PIONEER MARS 1979 MISSION OPTIONS

PRICES SUBJECT TO CHANGE

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PIONEER MARS 1979 MISSION OPTIONS

by

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Preface

This study was performed by the authors, between November 1973 and January 1974, as part of the advanced planning studies activity performed by Science Applications, Inc. for the Planetary Programs Division of OSS/NASA under Contract No. NASW-2494. The results are intended for advanced planning evaluation of an extended array of mission options for post-Viking Mars exploration.

Table of Contents

SUMMARY

Summary

As part of its continual planning effort, the Planetary Programs Division of OSS/NASA has been developing a number of mission options for post-Viking/75 Mars exploration. For the two remaining Mars launch opportunities in this decade, i.e. 1977 and 1979, planning emphasis to date has been placed on derivatives of Viking/75 hardware. NASA's recent commitments to the development of the Space Shuttle in this same time frame could, however, reduce resources to a point where a follow-on Viking mission might not be possible until the early 1980's. If this were to happen, rather than completely abandoning Mars opportunities in the late 1970's, OSS/NASA would like to have several lower cost mission concepts available for consideration as alternatives.

The purpose of this study was to conduct a preliminary investigation of lower cost (<$100M) Mars missions which perform useful exploration objectives after the Viking/75 mission. As a study guideline, it was assumed that significant cost savings would be realized by utilizing Pioneer hardware currently being developed for a pair of 1978 Venus missions. This in turn led to the additional constraint of a 1979 launch with the Atlas/Centaur launch vehicle which has been designated for the Pioneer Venus missions.

Selection of science-effective Pioneer mission concepts which would follow the Viking/75 mission without competing with future Viking missions in the early 1980's was accomplished by a process of elimination. Flyby concepts, e.g. a probe/relay bus, a remote sensor platform, or an atmospheric aeronomy platform, were all rejected because of the inadequate sampling time available considering the advanced state of Mars exploration. Low cost atmospheric entry probes and rough landers were rejected because their science potential is largely redundant to Viking/75 objectives. Two concepts, using an orbiter bus platform, were identified which have both good science potential and mission simplicity indicative of lower cost. These are: a) an aeronomy/geology orbiter, and b) a remote sensing orbiter with a number of deployable surface penetrometers.

Mission A, the Aeronomy/Geology Orbiter, would perform in situ aeronomy measurements in the Martian ionosphere by using low periapse altitude (≈ 100 km) elliptical orbits. The low altitudes in the region of periapse also permit the inclusion of several remote sensing instruments capable of performing geologic surface mapping, e.g. a radar altimeter and a γ-ray spectrometer. Key mission parameters

developed in this study are summarized in the Summary Table. Both the aeronomy and geology measurements would extend similar Viking entry/lander science data to a global scale. The trade-off for this capability is sterilization of the entire Pioneer orbiter spacecraft in order to meet Mars planetary quarantine requirements. Because the spacecraft passes through the upper atmosphere every orbit, its lifetime, even with periapse control, is only several years at best. The cost of this mission, excluding science, is estimated to be about $31M (FY '74 dollars). This assumes the modification of an additional Pioneer Venus orbiter flight article, including sterilization, for a single launch in 1979. Suitable aeronomy instruments are readily available from many earth satellite programs, some of which have already been proposed for the Pioneer Venus orbiter mission in 1978. Appropriate remote sensing geology instruments are much more questionable, especially the γ-ray spectrometer, and could require significant development. Still, a total mission cost of $40-50M dollars seems reasonable.

Mission B, the Remote Sensing/Penetrometer Orbiter would sequentially deploy a number of surface penetrometers to preselected impact sites distributed in either the northern or southern hemisphere of the planet. In addition to being a communications relay station between a deployed penetrometer and the earth, the orbiting bus could carry a complement of remote sensing instruments for orbital investigation of the Martian atmosphere and surface. Key mission parameters developed in this study are given in the Summary Table. A total of four sterilized penetrometers would be carried by a modified Pioneer Venus orbiter bus. These would be deployed one at a time from an elliptical polar orbit over a period of time as long as one Mars year. Each penetrometer would have its own deorbit motor and entry/ descent system. Penetrometer design and descent velocity specification provide for a minimum penetration of 1 m in rock without destruction. During a 1-week surface lifetime each penetrometer would identify soil penetrability, search for subsurface water, and perform an elemental chemical analysis of the subsurface material at its impact site. The data collected from its instruments would be transmitted to the orbiter once each Mars day for relay back to earth. Between the four one-week penetrometer missions the orbiter could perform remote sensing measurements with its own science package. The factors of low cost, low power, low data rate, and high minimum altitudes (>1000 km) probably restrict these measurements to atmospheric studies with existing or slightly modified instruments. The scientific merit of such experiments in 1980 requires further study. The cost of this mission, excluding orbiter science, for a single 1979 launch is estimated to be about $63M. This figure includes $24M for the development

4

Summary Table

SELECTED PIONEER MARS MISSION CONCEPTS

- Mission A: Aeronomy/Geology Orbiter

  o 50-70 kg science payload

  o Aeronomy and surface geology science instrumentation

  o 300-350 kg orbited payload

  o $\geq$ 100 km periapse altitude

  o 24 hour initial orbit period

  o 45° orbit inclination

  o $\approx$One Mars year orbit lifetime

  o Entire spacecraft sterilized

- Mission B: Remote Sensing Orbiter/Penetrometers

  o 40-60 kg orbiter·science payload

  o Four impact penetrometers @ 40 kg each

  o Penetrability, water detection, and soil chemistry impact science instrumentation

  o 500-550 kg orbited payload

  o 1000 km periapse altitude

  o 24.6 hour controlled orbit

  o 90° orbit inclination

  o >42 year orbit lifetime

  o $\approx$One week penetrometer lifetime

  o Penetrometers sterilized

5

and fabrication of four penetrometers (including penetrometer science), one flight spare and a PTM. Depending on the selected orbiter remote sensing experiments, total cost (excluding launch vehicle) for the Remote Sensing/Penetrometer Mission could have a range of $70-80M (FY '74 dollars).

This exploratory analysis has identified and outlined at least two 1979 Mars mission concepts, based on Pioneer Venus technology and hardware, which have the potential for performing relevant post-Viking/75 science at a cost of less than $100M. Mission A, the Aeronomy/Geology Orbiter, represents a minimum development/cost mission estimated at less than $50M. Yet the broad sampling of ionospheric composition and heat balance performed by this mission would greatly expand the data base from which scientists are trying to understand the evolution of the Martian atmosphere. Further, its potential for performing global geologic mapping from low altitude, gained by sterilizing the entire spacecraft, is not possible with the present Viking orbiter design.

Mission B, the Remote Sensing/Penetrometer Mission, is a somewhat more expensive mission, with in situ surface objectives, estimated at a cost of $70-80M. This mission requires the development of high impact (≈ 150m/sec) penetrometers for which there exists an impressive history of earth-based experience. Pioneer Venus orbiter modifications would also be more significant than for Mission A. The science highlights of this mission are a) global exploration for subsurface water and b) establishment of a basis for extension of Viking Lander geologic data to global interpretations. The orbiter has the capability to perform continued non-imaging remote sensing studies of Mars from a polar orbit. The penetrometer concept also is a viable candidate for additional missions after 1979. Besides deploying the same penetrometers to more sites, there is the potential for a penetrometer/seismometer experiment pending development of a longer life (≈ 90 day) power source.

It is important to point out that neither of these concepts should be considered feasible on the basis of this study. Many engineering questions exist for both concepts which require further study. Indeed, the actual Pioneer Venus Orbiter spacecraft design was not known at the time this analysis was performed. Undoubtedly there are solutions for each engineering problem which can be developed in a spacecraft systems study. The important question to be answered is: "How do these solutions change the definition and cost of the missions?"

It is equally important to note that the potential role of Pioneer-class Mars missions has not been thoroughly explored by a NASA science advisory group.[1] This potential should be refined for various post-Viking/75 Mars exploration scenarios as more and better definitions of Pioneer Mars mission concepts are developed.

INTRODUCTION

9

Study Guidelines

The Planetary Programs Division of OSS/NASA is in the process of refining its mission plans for the next step in Mars exploration after the Viking/75 mission. This planning presently favors another Viking mission in 1979 following the recommendations of the Mars Science Advisory Committee given in their report [1], "Mars: A Strategy for Exploration". There has, however, been a growing concern over available resources for later 1970's planetary missions due to NASA's commitment to development of the Space Shuttle in this time frame.

It is the Planetary Program Division's intent to vigorously pursue support for a Viking/79 mission. Yet it is also true that should this goal become unrealistic in the current constrained budgetary climate, less costly Mars mission concepts must be available as alternatives in order to avoid complete disruption of Mars exploration after the Viking/75 mission. The purpose of this study is to identify such alternatives, based on Pioneer Venus technology. Obviously, these alternatives should be less complex than Viking missions, resulting in lower cost. It is equally important that these missions be capable of building on our understanding of Mars after Viking/75, not merely recollecting existing data. In other words, science-effectiveness should not be sacrificed for cost-effectiveness.

STUDY GUIDELINES

- PLANNING RATIONALE FOR STUDY

    o NASA is planning a Viking/79 follow-on
       mission;

    o a 1979 Mars mission will compete with
       the Space Shuttle for development funds;

    o in the event of insufficient funds for a
       Viking/79 mission, is there a cost-
       effective Pioneer/79 mission that is a
       science-effective substitute for the
       continued exploration of Mars?

- STUDY OBJECTIVE

    o To identify and describe potentially
       feasible Pioneer Mars missions that
       perform relevant post-Viking/75 science
       in a cost-effective manner.

Study Constraints

A number of constraints, listed on the facing page, were adopted for this exploratory mission study. As much a guideline as a constraint, $100M was accepted as a cost ceiling for "lower cost" missions. The Planetary Programs Division directed that the mission concepts be compatible with Pioneer Venus design hardware. In this way, it would be possible to save spacecraft costs through direct hardware inheritance from the Pioneer Venus missions planned in 1978. This constraint led in turn to a 1979 launch opportunity and use of the Atlas/Centaur, the Pioneer Venus launch vehicle. Mission concepts were arbitrarily constrained to single launches, primarily to minimize cost but also for analysis convenience. Adjustments in cost and mission operations for dual launches could be subsequently added if desired. Finally, it was essential that planetary quarantine requirements established for Mars and observed by previous missions also apply for proposed Pioneer Mars mission concepts.

It should be noted that NASA was in the process of evaluating competitive bids by Hughes Aircraft Co. and TRW Systems, Inc. for the Pioneer Venus Program at the time of this study. Hence, it was not possible to incorporate a specific spacecraft design into the development of Pioneer Mars mission concepts based on Pioneer Venus technology. Rather, a generalized spacecraft design was assumed based on pre-proposal contractor and NASA studies of the Pioneer Venus and Venus Explorer concepts. However, since it was known that the bidders proposed significantly different spin-axis reference systems, e.g. one design incorporates a despun antenna with the spin axis perpendicular ($\perp$) to the ecliptic while the other design uses a fixed antenna on the spin axis which is parallel ($//$) to the ecliptic plane pointing toward the earth, the impact of spacecraft orientation on power and communications was briefly treated for each configuration. Hence, Pioneer Mars spacecraft results, in terms of subsystem weight, are presented in this report for both ($\perp$) and ($//$) designs. We hasten to add that these data were generated entirely independent of NASA's Pioneer Venus selection process and in no way affected the outcome of that evaluation.

STUDY CONSTRAINTS

- Lower Cost Mission Concepts (<$100 M)

- Pioneer Venus spacecraft/science inheritance

- 1979 Mars launch opportunity

- Atlas/Centaur launch vehicle

- Single launch

- Observation of Mars planetary quarantine

Candidate Pioneer Mission Concepts

There are within the spectrum of planetary mission plans a wide variation of mission concepts. A condensed list of concepts, relevant to continued Mars exploration, which are possibly compatible with Pioneer Venus spacecraft are listed on the facing page. Aeronomy missions imply in situ measurements within the Mars ionosphere. Upper atmospheric probes are unshielded entry probes which also perform aeronomy measurements until communication blackout and/or probe disintegration occur. Lower atmospheric probes are shielded probes which survive entry deceleration and transmit in situ atmospheric measurements until surface impact occurs. Rough lander probes survive entry and surface impact (≈ 50 m/sec) to perform limited in situ surface experiments. Surface penetrometers survive entry and impact (≈ 150 m/sec) to penetrate intact to several meters and transmit subsurface in situ measurements to an orbiter or flyby spacecraft by means of a transmitter left at the surface.

Each of these concepts, along with the supporting Pioneer spacecraft bus (flyby or orbiter), was briefly examined, a) for science potential after the Viking/75 mission, and b) for complexity relevant to Pioneer Venus hardware, and low cost. By a process of elimination the list was reduced to two missions of most interest for a 1979 launch. These are: Mission A) a combined aeronomy and remote sensing orbiter, and Mission B) a remote sensing orbiter with multiple surface penetrometers.

The lower atmospheric probes and rough lander probes were judged to be too complex and expensive if designed to perform relevant post-Viking/75 science. Science experiment flexibility of the orbiter bus mode is preferred to the flyby mode. The additional propulsion requirements of orbiter missions for the 1979 launch opportunity are reasonable compared with available Pioneer Venus orbiter solid rocket motors. Possible alternatives to Mission A, particularly if orbiter sterilization isn't possible, would be either an orbiter with an aeronomy subsatellite or an orbiter with multiple upper atmospheric probes. However, both of these concepts are more complex, have less science potential and probably would cost more than Mission A. A brief definition of the selected missions (A and B) is given on the next page.

CANDIDATE PIONEER MISSION CONCEPTS

- Aeronomy Flyby

- Flyby with Upper Atmosphere Probe(s)

- Flyby with Lower Atmosphere Probe(s)

- Flyby with Rough Lander Surface Probe(s)

- Flyby with Surface Penetrometer(s)

- Remote Sensing Orbiter

- Aeronomy Orbiter

- Orbiter with Aeronomy Subsatellite

- Orbiter with Upper Atmosphere Probe(s)

- Orbiter with Lower Atmosphere Probe(s)

- Orbiter with Rough Lander Surface Probe(s)

- Orbiter with Surface Penetrometer(s)

Selected Mission Concepts

The characterizing parameters of Mission A, the Aeronomy/Geology Orbiter, are listed on the first half of the facing page. The key difference between this mission and previous Mars orbiters is that it would orbit Mars with much lower periapse altitudes, 100-200 km. These low altitudes are necessary, of course, in order to perform in situ sampling of the ionosphere with aeronomy instrumentation. Low orbiter altitudes are also attractive for certain types of remote sensing sensors, e.g. a radar altimeter and a γ-ray spectrometer. Hence, global geologic mapping can be added as an important mission objective of a low altitude Mars orbiter. Orbited payload, following the Pioneer Venus Orbiter (PVO) design, would be between 300-350 kg. Preliminary analysis of Mars transfer characteristics for 1979 indicate that a 24-hour initial orbit is achievable with the baseline PVO retro motor. An orbit inclination between 30° - 60° will probably be necessary to resolve the competing objectives of diurnal and surface coverage. The low periapse altitudes will cause the orbit to decay in a relatively short period of time. Spacecraft sterilization is therefore essential to meet planetary quarantine requirements. An active orbit lifetime of one Mars year is a reasonable goal for this mission concept.

The key parameters of Mission B, the Remote Sensing Orbiter with Penetrometers, are listed on the second half of the facing page. Characteristic of all unsterilized Mars orbiters, the orbit periapse is set at about 1000 km altitude. The remote sensing science package (excluding imaging due to cost and communication limitations) on the orbiter may not be very different than the non-imaging instrumentation of the Mariner ('71) and Viking ('75) Mars orbiters. The really new and different science of Mission B would be carried in 4-6 surface penetrometers sequentially deployed by the orbiter to preselected impact sites. Subsurface water detection, soil chemical analysis, and seismometry are some of the relevant science measurements within the capability of penetrometers. Orbited payload, including a modified PVO bus and penetrometers, is expected to weight 500-550 kg. A Mars synchronous orbit period of 24.6 hours is possible with larger developed retro motors compatible with the basic PVO design. A polar synchronous orbit is desired to maximize impact site selection for the penetrometers and to receive one-way data transmissions from a deployed penetrometer on a daily basis. The orbiter, following prescribed planetary quarantine requirements would have at least a 42-year orbit lifetime. Each penetrometer would be sterilized and have an operating surface lifetime of about one week using battery power.

SELECTED PIONEER MARS MISSION CONCEPTS

- Mission A: Aeronomy/Geology Orbiter

    o 100-200 km periapse altitudes

    o Aeronomy and surface geology science

    o 300-350 kg orbited payload

    o 24 hour initial orbit period

    o 30°-60° orbit inclination

    o $\approx$One Mars year orbit lifetime

    o Entire spacecraft sterilized

- Mission B: Remote Sensing Orbiter/Penetrometers

    o 1000 km periapse altitude

    o Remote sensing orbiter science and subsurface
       penetrometer science

    o 500-550 kg orbited payload including 4-6 penetrometers

    o 24.6 hour controlled orbit

    o 90° orbit inclination

    o >42 year orbiter lifetime; $\approx$1 week penetrometer lifetime

    o Penetrometers sterilized

Candidate Pioneer Mars Orbiter Instruments

A summary of candidate orbiter experiments for a Pioneer Mars mission are presented in the table below. For each of four classes of experiments a set of relevant instruments is listed along with related flight projects and spacecraft requirements (weight, power, and average date rate). The solar wind/ionospheric interaction instruments, in addition to being proposed for PVO-78, have flown on many Pioneer, Imp, Mariner and Vela satellite missions. The aeronomy instruments have all been flown on many earth satellites including Explorers 17 and 31, Orbiting Geophysical Observatories, and Atmospheric Explorer/C, which has just been launched. They are included in the tentative payload of the Pioneer Venus Probe mission and some have been proposed as indicated for the PVO-78 mission. A good description of aeronomy instruments selected for the Atmosphere Explorer missions is presented in the April 1973 issue of Radio Science[2].

For remote sensing of surface properties only one of the three listed instruments has previously flown on an unmanned spacecraft. The IRIS was flown on the attitude-stabilized MMO-71 mission with success. It's applicability to another Mars orbiter mission on a spinning platform needs further study. The suggested data rate for the IRIS implies the lower duty cycle of a spinning platform. The radar altimeter has been proposed for the PVO-78 mission and would be a new development. The proposed design could be adapted to Mission A with little modification; on Mission B the radar antenna area would have to be increased by a factor of 10 in order to maintain the required gain at constant power. Power is down from 15 watts at Venus to 11 watts at Mars due to the planets' lower noise temperature. Weight of the experiment with the baseline 39 cm antenna is 9 kg; with the larger antenna this weight is increased to 15 kg. The most significant measurement returned by this radar system would be mapping of the surface dielectric constant. Although altitude measurements on Mission A could also be important the instrument is not too good for topographic mapping. A small quasi-specular spot size (.5-2.5 km diameter) separated by up to 50 km between consecutive measurements without the support of an imaging instrument for site identification combine to degrade the value of the instruments topographic mapping capability. Vertical resolution would be 150 m. One of the more exciting possibilities for global geologic exploration from orbit is a γ-ray spectrometer. Unfortunately the development of a suitable sensor for a small unmanned spacecraft is only in an embryotic stage of development. Sensor weight, deployment and separation of signal flux from background flux are some of the problems which need to be ironed out. The weight of 10 kg below is an adaptation of the sensor weight used on the γ-ray spectrometer of Apollo 15/16; it is not a complete experiment weight.

Although the instrumentation for atmospheric structure experiments is well-developed this objective will have been well-studied by 1979. The additional objectives available to these instruments on a spinning spacecraft at that time requires further study. Certainly a dual-frequency (S and X band) occultation experiment would support the aeronomy objectives of Mission A and could be of supportable value on Mission B as well.

CANDIDATE PIONEER MARS ORBITER INSTRUMENTS

(1978 PVO Science Payload $\leq$ 45 kg)

Experiment	Objectives	Instruments	Flt. Proj.	Weight (kg)	Power (watts)	Data rate (bps)
Solar Wind/ Ionosphere Interactions	Interaction of interplanetary medium with planet; Ionospheric penetration a definite advantage	Solar Wind/ Photoelectron Anlyzr	PVO-78*	5	2.5	36
		Electron/Ion Temp Probe	AE/C-73	2	3	25
		Magnetometer/Boom	PVO-78*	3.5	3	32
Aeronomy	Composition, photochemistry and escape processes of upper atmosphere; In situ density measurement helpful; Minimum altitude diurnal/latitude sampling desired	Neutral Mass Spectrometer	AE/C-73	5.5	10	100
		Ion Mass Spectrometer	AE/C-73	1.5	2	100
		Tri-axial Accelerometer	AE/C-73*	10	19	200
Surface Properties	Distribution of surface properties: topography, dielectric constant, composition, structure, and geology; Low altitude an advantage	Radar Altimeter	PVO-78*	9	11	125
		γ-Ray Spectrometer	Apollo 15/16	10+	10	1
		IR Spectrometer (IRIS)	MMO-71	17	16	750
Atmospheric Structure	Vertical/Global variations in atmospheric temperature, pressure, and aerosols	IR Radiometer	PVO-78*	5.5	6	100
		UV Spectrometer	PVO-78*	5.5	6	30
		Dual Frequency Occultation	PVO-78*	3	10	-

*Proposed for Pioneer Venus Orbiter - 1978
+Excludes experiment boom (Apollo 15/16 boom = 7.6 m)

Mars Transfer Characteristics

The table on the facing page summarizes the relevant trajectory data for Earth-Mars transfers launched during Oct.-Nov. 1979 and arriving Aug.-Oct. 1980. Type II transfers (travel angle >180°) are used in order to minimize the Mars approach velocity (VHP) and hence the retro ΔV requirements for orbit insertion. Nominal trajectory conditions are indicated (boxed in) for a Mars arrival on Sept. 7, 1980 (JD 2444490) and a 20-day launch window centered about Nov. 2, 1979 (2444180). The maximum launch energy C3 requirement of 10 (km/sec)² can be satisfied by the Atlas/Centaur launch vehicle capability. Declination of the launch asymptote (DLA) varies between 23° and 37° which indicates that range safety limitations would generally be satisfied. Nominal approach velocity is 2.63 km/sec with a variation of about 40 m/sec.

