4. THE PIONEER VENUS PROGRAM

LAWRENCE COLIN (Project Scientist) and CHARLES F. HALL (Project Manager)
NASA/Ames Research Center, Moffett Field, Calif. 94035, *U.S.A.*

Abstract. The Pioneer Venus program encompasses two spacecraft missions, Orbiter and Multiprobe, to be launched and to encounter Venus during the 1978 Venus mission opportunity. The missions are described in detail including mission and spacecraft descriptions, scientific objectives and payloads. The ways in which the payloads address the major scientific questions concerning Venus are treated in subsequent papers.

1. Introduction

The Pioneer Venus program managed by NASA's Ames Research Center encompasses two spacecraft missions, Orbiter and Multiprobe, both planned to encounter Venus during the 1978 mission opportunity. The Earth-to-Venus transit trajectories are sketched in Figure 1. The Orbiter spacecraft will be launched by an Atlas (SLV-3D)/Centaur (D-1AR) vehicle from the CCAFS, Complex 36, during the period May 20–June 2, 1978 on a Type-II* transit trajectory to Venus. Hughes Aircraft Company is the prime contractor for the Orbiter spacecraft. The launch weight capability is about 1300 lbs (590 kg) for this launch period. The spacecraft will encounter Venus and will be inserted into orbit on about December 4, 1978 after a transit time of 186–197 days. The dry weight-in-orbit will be approximately 790 lbs (360 kg) including a maximum scientific payload weight of 105 lbs (47.7 kg). The Earth–Venus distance at orbit insertion is 53×10^6 km. The nominal mission lifetime in orbit is planned to be 243 days (one Venus sidereal day) extending to August 4, 1979. The Earth–Venus distance increases monotonically during this period to 257×10^6 km.

The Multiprobe mission consists of a bus spacecraft (with many design features in common with the Orbiter spacecraft) transporting four atmospheric entry probes (one large and three identical small probes). Hughes Aircraft Company is the prime contractor for the Multiprobe spacecraft and General Electric Company, under subcontract to Hughes, is providing the entry systems for the probes. The Multiprobe spacecraft will be launched by an Atlas (SLV-3D)/Centaur (D-1AR) vehicle from the CCAFS, Complex 36, during the period August 7–24, 1978 on a Type-I transit trajectory to Venus. The launch weight capability is 2030 lbs (920 kg) for this launch opportunity.

The bus and four entry probes, the latter separated from the bus some three weeks prior to Venus encounter, will be separately targeted to arrive at Venus on about December 9, 1978 some five days following Orbiter insertion and after a

* Interplanetary spacecraft trajectories are classified as one of two types, Type I or Type II. They are distinguished by their heliocentric transfer angle, which is the angle between the Sun–Earth line at launch and the Sun–planet line at encounter. Type I trajectories are those for which the transfer angle is less than 180°. Type II trajectories have transfer angles between 180° and 360°.

Space Science Reviews **20** (1977) 283–306.

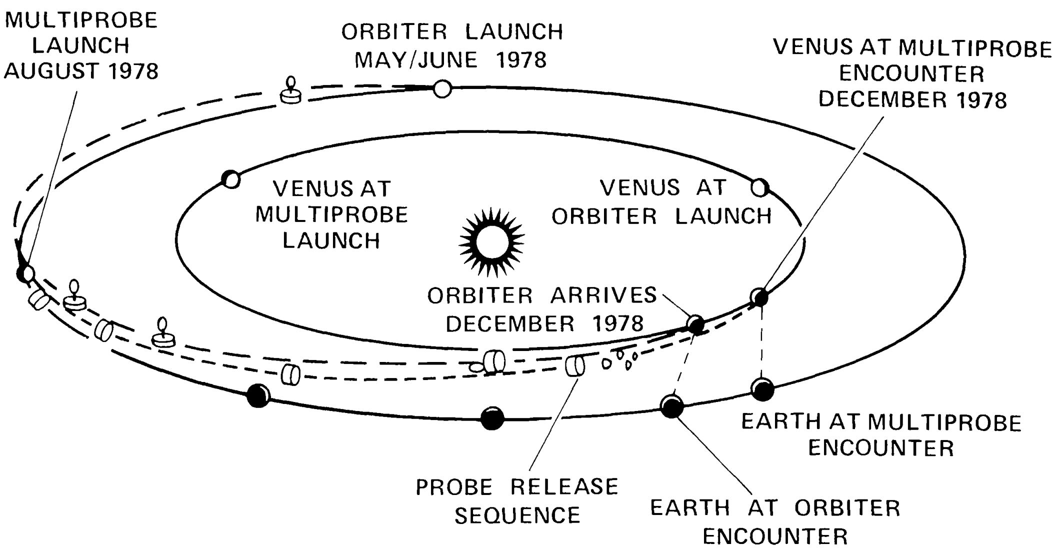

Fig. 1. Earth–Venus transit interplanetary trajectories.

