduct the deflection maneuver in a single thrusting operation.

5.2 Bus/Probe Integration

The integration of the probe with the spinning Pioneer class bus and the 3-axis stabilized Mariner class bus results in several operational differences.

For the Pioneer class bus, the probe must be mounted on the bus axis at the lower end of the bus when it is in the launch configuration (as shown in Figure 5-2). The axisymmetric mounting is essential to maintain the bus spin orientation both before and after probe release. During interplanetary transit, the probe is in the shadow of the large Earth oriented communications antenna. At probe separation, the bus may not make an attitude maneuver to release the probe. The spinning probe is, therefore, released in a non-zero (however quite small) angle of attack attitude and must be designed to accommodate the resulting angle of attack at entry.

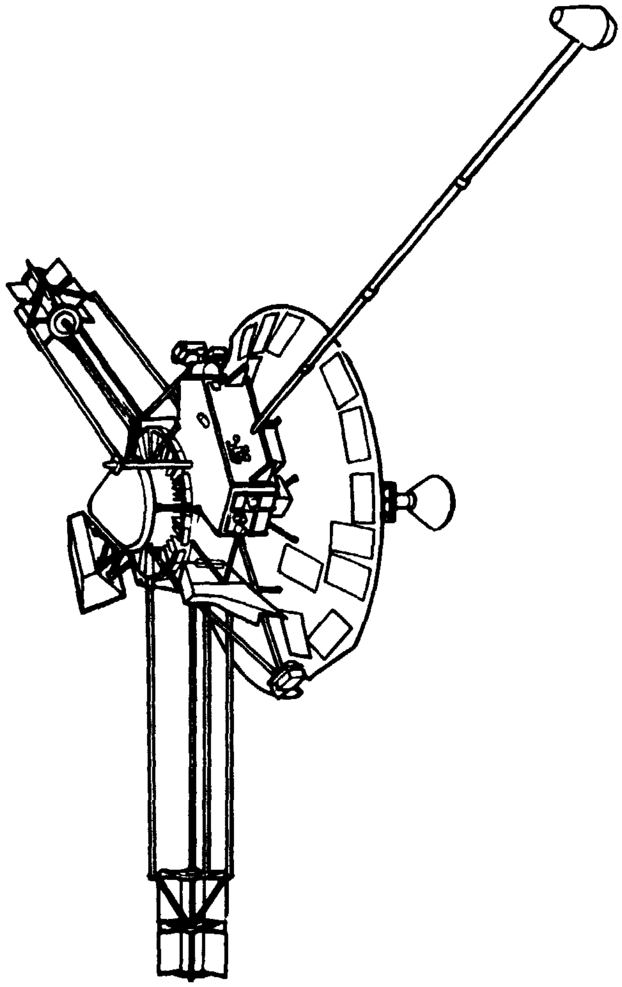


Figure 5-2. Probe on Pioneer Bus

The 3-axis stabilized Mariner class bus allows more flexibility in mounting attitude as shown in Figure 5-3. The probe would be mounted on the lower end of the spacecraft, in the launch configuration. However, it need not be mounted on the bus axis. In fact, the probe is mounted on the anti-sun side of the bus, off the centerline but parallel to it. During interplanetary transit, the probe is similarly in the shadow of the large Earth directed communications antenna. At separation, the 3-axis stabilized bus would make an attitude maneuver to release the probe and the probe can then be released in a zero

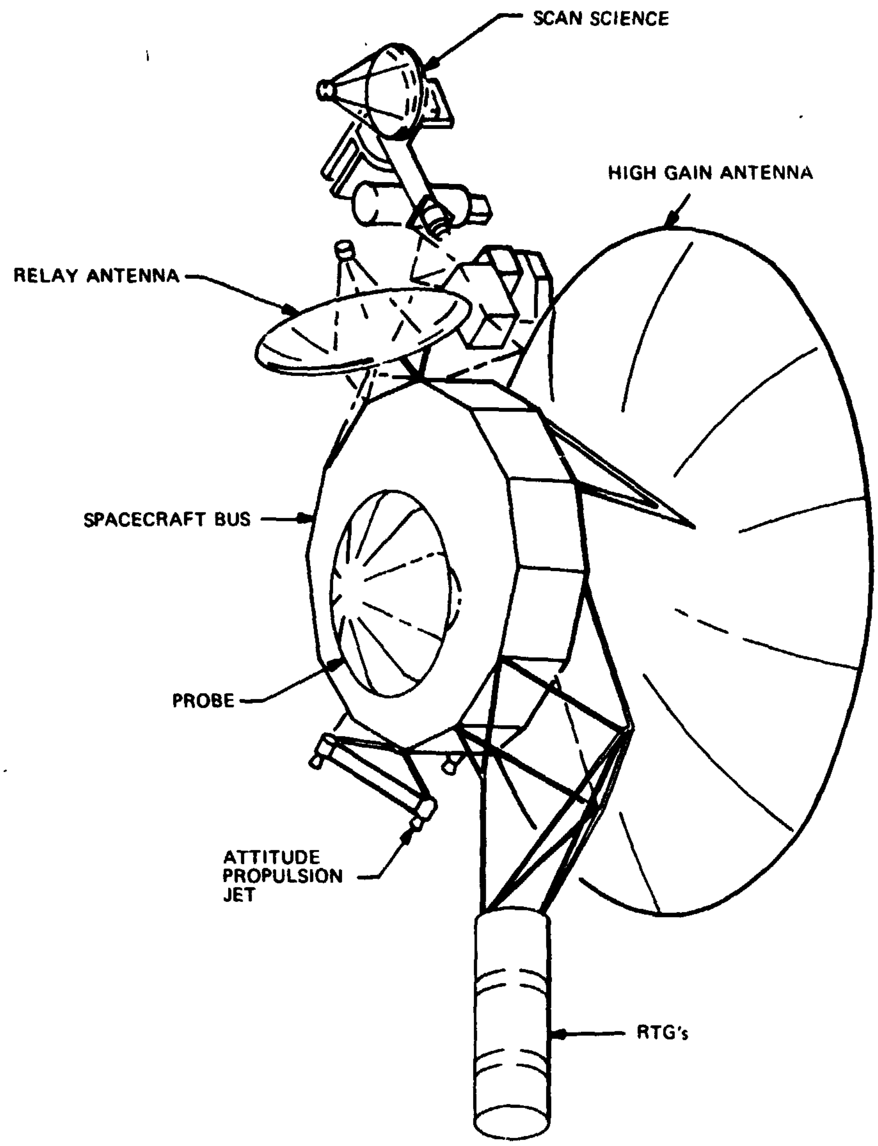


Figure 5-3. Probe on Mariner Bus

angle of attack attitude. However, before or after release, the probe must be spun up to maintain attitude stability through the long autonomous probe coast phase.