MARS 1979 TYPE II TRANSFER CHARACTERISTICS

|	Launch Date (JD-2440000)	TF (days)	C3 (km/sec)^2	DLA (deg)	VHP (km/sec)
Arrival Date = 2444460	4140	320	20.5	14.2	2.902
	4150	310	15.8	14.0	2.922
	4160	300	12.3	14.8	2.957
	4170	290	10.0	16.8	2.994
	4180	280	9.0	20.6	3.021
	4190	270	10.5	38.2	2.999
	4200	*	*	*	*
	4210	*	*	*	*
Arrival Date = 2444490	4140	350	21.9	16.9	2.927
	4150	340	16.4	17.6	2.792
	4160	330	12.5	19.6	2.707
	4170	320	10.0	23.4	2.657
	4180	310	9.1	29.1	2.630
	4190	300	9.9	37.6	2.630
	4200	290	16.0	55.9	2.833
	4210	*	*	*	*
Arrival Date = 2444520	4140	380	26.8	17.4	3.620
	4150	370	19.6	18.2	3.329
	4160	360	14.5	20.2	3.123
	4170	350	11.2	23.9	2.986
	4180	340	9.6	29.3	2.907
	4190	330	9.6	35.8	2.882
	4200	320	11.8	43.0	2.931
	4210	310	18.2	53.9	3.180

* Type I Transfers

Mars Arrival Conditions

Interplanetary transfer energy characteristics dictate a Mars arrival date on or near 7 September 1980 (Julian date = 2444490; see previous page) for low energy 1979 orbiter missions. The Mars arrival conditions on this date are depicted in the stereographic projection on the facing page. For purposes of this discussion the Eastern hemisphere (right half of planet as seen from the sun) is presented. The subsolar point (local noon) is on the left side of the primitive circle, 7.9° above the Mars equator. The motion of the subsolar point along the Mars ecliptic plane with passage of time is shown as arrows within the hemisphere. It crosses the Mars equator in a southerly direction (Autumnal equinox) approximately 40 days after arrival. The northern season at arrival is late Summer just before formation of the North polar hood. Recession of the South polar cap has not started yet. The evening terminator is shown as a double line in the center of the diagram. Lines of constant planetocentric latitude have also been included for completeness.

The orientation of minimum capture energy orbits is closely related to the asymptotic approach direction of the spacecraft to Mars. In particular the exit point of the asymptotic approach velocity ($\overline{VHP}$) vector on the planet is of interest. Three of these $\overline{VHP}$ positions are shown near the center of the hemisphere, one for each of three launch dates. The middle (solid) dot corresponds to the approach direction for a 2 November 1979 (2444180) launch date. The open dots correspond to launch dates 10 days before and after this reference date. Once a reference $\overline{VHP}$ position has been established, specifying the capture periapse altitude (assuming minimum energy periapse capture) determines the locus of all possible initial orbit periapse positions. This locus of points appears as a circle around the reference $\overline{VHP}$ position on a stereographic projection. It is shown on the facing page for an orbit periapse altitude of 500 km. A variation of 500 km in this value would make almost no change in the size of this circle.

Note the significance of the orbit periapse locus. All initial periapse locations are in the vicinity of the evening terminator with a bias toward the southern hemisphere. The subsatellite trace of a sample orbit has been constructed (solid curve with widely spaced arrows indicating direction of motion) for illustration. This is a 45° inclined orbit whose ground trace would eventually traverse 70% of the planet. Initial periapse (Rp_0) is located in the southern hemisphere, mid-afternoon local time. On the right-hand side of the hemisphere a dash-dot circle has been added indicating the earth occultation boundary for a satellite at 1000 km altitude above the Mars surface. Assuming elliptical synchronous (period = 24.6 hrs.) orbits are most practical for a Pioneer 1979 Mission the sample orbit illustrated would not experience earth-occultation even though its trace passes within the occultation boundary. This is due to the fact that by the time the spacecraft reaches the boundary its altitude has already increased from 500 km at periapse to more than 1000 km. For this orbit earth-occultations would not occur until 3-4 months after orbit capture.
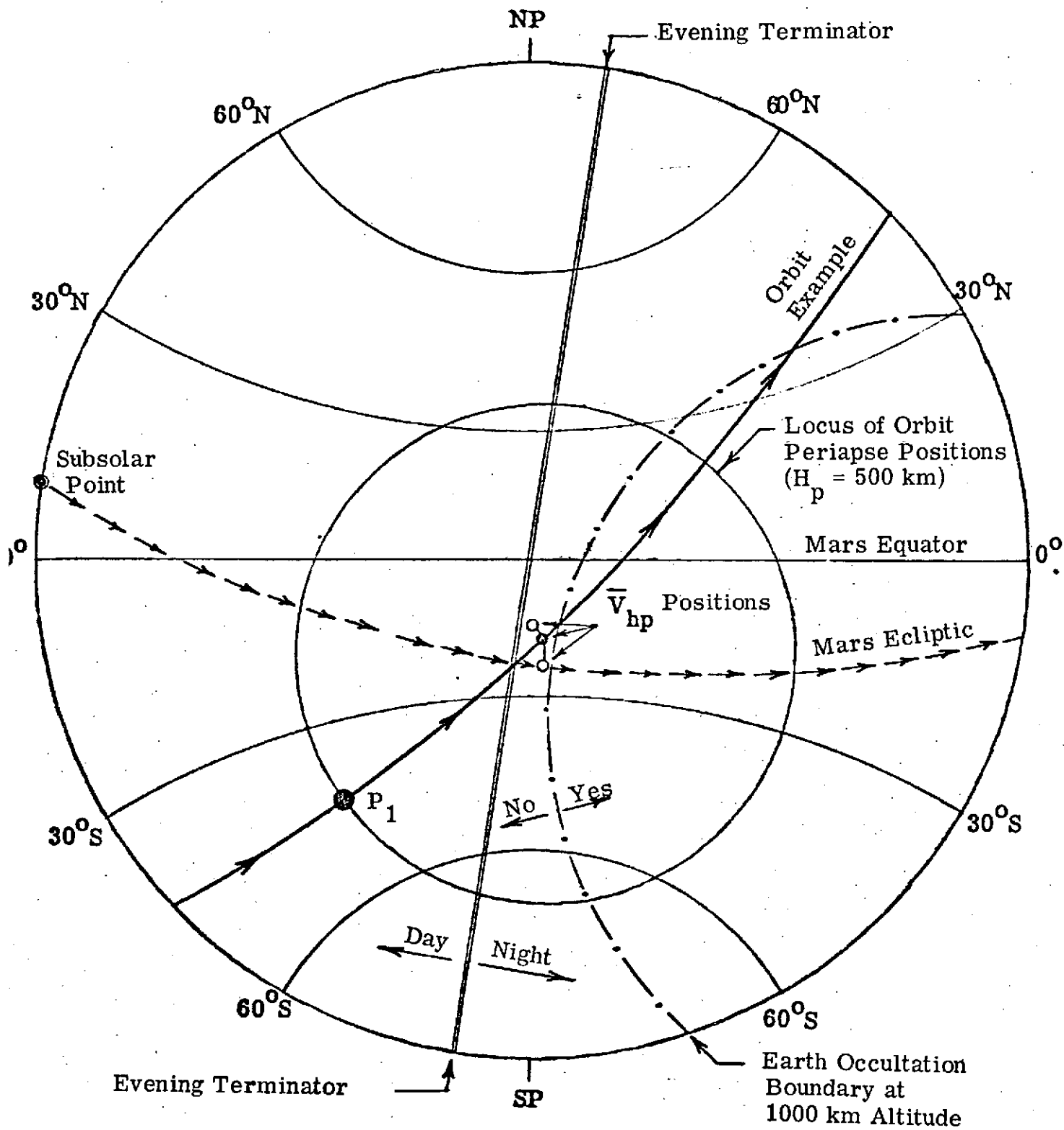
MARS AT ARRIVAL (9/7/80) OF PIONEER MARS ORBITER MISSIONS

Mission A

AERONOMY/GEOLOGY ORBITER

Rationale for an Aeronomy/Geology Orbiter

The first of the two Pioneer Mars mission concepts proposed as 1979 launch candidates is the Aeronomy/Geology Orbiter (Mission A). An introductory description of this mission is given above (pages 16 & 17). The following role of a Mars aeronomy orbiter is offered by D. M. Hunten:

"An aeronomy mission on an orbiter would emphasize measurement of the neutral and positive-ion composition of the ionosphere and its heat balance. Such information is interesting to a few specialists; its broader interest, though real, is less obvious. There is good reason to think that the basic nature of the Mars atmosphere has been determined by escape mechanisms that are a by-product of its ion chemistry. Without these mechanisms, the atmosphere might be much denser, and consist predominantly of N_2 and O_2. The implications for biology are obvious. As for geology, the Mariner 9 photography shows numerous examples of features most easily explained by running water. Yet we see no sign of this water today, nor is the atmospheric pressure great enough to permit liquid water. Perhaps great masses of H_2O and CO_2 are locked up in the polar regions; or perhaps major amounts have been lost.

Non-thermal escape mechanisms are important for Mars because of its low gravitational potential; the escape energy from a height of 200 km is 1/8 eV per atomic mass unit. An oxygen atom, for example, thus requires only 2 eV to escape, and several processes exist that could yield such an energy. According to McElroy (Science 175, 443, 1972, and later work), the most important processes are dissociative recombination of O_2^+, N_2^+, and CO^+, as well as photon and electron impact on N_2, CO_2, and CO. The loss rates, averaged over the planet, are:

$$\text{for O, } 6 \times 10^7 \text{ cm}^{-2} \text{ sec}^{-1}$$

$$\text{for N, } 3 \times 10^5$$

$$\text{for C, } 8 \times 10^5.$$

In addition, the thermal flux of H is $1.2 \times 10^8 \text{ cm}^{-2} \text{ sec}^{-1}$, and is probably regulated in the lower atmosphere to be equal to twice the O flux. The amount of water lost at this rate during the age of the solar system is 250 g cm^{-2}, and the amount of carbon more than half what is now present in the atmosphere as CO_2. The present rarity of nitrogen finds a natural explanation.

All these estimates are uncertain because they must be based on theoretical models of the Martian upper atmosphere and ionosphere. The only direct information we have to check the models are electron-density profiles and airglow data from Mariners 4, 6, 7, and 9. The loss rate of 0 is particularly sensitive to the degree to which CO_2^+ is converted to O_2^+ by ionospheric reactions. The complicated aeronomy of the earth's ionosphere suggests that great uncertainties remain. Direct, in-situ measurements of the relevant quantities (especially ion densities and charged-particle temperatures) would enormously increase our confidence in the models. Such measurements will be made during entry by the Viking landers in 1976, but the instrumentation is limited and only two geographical locations will be sampled. Again, our experience with the earth suggests that a well-instrumented orbiter, with wide coverage in latitude and local time and a low periapsis, will be required for a satisfactory and convincing description of the upper atmosphere."

Motivation for a geology orbiter lies in the fact that this would be the first Mars mission proposed which would directly address the objective of global surface geology with dedicated orbiter instruments, i.e. a radar altimeter and a γ-ray spectrometer. The prerequisite for these instruments is low orbit altitudes. Low Mars orbit altitudes in turn require spacecraft sterilization and hence aeronomy and global geology instruments can become companion payloads on a common orbiter bus.

The appropriate aeronomy instruments for this mission are well developed and have been flown many times on earth satellites. They are well suited for Pioneer spacecraft and several have been proposed for the PVO-78 mission. Unfortunately, at the present time the same is not true for the geology instruments. The simplicity of the proposed PVO-78 radar altimeter may be inadequate for global surface geologic objectives of a 1979 Mars mission. Although γ-ray detectors have been flown in aircraft and on Apollo lunar missions, a suitable instrument for an unmanned planetary orbiter has not yet been developed.

Other remote sensing geology instruments are probably not appropriate for (or unique to) a low altitude Mars orbiter. X-ray fluorescence from the surface would be largely absorbed by the Martian atmosphere. Infrared spectral techniques, although possibly useful, do not benefit from low altitude orbits. Such an instrument (IRIS) has already been flown on Mariner 9 and is being considered for a future Viking mission. Hence, the geology objectives of Mission A are proposed on the basis of potential rather than proven existing hardware. The cost-effective realization of science-effective geology instruments for this mission will require considerably more development.

Orbit Payload Capability: Mission A

Orbit payload capability is presented in the double figure on the facing page for 1979 Mars transfers. The left figure presents the relationship between launch date and orbit capture velocity (retro ΔV). Three curves are shown for 12, 24 and 48 hour orbit periods. A constant capture periapse altitude of 500 km is assumed. These data are for the reference arrival date, 7 September 1980, discussed above. Orbit capture velocity can also be related to useful orbit payload once a fixed retro state is specified. The right hand figure shows this relationship for the PVO-78 baseline solid retro motor.

The two figures make it possible to select a consistent combination of launch window, orbit period and useful orbit mass. The useful orbit mass for this mission should be at least 300 kg based on available PVO-78 data. [3,4] A nominal 24 hour orbit period provides a maximum useful orbit mass of 335 kg as shown. Expanding the launch period to a 30 day window increases the required orbit capture ΔV by as much as 30 m/sec in order to maintain this 24 hour orbit period. Since a solid retro motor with a fixed propellant loading would be used for this mission it will have a fixed retro ΔV capability if the spacecraft mass is also fixed, which it should be. Hence there is an inconsistency between a fixed spacecraft/propulsion system and a fixed orbit period due to capture impulse variations across the launch window. Rather than follow fixed arrival speed transfers across the launch window (which result in higher launch energy requirements), it is suggested that the actual period of the capture orbit be allowed to float to match the retro ΔV capability of the spacecraft. For the Aeronomy/Geology orbiter this makes sense because the orbit period will be constantly changing anyway. Choosing a 335 kg useful mass the retro ΔV capability of the SVM-2 motor is seen to be 915 m/sec. The resultant variation in orbit period across a 30 day launch window, assuming a fixed orbit periapse altitude of 500 km is 24.0-29.3 hours, with the larger value occurring at the beginning and end of the window.

Type II Mars Transfers
Trip Time = 310 days
Periapse Alt. = 500 km

12^h Period Orbit

24^h

48^h

30 m/sec

30^d
←Launch→
Window

Retro ΔV
(m/sec)

1500

1400

1300

1200

1100

1000

900

800

700

SVM-2 Motor
(PVO Baseline)
M_o = 159 kg
MF = 0.873
I_{sp} = 284 sec.

335 kg

9/23   10/13   11/2   11/22

1979 Launch Date (M/D)

200   300   400   500

Useful Orbit Mass (kg)
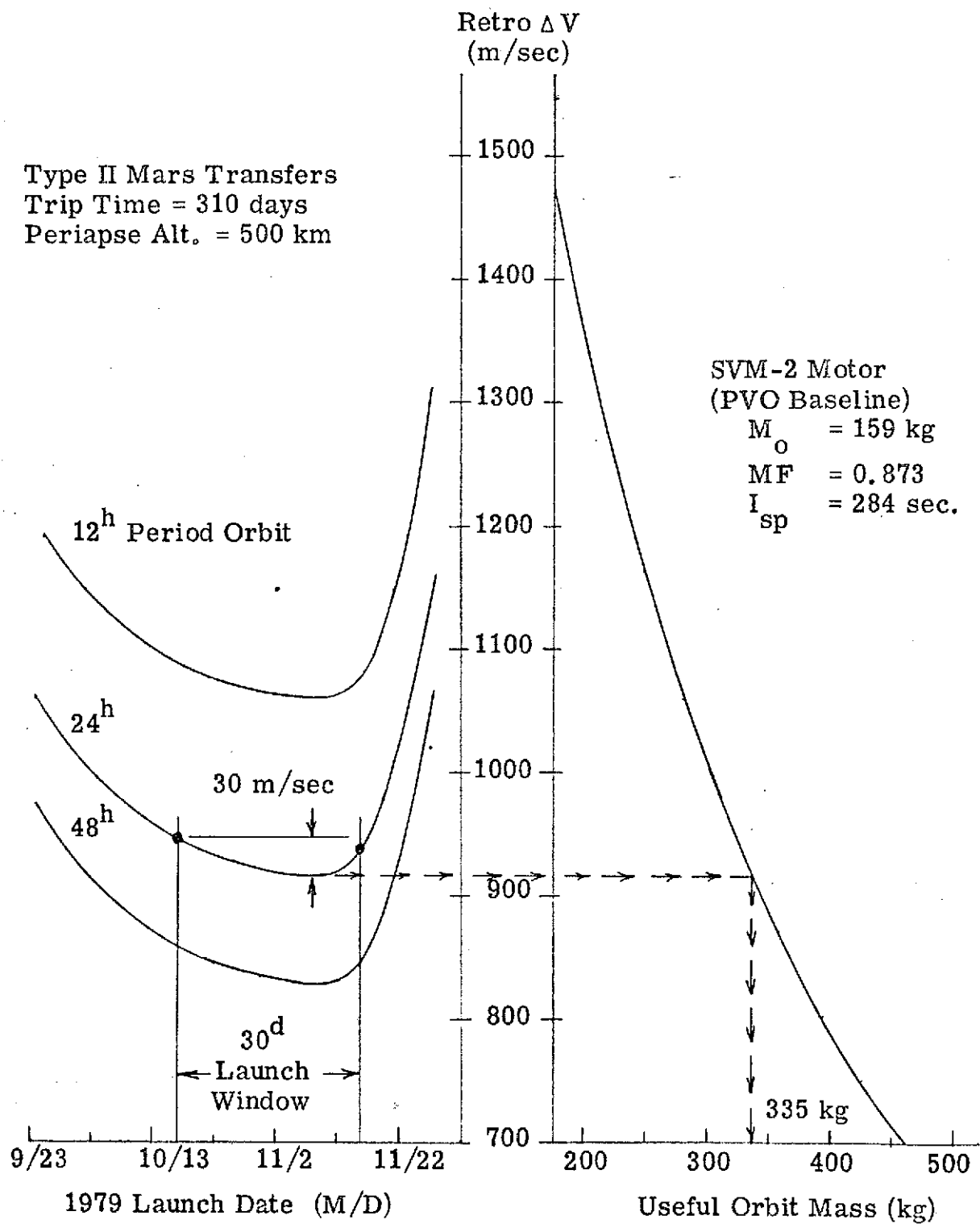
1979   PIONEER MARS ORBIT PAYLOAD CAPABILITY
AERONOMY/GEOLOGY MISSION

Aeronomy/Geology Payload Summary

Orbiter and science payload masses are summarized in the facing table for the Aeronomy/Geology mission (A). Two proposed spacecraft designs are presently under consideration by NASA for the PV-78 mission. As described above, both concepts use spin stabilization, one with the spin axis parallel (//) to the ecliptic plane with a fix-mounted earth pointing antenna, the other with the spin axis perpendicular ($\perp$) to the ecliptic plane with a despun earth pointing antenna. ARC/NASA provided SAI with a reference basic orbiter bus weight of 230 kg for either concept. Modifications were then examined for each concept to make it perform adequately at Mars. The // concept required an additional 23 kg of system modifications for Mission A. The 23 kg is divided into 17 kg for additional ACS propellant and tanks, 4 kg for solar arrays (entire system re-oriented) and 2 kg for an additional medium gain antenna. The $\perp$ concept required an additional 44 kg of system mods. These are divided into 17 kg for ACS propellant and tanks, and 27 kg for additional solar cells using the same cylindrical mounting. Both concepts with these modifications should provide a minimum power of 150 watts and communicate with earth at a minimum rate of 32 bps (at 2.44 AU) with the 26 m DSN receivers.
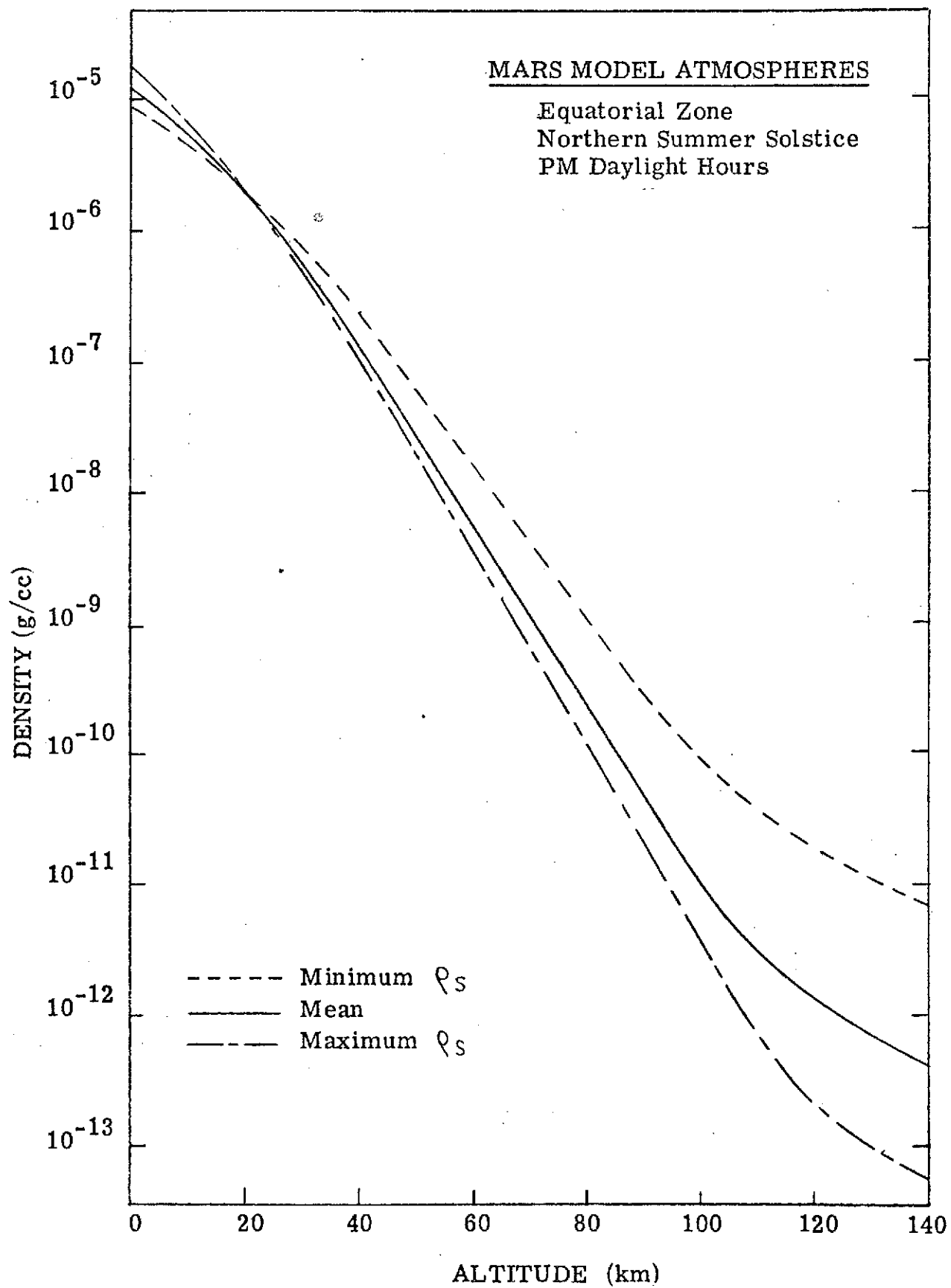
The maximum available useful orbit mass, as shown on the previous page, is 335 kg for Mission A. Hence, with modifications for Mars, the available science payload mass is 82 and 61 kg for the // and $\perp$ concepts, respectively. A science payload example for aeronomy/ geology objectives is also given on the facing page. The aeronomy instrumentation is similar to that proposed for PVO-78 with the single addition of the tri-axial accelerometer which also provides necessary engineering measurements for control of atmospheric drag perturbed orbits. The only geology type instrument included is the PVO-78 radar altimeter for which there is some design certainty. Depending on the selected PVO bus concept there is an additional 20-40 kg available for other geology instruments. Clearly better geology instruments can be added to this payload list as they become better understood. It should be noted that the limiting factors of the PVO science support capability for Mars are probably power and data rather than weight. To increase available power and communications data rate to a level comparable with the PVO-78 mission could require substantial bus modifications inconsistent with the ground rules for a 1979 Pioneer Mars mission.