transit time of 107–124 days. The five vehicles will enter the Venus atmosphere within a maximum arrival-time spread of about 90 minutes, and subsequently make upper-atmospheric (bus) and lower-atmospheric (probes) measurements along pre-determined descent trajectories which are widely separated spatially (longitude and latitude) within the hemisphere visible to Earth. The total entry weight will be approximately 1950 lbs (886 kg) including a maximum scientific payload weight of 133 lbs (60 kg). The bus will traverse its primary measurement altitude regime (300–115 km) in about one minute before becoming inoperative and finally destroyed. The large and small probes will traverse their measurement altitude regime (200–0 km) in about 57 min. Survival on the surface is not planned.

The data from all the Pioneer Venus spacecraft will be transmitted directly to Earth using the 26-meter and 64-meter diameter antennas of NASA's Deep Space Network (DSN), which is managed for NASA by the California Institute of Technology's Jet Propulsion Laboratory. The DSN antennas are located in California, Australia and Spain. The 64-meter antennas will be used when higher rates of data return are required and during critical events.

2. Orbiter Mission

The key elements of the currently planned orbit are listed in Table I. Most of these elements were selected by the Pioneer Venus Science Steering Group (PVSSG, see Section 4) to optimize the overall science return from the mission. The broad scientific objectives of the Orbiter mission are: (1) Global mapping of the clouds, atmosphere and ionosphere by remote sensing and radio occultation to extend the information obtained on the vertical structure from the Multiprobe mission; (2) Global studies by *in situ* measurements of the upper atmosphere,

TABLE I
Pioneer venus orbiter data

Orbit	
Periapsis altitude	200 km nominal[a]
Apoapsis altitude	66 614 km
Orbital period	24 hr[b]
Orbital inclination	105° (75° retrograde) ref. to ecliptic
Periapsis celestial latitude	16–28° North[c]
Periapsis celestial longitude	203–223°[d]
Mission lifetime	≥243 days on-orbit
Spacecraft	
Stabilization	Spin-stabilized; spin-axis perpendicular to ecliptic[e]
Spin-rate	5 RPM nominal (on-orbit) 15 RPM nominal (cruise)

[a] 7–8 apoapsis trim maneuvers during nominal mission lifetime planned to maintain periapsis altitude in the range 150–260 km (assumes Venus sphere of radius 6050 km).
[b] 8 periapsis trim maneuvers during nominal mission lifetime to maintain the 24 hr period to ±10 mins.
[c] Actual periapsis latitude depends on launch date: 16°N and 28°N appropriate to May 20 and June 2 launches, respectively.
[d] At orbit insertion, the subearth longitude is 38°; subsolar longitude is 268°.
[e] Positive spin-axis aligned with south ecliptic pole on-orbit; aligned with north ecliptic pole in cruise mode.

ionosphere and solar wind–ionosphere interaction region to extend and supplement the information obtained from the Multiprobe mission; (3) Studies of the planetary surface by remote sensing; (4) Determination of gravitational field harmonics from perturbations of the spacecraft orbit.

To meet objective (2) a periapsis altitude as low as possible is desirable. At this time, 150 km appears to be the lowest safe limit to assure the planned mission lifetime. Solar gravitational perturbations raise periapsis altitude with time, necessitating the trim maneuvers indicated in Table I to maintain periapsis altitude in the range 150–260 km. The dynamics of orbit insertion dictate a highly eccentric orbit and a large apoapsis distance. The actual apoapsis altitude was chosen to yield a 24-hr period, convenient for repetitive ground operations. The latitude of periapsis was selected for scientific purposes to be non-equatorial. A low-middle or middle latitude periapsis was considered most desirable. Launch constraints actually dictate the latitude range achievable for a northern hemisphere periapsis as indicated in Table I. For the scientific purposes of this mission, a northern or southern hemisphere periapsis location is immaterial. However, constraints imposed by instruments made the northern periapsis more desirable. A near-90° inclination is most desirable for global mapping of the surface, whereas a more equatorial inclination is more desirable for the radio occultation and *in situ* sampling devices. The 105° inclination chosen represents the only substantial compromise that had to be made. Figure 2 shows the altitude of the orbit above