Several of the subsystems on each of the bus spacecraft must be upgraded and/or modified to support the additional requirements imposed on the bus by the probe as shown in Table 5-2.

Table 5-2 Requirements of Probe on Bus

(1)	Structurally support 114 kg (250 lb) probe.
(2)	Separate probe from bus in proper attitude and at minimum tipoff rates – mechanical and electrical.
(3)	Provide power and command functions to fire pyrotechnics for (2).
(4)	Command probe functions and probe related bus functions while probe is attached to bus.
(5)	Accept probe telemetry via hard-line or rf link.
(6)	Charge probe bootstrap batteries during interplanetary cruise.
(7)	Control and supply power adapter mounted to probe electric heaters while probe is on bus.
(8)	Control probe targeting before probe separation and deflect bus to properly phased flyby trajectory after separation

Pioneer Class Bus

The propellant tank must be enlarged to 56 cm diameter to accommodate the increased propellant required by the larger mass and moments of inertia of the bus and probe spacecraft configuration and the additional ΔV requirement to deflect the bus to a properly phased flyby trajectory after the probe is released. One radial thruster must be added for the post probe-separation maneuver and for the post-Saturn maneuver or post-Jupiter maneuver on multiplanet missions. The propellant lines and heaters must be modified to accommodate the larger tank and added thruster.

The thermal control system must be modified to add probe interstage heaters and insulation for probe thermal control. The equipment compartment insulation and louvre design must be changed to account for the probe and interstage shadowing of existing reflecting surfaces and to allow for the addition of probe related equipment on the bus.

A new interstage structure must be added to carry and separate the probe and to transmit spacecraft loads around the probe to the launch vehicle adapter. The structure of the bus must also be modified to accommodate larger RTG power sources and to balance the larger RTG's by ballasting the magnetometer boom.

The communications subsystem must be augmented by the addition of a relay link antenna, receiver and data storage. Also an X-Band driver, transmitter, waveguide and switches must be added to improve the bus-Earth communications link. The high gain Earth Antenna feed must be modified to include a dual S-X Band Feed and the S-Band receiver must be modified to provide a coherent S-Band up link and X-Band down link. The data handling formats must be modified to include probe data and probe data buffers must be added. The command system must be modified to accommodate probe commands and bus commands relative to probe functions.

The power system must be modified to accommodate larger RTG's which allow charging of the probe bootstrap batteries and which can supply increased power to communications subsystems. The spacecraft battery and its related charging equipment are also replaced.

The weight impact of the required modifications is about 113 kg (250 lb) in the spacecraft and launch vehicle adapter.

<u>Mariner Class Bus</u>

Modifications to the Mariner class bus to support the entry probe requirements are relatively minor although several are required. The basic MJS spacecraft must add propellant to the ACS system to support the longer duration mission and perform the bus deflection maneuver. While the tank volume is more than adequate to permit this, the impact on the design is operation at slightly higher tank pressure and blow down ratio. However, the ACS tank must also be moved slightly forward and outboard to provide clearance for the probe support structure. The spacecraft to propulsion module adapter must be slightly enlarged to provide adequate clearance for the probe.

There are essentially no modifications requir for communications or thermal control. Since the data han ng system will be changed to support the requirements of new flyby science, changes to support the probe data requirements are easily accommodated. In fact, the probe data rates are so low they represent a very minor impact.

The relay link will necessitate incorporation of the receiver and attention to configuration and size of the relay antenna. Mariner can accommodate either a UHF or an L-Band link. Depending on choice, the antenna could be either a low or moderate gain fixed dish.

The demands on the Mariner power system to provide power for thermal control during cruise, periodic health checks, probe battery charging/discharging, pre-separation checkout and arming of the probe ordinance are well within the capability without any increase in size or number of RTG's.

Mariner must be modified to provide capability to spin-up the probe prior to release. However, direct adaptation of the Pioneer probe adapter and release mechanization can be accommodated with the addition of a small motor to perform the spin-up.

The weight impact of the required modifications to the Mariner class bus is approximately 10 kg (22 lb) in the spacecraft and launch vehicle adapter hardware and 20 kg (44 lb) in fuel.

6.0 PIONEER SATURN/URANUS PROBE FROM PIONEER VENUS (PV) PROGRAM

This section presents a summary of the results of the Reference 12 study and a series of private communications with the Hughes Aircraft Company Pioneer Venus Program Office relating the Reference 14 study to the outer planets probe. The Reference 12 study's objective was to define a common PS/U probe, compatible with the outer planets science objectives, using designs based on existing hardware from the Martin-Marietta Pioneer Venus (PV) Program and/or to assess the effect of modifying their PV hardware designs to make them compatible with outer planet missions requirements. An iterative study was conducted using nominal atmospheric models which resulted in a "baseline" and an alternative probe configurations. The study was later to consider the cool, nominal and warm atmospheric models and a "final" probe configuration was defined to meet the worst case atmospheric models. This section will summarize only the "final" configuration of the study. The Appendix A summary charts present a convenient comparison of this probe and those configured in other outer planet missions studies.

The Hughes Aircraft Company data reflects an evaluation of the application of their Pioneer Venus hardware now under development toward the outer planet probe.

6.1 Science Objectives and Instruments

The science constraints, basic science objectives and instrument complement are essentially the same as shown in Tables 3-3 and 4-4 herein. Some differences from the baseline probe of Section 3 exist in the characteristics of the particular instrument equipment selected with regard to weight, volume, power, data rate, etc. The PV science instruments can be used, with modification, for the Saturn/Uranus missions. Table 6-1 summarizes the instrument availability and modifications required for the SU missions.