AERONOMY/GEOLOGY PAYLOAD SUMMARY

- SPACECRAFT ORIENTATION TO ECLIPTIC   //   ⊥
- MASS SUMMARY

	//	⊥
Assumed PVO Bus Weight	230 kg	230 kg
Required Modifications	+23	+44
Modified Bus Weight	253	274
Maximum Useful Orbit Weight	335	335
Available Science Payload	≤ 82 kg	≤ 61 kg

- SCIENCE PAYLOAD EXAMPLE

	kg	watts	bps
Solar Wind/Photoelectron Analyzer	5	3	32
Electron/Ion Temperature	2	2.5	36
Magnetometer	3.5	3	25
Neutral Mass Spectrometer	5.5	10	100
Ion Mass Spectrometer	1.5	2	100
Tri-Axial Accelerometer	10	19	200
Radar Altimeter	9	11	125
Dual-Frequency Occultation	3	10	–
Totals	39.5	60.5	618

- SCIENCE SUPPORT CAPABILITY

With // Concept	82	≈ 50	10^6/orb
With ⊥ Concept	61	≈ 50	10^6/orb

Mars Model Atmospheres

The three atmospheres described by the density-altitude profiles are taken from the Viking Project engineering models. Note that the minimum and maximum surface density curves cross over at about 20 km. Hence, the minimum ρ s is actually the high density model in the upper atmosphere which is the region of interest for the aeronomy orbiter mission. The mean atmosphere profile is used as the nominal design condition for the present analysis. However, the effect of each of the bounding models will be described for the aeronomy orbiter and entry probe (penetrometer) missions. The uncertainty represented by the bounding models is probably conservative even at the present time, and will certainly be decreased after the Viking 75 mission.

MARS MODEL ATMOSPHERES

Equatorial Zone
Northern Summer Solstice
PM Daylight Hours

- - - - - Minimum ρ_S
———— Mean
— - — Maximum ρ_S

DENSITY (g/cc)

ALTITUDE (km)

Orbiter Size and Characteristics

The effective drag diameter is shown as a function of ballistic coefficient $B = m/C_D A$, where m is the spacecraft mass and $A = \pi d^2/4$. In the subsequent analysis of orbit perturbations due to drag, a nominal ballistic coefficient of 50 kg/m^2 is assumed. The figure indicates an effective drag diameter of about 2 meters for a 300 kg spacecraft. The actual spacecraft diameter could be as large as 2.7 meters to allow for sufficient solar cell area to meet the power requirements at Mars heliocentric distance. Since the orbiter configuration and orientation relative to the atmosphere are not well-defined for this preliminary study, the actual ballistic coefficient is uncertain, but very likely lies in the range 25 - 75 kg/m^2. The effect of this variation about the nominal value assumed will be described.

$$\underline{C_D = 2}$$

Spacecraft mass (kg)

400

300

200
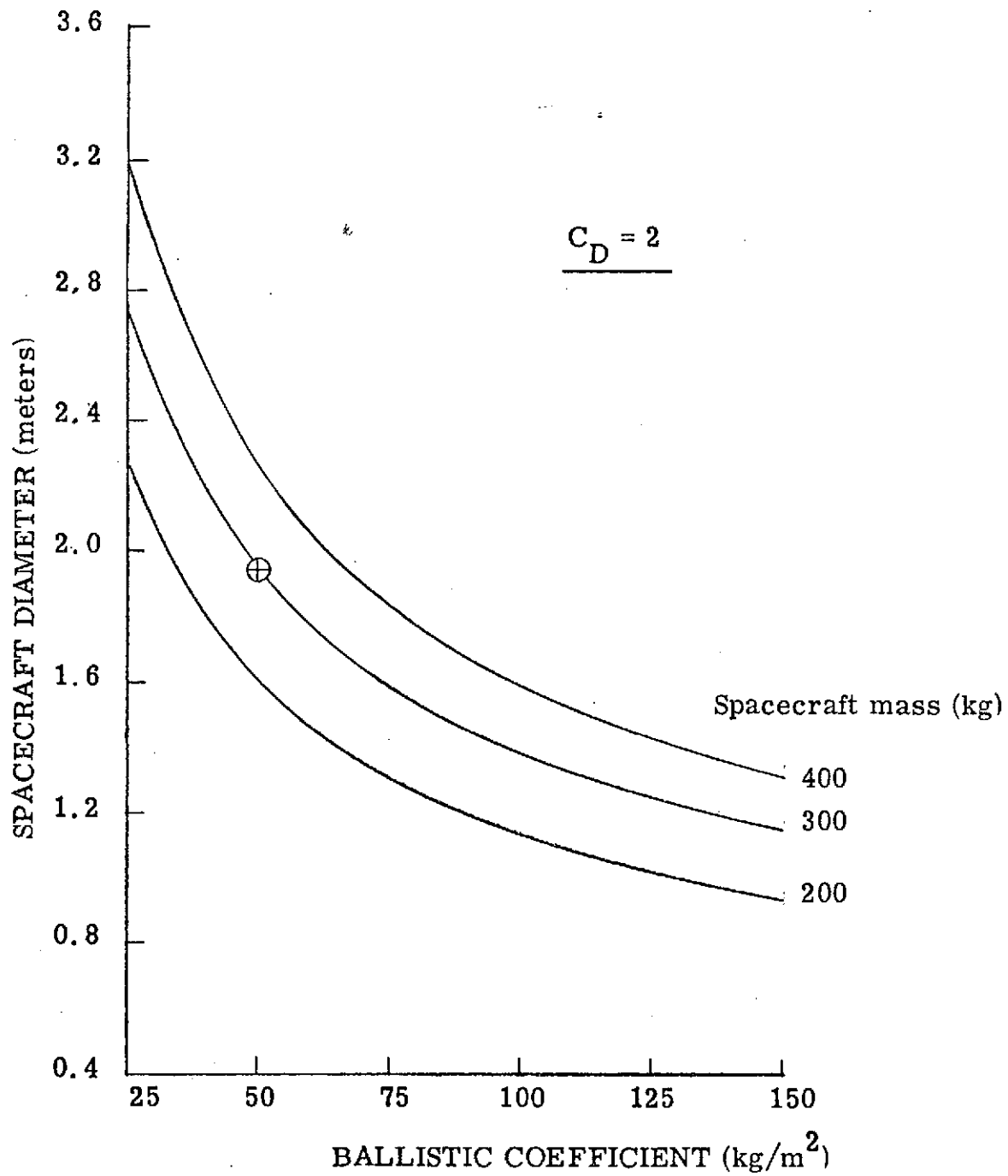
SPACECRAFT SIZE AND DRAG CHARACTERISTICS FOR
MARS AERONOMY/GEOLOGY ORBITER MISSION

Nominal Conditions for Aeronomy/Geology Orbiter

The nominal initial orbit is inclined 45° to the Mars equatorial plane, has an orbital period of 24 hours and a periapse altitude of 100 km. Initial apoapse altitude and eccentricity are, respectively, 33,277 km and 0.82604. Longitude of the node is measured in a local solar time reference at the initial epoch, i.e., the reference X -axis is the projection on the equatorial plane of the anti-solar direction at initial epoch. Periapse is located 282.9^O from the ascending node measured in the orbit plane. These orientation angles determine an initial periapse location of 43.6^O south latitude and 218.5^O longitude (mid-afternoon).

Subsequent evolution of the orbit size and orientation is determined by the influence of three perturbing effects, namely; 1) atmospheric drag, 2) Mars oblateness, and 3) solar gravitation. These perturbations can be highly inter-related and result in a rather complicated orbit evolution. Atmospheric drag acts principally to reduce apoapse distance (also period and eccentricity) with only very small changes in periapse altitude; the effect depends upon the atmospheric density in the vicinity of periapse passage and the ballistic coefficient. The major effect of oblateness is precession of the orbit via regression of the node and advancement (or regression) of periapse argument; this effect depends on orbit size and is thus coupled to drag perturbations. The main influence of solar gravitation is to increase or decrease periapse altitude; this effect depends on the position of the sun relative to the orbit's major axis and the size of the orbit, and hence is coupled to both drag and oblateness perturbations.

NOMINAL CONDITIONS FOR AERONOMY/GEOLOGY ORBITER

- **INITIAL ORBIT**

EPOCH	7 SEPT. 1980
ORBIT PERIOD	24 hours
PERIAPSE ALTITUDE	100 km
INCLINATION	45°
LONGITUDE OF NODE	290.5^O (Local time)
ARGUMENT OF PERIAPSE	282.9^O

- **SECULAR PERTURBATIONS**

1) ATMOSPHERIC DRAG

DENSITY MODEL	MEAN
BALLISTIC COEFFICIENT	50 kg/m^2

2) MARS OBLATENESS (J_2) 1.96×10^{-3}

3) SOLAR GRAVITATION

Periapse Location Coverage

The achievement of wide planetary coverage is an important scientific objective for both aeronomy and geology aspects of the mission. The nature of this coverage is indicated by the facing illustration (and subsequent figures) which shows the "free" or uncontrolled motion of periapse location in a grid of latitude and longitude (local time) coordinates. Latitude motion is caused by the changing argument of periapse due to oblateness perturbations and is, of course, bounded in magnitude by the orbit inclination which remains nearly constant throughout orbit evolution. Longitude motion is a function of the apparent solar motion in the Mars ecliptic plane as well as changes in nodal longitude and argument of periapse caused by oblateness. A general characteristic is that the rate of latitude motion increases as the orbit decays and eventually predominates over longitudinal motion.

Orbit evolution can be highly sensitive to inclination as indicated by the figure. For $i = 30^{\circ}$, the scenario that developed was as follows: solar gravitation acted initially to reduce periapse altitude at a high rate (0.3 km/orbit) from which free recovery became impossible due to the increasing drag perturbation at lower altitudes. The orbit decayed to end-of-life entry conditions in only 135 days. In the case of $i = 60^{\circ}$, solar gravitation resulted in an initial dip to only 98 km and then acted to increase periapse altitude to over 150 km (after 300 orbits) thereby ensuring a long lifetime. The low rate of latitudinal motion is a result of the inclination being close to the critical value of 63.5°.

The nominal selection of 45° inclination seemed a suitable tradeoff between rate of longitude motion, latitude coverage and orbit lifetime. It is desirable to sample atmospheric properties over the full extent of local time conditions with latitude coverage being a secondary objective. Periapse location in the nominal orbit starts in mid-afternoon near 45° southern latitude and traverses a northward retrograde path. After 500 orbits (350 days) periapse is located near 30° N latitude and midnight local time. Sufficient orbit lifetime remains at this point to continue through maximum northern latitude and then southward towards the dusk terminator. The decaying aeronomy orbit is probably not best suited for geology measurements which may prefer a more stable orbit configuration. This could be achieved on an intermittent basis by raising the periapse altitude and holding a nearly drag-free orbit for a duration of several weeks. A modest ΔV expenditure of 25-50 m/sec would be required for such maneuvers.

MIDNIGHT   DAWN   NOON   DUSK
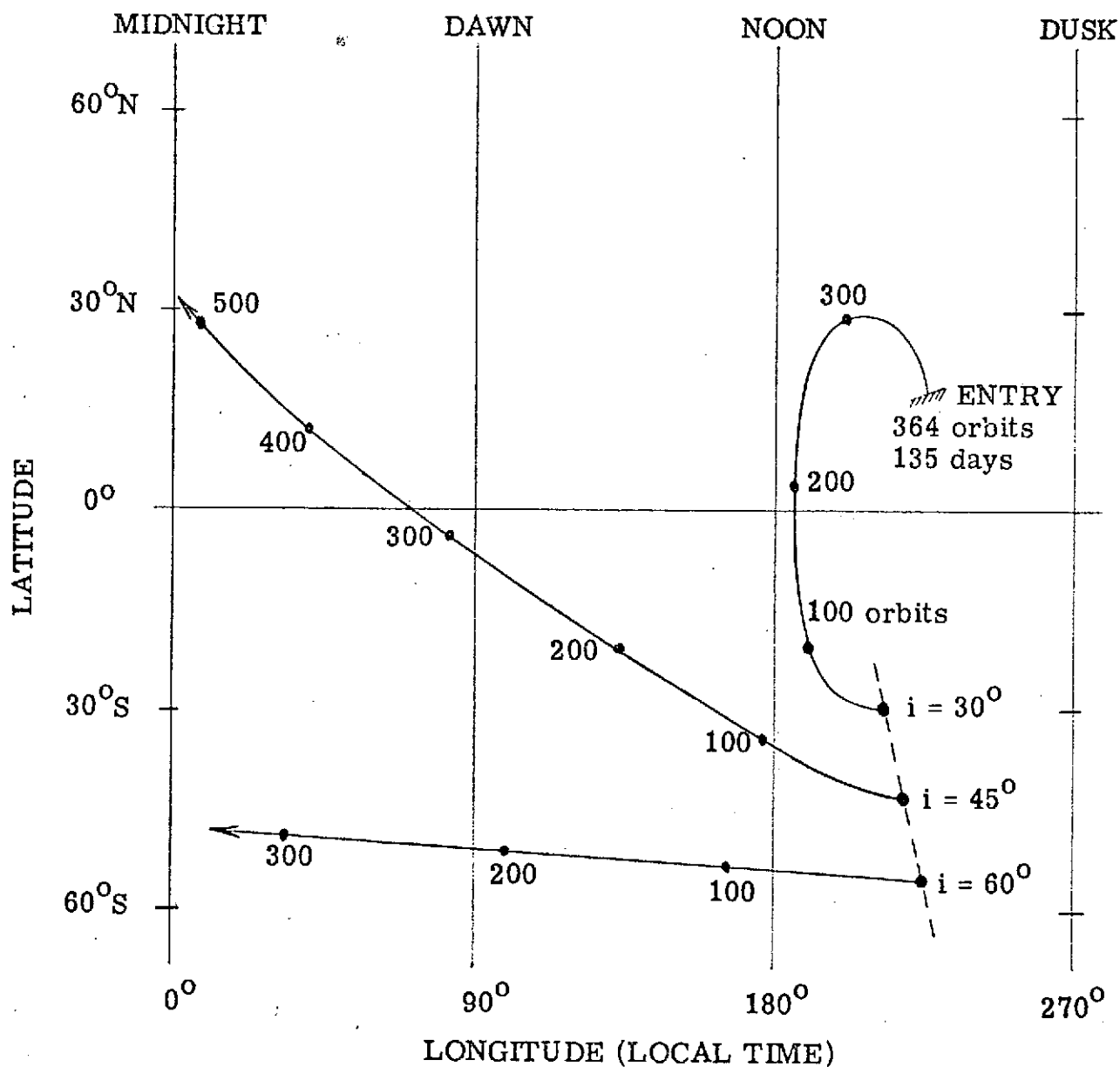
EFFECT OF INCLINATION ON PERIAPSE LOCATION COVERAGE

$(Po = 24 \text{ hrs.}, \ h_{po} = 100 \text{ km}, \ B = 50 \text{ kg/m}^2)$

Orbit Period and Periapse Altitude History

The facing illustrations present additional information about the nominal orbit evolution. Solar gravitational perturbations cause an initial decrease in periapse altitude to about 93 km after 60 orbits. This effect is then reversed and periapse increases to a maximum of 117 km after 300 orbits. The periodic nature of this perturbation is due to the apparent motion of the Sun around Mars relative to the periapse direction, the period being approximately one-half of a Mars year. The amplitude of the periapse altitude perturbation is damped with time as the orbit size decays due to atmospheric drag. This decay is indicated by the decreasing orbital period which, at the end of 500 orbits, has been reduced to 14.3 hours. Coupling between solar gravity and drag perturbations is clearly evidenced by correlating the rate of change of orbit period with periapse altitude.

PERIOD & PERIAPSE ALTITUDE HISTORY
$i = 45°$, $B = 50$ kg/m^2
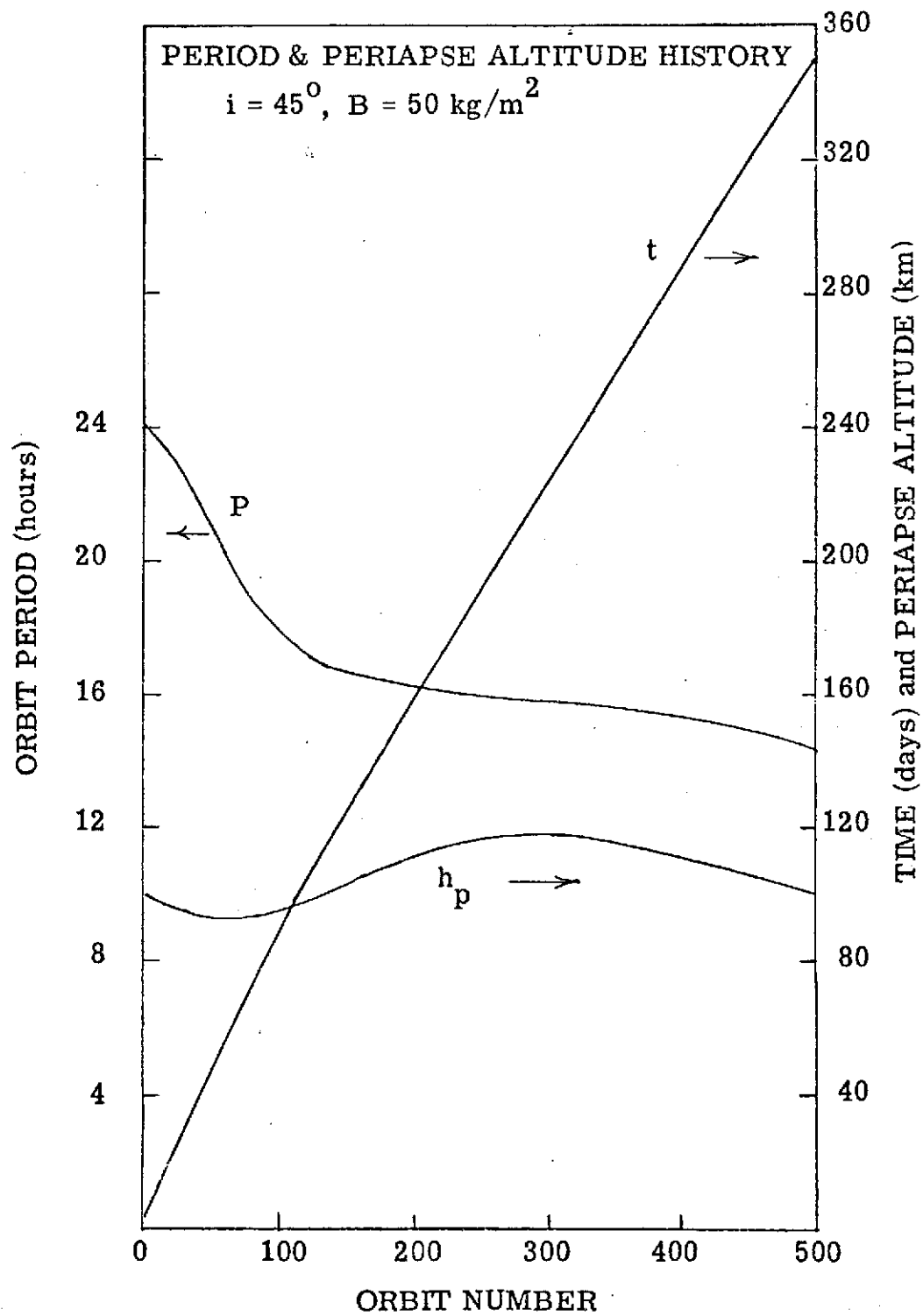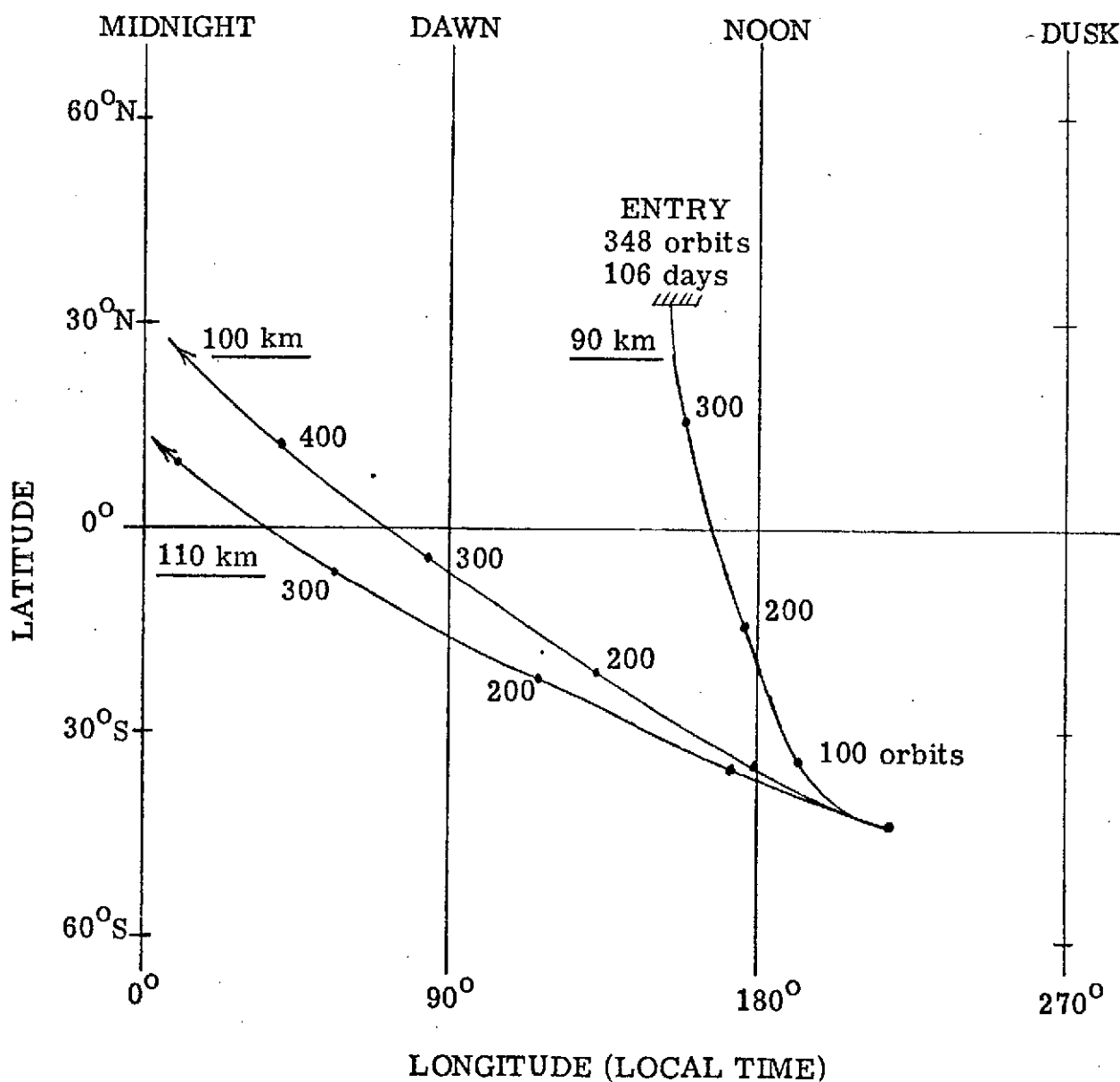
Supporting Parametric Data