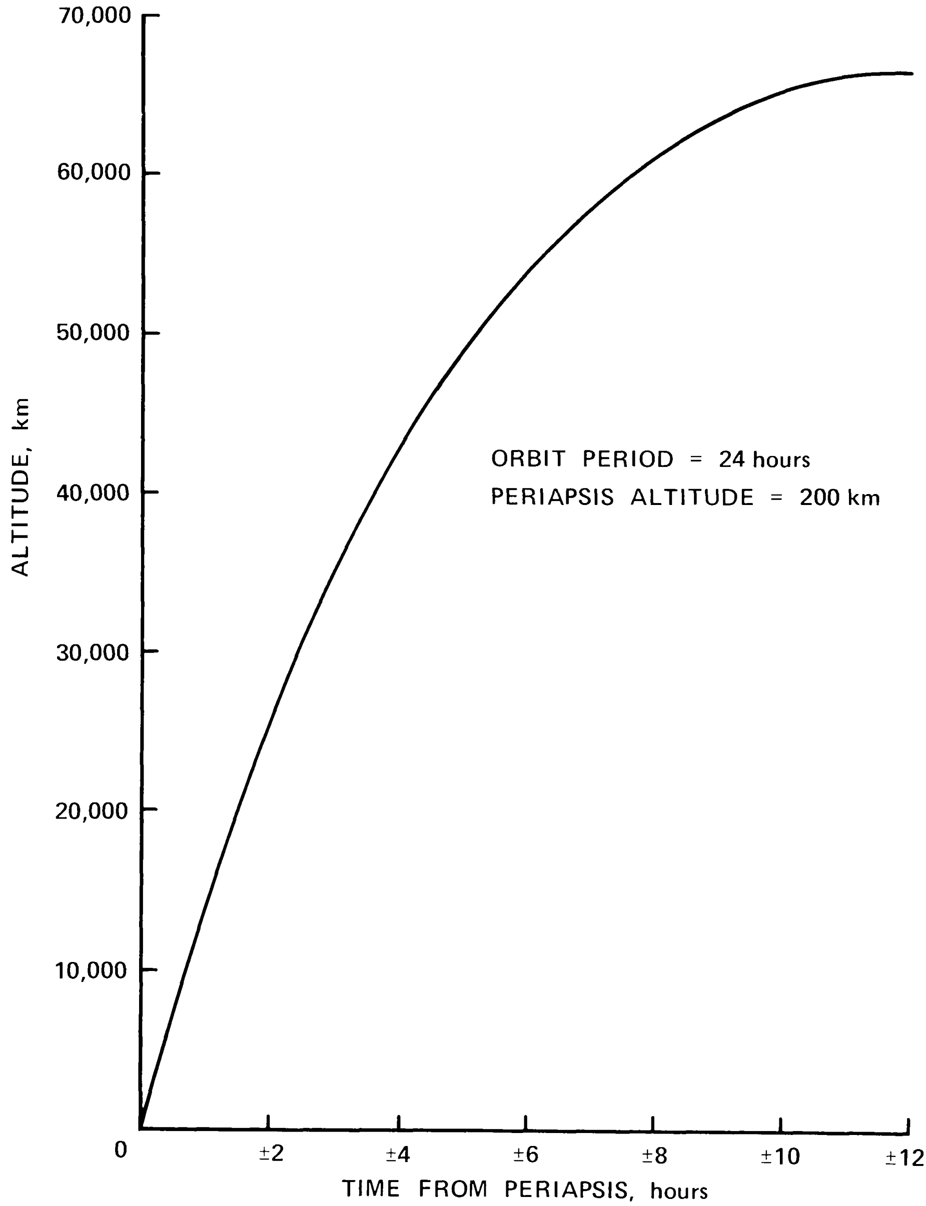

Fig. 2. Orbiter altitude versus time from periapsis.

the planet's surface as a function of time from periapsis. A total of 15 minutes is spent at altitudes below 900 km and about 30 minutes is spent below 2500 km.

At orbit insertion on December 4, 1978, periapsis occurs on the far, invisible side of Venus as viewed from Earth, in daylight some 25–44° from the evening terminator (assuming a nominal 24 May 1978 launch, this angle is 34° yielding a solar zenith angle of 60°). There are two seasons during the Orbiter's 243-day mission when the spacecraft will pass behind Venus and be occulted from Earth view. The first occurs during the initial 77–88 days in orbit, when the occulted section of the orbit is near periapsis. The second, shorter season, occurs near the middle of the in-orbit mission, when the occultations are longer because they occur when the spacecraft is near apoapsis. The duration of occultation during the first season varies up to 24 min and periapsis itself occurs in occultation throughout most of the season. The duration of occultation reaches as much as 3.5 hr in the second season. In the first season, the distance from spacecraft to occultation point is always less than 4000 km at immersion and 7000 km at emersion. In the second season, the corresponding distances are 68 000 km and 29 000 km. Throughout the first occultation season, the emersion occultation points vary with latitude over a broad range (60°S to 20°N). However the points are all in darkness (solar zenith angle range 110–170°). The immersion occultation points vary from about 30°N to 85°N in latitude and from about 80° to 150° in solar zenith angle. However the daytime points are only at the highest latitudes. The second occultation season offers much improved nighttime and daytime coverage over a broad range of latitudes.