TABLE 6-1

SCIENCE INSTRUMENT AVAILABILITY

Instrument	PV Source	Remarks
Nephelometer	Large or Small Probes	No Modification
Pressure Gauge (10^3 to 10^7 N/m^2)	Large or Small Probes	No Modification
Accelerometers (10^{-2} to 400G)	Large Probe	Modified Range (10^{-2} to 600G)
Temperature Gauge (200° to 850°K)	Large or Small Probes	Modified Range (40° to 450°K)
Neutral Mass Spectrometer (1 to 254 AMU)	Large Probe	Modified for Mass Range (1 to 40 AMU). Porous plug leaks replaced. Vent tube added for outgassing. Repackage to fit probe. Possible replacement of ion pump system for helium. Alternate sources possible.

6.2 System Design

Figure 6-1 is a cutaway pictorial of the Martin-Marietta probe concept and its major subsystems. The mission sequence of events for this design is the same as that shown on Figure 3-4 herein, with the addition of the staging event for jettison of the probe nose cap after entry to expose the science instrument ports (Figure 6-2). This staging event may present an attractive compromise to the conflicting requirements of ballistic coefficient/weight/reliability as discussed in Paragraph 4.2 herein. This configuration should be among those evaluated in the advanced development programs recommended in Paragraphs 7.1 and 7.4. Figure 6-3 presents geometric and ballistic information