Graphical results presented on pages 44 through 51 describe the effects of variations in nominal parameters on orbit evolution. Four parameter variations are considered:

    1) Initial periapse altitude
    2) Ballistic coefficient
    3) Atmosphere density model
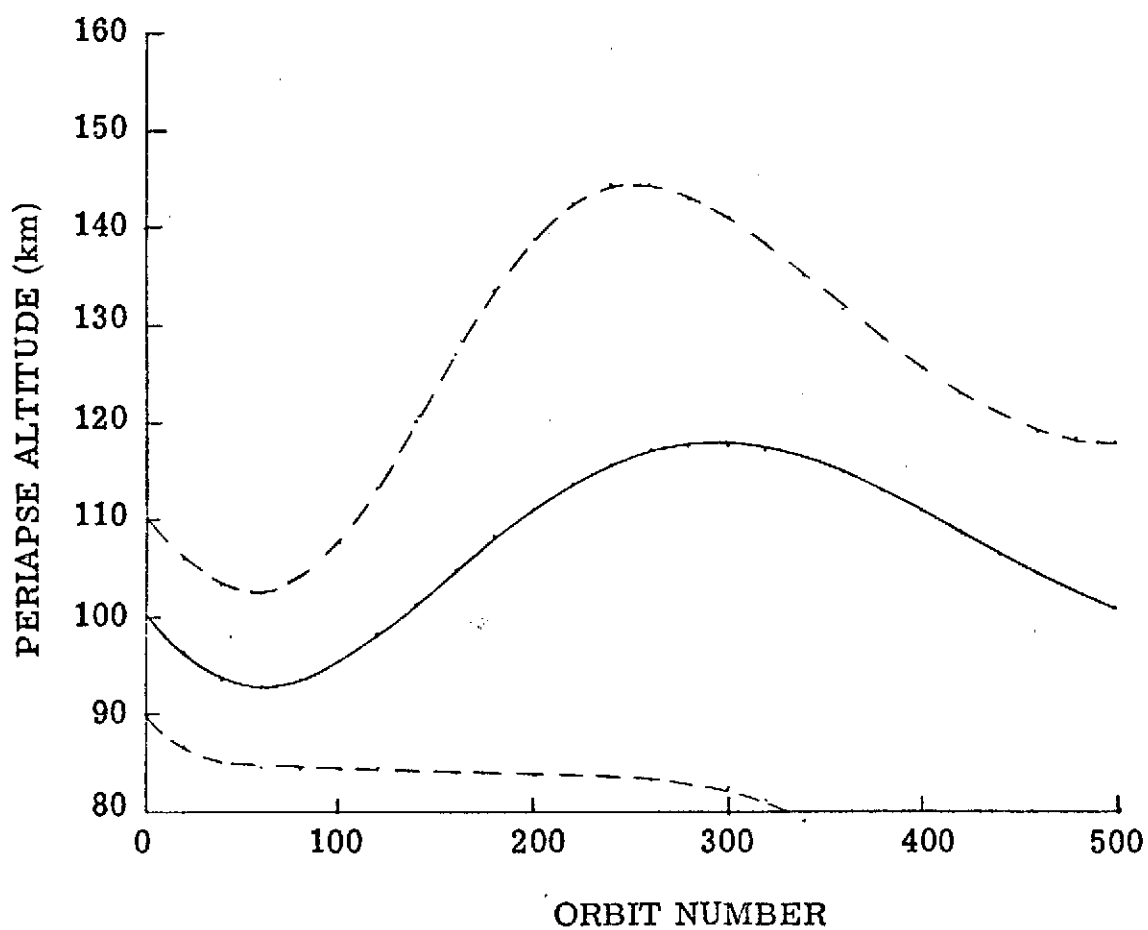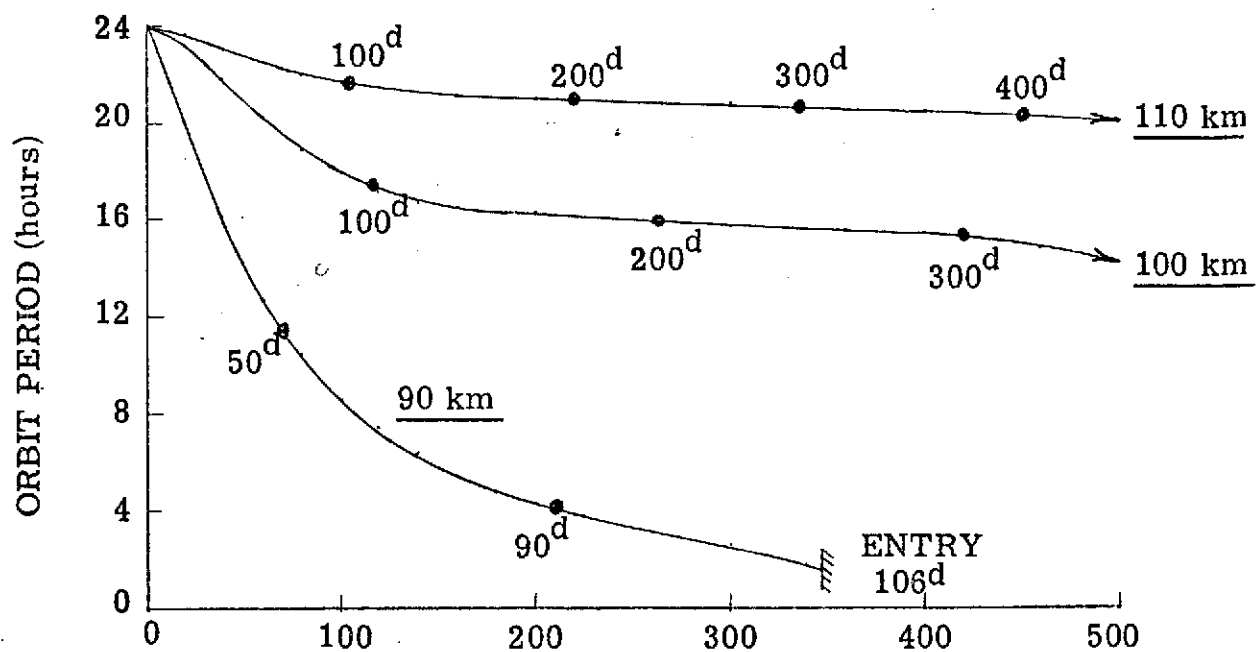   4) "Worst" case adaptive policy example

The data for each effect is presented on facing pages with the left-hand side showing the periapse location coverage and the right-hand side showing the orbit period and periapse altitude histories. The results may be summarized as follows:
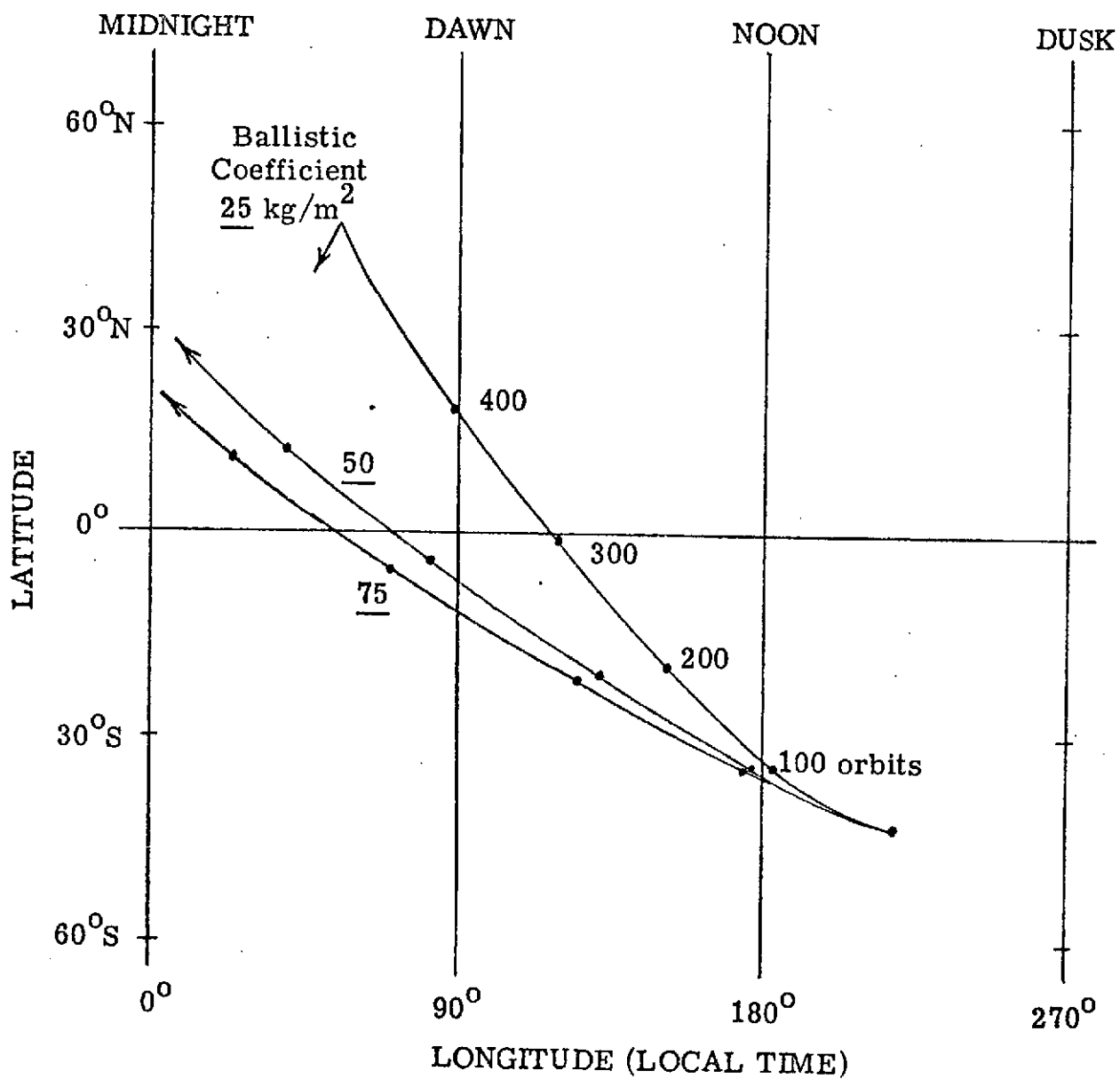
1. Given the mean atmosphere model, initial periapse altitude should not be placed lower than about 100 km in order to ensure adequate orbit lifetime and coverage.

2. A reduction in ballistic coefficient to 25 kg/m^2 has a significant effect on orbit evolution and results in approximately one-half of the nominal orbit lifetime. This can be compensated for by raising the initial periapse altitude to about 110 km in the mean atmosphere.

3. Operation at 100 km altitudes would be catastrophic in terms of orbit lifetime if the high density upper atmosphere (Min ρ s) model were encountered. In this case the altitude should be increased to 140-150 km in order to obtain an equivalent (nominal) orbit evolution profile.

4. If the density uncertainty were still significant at the time of mission/spacecraft design, then an adaptive operational policy would seem most appropriate. Periapse altitude should be set high enough to ensure a low orbit decay rate until the density and orbiter drag characteristics are better defined. On-board propulsion would allow the altitude to be adjusted as appropriate to the environment found. An example of such a policy is shown on pages 50 and 51 where the conditions encountered are a high density upper atmosphere and a lower than nominal ballistic coefficient. Periapse altitude is adapted to the region near 150 km, however, six short-term sampling excursions to 110 km are executed for a total ΔV cost of 54 m/sec.

MIDNIGHT   DAWN   NOON   DUSK

ENTRY
348 orbits
106 days

90 km

100 km

400

110 km

300

300

300

200

200

200

100 orbits

LONGITUDE (LOCAL TIME)

EFFECT OF INITIAL PERIAPSE ALTITUDE
ON PERIAPSE LOCATION COVERAGE
$(i = 45^{\circ}, \ Po = 24 \ hrs., \ B = 50 \ kg/m^2)$

MIDNIGHT   DAWN   NOON   DUSK

Ballistic
Coefficient
$\underline{25}$ kg/m^2

400

$\underline{50}$

300

$\underline{75}$

200

100 orbits

LATITUDE

60°N

30°N

0°

30°S

60°S

0° 90° 180° 270°

LONGITUDE (LOCAL TIME)

EFFECT OF BALLISTIC COEFFICIENT
ON PERIAPSE LOCATION COVERAGE

$(i = 45^{\circ}$, Po = 24 hrs., $h_{po} = 100$ km)

MIDNIGHT   DAWN   NOON   DUSK

LATITUDE

LONGITUDE (LOCAL TIME)
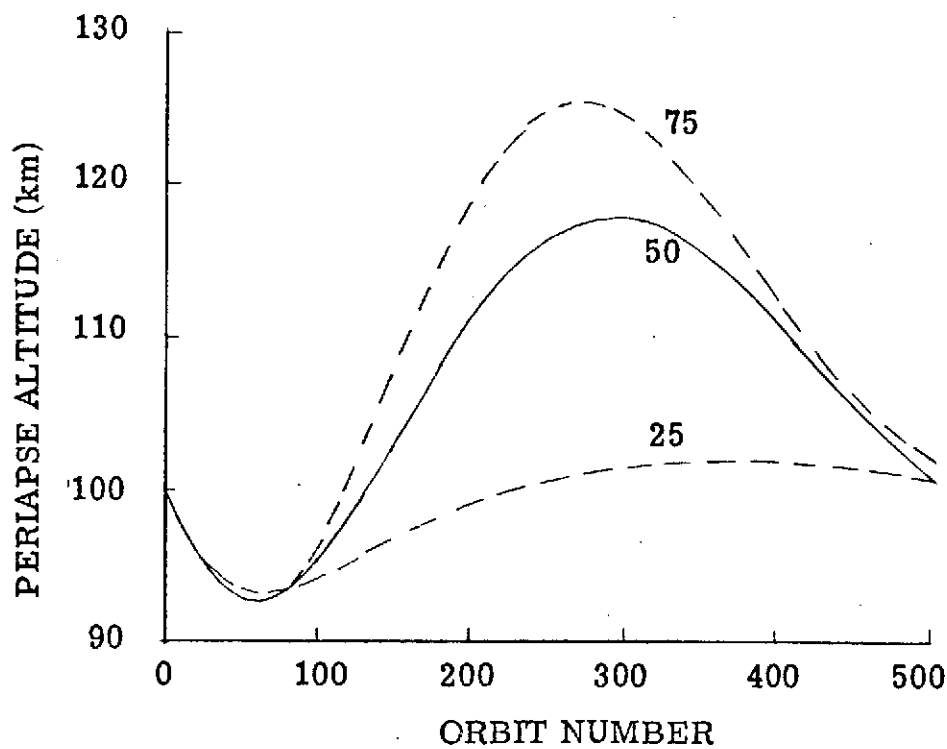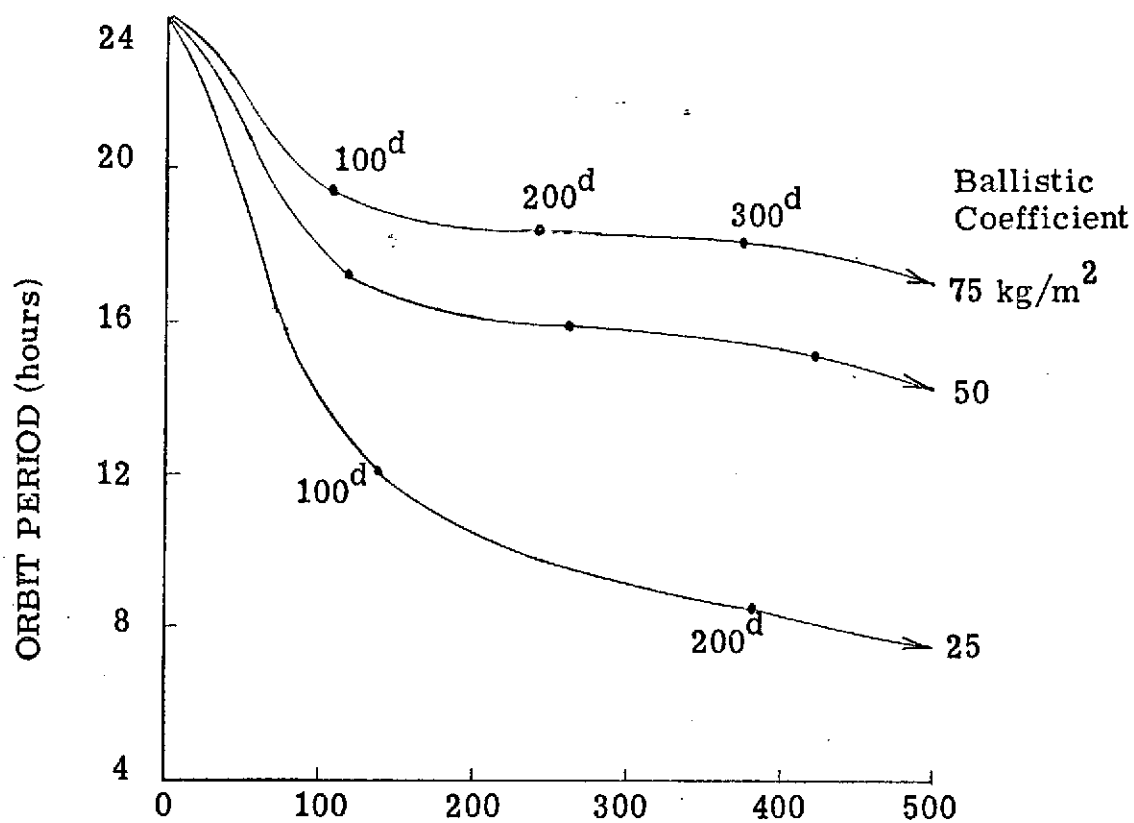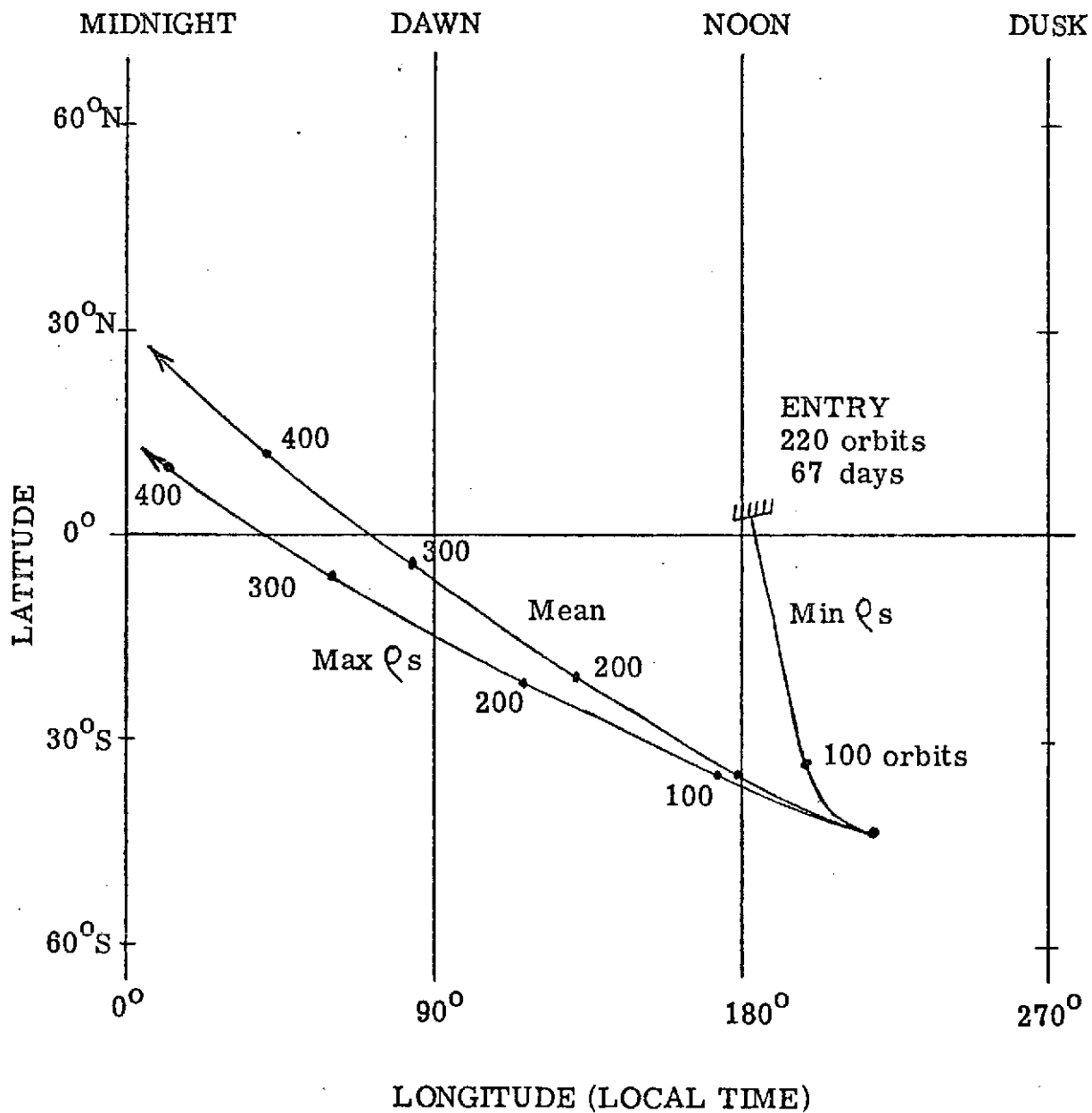
EFFECT OF ATMOSPHERE DENSITY MODEL
ON PERIAPSE LOCATION COVERAGE

$(i = 45^O,\ P_o = 24\ hrs.,\ h_{po} = 100\ km,\ B = 50\ kg/m^2)$

Top figure: ORBIT PERIOD (hours) vs ORBIT NUMBER. Curves labeled Max ϱ s, Mean, Min ϱ s, with day markers 100ᵈ, 200ᵈ, 300ᵈ, 400ᵈ, and ENTRY (67ᵈ).