Twelve scientific on-board instruments are included in the Orbiter payload. The experiments, Principal Investigators, and their current requirements for mass, volume, shelf footprint and average power are listed in Table II. Mnemonics identifying each experiment are also listed. Succeeding papers make extensive use of these shorthand titles to simplify discussions. In addition to the above experiments, six radio-science experiments will be performed using the S-band telemetry system and X-band beacon especially provided for this purpose. The

TABLE II
Pioneer Venus scientific instrument accommodations
Orbiter Mission

Instrument/Mnemonic	PI/Inst.	Mass lb (kg)	Volume in^3 (cm^3)	Shelf area in^2 (cm^2)	Avg. P watts
Neutral Mass Spectrometer/ONMS	H. Niemann/GSFC	9.5 (4.3)	260 (4260)	35 (229)	11.2
Ion Mass Spectrometer/OIMS	H. Taylor/GSFC	6.4 (2.9)	270 (4425)	33 (212)	1.5
Retarding Potential Analyzer/ORPA	W. Knudsen/LMSC	6.0 (2.7)	270 (4440)	28.5 (186)	2.8
Electron Temperature Probe/OETP	L. Brace/GSFC	4.4 (2.0)	125 (2055)	33 (216)	4.0
Ultraviolet Spectrometer/OUVS	A. Stewart/COLO	6.9 (3.1)	234 (3842)	42 (276)	1.7
Solar Wind Plasma Analyzer/OPA	J. Wolfe/ARC	8.1 (3.7)	550 (9015)	50 (330)	4.5
Magnetometer/OMAG	C. Russell/UCLA	4.4 (2.0)	240 (3934)	40 (258)	3.0
Infrared Radiometer/OIR	F. Taylor/JPL	11.0 (5.0)	540 (8850)	72 (471)	5.5
Cloud Photopolarimeter/Imager/OCPP	J. Hansen/GISS	8.8 (4.0)	300 (4916)	70 (452)	5.4
Radar Altimeter/ORAD	Team	24.0 (10.9)	700 (11475)	130 (850)	23.0
Electric Field Detector/OEFD	F. Scarf/TRW	1.5 (0.7)	67 (1092)	22 (144)	0.6
Gamma Burst Detector/OGBD	W. Evans/LASL	6.0 (2.7)	110 (1803)	26 (168)	1.8
TOTAL		97.0 (44.0)	3666 (60107)	581.5 (3692)	67.0

primary use of the latter is for the dual-frequency radio-occultation experiment. The radio scientists (mnemonics) involved are T. Croft, Stanford University (OGPE); A. Kliore, JPL (ORO); R. Phillips, JPL(OIUD); R. Woo, JPL(OTUR); I. Shapiro, MIT(OCM); G. Keating, LRC(OAD).

The spacecraft will accept information from the scientific instruments in serial digital, analog and bilevel form, convert the analog and bilevel information to serial digital form, and arrange all information in an appropriate format (major telemetry frames, composed of 64 minor frames) for time-multiplexed transmission to Earth. Each minor frame is composed of a series of 64 eight-bit words (512 bits per minor frame). The words in a frame are arranged into preprogrammed formats, selectable by command and independent of bit rate and mode of operation. Each frame will contain frame synchronization, sub-commutated data, high rate scientific instrument data and spacecraft data. Table III lists the word allocations for each of the Orbiter minor frame formats for each scientific instrument. Low-rate science and housekeeping data will be contained within a subcommutated format. One of the three available subcommutated formats contain science words as shown also in Table III.

The data subsystem will be capable of operating in two basic modes: real time and telemetry storage. An on-board data storage capacity of 1 048 576(2^{20}) bits will be provided by the data handling subsystem for both scientific and engineering data. Eleven telemetry storage playback and real-time data transmission rates between 8 and 2048 bps are available on-orbit; a transmission rate of 1024 bps is used during cruise. It should be noted that no restrictions are imposed on the use of any of the 'Periapsis' or 'Apoapsis' formats listed in Table III. Any format may be selected by command (at any of the 11 transmission rates between 8 and 2048 bps) for use over any segment of the orbit. However, because of instrument design limitations certain format/bit rate combinations are preferred and probably will be used predominantly.