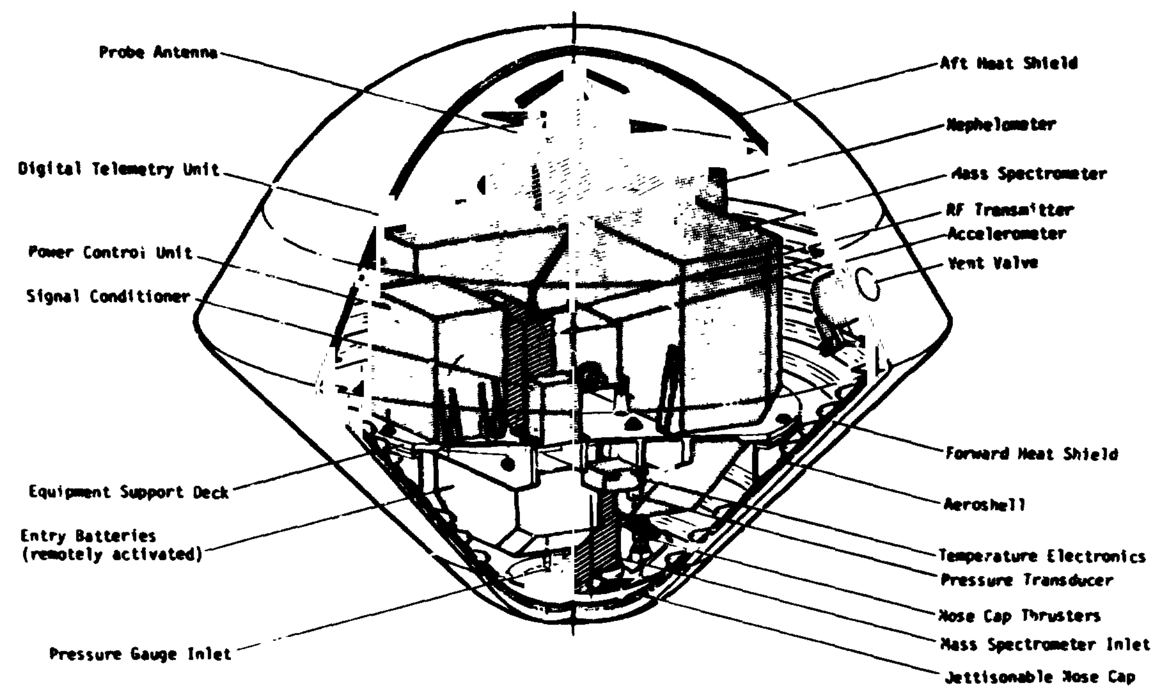


Figure 6-1 Probe Configuration

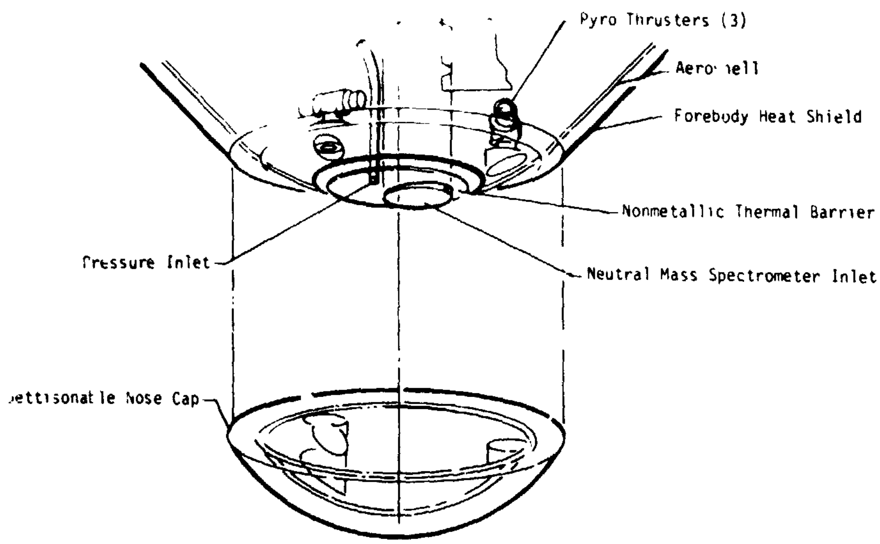


Figure 6-2 Nose Cap Jettisoning System

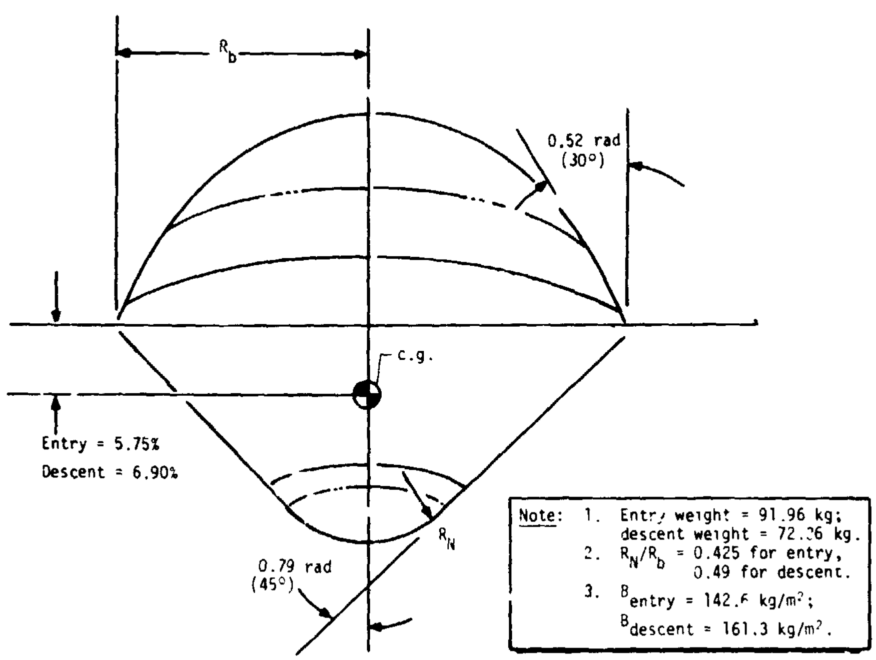


Figure 6-3 Probe Deceleration System

with regard to the probe deceleration system. No deployable devices are required for aerodynamic stability or to control rate of descent.

Figure 6-4 shows the Pioneer Venus probes presently being developed by Hughes. Many of the subsystems are directly applicable with little or no modification

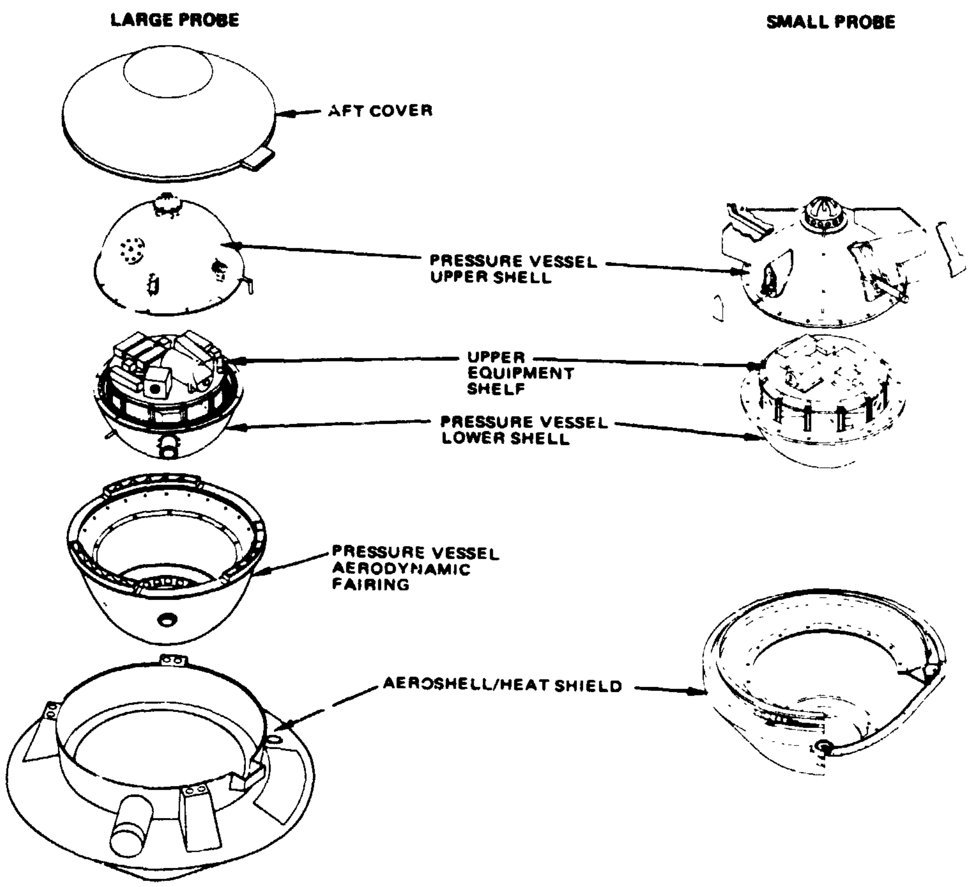


Figure 6-4 Exploded Views of Pioneer Venus Large and Small Probes (Not to scale -- Large probe actually twice size of small probe)

6.3 Electrical/Electronic Design

The communication requirements on this probe are summarized in Section 3.0.

Table 6-2 summarizes the hardware source and availability for the Electrical/Electronic Components based upon the Reference 12 Martin Marietta Study. Table 6-3 summarizes the hardware source and availability based upon the Hughes Reference 14 Study.

Table 6-2 Hardware Availability for Electrical/Electronic Components—MMC

Electrical & Electronic Design	PV Commonality or Other Source	Remarks
DATA		
Digital Telemetry Unit	PV, Either Probe	Minimal modification to replace two programmed ROMs
Power Control Unit	PV, Either Probe	Modify coast timer decoding logic. Replace 6 programmed ROMs.
Signal Conditioner	PV, Either Probe	No change.
POWER		
Power Control Unit	PV, Either Probe	Update unijunction transistors. Modify wiring to add 29.4 m/sec^2 (3 G) switch.
Battery	New Design & Build	Existing technology.
COMMUNICATIONS		
Transmitter	PV, Either Probe	Modified for frequency and modulation change.
Antenna	PV, Either Probe	Modified for frequency change.
GENERAL		
G Switches	PV, Either Probe	Change 49 m/sec^2 (5 G) switch to 0.98 m/sec^2 (0.1 G). Add 29.4 m/sec^2 (3 G) (decreasing) switch.