Bottom figure: PERIAPSE ALTITUDE (km) vs ORBIT NUMBER. Curves labeled Max ϱ s, Mean, Min ϱ s.
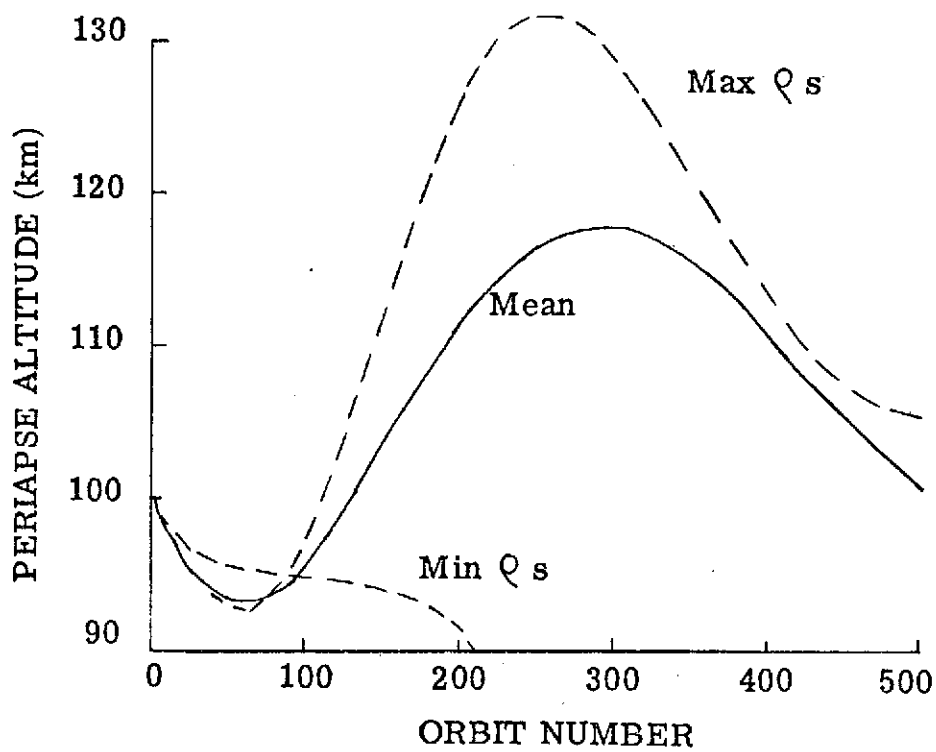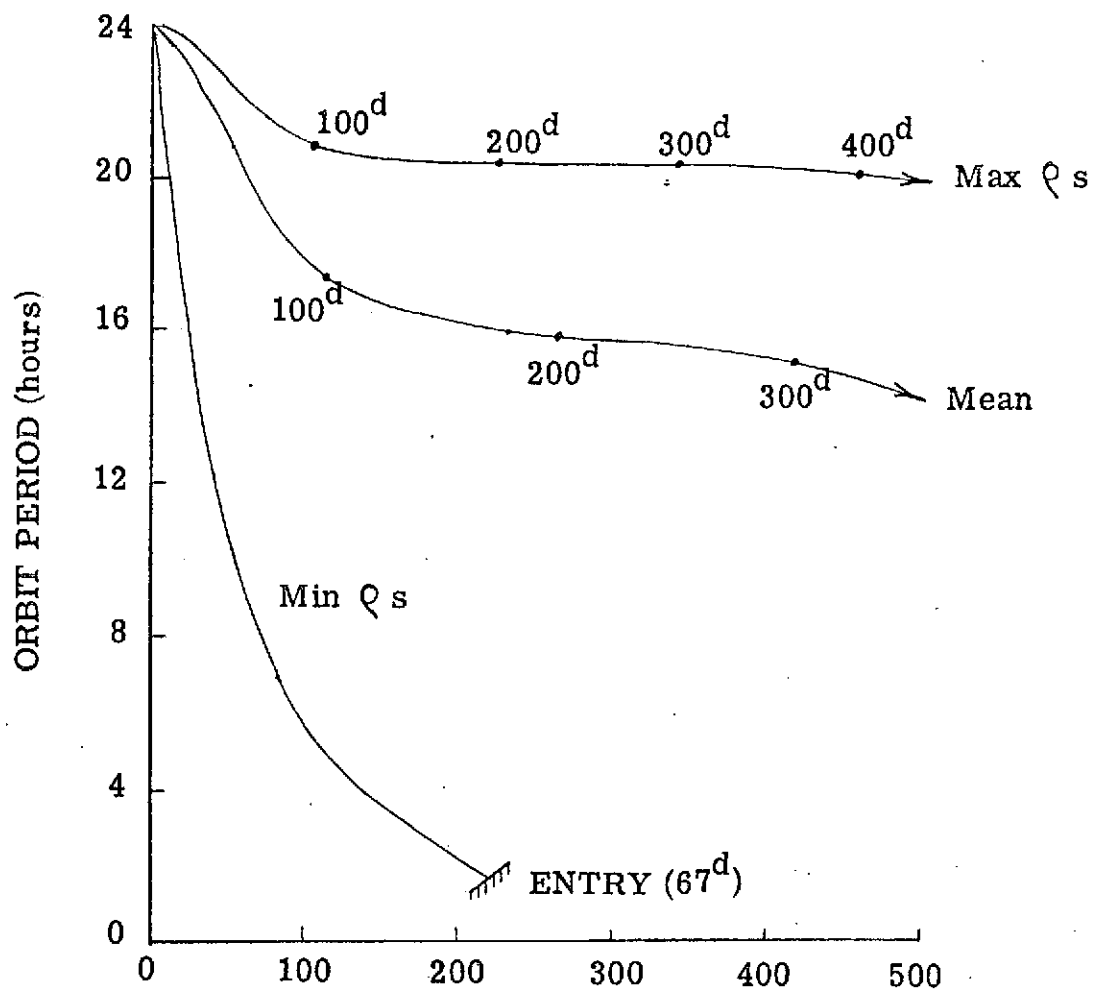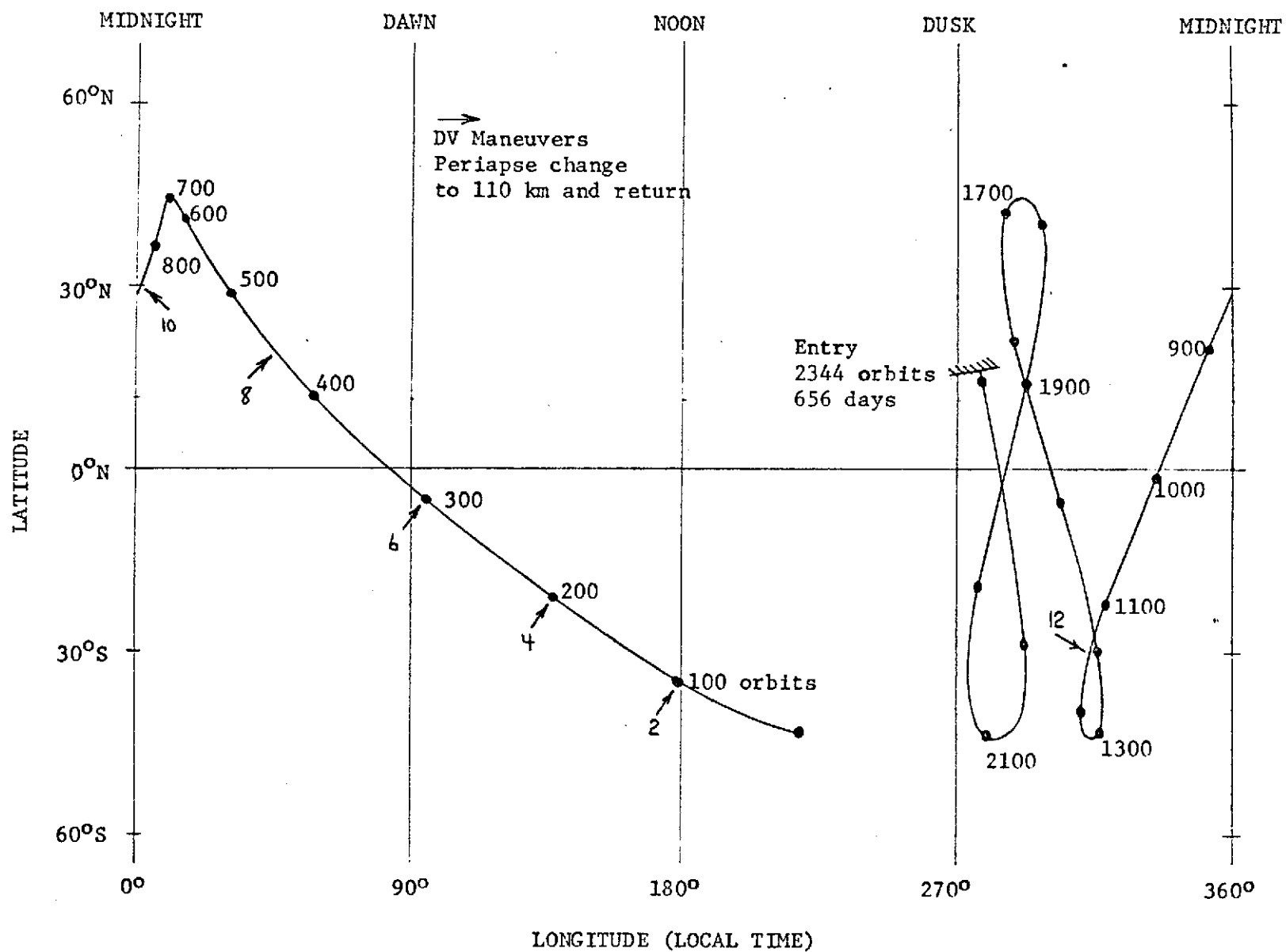
PERIAPSE LOCATION COVERAGE FOR "WORST" CASE ADAPTIVE POLICY

(High density upper atmosphere, P_o = 24 hrs, h_{po} = 150 km, B = 25 kg/m^2)

High Density Upper Atmosphere
$B = 25$ kg/m^2

ORBIT PERIOD (hours)

100^d
200^d
300^d
400^d
500^d
600^d
Entry
656^d

PERIAPSE ALTITUDE (km)

ΔV Maneuvers
54 m/s Total
1 or 2 Orbit Stay
at 110 km

ORBIT NUMBER
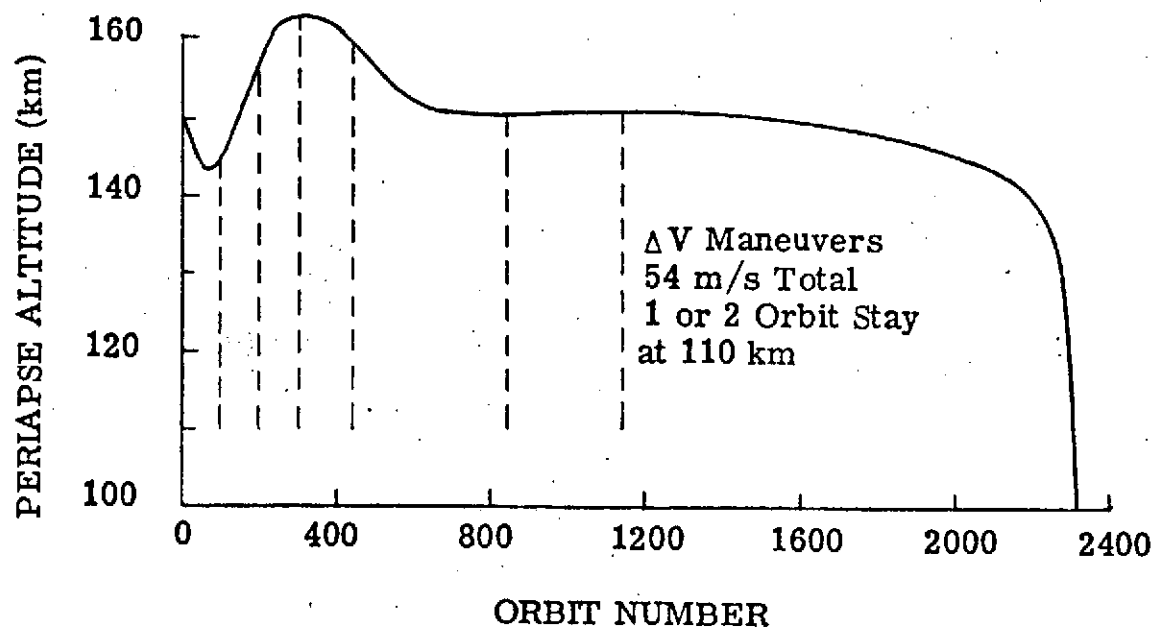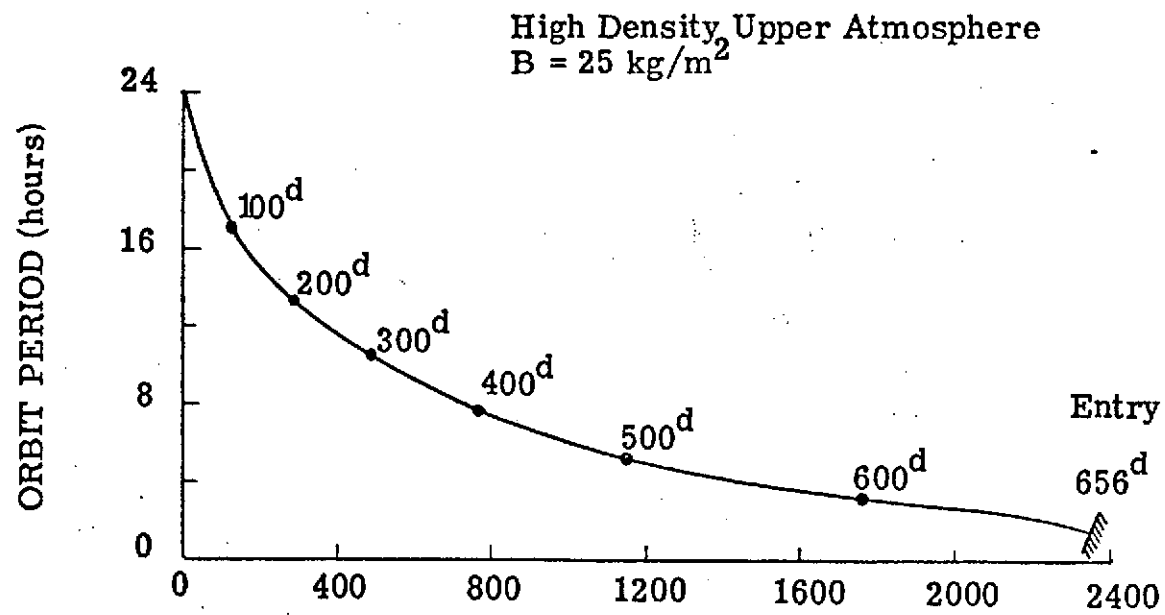
PERIAPSE ALTITUDE AND ORBIT PERIOD HISTORY
FOR "WORST" CASE ADAPTIVE POLICY

Heat Transfer in Mars Atmosphere

Convective heating rate is shown as a function of altitude in the mean Mars atmosphere for a typical velocity of 4.5 km/sec near periapse passage. The continuum and free molecule solutions are calculated from the following expressions:

$$\dot{Q}_{continuum} = C_H \, \rho^{0.5} V^3$$

$$C_H = \frac{3}{2\sqrt{2}} \sqrt{\frac{\eta}{V_s}} \, \frac{1}{\sqrt{R_n}}$$

$$\dot{Q}_{free \; molecule} = 0.4 \, (1/2 \, \rho \, V^3)$$

where ρ = atmospheric density, V = velocity, η = coefficient of viscosity, V_s = speed of sound, and R_n = spacecraft radius which is taken as 1 meter. It is assumed that 40% of the kinetic energy is transferred as heat when the spacecraft is in the free molecule region. An approximate indication of the heat transfer-altitude regimes is given based on a Knudsen number criterion. The transition regime includes "first collision" flow ($\approx$85 to 107 km). The aeronomy orbiter mission operation lies mainly in the free molecule region and partly in the first collision regime for lower altitude periapse passages. At 100 km the free molecule solution gives a heating rate of 162 joules / m^2/ sec = 0.014 Btu/ ft^2/sec.

Mean Atmosphere
V = 4500 m/sec

Continuum Solution

Free Molecule Solution

Continuum | Transition | Free Molecule

CONVECTIVE HEATING RATE ($\frac{joules}{m^2 \cdot sec}$)

ALTITUDE (km)

HEAT TRANSFER IN MARS ATMOSPHERE

53
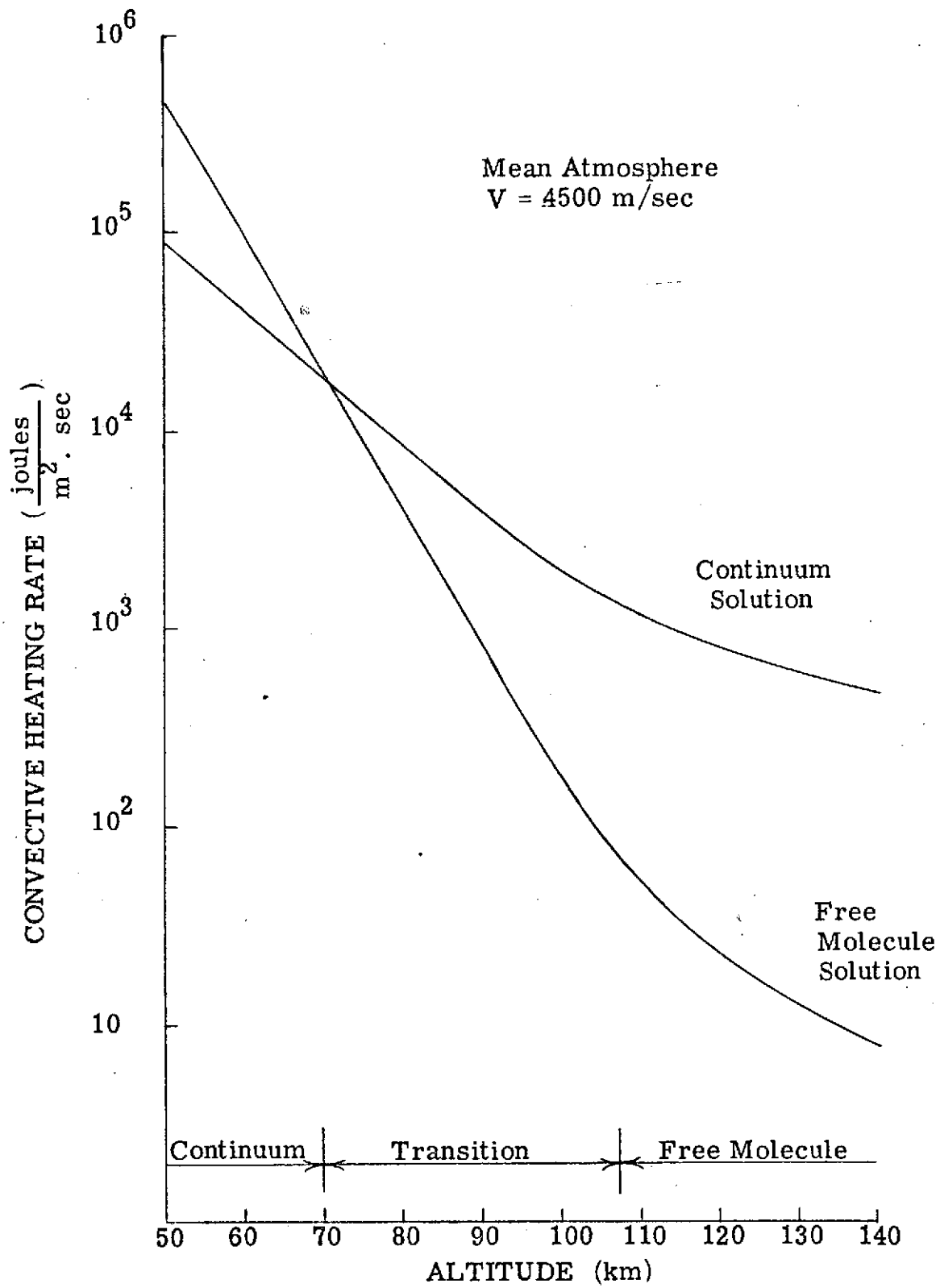
Equilibrium Temperature in Mars Atmosphere

The heat transfer data from the preceeding page is converted to radiative equilibrium temperature

$$T_r \, (^{\circ}k) = \left(\frac{\dot{Q}}{\sigma \, \mathcal{E}} \right)^{1/4}$$

where $\sigma = 5.669 \times 10^{-8}$ and $\mathcal{E}$ is the spacecraft surface emmisivity. This provides a measure of the severity of the heating problem. For typical emmisivity values (>0.5) it appears that operation above 100 km in the mean atmosphere will cause no thermal control problem due to atmospheric friction. Some thermal protection may be required for operating in the 90-100 km region, and excursions below 90 km should be avoided. Clearly a detailed heat transfer analysis should be undertaken, but this was beyond the scope of the present study.