TABLE III
Pioneer Venus Orbiter data word assignments
Words/Frame

Format Exp	Periapsis A (Aeronomy)	Periapsis B (Aeronomy)	Periapsis C (All up)	Periapsis D (Optical)	Periapsis E (Mapping)	Apoapsis A (Imaging)	Apoapsis B (General)	Playback	Launch Cruise	Bits/ Frame Subcom D
ONMS	—	14	6	—	—	—	1	—	—	17
OIMS	18	9	9	—	—	—	2	—	1	65
ORPA	5	3	3	—	—	—	2	—	1	42
OETP	10	8	5	—	—	—	3	—	—	18
OUVS	11	7	7	—	18	—	1	1	7	41
OPA	3	2	3	—	—	3	12	5	3	17
OMAG	4	4	4	—	4	4	12	4	4	41
OIR	—	—	4	47	4	—	—	—	—	41
OCPP	—	8	—	8	—	43	—	—	—	33
ORAD	—	—	10	—	28	—	1	—	—	54
OEFD	4	—	4	—	1	4	4	4	4	1
OGBD	—	—	—	—	—	1	17	1	1	82

3. Multiprobe Mission

The broad scientific objectives of the Multiprobe mission are to study (1) the nature and composition of the clouds, (2) the composition and structure of the atmosphere from the surface to high altitudes, (3) the general circulation pattern of the atmosphere, and (4) the characteristics of the planetary environmental interaction with the solar wind. The scientific instruments and the targeting of the bus, large probe and three small probes are presented below. The major scientific questions associated with the above broad objectives and the way that the scientific instruments address those questions are discussed in subsequent papers.

The targeting capabilities at Venus of the five entry vehicles are quite flexible and final selection of the target entry points will be made later in the program. The most likely of many options according to current planning is described herein. Probe targeting and release from the bus are illustrated in Figure 3. About 28 days from the planet the bus is reoriented from its cruise attitude to accommodate spacecraft communication. Four days later, the explosive devices attaching the large probe to the bus are fired, and separation springs jettison the probe. Then about 20 days from the planet and after reorienting the bus to accommodate the small probe targeting, the bus spin rate is increased to a value appropriate to target selection and the three small probes are released simultaneously at a precise point in the spin cycle. Finally the bus is retargeted for its entry.

The probe entry sequences are quite complex. Some 20 min before entry all subsystems and instruments are activated, direct communication links with Earth are established, and the probe mission begins. The initial large probe sequence is illustrated in Figure 4. At about 100 km altitude, atmospheric braking has already begun, and a data storage mode is activated in which 3-axis acceleration and heat shield temperature data are stored for later playback. Peak deceleration, 320 g, occurs at about 78 km. On the down side of the deceleration pulse, a g-switch ends the storage mode and initiates a timing sequence in which the aeroshell and heat shield are jettisoned as follows. Just below 68 km altitude a pilot chute is mortared from the side of the vehicle. The pilot chute in turn removes an aft protective cover and extracts the main parachute. After a suitable time delay for stabilization, mechanical and electrical ties to the aeroshell are broken and the

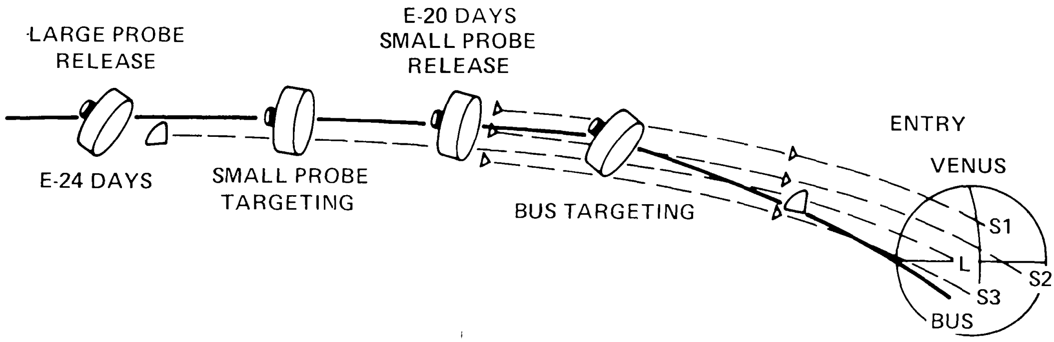

Fig. 3. Multiprobe mission probe release sequence.