Table 6-3 Hardware Availability for Electrical/Electronic Components—HAC

ELECTRICAL & ELECTRONIC DESIGN	P.V. COMMONALITY OR OTHER SOURCE	REMARKS
COMMAND & DATA		
Command/Data Unit	P.V., Either Probe	Reprogram ROM. Increase memory storage capacity
Command Output Module	P.V., Either Probe	No Change
Pyro Control Unit	P.V., Either Probe	P.V. unit is modular; no change
Acceleration Switches	P.V., Either Probe	No Change
POWER		
Power Interface Unit	P.V., Large Probe	No change or remove unneeded fused circuits
Battery	New	Based on P.V. High-G Technology
Current Sensor	P.V., Either Probe	No Change
COMMUNICATIONS		
Transmitter	New	Based on P.V. solid state modular concept; modify for frequency change.
Antenna	New	P.V. Antenna has hemispherical coverage.

6.4 Structural and Mechanical Design

The structural and mechanical requirements on this probe are also summarized in Section 3.0. Table 6-4 summarizes the hardware source and availability for the Structural/Mechanical Components based upon the Reference 12 Martin Marietta Study. Table 6-5 summarizes the hardware source and availability for the Structural/Mechanical Components based upon the Reference 14 Hughes Study.

Table 6-4 Hardware Availability for Structural/Mechanical Components—MMC

Mechanical & Structural Design	PV Commonality or Other Source	Remarks
<u>CONFIGURATION</u>	PV, Small Probe	Use aeroshell tooling.
<u>AERODECELERATOR & AERODYNAMICS</u>	PV, Small Probe	Use aerodynamic test data.
<u>HEAT SHIELD</u>	New Design & Build	Existing technology.
<u>MECHANISMS</u>		
Pin Pullers	Viking, PV	No modification.
Ball-Lock Release Pins	TRW Programs, Minuteman	No modification.
Cable Cutter	PV, TRW Programs	No modification.
Pyro Thrusters	Hi Shear	No modification.
<u>THERMAL CONTROL</u>		
Isotope Heaters	Pioneer Spacecraft	No modification.
Thermal Blanket	Pioneer Spacecraft	No modification.
Foam Insulation	Saturn II	No modification.
Argon Gas Assembly	New Design & Build	Existing technology.
<u>STRUCTURE</u>		
Aeroshell	PV, Small Probe	Modified for larger diameter.
Remaining Structure	New Design & Build	Existing technology.

Table 6-5 Hardware Availability for Structural/Mechanical Components—HAC

STRUCTURAL AND MECHANICAL DESIGN	P.V. COMMONALITY OR OTHER SOURCE	REMARKS
<u>CONFIGURATION/AERODYNAMICS</u>	P.V., Small Probe	Use P.V. Aero. Test Data
<u>HEAT SHIELD</u>		
Material	P.V., Either Probe	No Change
Insulation	Military Programs	Existing Te~hnology
<u>STRUCTURE</u>		
Aeroshell	P.V., Small Prove	Larger Diameter
Equipment Support	P.V., Small Probe	Minor Modification
<u>THERMAL CONTROL</u>		
Isotope Heaters	Pioneer	No Change
Thermal Blanket	Intelsat IV, etc.	Configuration – Dependent
Internal Insulation	P.V., Either Probe	Configuration – Dependent
<u>MECHANISMS</u>	P.V., Military Programs	No Change

6.5 Conclusions

These studies conclude that there exists approximately 85% commonality between the hardware being used on the Pioneer Venus probes and the hardware which would be used on a probe designed to accomplish the first-order science objectives through entry into the expected range of environments at Saturn and Uranus. With few exceptions, the remaining equipment is available either from other programs or new designs based on existing technology. Additional economies appear to be obtainable by incorporating the Saturn/Uranus requirements into the PV equipment design -- with little or no cost or performance impact on the PV Program.

7.0 KEY TECHNOLOGY ELEMENTS

During the conduct of this study, and the assessment of the work of prior studies, several areas have been identified which should receive priority attention in the subsequent phases of the design and development of the Probe. This section provides a summary of these key technology areas. Not all of these elements are discussed in the body of the report

7.1 Verification of Aerodynamic Descent Configuration

The baseline probe concept should be exposed to detailed design and model testing to refine and verify its suitability for the data gathering missions into the varied planetary atmospheres. Advanced development work is suggested for examining the pivotal issues on the entry and descent configurations. Among these is the evaluation and verification of ballistic coefficients and stability in the outer planet atmospheres and to investigate associated aerothermal and operational sequencing problems (instrument exposure and deployment, requirement for staging, etc.) The overriding objective would be to eliminate or at least to minimize, the need for staging sequences. Wind tunnel testing is suggested as a screening process to assist in selecting the most aerodynamically promising configuration for more comprehensive investigation. If found to be necessary, follow-on activities would include obtaining performance and stability and control information on the use of various aerodynamic devices (ballutes, parachutes, etc.)

7.2 Selection of Bus Concept

This study, and those conducted previously, considered the spin-stabilized bus as represented by the Pioneer 10 configuration, and the three-axis stabilized bus, as represented by the Mariner J-S configuration. Because of the considerations of probe separation, probe thermal control, bus deflection maneuvers after probe separation, relay link communications, and weight, it is recommended that detail design activity be initiated to:

- o Establish the Pioneer and Mariner design modifications
- o Investigate alternatives to minimize the weight, power and subsystem impacts of these modifications

In parallel with the above, a design concept study is recommended which would seek to configure a three-axis stabilized bus, lighter than the Mariner, from existing, space-qualified hardware.

7.3 Long Shelf Life Battery

The baseline power subsystem for the probe represents a compromise of the requirements for short term high power drains, long term low power drains, and high peak currents (for pyrotechnic initiation). In addition, lifetimes up to eleven years and decelerations up to 750 G_E must be considered. NASA has been conducting advanced development activities in this field at the Ames Research Center, the Goddard Space Flight Center, and the Jet Propulsion Laboratory in advanced battery material development, improved cell and cell components and in the development of long life batteries. An assessment is recommended of the current state of NASA battery developments and the application of these to the baseline detail probe design. Detail of battery configurations for this probe should be initiated and would consider:

- o separator materials
- o plate composition
- o volume (to contain residual gas)
- o fill mechanism

A verification test program including wet stand life and deceleration tests should be a part of this activity.