Because of potential thermal and orbit decay problems at lower altitudes, it becomes obvious that the most critical phase of an aeronomy mission is the adjustment (lowering) of periapse altitude via ΔV maneuvers. Such maneuvers are executed near orbit apoapsis which leaves a maximum of 12 hours to identify and react to an "overburn" situation. The time interval is more likely to be only a few hours due to ΔV limitations. This problem was addressed in the Atmosphere Explorer - C project with the resultant design solution; on-board accelerometers to monitor the ΔV maneuver and execution of a programmed, automatic contingency maneuver to raise periapse if necessary.
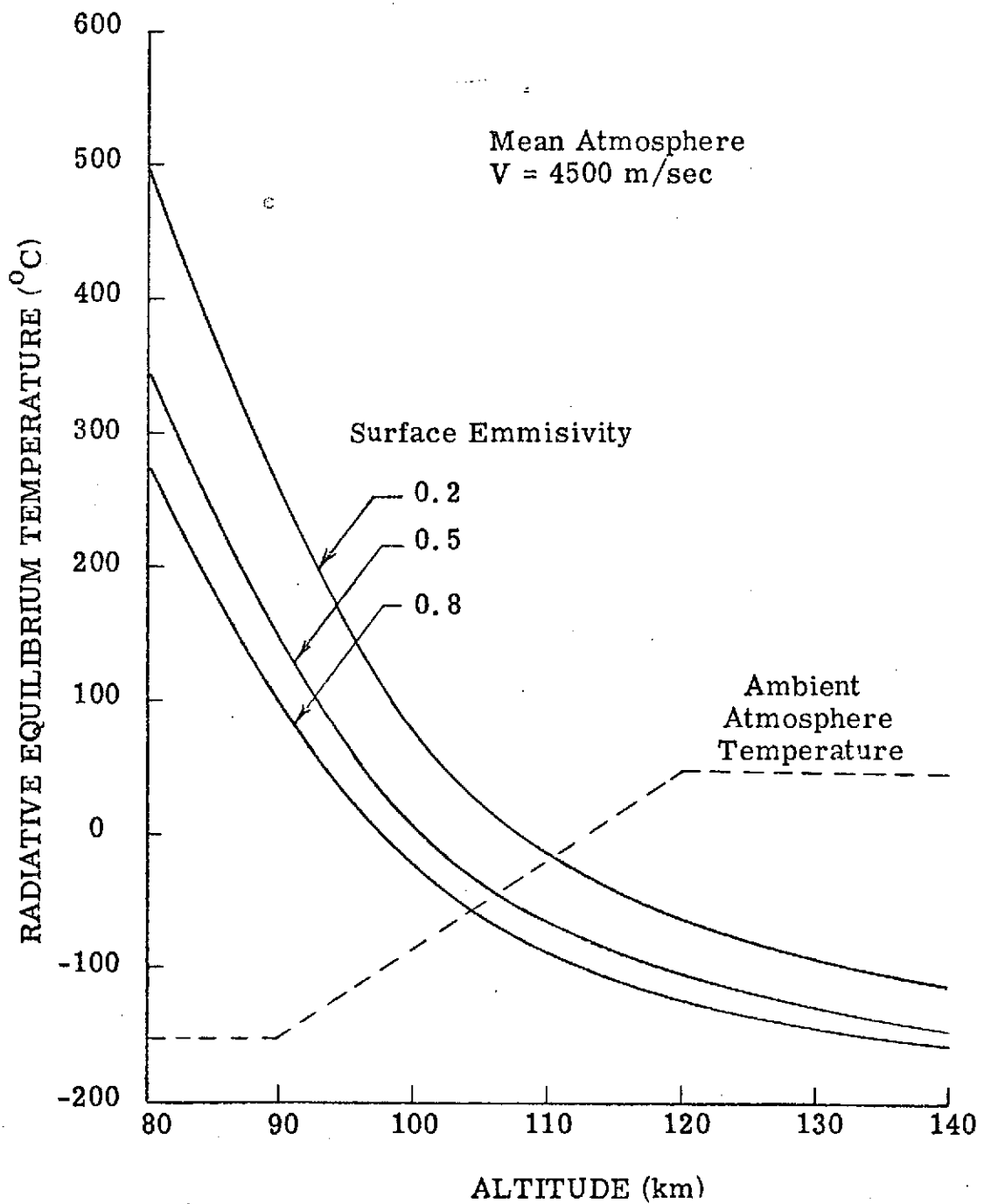
EQUILIBRIUM TEMPERATURE CONDITIONS IN MARS
ATMOSPHERE FOR ORBITER MISSION

55

SAI Aeronomy/Geology Mission Cost Estimate

A preliminary estimate of the cost of a single 1979 Pioneer Aeronomy/Geology Orbiter mission based on PVO-78 hardware is presented below. The cost for an additional PVO-78 flight article (less science) of $9.5M assumes an FY 76 order date. System modification costs have been minimized according to study ground rules and are estimated to cost less than $5M ($\Delta$ 1). Sterilization is assumed to follow Viking/75 procedures. Of the $7M allocated for sterilization, $2.5M is for additional hardware for qualification, $2M for redesign of sterilization failures, $1M for design of the bioshell, and the remaining $1.5M is for actual spacecraft testing and qualification. Mission operations costs (Δ 3) of $8.5M include launch, flight operations, guidance/navigation, and DSN services for a one Mars year orbit mission. Total cost (less science) with 10% contingency is about $31M. The cost of science, including hardware, team support and data analysis is, of course, dependent upon selection of the science payload. It might be pointed out that the science payload for PVO-78 has not yet been selected. Nonetheless, some preliminary bounds on the range of the science cost have been estimated. A completely inherited PVO-78 payload could cost as little as $5M. Adding different instruments and modifying some of the existing ones could easily escalate this cost to $10M with $15M being a probable upper bound. Hence, it appears reasonable to conclude a total mission cost for the Aeronomy/Geology Orbiter (Mission A) of $40-50M.

56

SAI AERONOMY/GEOLOGY MISSION COST ESTIMATE*

Cost Item	Concept	
	//	⊥
Procure 1 additional PVO-78 flight article	$9.5M	$9.5M
Δ1: Modify this flight article (Pwr, Comm, ACS)	3.0	4.1
Δ2: Sterilize spacecraft	7.0	7.0
Δ3: Add cost of mission operations	8.5	8.5
Spacecraft Net Totals	28.0	29.1
Contingency (10%)	2.8	2.9
Total (less science & data analysis)	30.8	32.0

*FY 1974 dollars

57

Subjects for Further Study (Aeronomy/Geology Orbiter)

A number of subjects identified for further study in this preliminary analysis of a 1979 Pioneer Mars Aeronomy/Geology Orbiter Mission are summarized on the facing page. First and foremost the science objectives achievable with this mission should be reviewed by a Pioneer Mars Science Advisory Group. Assuming this group endorses this mission, baseline science payload should be one of their key outputs. On the basis of that output, specific analyses of PVO-78 instrument modifications, new instrument development, and science payload cost should be performed.

Regarding spacecraft systems, a number of subjects are identified which will require at least a pre-Phase A systems study of the related questions of feasibility and cost. New problems for the PVO-78 design applied to this mission include sterilization, attitude and thermal control during ionosphere penetration, and communications-link capability at maximum ranges approaching 2.5 AU. Modifications to the power system also require more study. Finally, based on explicit solutions to these and other problems, the total mission cost should be re-estimated.

SUBJECTS FOR FURTHER STUDY

(Aeronomy/Geology Orbiter)

- Applicability of PVO-78 science instruments

- Design and cost of orbiter geology science

- PVO-78 design changes (power, communications and attitude control requirements)

- Response of PVO-78 hardware to sterilization

- Effective spacecraft ballistic coefficient

- Spacecraft attitude and thermal stability during atmospheric periapse passages

- Mission pointing requirements compatibility with PVO-78 design capability

- Refined mission cost estimate

Mission B

REMOTE SENSING ORBITER WITH PENETROMETERS

Rationale for a Pioneer Mars Orbiter with Surface Penetrometers

The second selected mission concept for a 1979 Pioneer Mars mission, Mission B, is a remote sensing orbiter capable of carrying 4-6 surface penetrometers. The orbiter would be maintained in a polar synchronous elliptical Mars orbit with a periapse altitude of 1000 km. This orbit satisfies planetary quarantine constraints for the orbiter without sterilization. From the vantage point of a polar orbit, the orbiter would sequentially deploy its surface penetrometers (sterilized) to selected impact sites over the operational lifetime of the mission, i.e. one Mars year. Each penetrometer would be capable of subsurface water detection and chemical analysis measurements, and have a deployed life of about one week. Between penetrometer activities (about 5% of active orbit lifetime) the orbiter could conduct remote sensing mapping studies of the Martian atmosphere and surface.

The unexpected complexity and evidences of water revealed on the surface of Mars by Mariner 9 make it extremely important that post-Viking missions have maximum flexibility in terms of latitude, longitude, altitude, and surface geology of landing sites. This is dramatically illustrated by considering possible outcomes of the Viking mission if the two Viking spacecraft, launched in 1975, land in two different sites and report slight differences in soil chemistry, as is probable. Scientific study of Mars will be thrown into a quandry, because it will be unclear whether the differences are correlated with albedo, elevation, etc. The first Viking site is likely to be somewhat different in albedo than the second; should environmental differences be correlated with albedo and possibly with rock type? The question is unanswerable on the basis of only two data points. Mission B, a Pioneer orbiter with surface penetrometers, offers the possibility of exploratory surface investigation of many (4-6) different geologic units with one mission. Furthermore, it's premise of lower cost offers the opportunity to expand the concept to a program of several missions, thus rapidly expanding the number of impact sites at a reasonable cost. Hence, it can be argued that this Pioneer mission concept retains its relevance to global Mars exploration regardless of the outcome of Viking/75 explorations.

There are a number of reasons for selecting a surface penetrometer as opposed to other probe designs (particularily rough landers) for multi-site surface science with a Pioneer bus. One important favorable characteristic of the penetrometer is an impressive history of earth experience with the concept. Not only is engineering data

available from literally thousands of tests[5], but the concept*, developed principally by Sandia Laboratories, has been applied to exploration in polar ice, deserts, lava flows, and other environments relative to Mars.

A second favorable aspect of the penetrometer is that it is especially suited to post-Viking/75 needs. Pioneer-class surface landers could not hope to compete with soil analyses, biochemical studies, or surface imaging carried out by Viking landers. But multiple subsurface investigations are suggested by Mariner and earth-based observations and are not presently planned with Viking landers. The hypothesis that the subsurface may be a major reservoir of volatiles can be directly tested by a penetrometer capable of several meters penetration. Simple tests for subsurface water and soil chemistry can be carried out in situ.

A third advantage of the penetrometer is its ability to gather data on the strength, coherence and decimeter-scale stratigraphy of Martian surface layers. A series of probes penetrating at 4-6 landing sites could begin to give information as to whether Mars is covered by a mega-regolith of loose material, or whether hard-frozen, ice-bonded layers exist near the surface or whether rock outcrops are common. In the Tharsis volcanic area, for example, we see the outlines of lava flows in spite of the fact that the surface colors and albedos are typical of the supposed Martian dust. Are these flows mantled with only a few meters of dust?

A fourth area of special applicability is polar investigation. Present data suggest that the Martian caps may have a CO_2-H_2O layering with CO_2 forming preferentially on the surface, buffering the atmosphere, and accounting for some earth-based spectra. Added to this is the possibility of dust layers mixed with the polar snows and forming a historical record of dust-storm episodes. A penetrating probe could, by accelerometer measures, detect significant strata in the polar snows, and could analyze elemental chemical composition of these snows in the upper meters of the subsurface.

The penetrometers are undoubtedly the key feature of Mission B. A comparable role of remote sensing experiments on the orbiter is much less certain. Imaging appears ruled out by limited spacecraft power and communications. Radar altimetry, to be meaningful, is probably too complex at the large altitudes of the bus' orbit (>1000 km). IR and UV spectroscopy could be redundant experiments, this time on a spinning spacecraft. A repeat performance for them in a different Mars season requires further study for justification. However, despite the lack of a leading experiment for the orbiter, its polar orbit platform should be attractive for orbital surface measurements which specifically support the objectives of the penetrometers.

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*Known as Terradynamic Probes

Penetrometer Science Payload

A suggested package of instruments for each penetrometer is presented below. These experiments encompass the objectives of surface penetrability, subsurface layering, soil water content, atmospheric humidity, and elemental soil chemistry. The first experiment, an axial high-g accelerometer, measures the deceleration history of impact. The instrument has a range to 10,000 g's with a resonant frequency of 20 kHz. 8000 bits of data are generated by the instrument in approximately 15 msec. These data would be stored for subsequent transmission to the orbiter at a much slower rate. This instrument has been flown repeatedly on Sandia Terradynamic Probes to demonstrate its application to exploration of subsoil conditions at inaccessible sites.[6]

The next three instruments relate to the measurement of water in the Mars soil and atmosphere. The moisture chamber would be designed to analyze H_2O and CO_2 content of a small subsurface soil sample. The sample (100-200 mgm) would be collected through a sidewall port of the penetrometer by a non-reactive explosive charge (metallic azides) into a subsequently sealed chamber. The sample would be weighed and heated with pressure and temperature closely monitored to reproduce a composite vapor pressure curve of released volatiles. Maximum sample temperature is expected to be about 200°C. The dewpoint of the released vapor could also be monitored as the chamber subsequently cools. A total of approximately 3600 bits of data are anticipated from this once only measurement which would be performed shortly after impact. The chief drawback of this experiment is that it isn't developed. This naturally implies a number of possible problems including soil acquisition, contamination and measurement error. The inaccuracy of $\pm 50\%$ shown below results from uncertainties in weighing the sample. The hygrometer experiment, consisting of an aluminum oxide element and temperature sensor, would measure atmospheric water vapor at the surface of the impact site and at the depth of penetration. Instrument characteristics are given below. This experiment would be operated six times a Mars day for 10 second periods. A total data load of 1000 bits/day is anticipated. The hygrometer instruments are developed but have not been tested on a penetrometer.

The last experiment shown is a miniature alpha, proton backscatter instrument. Similar to the Surveyor α-backscatter experiment, this instrument is smaller and more accurate than its predecessor. The detectors have been successfully tested to over 1800 g's. A detailed description of the experiment was presented at a recent PPPI Meeting by T. Economou and A. Turkevich.[7] For the penetrometers, two sets of detectors would examine two samples through sidewall ports over a period of 8 days, generating 5100 bits of data per day. The instrument is presently in a breadboard stage and would require further development for a penetrometer mission.

PENETROMETER SCIENCE PAYLOAD

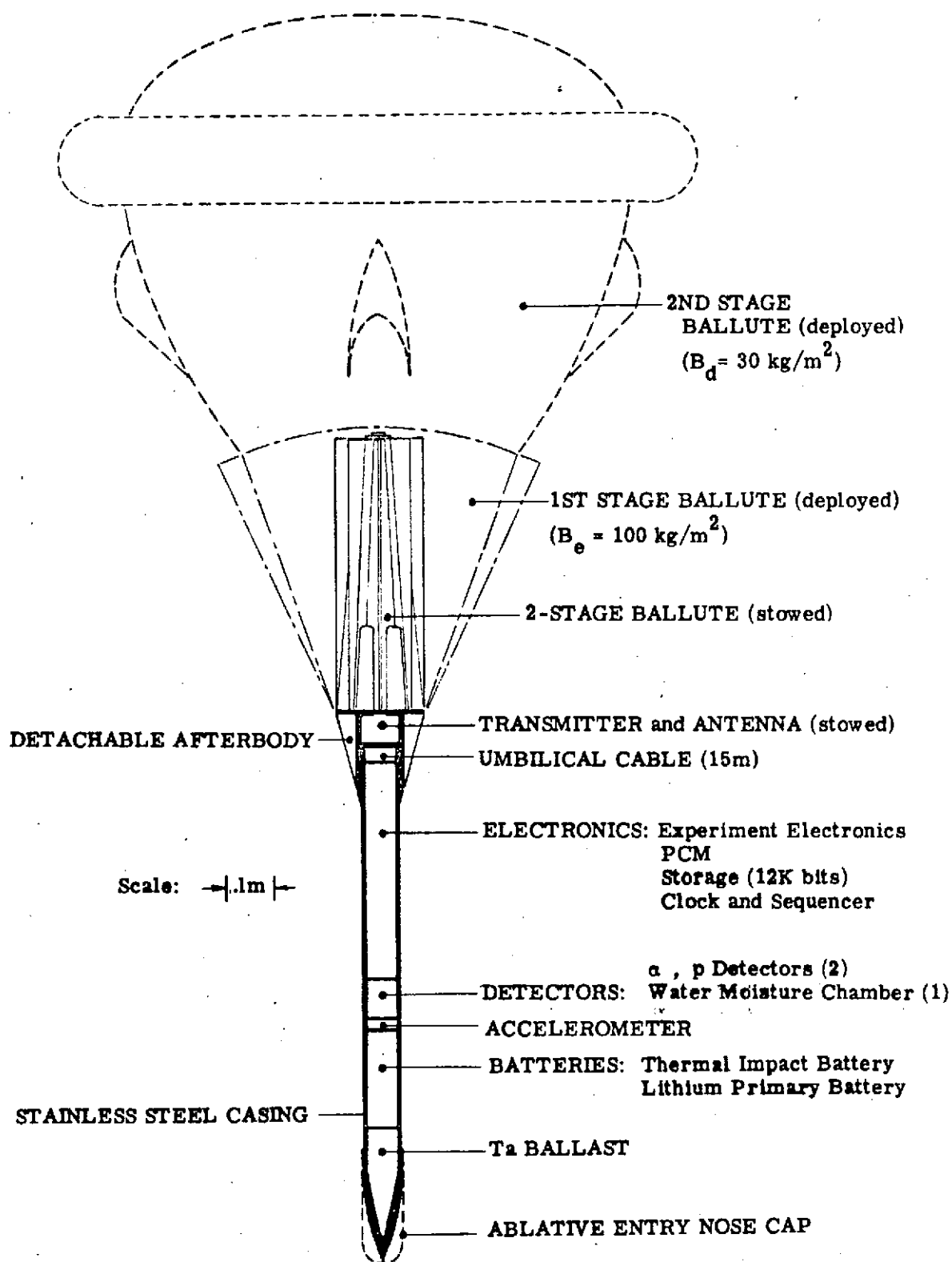| Instrument	Objective	Range	Accuracy	Mass(kg)	Pwr(w)	Data (bits)
Accelerometer	Soil density, composition, layering	10,000 g	$\pm$ 2%	0.1	0.05	8000 total
Moisture Chamber	Soil moisture content; H_2O/CO_2 distinction	10 μgm to 0.1 gm per gram	$\pm$ 50%	0.5	1.0	3600 total
Aluminum Oxide Hygrometer	Equilibrium free H_2O content	0.01 μgm to 20,000 μgm per liter (-100° to 30°C)	$\pm$ 5%	0.1	0.01	500/day*
Sensistor Temperature Sensor	Gas temperature	-120° to 30°C	$\pm$ 3°C	0.1	0.01	500/day*
Mini-α, p Backscatter	Soil chemistry; H_2O content	C, N, O, F, Na, Mg, Al, S, Si, K/Ca, Ti, Fe	.03 to .5 atom %	1.2	1.2	5100/day*

*Surface lifetime = 8 days

1979 Pioneer Mars Penetrometers

A schematic diagram of the penetrometer design envisioned for a 1979 Pioneer Mars mission is shown on the right. The forebody, or main penetrating section, of the penetrometer consists of a stainless steel casing, tantalum nose ballast, batteries, experiments and electronics, and trailing umbilical cable. The detachable afterbody consists of the surface to orbiter transmitter, antenna, 2-stage ballute and related structure. An ablative nose cap shrouds a 6.0 CRH tangent ogive nose cone for thermal protection during entry.

The same operations profile would be followed for each penetrometer. The sterilized penetrometer would be released from the orbiter in its bioshell, spinning in the correct attitude for a deorbit impulse in the vicinity of orbit apoapse. After jettisoning the bioshell a short retro impulse would be imparted to the penetrometer placing it on an entry trajectory. The spent retro motor would then be jettisoned, the penetrometer despun and the first stage of the ballute deployed. Following a 6-8 hour coast the penetrometer enters the atmosphere at a random angle of attack, quickly weathervaning to near zero angle of attack conditions. (An aerodynamic stability analysis of entry needs to be performed to verify this characteristic.) Following peak-heating and deceleration loads the second stage of the ballute is deployed to slow the penetrometer to an impact velocity of about 150 m/sec. At impact the penetrometer drives through the nose cap beginning surface penetration. When the afterbody reaches the surface it separates from the main forebody section remaining at the surface due to its larger cross-section. The forebody, trailing an umbilical line to the afterbody, continues to penetrate the soil to a depth of several meters. Depth of penetration can vary from 1 to 15 meters (umbilical line limit), depending on soil penetrability, without failure. This should encompass 95% of known soil conditions. The second stage of the ballute ruptures at impact providing space for presprung deployment of a double cross dipole antenna. Experiments are initiated at impact, as described on the previous page, and continue for the life of the primary battery which is about 8 days. Immediately after impact, and once each day thereafter, stored data is transmitted to the orbiter as it passes overhead.

It is important to note that all of the penetrometer subsystems, with the exception of the experiments, are based on existing hardware which has been used under similar conditions. Final design and assembly, as well as sterilization, would of course be a new effort. As far as has been possible to ascertain at the present time, sterilization does not appear to represent any special problems for this proposed design.

2ND STAGE
BALLUTE (deployed)
$(B_d = 30 \text{ kg/m}^2)$

1ST STAGE BALLUTE (deployed)
$(B_e = 100 \text{ kg/m}^2)$

2-STAGE BALLUTE (stowed)

TRANSMITTER and ANTENNA (stowed)

DETACHABLE AFTERBODY

UMBILICAL CABLE (15m)

ELECTRONICS: Experiment Electronics
PCM
Storage (12K bits)
Clock and Sequencer

Scale: $\vdash$.1m$\dashv$

α , p Detectors (2)
DETECTORS: Water Moisture Chamber (1)

ACCELEROMETER

BATTERIES: Thermal Impact Battery
Lithium Primary Battery

STAINLESS STEEL CASING

Ta BALLAST

ABLATIVE ENTRY NOSE CAP

1979 PIONEER MARS PENETROMETER

67

Penetrometer Mass Summary

A mass summary of the proposed penetrometer design is presented on the facing page. Total mass attached to the Pioneer orbiter would be 40.8 kg per penetrometer. All of this is separated from the orbiter at penetrometer deployment. If the penetrometers are carried circumferentially on the drum of the orbiter bus they can be deployed radially by centrifugal force of the spinning spacecraft. Since the penetrometers are deployed one at a time, and mounted in pairs diametrically opposite each other, a lateral CG offset occurs whenever an odd number of penetrometers remain on the bus. The spin stability should be preserved but considerably more analysis of spacecraft attitude control needs to be performed.

Of the 40.8 kg which leave the orbiter with each penetrometer, 10 kg of this is jettisoned or expended prior to impact. Impact mass of the penetrometer is just under 31 kg. Most of the individual weights are reasonably secure. The science package, the most significant development item, could vary by several kg. Changes in its mass would simply be traded off with the ballast mass to preserve the penetration characteristics of the proposed design.

PENETROMETER MASS SUMMARY

- **FOREBODY**   22.5 kg

Science	2.5
Power	2.5
CCS/Data Handling	0.9
Structure	7.8
Ballast	8.7
Umbilical Cable	0.1

- **DETACHABLE AFTERBODY**   8.1

Transmitter	2.3
Antenna	0.5
Two-Stage Ballute	3.0
Structure	2.3

- **ADDITIONAL EXTERNAL EQUIPMENT**   10.2

Entry Nose Cap	0.3
Deorbit/Despin Motors	3.0
Bioshell Cannister	4.4
S/C Attachment and Spin Motors	2.5

- **TOTAL MASS/PENETROMETER**   40.8 kg

Penetrometer Mission Characteristics

A summary of pertinent parameters of the baseline penetrometer mission design are listed on the facing page. Beginning with penetrometer deployment a retro impulse of 80 m/sec is required for deorbit. The entry conditions and ballistic coefficients provided by the first and second stages of the ballute result in a nominal impact velocity of 144 m/sec. At this speed a minimum penetration of 1 m is possible in rock. Penetration is based on empirical data presented by C. W. Young [8] of Sandia Laboratories.

The impact accelerometer generates approximately 8000 bits of deceleration data during an average 15 msec penetration. The duty cycles of the other instruments are also presented and have been discussed above. The penetrometer can store up to 12 K bits of data between daily transmissions to the orbiter. Maximum anticipated data load is approximately 10K bits/day during the first day after which it drops to less than 6K bits/day.

A 400 MHz, 0.5 watt transmitter can send data to the orbiter at a rate of 50 bps at a range of 2000 km with 10 db margin. The orbiter is within the antenna beamwidth of 90° for 12 minutes each orbit. A similar transmission period once per day, triggered by a penetrometer clock, makes three passes through the stored data, insuring at least one complete transmission received open-loop by the orbiter. The communication link is based on a similar design proposed for the Saturn-Uranus Pioneer entry probes. Daily power consumption of 6 watt-hours results in an 8-day lifetime based on a 50 watt-hour rated lithium primary battery.