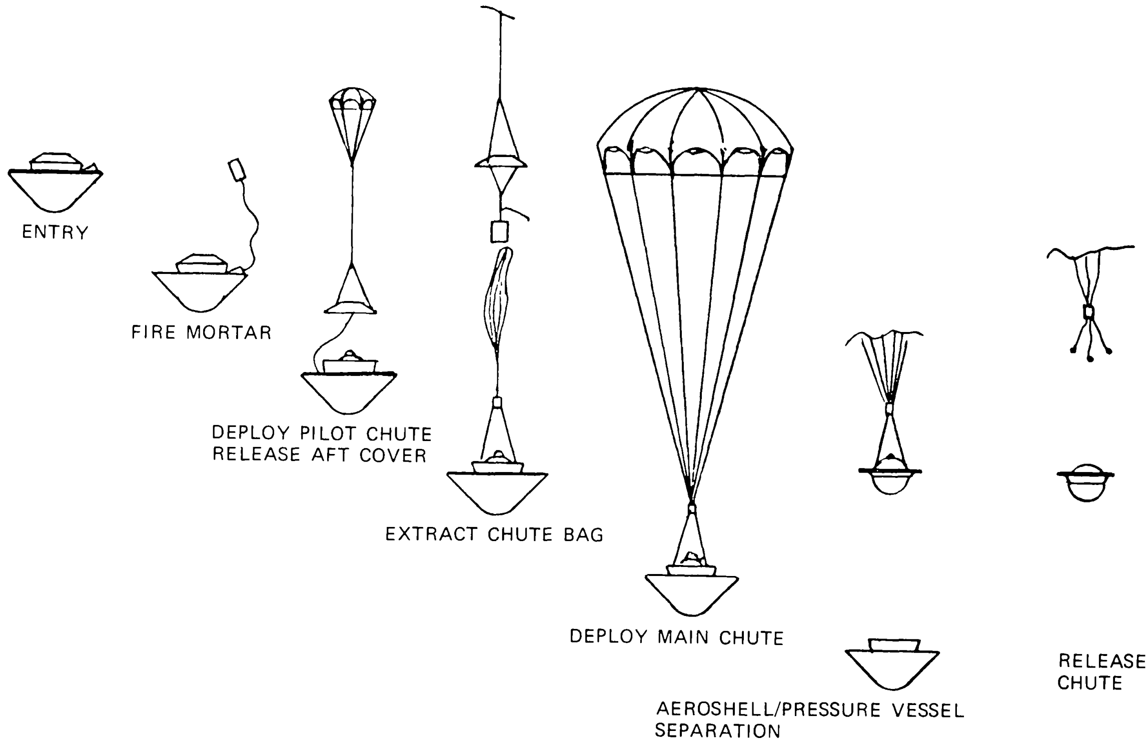

Fig. 4. Large probe parachute deployment sequence.

spherical pressure vessel which houses all of the instruments and probe subsystems is extracted by the parachute. This separation gives instruments access to the atmosphere, so that at an altitude of 67 km *in situ* scientific examination of the atmosphere has begun. The parachute remains attached to the pressure vessel for the next 17 min to about 47 km altitude, at which point the parachute must also be jettisoned. The pressure vessel then free falls in a stable configuration to the surface, which it impacts about 39 min later.

The descent sequence for the small probes differs from that of the large probe in that the small probes are despun just before entry from spin rates necessary for targeting to rates low enough to permit rapid convergence of angle of attack. This is necessary to prevent excessive base heating and is accomplished using a 4:1 yo-yo despin system. The descent also differs from that of the large probe in that the pressure vessel is not separated from the aeroshell nor is a parachute employed. Instrument access to the atmosphere is provided by opening covers on the vehicle afterbody. As in the large probe sequence, a data storage mode is terminated by a g-switch during the entry deceleration pulse. Peak decelerations range from 200 g to 565 g, because of the range in entry flight path angles. Instrument deployment occurs after peak deceleration, initiated by a g-switch signal, in the altitude range 65 to 72 km, depending on probe targeting. Descent from this point to the surface requires about 57 min. Typical altitude-time histories of the large and small probes are shown in Figure 5.

A prime example of the many targeting possibilities for the bus, large probe and small probes is shown in Figure 6. Data for this case are given in Table IV.