7.4 Instrument Inlet Design

A primary mission science objective for the outer planet probe, independent of the planet of interest, is the examination of the planetary atmosphere. In-situ measurement of the atmospheric composition and structure will contribute significant information concerning the solar system and planetary evolution. The neutral mass spectrometer and the pressure transducer require direct access to the atmosphere. Of primary engineering importance to the accuracy and success of these instruments is the design of the inlet ports. An RTOP project is recommended whose results will contribute significantly to the Probe design and the Pioneer Venus program. Beginning with the technology established by PAET, Viking, and advanced development projects at Goddard and Ames, a detail design and verification project is suggested. This project should evaluate inlet port locations and designs with the objective of minimizing the chance of sample contamination from heat shield out gassing or ablative products. The inlet design must also minimize the problems of mass discrimination through absorption, chemisorption, and condensation of the gas sample on the inlet system surfaces. Laboratory models of the inlet system concepts should be tested in simulated planetary atmospheres to provide data for detail design specifications. This project may lead to variations of inlet design as a function of the planet of interest (i.e. heaters, pumping, etc. as a function of gases to be measured.)

In paragraph 7.1 a project is recommended for detail design and verification of the probe descent configuration with the primary objective of establishing a design which eliminated or minimized the staging sequences. Retention of the entry heat shield influences the inlet designs and their locations. These projects should be conducted in parallel with an interchange of design and trade-off information.

7.5 Atmospheric Composition Instruments

In conjunction with the inlet design study above, a continuing evaluation of the basic science instrument development status and suitability should be conducted by NASA. For example, the mass spectrometer may fall short of a gas chromatograph for obtaining data in light atmospheres whereas for gasses such as N_2 and CO_2 the mass spectrometer may be satisfactory. The neutral particle mass spectrometer cannot distinguish between molecules and compounds that have the same atomic mass. There are two schemes that could help to resolve this problem. First, use of a gas chromatograph would make a unique unambiguous determination of the atmospheric composition. The main disadvantage of the gas chromatograph is the long process time. A second approach takes advantage of the vapor pressure fractionation of the atmosphere. With this approach, certain constituent gases of the atmosphere are frozen out. In this manner sufficient mass composition data is available to help unravel ambiguities. This scheme is dependent upon the ability to process enough data at the various cloud levels.

7.6 Heat Shield Verification

The selection of a baseline heat shield in this study, and previous studies, has been based on the performance data of heat protection systems demonstrated in laboratory tests and Earth reentry vehicle programs (NASA and USAF). The heating environment which must be survived for successful entry into the atmospheres of Saturn and Uranus extends the state-of-the-art demonstrated to date. Heat shield advanced development effort should continue in the evaluation of available materials in simulated environments. Particular attention is suggested in conducting material characterization tests using a number of atmospheric compositions and combined convective and radiative heating loads. This key technology area also identifies the need for test facilities capable of simulating the expected entry conditions.

The ability to test in these conditions is prerequisite to the development of new materials tailored to provide maximum heat protection per kilogram in the outer planet environments. This program should be conducted considering the ultimate goal of Jupiter entry.

7.7 Micrometeoroid Protection

The definition of activity required in this area will depend in large measure on the results of data from Pioneer 10 and Pioneer 11. Preliminary assessments of Pioneer 10's flight through the asteroid belt have indicated existing specification models of micrometeoroid flux to be conservative. Analysis of Pioneer 10 and 11 data and its impact on the baseline bus and probe subsystem design is appropriate.

7.8 Planetary Sterilization

The requirements of NASA document NHB 8020.12 were not considered in detail in this study or in any of the prior probe system studies. Nevertheless, the next phase of program planning and design of the outer planet probe should consider the provisions of this document and its impact on planetary mission designs (bus/probe targeting), bus systems, and bus/probe manufacture.

7.9 Shelf Life of Probe Electrical and Mechanical Equipment

A major design concern for the outer planet probe is successful turn-on and operation within specifications of all equipment. This concern is associated with all hardware programs, but is particularly severe for the outer planet probe due to the extremely long shelf-life requirements. Outer planet missions have flight times that range from 1126 days (∿3 years) to 4074 days (∿11 years). To this must be added about one to two years for the time from initial sub-assembly fabrication with subsequent integration into subsystem hardware until the launch date.

Therefore, the bounding shelf life times can range from about a minimum of four years to a maximum of about thirteen years after which time the equipment must operate without anomaly for about sixty minutes. An assessment must be made of the equipment design specifications that are necessary to achieve the equipment reliability requirements to satisfy the outer planet mission success goal.

7.10 Planetary Position Refinement

From the viewpoint of reduction in the outer planet probe heating and loads it would be valuable to reduce the uncertainty estimates in the planetary positions. This points to the value of continued telescopic work to refine the ephemerides of the outer planets, particularly Uranus. At present, the targeting requirements for Uranus, based upon the current ephemeris, set the design boundary for the "common" outer planet (Saturn-Uranus) probe. With better defined ephemerides, the width of the design entry corridor can be reduced. An increase in the entry angle on the shallow entry angle side of the corridor will reduce the integrated heating and resultant heat shield fraction. A decrease in entry angle on the steep entry angle side of the corridor will reduce the heating rates and loads. Reduction in heating rate will provide greater confidence in the heat shield design, and a reduction in loads will reduce the aeroshell structure and internal support structure weight fractions and also reduce the level of extensive development and testing of equipment.

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APPENDIX A

This appendix contains summary charts of information and data assimilated and used during the conduct of this study. They are based on the various studies conducted for NASA in the past (as referenced on the charts) and are presented for the information and use of the reader in comparing the characteristics of the major subsystems studied for probe missions to the outer planets.