PENETROMETER MISSION CHARACTERISTICS

- **DEORBIT/ENTRY/DESCENT**

    - o Deorbit Impulse   80m/sec

    - o Entry Conditions
            velocity   4.73km/sec
            path angle -15°

    - o Ballistic Coefficients
            entry 100 kg/m^2
            descent 30 kg/m^2

    - o Impact Velocity   144 m/sec

    - o Minimum Penetration   1 meter

- **SURFACE OPERATIONS**

    - o Experiment Duty Cycles
            impact accelerometer   15 msec
            moisture chamber   30 sec
            hygrometer/temperature   10 sec - 6x/day
            α , p backscatter   2 hr./day

    - o Data Management
            storage   12 K bits
            transmission rate   50 bps
            transmission period   12 min

    - o Power Consumption   6 watt-hrs./day

    - o Surface Lifetime   8 days

Orbit Payload Capability: Mission B

Similar to Mission A, orbit payload capability is presented in the double figure on the facing page for 1979 Mars transfers. The left figure is the plot of orbit capture velocity (retro ΔV) versus launch date for the three orbit periods 12, 24 and 48 hours. Note that, for this figure, the periapse altitude is constant at 1000 km. This higher value (compared to 500 km for Mission A) is chosen to insure a 42-year lifetime for the unsterilized orbiter bus. As before, these data are for the reference arrival date, 7 September 1980. Useful orbit payload is presented on the right-hand plot as a function of orbit capture velocity. In this case a larger solid rocket motor, the TEM-616, has been chosen to provide larger orbit payloads anticipated for Mission B. This motor has been flight qualified, is dimensionally compatible with the baseline PVO-78 bus design, and will make its first operational flight in 1975. However, a fully loaded TEM-616 and the payload it can place in Mars orbit would exceed the injected mass capability of the Atlas/Centaur launch vehicle for most 1979 Mars transfers. The boundary of the Atlas/Centaur performance at a C3=10 $(km/sec)^2$ is shown at the far right. Curves of constant fractional propellant loadings are shown to the left of this boundary, indicating how retro stage capacity must be matched to launch vehicle performance for Mission B.

The two figures provide a means of selecting a consistent combination of launch window, orbit period, maximum useful orbit mass, and correct retro motor propellant loading. A synchronous (24.6-hour) orbit period appears to be a good compromise between penetrometer deorbit retro requirements and post-impact communication characteristics with the orbiter. A nominal orbit period of 24 hours, and a 20 day launch window (maximum C3=10) provide a maximum useful orbit mass of 520 kg as shown. For this design point the TEM-616 would have about a two-thirds propellant loading.

The maximum retro velocity required for the above design point is 985 m/sec at the beginning of the launch window. Toward the end of the window this value drops by about 15 m/sec. Since the solid rocket motor would have a fixed propellant load once the mission design point is fixed, optimum capture maneuvers at Mars combined with late launches would result in orbit periods less than the synchronous value. This should be avoided for Mission B. An acceptable remedy of the retro velocity control problem would be use of off-periapse insertion for orbit capture. While the orbit apseline could be rotated 10-15° from nominal, this can be accomodated in the impact site selection process, and a fixed orbit period is maintained across the launch window.

RETRO DV
(m/sec)

TEM-616 MOTOR
M_O=355 kg
MF=0.923
Isp=293 sec

TYPE II MARS TRANSFERS
TRIP TIME = 310 days
PERIAPSE ALTITUDE = 1000 km

80% PROPELLANT
LOAD

70%

12^h PERIOD ORBIT

MAXIMUM
ATLAS/CENTAUR
CAPABILITY
@C3 = 10

60%

24^h PERIOD

50%

48^h

40%

20^d
LAUNCH
WINDOW

520kg

9/23   10/13   11/2   11/22

1979 LAUNCH DATE (M/D)

300   400   500   600

USEFUL ORBIT MASS (kg)
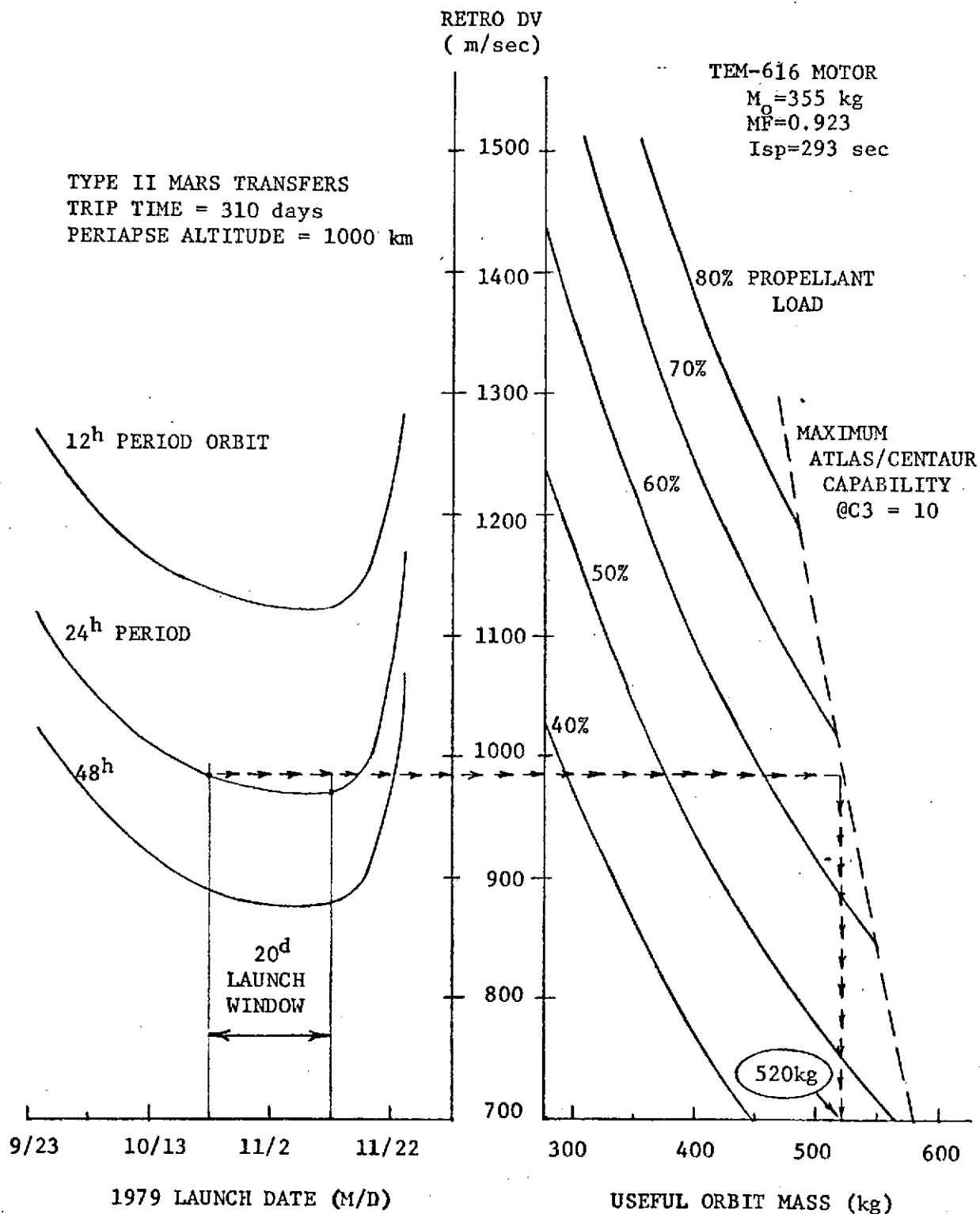
1979 PIONEER MARS ORBIT PAYLOAD CAPABILITY

REMOTE SENSING/PENETROMETER MISSION

73

Remote Sensing/Penetrometer Payload Summary

Orbiter and science payload masses are listed in the facing table for the Remote Sensing/Penetrometer Mission (B). Data for both the // and ⊥ PV-78 design concepts are presented. The initial assumed PVO bus weight and modifications are the same as required for Mission A (see page 30). An additional 40 kg is provided for anticipated structural improvements to carry the penetrometers. Four (4) penetrometers can be added to the fully modified bus bringing the orbited payload (less orbiter science) to 460 and 481 kg, respectively, for the // and ⊥ designs. Assuming a maximum orbit payload of 520 kg, as presented on the previous page, the available orbiter bus science payload is 60 and 39 kg, respectively for the // and ⊥ designs.

The example of an orbiter bus payload is given to indicate what the available payload means in terms of remote sensing instruments. As has already been mentioned the proposed PVO-78 radar instrument is not very meaningful for Mission B, and the IR instruments shown are basically repeat experiments from earlier Mariner and Viking Orbiter payloads. More study is needed to optimize the orbiter science instruments for this mission. Unlike Mission A, weight, as well as power and communications, will be a limiting factor in the payload selection.

REMOTE SENSING/PENETROMETER PAYLOAD SUMMARY

- SPACECRAFT ORIENTATION TO ECLIPTIC $\parallel$ $\perp$

- MASS SUMMARY

	$\parallel$	$\perp$
Assumed PVO Bus Weight	230 kg	230 kg
Required Modifications	+27	+48
Structural Improvement	+40	+40
Modified Bus Weight	297	318
Penetrometers (4 @ 40.8 kg ea.)	163	163
Orbit Weight Less Bus Science	460	481
Maximum Useful Orbit Weight	520	520
Available Bus Science Payload	$\leq$60 kg	$\leq$39 kg

- BUS SCIENCE PAYLOAD EXAMPLE

	kg	watts	bps
Radar Altimeter	15	11	125
IR Spectrometer (IRIS design)	17	16	750
IR Radiometer	5.5	6	100
Dual Frequency Occultation	3	10	–
Totals	40.5	43	975

o SCIENCE SUPPORT CAPABILITY

With $\parallel$ Concept	60	$\approx$50	9.6×10^5/orb
With $\perp$ Concept	39	$\approx$50	9.6×10^5/orb

75

Nominal Orbit Conditions

The orbiting bus is placed in a polar orbit having a period of slightly more than 24 hours and a periapse altitude of 1000 km. Since the period is synchronous with Mars' rotation, the bus will pass over the pene-trometer impact site on several orbits subsequent to the impact event. There will be a drift in periapse location due to Mars oblateness of the amount 0.099 deg/orbit regression in the argument of periapse. Over the eight day lifetime of the penetrometer the drift amounts to only 0.8° in latitude (longitude is stationary). It should not be necessary to control the orbit during this time interval to maintain the proper geometry and timing of the penetrometer-to-bus communications link. In the long run the orbit precession aids the multi-penetrometer mission in that other impact sites on the Mars surface become accessible.

Two nominal orbit orientations are called out in the table. For the north polar mission the initial periapse location is at 36.8° N latitude; subsequent drift due to regression is toward the north pole. For the south mid-latitude mission periapse is initially located at 71.6° S latitude and subsequent drift is toward the south pole (impact site latitude, being displaced from periapse, begins near the south pole and drifts toward the equator).

NOMINAL ORBIT CONDITIONS FOR
REMOTE SENSING/PENETROMETER MISSION

EPOCH	7 SEPT. 1980
ORBIT PERIOD	24.623 hours
PERIAPSE ALTITUDE	1000 km
INCLINATION	90°

$$\text{LONGITUDE OF NODE*} \qquad \begin{pmatrix} 92.3^O \\ 143.2^O \end{pmatrix}_A \quad \begin{pmatrix} 272.3^O \\ 288.4^O \end{pmatrix}_B$$

ARGUMENT OF PERIAPSE

*LOCAL TIME (MIDNIGHT REFERENCE)

A-NORTH POLAR MISSION

B-SOUTH MID-LATITUDE MISSION

Impact Sites for Penetrometer Missions

Traces of impact site accessibility for each of the polar orbits presented on the previous page are illustrated in the graph on the right. The figure is a plot of impact site latitude versus time in orbit. The traces are solid lines with arrows indicating direction of motion. Since both paths cross the Mars polar regions, polar changes are also included as a function of season. The lightly stippled areas indicate the presence of a polar hood, while heavy stippling represents the exposed polar cap. Polar cap behavior is taken from Glastone, "The book of Mars," NASA SP 179, page 100.

Candidate impact sites are marked along the traces for both the North Polar (N1, N2, ...etc.) and South Mid-Latitude (S1, S2, ...etc.) Missions. A brief description of each site is presented on the next two pages.

Curiously, the southern mission turns out to be a better polar mission in some ways than the northern mission, even though the latter spends more time near the pole. The southern mission has the advantage of starting over polar ice, instead of over a late Summer pole. For the same reason, the southern mission has the advantage of crossing over the receding edge of the cap (the most interesting place on the planet from the point of view of liquid water or soil moisture detection) only about 4-5 months after arrival, rather than 17 months after arrival as in northern mission. These time intervals are approximate estimates, but relative magnitudes should be right.

From the point of view of geology, the southern mission also has the advantage of proceeding directly to lower latitudes, where interesting features are more available. The northern mission stays in the small area north of 60°. On balance, the southern mission represents the best balance of polar cap science and geological sampling, available within a reasonable time after arrival.

Northern Hemisphere Seasons
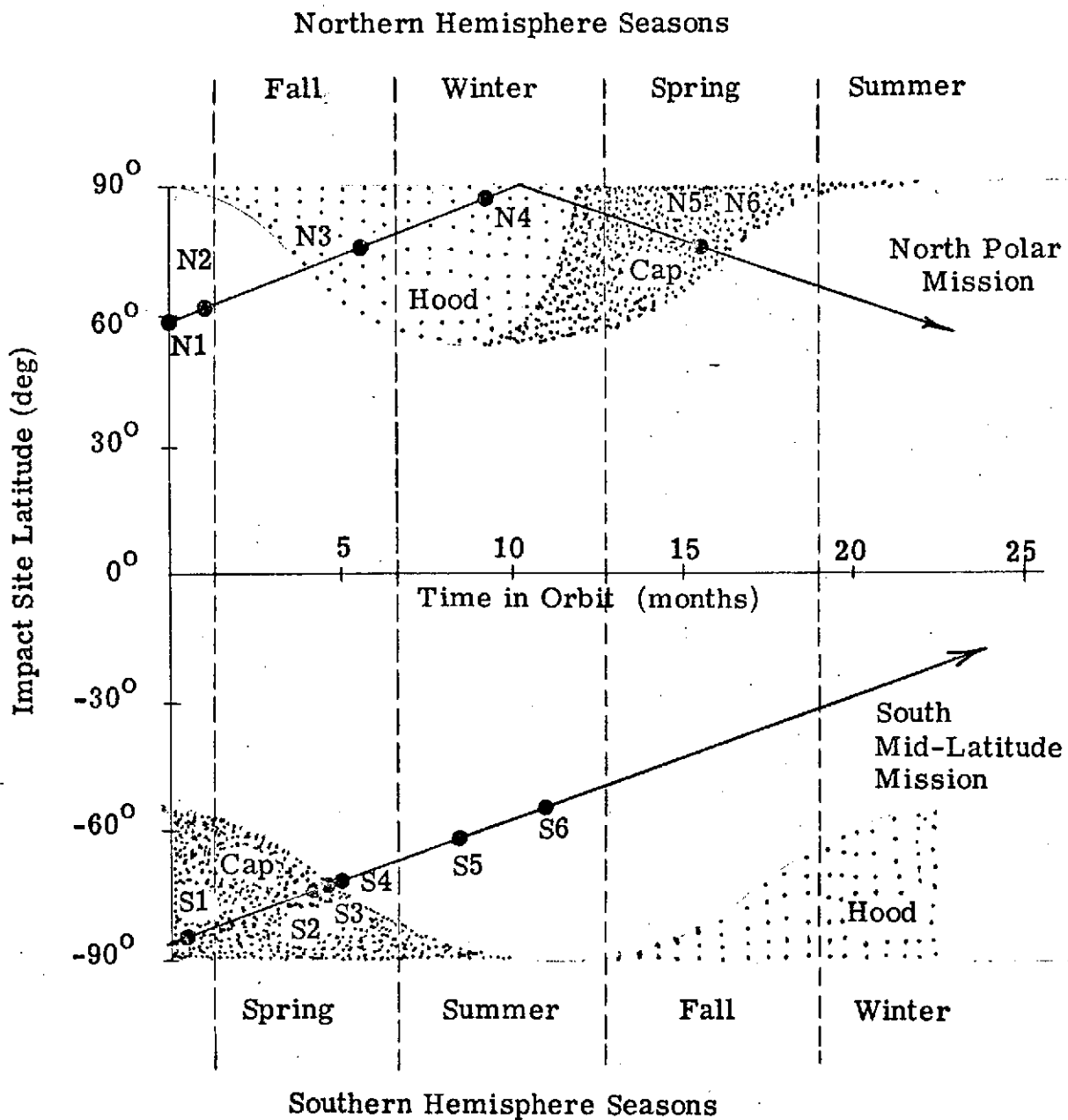
IMPACT SITES FOR PENETROMETER MISSIONS

(Initial Epoch: 7 September 1980)

IMPACT SITE EXAMPLE FOR PENETROMETER MISSION
(1979 Northern Polar Mission)

Months After Arrival	Site No.	Lat.	Long.	Geologic Province	Remarks
0.1	N1*	$+59^O$	38°	Mod. Cratered Plain (Acidalium)	Low Albedo Area - Test Soil Composition
1.4	N2*	$+62^O$	103°	Volcanic Plains	Small Volcanic Mountain; Test for Expected Basaltic Rocks
3	N3	$+76^O$	150°	Etch-Pitted Plains	Test Soil Composition & Water Content. See Also Site N5
8	N4	$+85^O$	16°	Layered Deposit (See JGR, 78, 4203, Fig. 6A)	Test for Thin Laminae of Soil & Other Sedimentary Structure
16	N5	$+76^O$	150°	Same as N3, Near Edge of Retreating Cap	Repeat N3 Tests but Note Site Now Covered by Polar Snows. Liq. Water May Exist Near Cap Edge.
17	N6	$+76^O$	150°	Plain, Near Edge of Retreating Cap	Test for Liquid Water. Pimental Data Suggest Max. Probability Here.

*N1 and N2 are lowest priority; if only 4 sites available, use N3-N6 for polar cap studies, but note long delay required; South Mission may be better.

IMPACT SITE EXAMPLE FOR PENETROMETER MISSION

(1979 Southern Mid-Latitude Mission)

| Months After Arrival	Site No.	Lat.	Long.	Geologic Province	Remarks
0.5	S1	-85°	270°	Polar Ice, "Australis Chasma"	Test for H_2O vs CO_2 in Valley
4.1	S2	-72°	-	Edge of Shrinking Cap	Pimentel Data Suggest Max. Probability of Water Moisture in Soil Near Edge of Shrinking Cap.
4.2	S3	-71°	-	" "	
4.3	S4	-70°	-	" "	Test H_2O vs CO_2, since CO_2 Cover may "Burn Off" to Expose H_2O as Cap Recedes.
8	S5	-62°	240°	Cratered Uplands	Low Albedo Unit; Test Soil Composition & Granularity
13	S6	-55°	290°	Floor of Hellas	Discriminate Loose Soil from Rocks. High Albedo Unit; Test Soil Composition & Granularity.

81

Probe Deflection Maneuver Design Map

The entry probe (penetrometer) will be separated from the orbiter bus near its apoapse location and then deflected toward the Mars surface via a ΔV retro maneuver. For purposes of generating the design tradeoff map, this maneuver is assumed to be made in the circumferential direction (normal to radius vector) which is near-optimal from a ΔV standpoint. The map describes the ΔV requirement as a function of entry angle and maneuver location, and also shows the time-to-entry and the orbiter location at probe entry. In general, the ΔV requirement increases with steeper entry angles and with later deflection times after apoapse (180°). For a maneuver made at orbiter true anomolies greater than 200° the ΔV becomes relatively very expensive. One can trade this effect off with the desire to minimize the transit time to entry. The nominal design selection places the entry angle at -15° and the orbiter true anomoly at deflection at 200°. The ΔV required is 80 m/sec, the time to entry is 5.8 hours, and the orbiter position at entry is 302° true anomoly. At entry (100 km altitude) the probe leads the orbiter in the plane of motion by about 29°.