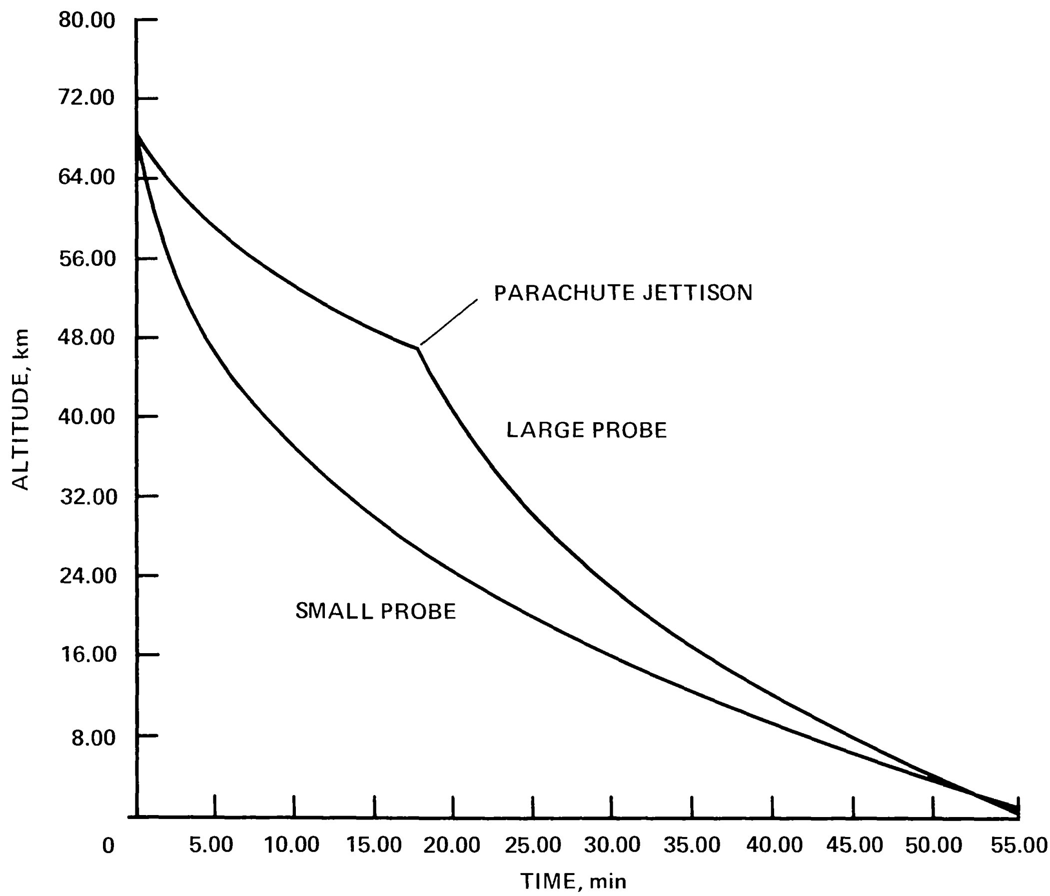

Fig. 5. Large and small Probe descent histories.

Regardless of the precise targeting, it is expected that three vehicles (bus, large probe, small probe 3) will be targeted for dayside entry and two vehicles (small probes 1, 2) will be targeted for nightside entry. The particular small probe targeting illustrated maximizes the latitude and longitude dispersion within the geometric and communications restraints of the mission and is the current 'strawman' option chosen by the PVSSG. The bus instruments will be turned on four hours before entry and measurements will be made in the upper atmosphere above its burn-up altitude of ~115 km. It will be targeted to a shallow ($\lesssim 10°$) entry flight path angle and will be oriented for 5° entry attack angle at 200 km to ensure adequate altitude resolution for its scientific measurements along its slant path at the high entry speeds (11.6 km s^{-1}).

Seven instruments are on the large probe, three on each of the identical three small probes, and two on the bus. The instruments, Principal Investigators and their requirements for mass, volume, shelf footprint, average power and data rate are listed in Table V. In addition to the above experiments, a set of radio-science experiments will be performed using all entry vehicles and their associated S-band telemetry systems. Each entry vehicle transmits directly to Earth; transponder links are provided on the bus and large probe, whereas stable oscillators are on each small probe. The radio scientists (mnemonics) involved are G. Pettengill,