Characteristics of Selected Science Instruments

	JUPITER		SATURN-URANUS			NEPTUNE	TITAN
	MARTIN-MARIETTA REF 3	JET PROPULSION LABORATORY REF 2	MARTIN-MARIETTA REF 5	McDONNELL DOUGLAS REF 4	PV to PS/U MARTIN-MARIETTA REF 12	MARTIN-MARIETTA REF 3	DYNATREND REF 31
Temperature Sensor							
Range $\sim ^{o}K$	100-400	100-400	50-350	50-500	40-450	40-190	–
Weight ~ kg	0.45	0.45	0 45	0.35	0 32	0 45	0 32
Volume ~ cm^3	426	409	426	115	98	426	197
Power ~ watts	1.4	1.5	1 4	1 0	0.5	1 4	0.2
Bits/sample	10	7	10	10	8	10	9
Sample interval ~ sec	3-5	3.5	3-5	50	24	3-5	1/5 km
Bit rate - bits/sec	1.7 - 3.3	2	1.7 - 3.3	0.2	0.3 to 0.5	1 7 - 3.3	–
Pressure Sensor							
Ranges ~ bar	$5x10^{-2}$-1 1-30	0-1 1-13	$5x10^{-2}$-1 1-30	0.1-20	10^{-2} – 10^{2}	$5x10^{-2}$-1 1-30	–
Weight ~ kg	0.68	0.68	0.68	0.2	0.45	0.68	0.36
Volume ~ cm^3	246	–	246	192	115	246	262
Power ~ watts	1 3	2.0	1.3	1.2	0.5	1.3	0 8
Bits/sample	10	7	10	10	8	10	9
Sample interval ~ sec	3-6	3 5	4	50	24	4	1/5 km
Bit rate - bits/sec	1.7 - 3.3	2	2.5	0.2	0 3 to 0 5	2.5	–
Accelerometer Triad							
Range - Longitudinal G_E	10^{-2}-1600	10^{-2}-1600	10^{-2}-400	0-800	10^{-2} – 600	10^{-2}-250	–
Range - Lateral G_E	±125	10^{-2}-25	±125	±10	–	±125	–
Weight ~ kg	1 55	1.55	1.55	0.3	1 13	1 55	0.68
Volume cm^3	916	–	916	101	656	916	492
Power ~ watts	2.8	2 0	2.8	2 0	2 3	2 8	2 0
Bits/sample - entry	30	10	30	10/7	40	30	27
- descent	60	60	60	10/7	40	60	27
Sample interval - entry-sec	0.1-0 4	0.1/0.2	0 1-0 4	0.2	0.2	0 1-0 4	0 1
- descent-sec	8-15	10	8-15	50	24	8-15	1/5 km
Bit rate - entry-bps	100-200	100	100-200	50/35	100	100-200	270
- descent-bps	4-7.5	50	4-7.5	0.2/0 14	1 7 to 2 1	4-7 5	–
Neutral Mass Spectrometer							
Range ~ MU	1-40	1-40	1-40	1-40	1-40	1-40	1-40
Weight ~ kg	5.43	5.45	5 43	6.4	9.07	5.43	6 4
Volume ~ cm^3	6756	5834	6756	8197	9830	6756	6555
Power ~ watts	16	16	16	11	14	16	12
Type	Quadropole	–	Quadropole	Quadropole	*	Quadropole	Quadropole
Bits/sample	400	463	400	5706	6400	400	290
Sample interval ~ sec	30-70	40	50	405	400	50	1/25 km
Bit rate - bits/sec	6-14	11 6	8	16	16	8	–
Pressure limit ~ bars	13	13	13	–	–	13	–
Nephelometer							
Weight ~ kg	1 13	–	1 13	0 5	0 49	–	0 91
Volume ~ cm^3	1312	–	1312	427	524	–	1311
Power ~ watts	3	–	3	1 2	1 2	–	3
Bits/sample	10	–	10	58	43	–	18
Sample interval ~ sec	3	–	3	30	12	–	1/5 km
Bit rate - bits/sec	3 3	–	3 3	2	3 6	–	–

* Quadropole or Magnetic sector Analyzer

Science Instrument Complement

	JUPITER		SATURN-URANUS			TITAN
	MARTIN-MARIETTA REF 3	JET PROPULSION LABORATORY REF 2	MARTIN-MARIETTA REF 5	McDONNELL DOUGLAS REF 4	PV to PS/U MARTIN-MARIETTA REF 12	DYNATREND REF 31
TEMPERATURE GAGE	X	X	X	X	X	X
PRESSURE GAGE	X	X	X	X	X	X
ACCELEROMETERS	X	X	X	X	X	X
NEUTRAL MASS SPECTROMETER	X	X	X	X	X	X
NEPHELOMETER			O	X	X	X
GAS CHROMATOGRAPH						X
ION MASS SPECTROMETER				O		X
LANGMUIR PROBE			O	O		X
ALPHA SCATTER						X
BETA SCATTER						X
UV PHOTOMETER						X
THERMAL RADIOMETER						X
ALTIMETER						X
GYRO						X
ION RETARDING POTENTIAL ANALYZER			O			X
NEUTRAL RETARDING POTENTIAL ANALYZER			O			
X = Primary Payload O = Expanded Payload						

Thermal Control Subsystem Characteristics

	JUPITER			SATURN-URANUS		TITAN
	MARTIN-MARIETTA REF 3	JET PROPULSION LABORATORY REF 2	MARTIN-MARIETTA REF 5	McDONNELL DOUGLAS REF 4	PV to PS/U MARTIN-MARIETTA REF 12	DYNATREND REF 31
Temperature Range						
Operating °K	272-320	250-340	280-320	278-336	255-348	255-305
Non-Operating °K	250-325	228-340	240-325	233-273	233-339	–
Heaters						
Type 1	RHU	RHU	RHU	RHU	RHU	RHU
Power ~ w	–	–	8	4	10	30
Type 2	–	–	–	SC/ELECTR	–	–
Power ~ w	–	–	–	–	–	–
Heater Weight ~ kg	–	1.8	1.6	1.8	0.6	3.5
Insulation Type	External Blanket	External Blanket	External Aluminized Mylar	External Goldized Mylar	External Mylar Blanket & Internal Foam	Internal Foam
Thickness ~ cm	–	–	–	1.3	2 5	2
Total Thermal Control Weight ~ kg	7.7	7.9	5.2	7	4.24	8 2