PROBE DEFLECTION MANEUVER DESIGN MAP

83
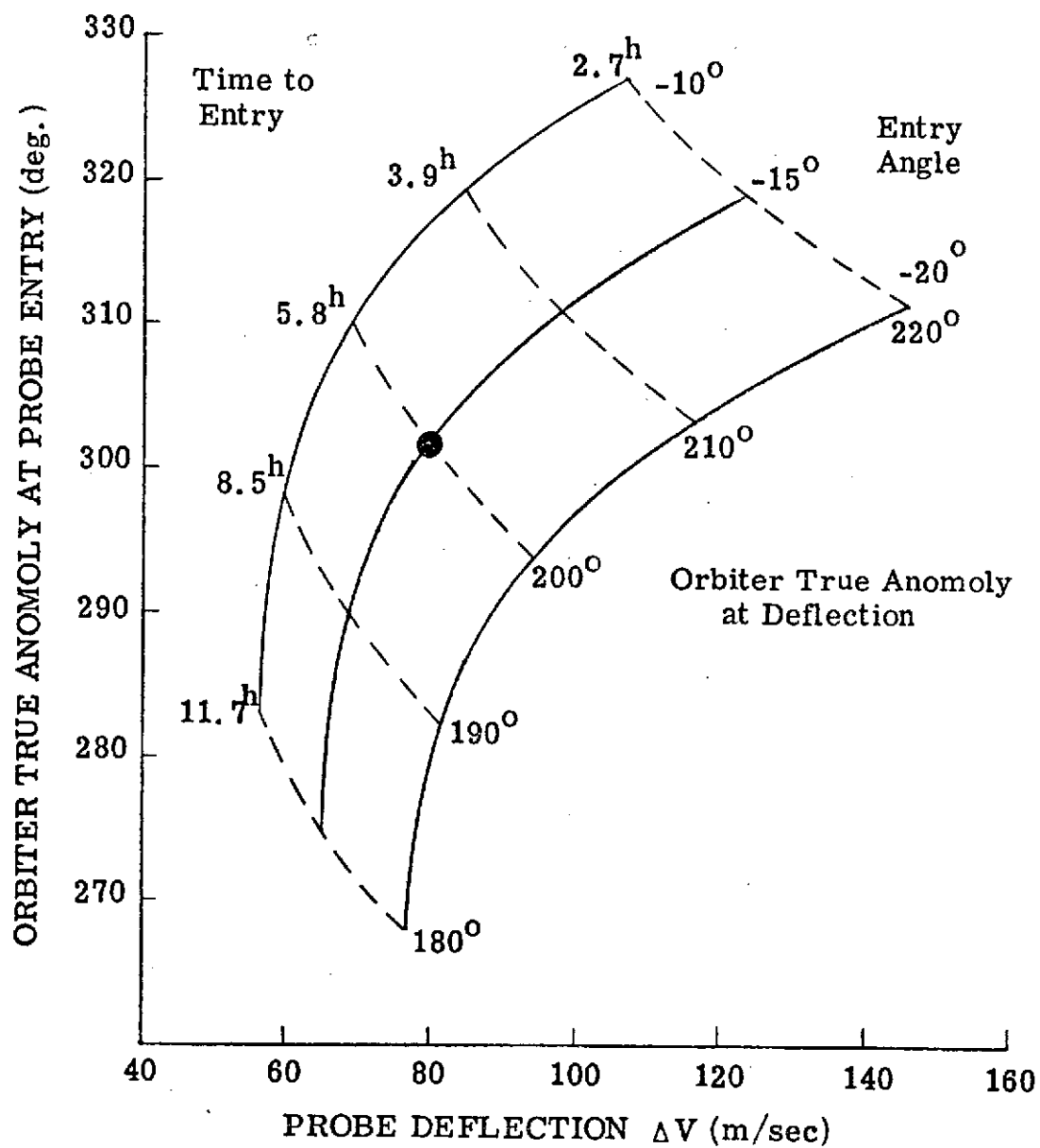
## Size and Drag Characteristics for Terminal Descent

Atmospheric entry and terminal descent is accomplished by a 2-stage, attached ballute decelerator system. The ballute system is sized by the second stage requirement to attain an impact velocity of about 150 m/sec. Assuming a drag coefficient of 0.76, curves of second stage ballute diameter are shown as a function of ballistic coefficient ($B = m/C_D A$) and descent mass. The impact velocity curve is derived from an approximate analytical formula assuming the mean atmosphere model. The nominal design point indicated gives a second stage ballistic coefficient of 30 kg/m^2 and a ballute diameter of 1.3 meters for a 31 kg descent mass. It should be noted that both entry and descent operations assume a zero-lift configuration. The first stage ballute has a ballistic coefficient of 100 kg/m^2 and a diameter of 0.54 meters assuming a drag coefficient of 1.35. The actual impact velocity derived from an integrated trajectory solution is 144 m/sec.
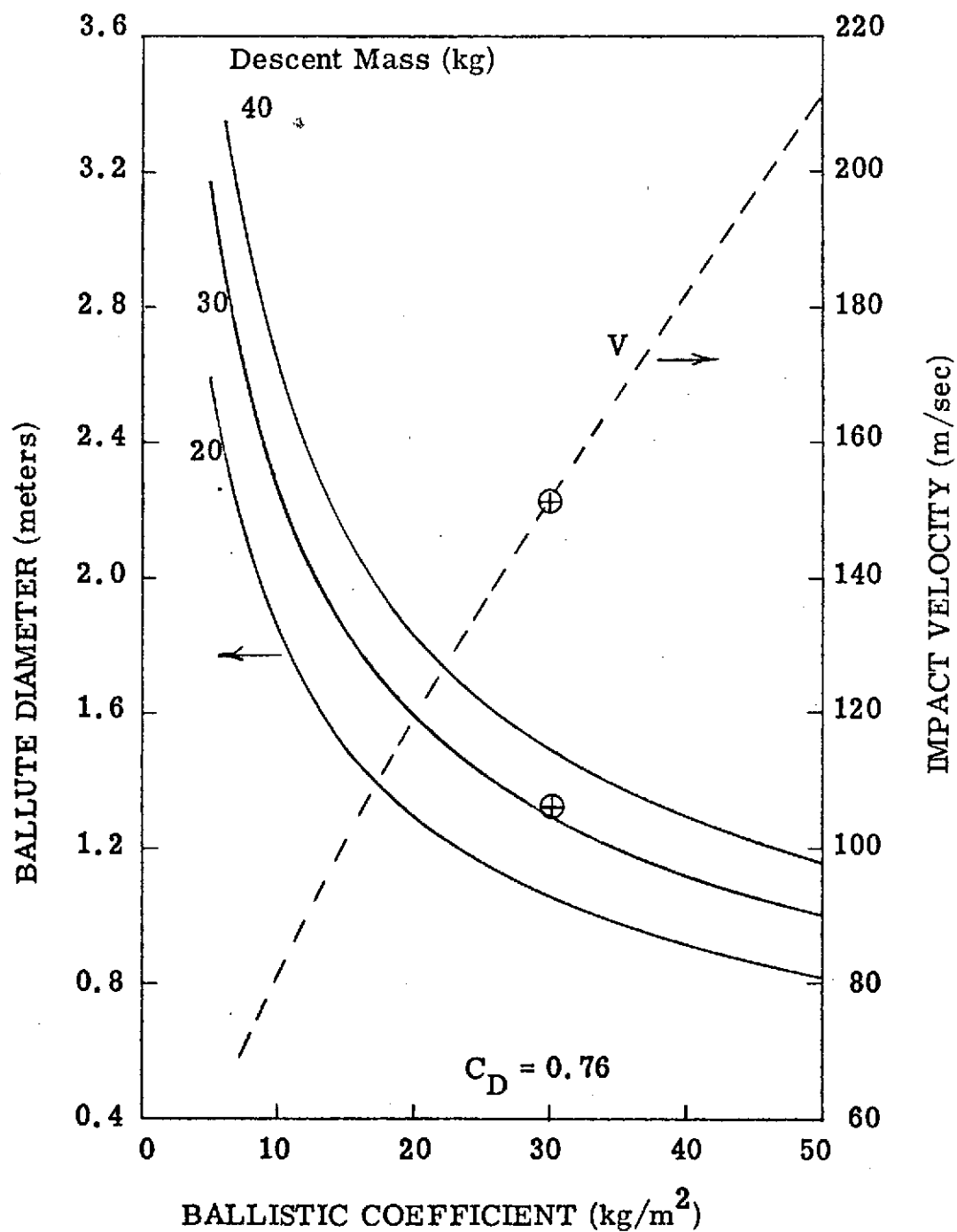
BALLUTE (2ND STAGE) SIZE AND DRAG CHARACTERISTICS
FOR MARS ATMOSPHERIC TERMINAL DESCENT
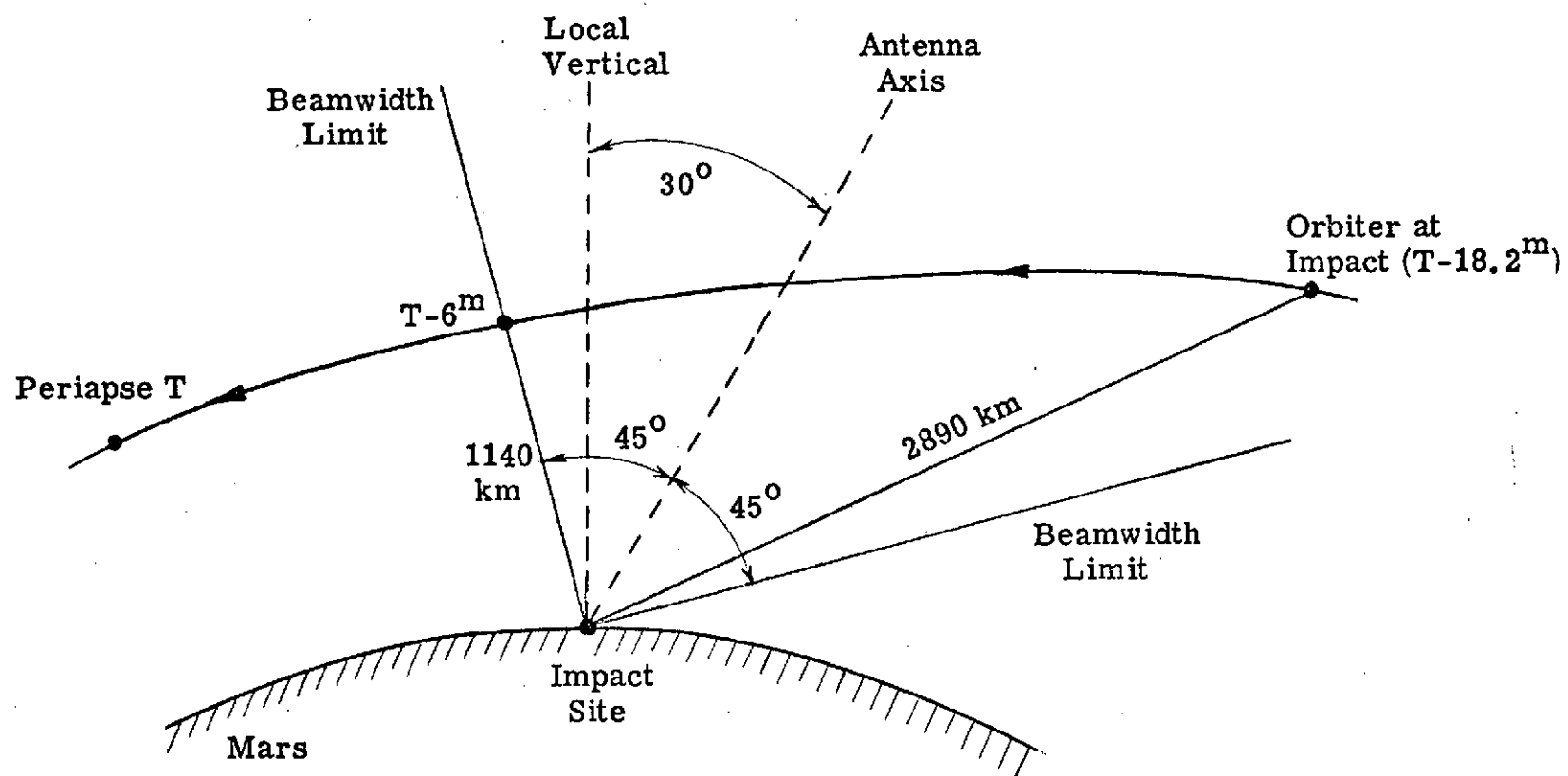
Mars Ballistic Entry and Descent Profile

Initial entry conditions are taken at 100 km altitude in the mean atmosphere. Velocity magnitude and entry angle are, respectively, 4.732 km/sec and -15°. Maximum convective heating rate occurs 66 seconds after entry at 26 km altitude and 4.1 km/sec velocity. The heat pulse has a half-width of about 28 seconds and its peak value (for a 1.5 inch stagnation point nose radius) is 2.14×10^6 joules/m^2/sec = 189 Btu/ft.^{2}/sec. Heating rate to the first stage ballute is significantly less than this value, it being of larger dimension than the penetrometer nose. Maximum dynamic pressure is 238 lb/ft.^{2} occurring 76 seconds after entry; the peak deceleration acting on the first stage is 11.6 earth gravity units. The second stage ballute is deployed 88 seconds after entry at an altitude of 11 km and a velocity of 1.88 km/sec (Mach 8.8). At this point the heat transfer is down to 20% of the peak value, the dynamic pressure is 171 lb/ft.^{2}, and the flight path angle is -13.3° from the horizontal. Terminal descent takes an additional 77 seconds with the impact occurring at a speed of 144 m/sec inclined about 30° off the vertical. The down-range angle from entry to impact is 6.5°.
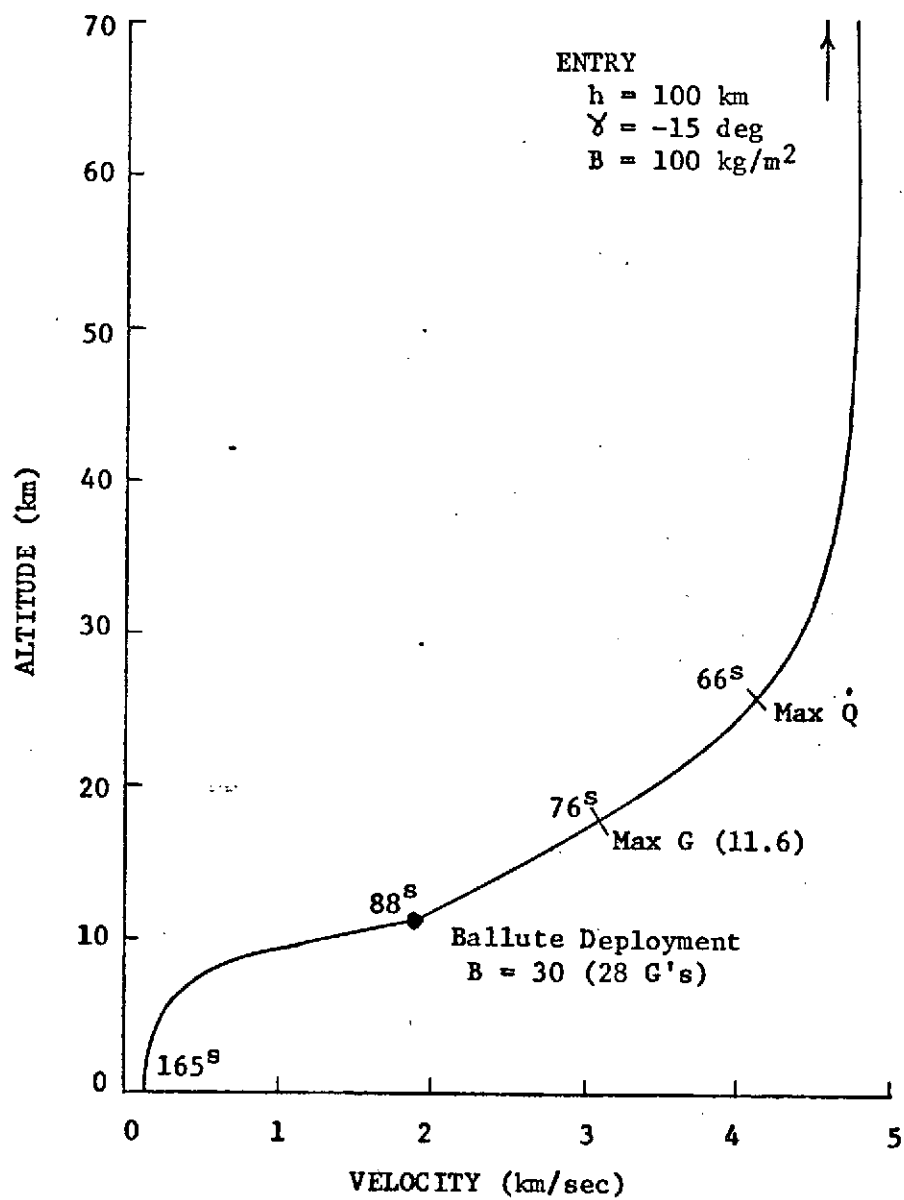
ALTITUDE-VELOCITY PROFILE FOR MARS ENTRY AND DESCENT

Penetrometer-Orbiter Communications Link

The penetrometer communications subsystem will relay to the orbiter bus approximately 12,000 bits of data at a rate of 50 bits/sec. Three repetitive transmissions of this data are made during the time interval (≈ 12 minutes) of orbiter overflight. Assuming that the transmission period is initiated by a pre-set "clock" contained in the penetrometer, the redundant transmissions serve to assure that a complete data set will be received, within the antenna beam width, in the face of position and timing errors. The antenna beamwidth is 90° centered about the penetrometer's longitudinal axis which is nominally oriented 30° off the local vertical. The illustration describes the communications geometry during the first orbiter pass; subsequent passes are approximately the same because of the orbiter's synchronous period. At the time of impact the orbiter is 18.2 minutes from periapse passage and the communications line-of-sight distance is 2890 km. The distance reduces to 1140 km when the orbiter crosses the leading beamwidth limit 6 minutes before periapse passage.

88

Local
Vertical

Antenna
Axis

Beamwidth
Limit

30°

Orbiter at
Impact (T-18.2^m)

$T-6^m$

Periapse T

68

1140
km

45°

2890 km

45°

Beamwidth
Limit

Impact
Site

Mars

PENETROMETER TO ORBITER COMMUNICATIONS LINK (1st PASS)

Entry Parameter Variations with Model Atmospheres

Shown in the facing table is the sensitivity of key entry parameters to the atmospheric density model in effect. The entry/descent ballistic coefficients are for the nominal (mean atmosphere) design. Most significant are the variations in impact velocity and path angle. Velocity increases by 21% in the Min ρ s atmosphere and decreases by 17% in the Max ρ s atmosphere. This would affect the depth of penetration as well as the interpretation of impact accelerometer data. The impact angle variation of about 10° results in similar uncertainty. Also, in the case of the Min ρ s atmosphere, angular displacement of the beamwidth limit causes a communications time loss of about 1 minute ($\approx 8\%$) at end of transmission. None of the above effects are thought to be serious enough to question basic mission feasibility. Furthermore, design parameters will become more certain after the Viking mission determines the atmosphere profile.

ENTRY PARAMETER VARIATIONS WITH MODEL ATMOSPHERES

PARAMETER*	MIN ρ's	MEAN	MAX ρ's
TIME TO IMPACT (sec)	157	165	171
IMPACT VELOCITY (m/sec)	174	144	120
IMPACT PATH ANGLE (deg)	-51	-62	-72
DOWNRANGE ANGLE (deg)	6.54	6.49	6.46
MAX. DRAG (Earth g's)	9.7	11.6	13.4
MAX. HEAT RATE $(\frac{joules}{m^2 \cdot sec})$**	1.99×10^6	2.14×10^6	2.24×10^6
MAX. TOTAL HEAT (joules/m^2)**	75.0×10^6	69.3×10^6	67.1×10^6

* Entry conditions: h = 100 km, V = 4.73 km/sec, γ = -15°

** For nose radius = 1.5 inches; $\dot{Q}$ and Q inversely proportional to $R_N^{0.5}$

Deflection Maneuver Error Analysis

Maximum (3σ) errors in executing the probe deflection maneuver are assumed to be 3% in ΔV magnitude and 3° in direction. The magnitude error contributes about 4 to 6 times the effect of the direction error on subsequent deviations at entry conditions. Angular position error at entry is 1.9^O and entry path angle error is 1^O. These two effects combine to cause a landing site error of 2.3^O or 135 km along the Martian surface. This error in turn results in a loss of 45 seconds or about 6% of the nominal communications time.

ERROR ANALYSIS FOR PENETROMETER MISSION

- PROBE DEFLECTION MANEUVER

    ΔV MAGNITUDE $\pm 3\%$ (3σ)

    ΔV DIRECTION $\pm 3^{O}$ (3σ)

- RESULTING ERRORS (3σ)

    ENTRY ANGLE $\pm 1^{O}$

    LANDING SITE $\pm 2.3^{O}$ $(\pm 135$ KM)

    ORBITER COMMUNICATIONS
    TIME LOSS (1ST PASS) 45^{S} $(6\%$ of $12^{m})$

93

SAI Orbiter/Penetrometers Mission Cost Estimate

A preliminary estimate of the cost of a single 1979 Pioneer Orbiter/Penetrometers Mission based on PVO-78 hardware inheritance is presented below. The cost for an additional PVO-78 flight article (less science) of $9.5M assumes an FY 76 order date. Structural changes to accomodate four penetrometers, each weighing 40 kg, is estimated to cost almost $5M (Δ1). System modifications, which in addition to those considered in Mission A also include a different orbit retro motor, are estimated at $5-7M. The total cost of the penetrometers is estimated at almost $24M (including 10% contingency) and includes the following: a) 1 PTM, b) 5 flight articles, c) science instrument development, and d) sterilization. Of the $24M for the penetrometers, about $6.5M is for science instrument development. Mission operations costs (Δ4) of $13.5M include launch and flight operations, guidance/navigation, DSN services, and data handling related to penetrometer science only. Total cost (less orbiter science) with 10% contingency is about $63M, with the difference due to PVO-78 design concepts (// vs ⊥) being negligible. Note that the double application of contingency applied to the penetrometers represents a padding of more than $4M on the base estimate of their cost. The cost of the orbiter science could vary considerably depending on the amount of instrument development required. To do really new orbital remote sensing of the Mars atmosphere and surface in 1979 with a Pioneer bus could cost $15-20M in instrument development alone. Depending on the outcome of further study of the orbiter science payload, it can be expected that the total cost of a Pioneer Orbiter/Penetrometers Mission (B) will be $70-80M.

SAI PIONEER ORBITER/PENETROMETER MISSION COST ESTIMATE*

Cost Item	Concept	
	//	1
Procure 1 additional PVO-78 flight article	$9.5M	$9.5M
Δ1: Modify S/C structure for penetrometers	4.8	4.8
Δ2: Modify flight article subsystems (Pwr, Comm, ACS, Prop)	5.3	6.4
Δ3: Add Penetrometers (1 PTM, 5 Flight Articles; 10% Contingency)	23.6	23.6
Δ4: Add cost of mission operations	13.5	13.5
Net Totals	56.7	57.8
Contingency (10%)	5.7	5.8
Total (less orbiter science & data analysis)	62.4	63.6

*FY 1974 dollars

Subjects for Further Study (Orbiter/Penetrometers Missions)

A number of items requiring further study in this analysis of a 1979 Pioneer Orbiter/Penetrometers Mission are listed on the facing page. An evaluation of the mission's science credibility by a Science Advisory Group is essential to the justification of continued in-depth analysis of this mission concept. Of particular interest is concurrence in the relevance of penetrometers to post-Viking/75 exploration, and the identification of a valid orbiter payload for the 1979 opportunity. Provided with positive outputs on both counts, analysis of instrument development requirements and associated cost should be performed.

For the spacecraft and penetrometer systems, a number of subjects are identified which require at least a pre-Phase A systems study to address the question of feasibility. For the penetrometers a systems analysis is necessary to verify the basic proposed design. Key questions regarding flight stability and science instrument operations need to be addressed. A good estimate of the amount of testing to verify the penetrometer design for Mars application is needed. Sterilization impact should also be reviewed. For the orbiter, the feasibility of altering the PVO-78 design to meet the demanding requirements of the penetrometer mission must be analyzed. In addition to those system changes required by the Aeronomy/Geology orbiter, structure and propulsion changes exist. A system for penetrometer deployment needs to be worked out. Spacecraft stability with variable numbers of penetrometers onboard must be studied. Finally, based on explicit solutions to these problems, the total mission cost should be re-estimated.

During the course of this preliminary study, seismometer experiments with the penetrometers emerged as an important exciting measurement technique for study of active Martian surface processes and internal structure. Unfortunately, the lifetime and power requirements of seismometers appeared unrealistic for a first mission. The concept, however, is quite valid for a follow-on mission and should be explored further.

SUBJECTS FOR FURTHER STUDY
(Orbiter/Penetrometers Mission)

- Design and cost of penetrometer science

- Selection, design and cost of orbiter science

- Penetrometer systems analysis (including sterilization)

- Penetrometer stability analysis (entry, descent and impact)

- PVO-78 redesign analysis

- Modified Pioneer orbiter attitude control and stability analysis

- Refined mission cost estimate

- Evaluation of penetrometers for future seismometry mission

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4. "Pioneer Venus Mission Description", Doc. No. PV-1009, Ames Research Center, NASA, October 1973.

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6. "The Use of Instrumented Penetrometers to Determine Subsoil Conditions at Inaccessible Sites", J. L. Colp, Sandia Laboratories, October 1969.

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