Power & Pyrotechnics Subsystem Characteristics

	JUPITER		SATURN-URANUS			TITAN
	MARTIN-MARIETTA REF 3	JET PROPULSION LABORATORY REF 2	MARTIN-MARIETTA REF 5	McDONNELL DOUGLAS REF 4	PV to PS/U MARTIN-MARIETTA REF 12	DYNATREND REF 31
Main Battery						
Type	Remote AgZn	Remote AgZn	Remote AgZn	Remote AgZn	Remote AgZn	Remote AgZn
Weight ~ kg	1.9	4.5	2.4	4.0	7.3	9.1
Volume – cm^3	361	871	1200	1277		–
Energy Capacity W-H	–	179	160	239	270	600
Pyrotechnic Battery						
Type	HgZn	HgZn	HgZn	NICAD	NONE	NONE
Weight - kg	Contained	0.4	Contained	1.4		
Volume - cm^3	in timer	10	in timer	400		
Energy Capacity W-H	0.2	0.2	0.2	12		
Post Separation Battery						
Type	Remote AgZn	–	Remote AgZn	– NONE	NONE	Sealed NICAD
Weight kg	3.1	–	0.1			3.6
Volume cm^3	606	–	94			–
Energy Capacity - W-H	–	–	7.5			21
Power Conversion Pyrotechnic Initiate	Remote Capacitor	Central Capacitor	Remote Capacitor	Remote Battery-Relays	Remote Battery	Remote Battery-Solid St. Sw.
Coast Timer						
Type	Accutron	Accutron	Accutron	Crystal Oscillator	Accutron	Accutron
Power ~μW	8	15	9	140		4
Pyrotechnic System Weight ~ kg	–	7.3	2.5	2.5		1.6

Communications Subsystem Characteristics

	JUPITER			SATURN-URANUS		TITAN
	MARTIN-MARIETTA REF 3	JET PROPULSION LABORATORY REF 2	MARTIN-MARIETTA REF 5	McDONNELL DOUGLAS REF 4	PV to PS/U MARTIN-MARIETTA REF 12	DYNATREND REF 31
Communications Range (max) ~ 10^3 km	197	89	180	100	108	64
Frequency - MHz	860	860	860	400	560	1,000
Transmitter Power ~ watts	25	10	25	40	18	30
Probe Antenna Type*	Turnstile/Spiral	Turnstile/Slot	Turnstile/Cone	Microstrip	Turnstile/Cone	Circular Aperture
Gain ~ dB	6.2/5.2	3/6	6.5	7.5	6.5	7
Beam ~ deg	35/110	30/90	100	66	100	30
Probe Antenna(s)						
Weight ~ kg	0.68	2.73	0.45	0.91	0.54	1.1
Bus Antenna Type	Helix	Helix	Dish	Loop-vee	Loop-vee	Turnstile
Gain ~ dB	12.3	12	18.3		3.1	1.5
Beam ~ deg	45	35	20	50	55	70
Bit Rate	28	5	28	44	32	32
Modulation	PCM/PSK/PM	FSK	FSK + tone	FSK	FSK + tone	PCM/PM
Transmitter						
Weight ~ kg	2.7	2.7	2.7	1.4	1.1	2.7
Volume ~ cm^3	1,394	–	1,355	555	675	–
Power ~ watts input	56	25	56	90	64	100
Storage Capacity ~ k bits	13	–	60.2	17.4	15	–
Descent Time ~ min	33	33	60	60	29 to 72	30

*Two Values Shown entry/descent.

Mechanical Characteristics

DIAMETER	JUPITER: MARTIN-MARIETTA REF 3	JUPITER: JET PROPULSION LABORATORY REF 2	SATURN-URANUS: MARTIN-MARIETTA REF 5	SATURN-URANUS: McDONNELL DOUGLAS REF 4	SATURN-URANUS: PV – PS/U MARTIN-MARIETTA REF 12	TITAN: DYNATREND REF 31
Diameter ~ cm	94	100	87	89	87.4	71
Ballistic Coefficient kg/m^2	102	104	104	124	143	478-796
Weight ~ kg	157.5	122	89	113.6	92	123
Weight Contingency %	15	–	15	20	15	40
Hypersonic Drag Coefficient	1.51	1.51	1.51	1.22	–	0.71
Forebody Shape	60° Cone	60° Cone	60° Cone	60° Cone	45° Cone	Discoverer
Forebody Structural Concept	RSM	RSM	RSM	HC	RSM	HC
Forebody Structural Material	Alum.	Titanium	Titanium	Fiberglass Alum.	Alum.	Fiberglass Beryllium Nose
Forebody Heatshield Material	ATJ	ATJ	ATJ	Carbon Phenolic	Carbon Phenolic	Elastomeric
Afterbody Shape	Spherical	–	Spherical	Spherical	Spherical	Cone
Afterbody Structural Concept	Segmented	–	Segmented	HC	Segmented	–
Afterbody Structural Material	Alum.	Alum.	Alum.	Fiberglass	Laminates	–
Afterbody Heatshield Material	Elastomeric	Elastomeric	Elastomeric	Elastomeric	Teflon	Elastomeric
Staged	Yes	Yes	Yes	No	Yes*	Yes
Parachute Type	disc-gap-band	disc-gap-band	disc-gap-band	NONE	NONE	–
Parachute Diameter ~ m	2.46	2.59	2.28	NONE	NONE	3.35
Parachute Material	Dacron	–	Dacron	NONE	NONE	Dacron
Secondary Parachute Type	Circular	NONE	Circular	NONE	NONE	NONE
Secondary Parachute Diameter ~ m	0.45	NONE	0.73	NONE	NONE	NONE
Secondary Parachute Material	Dacron	NONE	Dacron	NONE	NONE	NONE
Descent Capsule Shape	Cone/cylinder	–	Cone/cylinder	NONE	Cone/Sphere	NONE
Diameter ~ cm	48.3	–	47	NONE	84.6	NONE
Weight kg	41.9	–	40.5	NONE	69.4	NONE
Descent Ballistic Coefficient kg/m^2	236	–			160	

HC - Honeycomb
RSM - Ring Stiffened Monocoque
ATJ - ATJ Graphite

*Deployable Nose Cap