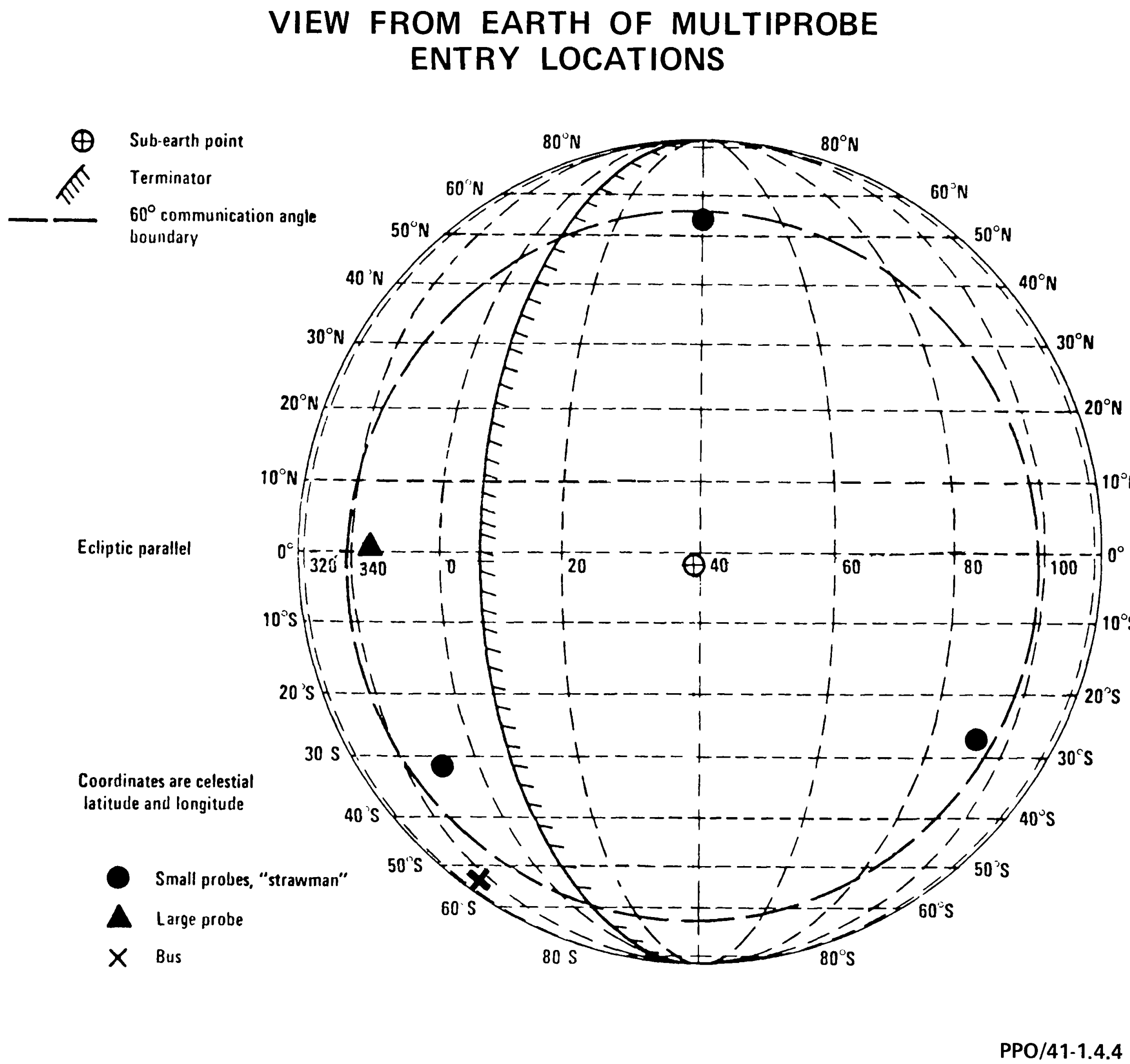

Fig. 6. View from Earth of Multiprobe entry locations.

TABLE IV
Pioneer Venus Multiprobe targeting data

Vehicle	Celestial[a] latitude–deg.	Celestial[a] long.–deg.	Solar zenith angle–deg.	Comm. angle–deg.	Relative[d] entry time
Bus	−52.7	334.2	70.5[c]	2.6[c]	E+90 min
Large Probe	−3.5	347.0	71.1	52.3[b]	E+0.0
Small Probe 1	56.5	40.2	109.1	58.7[b]	E+3.0 min
Small Probe 2	−28.3	93.7	150.5	58.0[b]	E+8.7 min
Small Probe 3	−31.7	353.7	79.0	52.0[b]	E+4.0 min

[a] 70 km altitude; Sub-solar point longitude = 275.9°; Morning terminator longitude = 5.9°; Anti-solar point longitude = 95.9°.
[b] Vertical descent assumed; angle between vertical and direction to Earth.
[c] Slant entry at 180 km; communication angle between spin axis direction and direction to Earth.
[d] E = 0 at 200 km.

TABLE V
Pioneer Venus scientific instrument accommodations
Multiprobe Mission

V	Instrument/Mnemonic	PI/Inst.	Mass lb (kg)	Volume in³ (cm³)	Shelf area in² (cm²)	Avg. P watts	Descent Descent bps
LP	Neutral Mass Spectrometer/LNMS	J. Hoffman/UTD	22.9 (10.4)	659 (10800)	155 (1000)	14.0	44
LP	Gas Chromatograph/LGC	V. Oyama/ARC	13.8 (6.3)	576 (9440)	87 (561)	42.0	30.5
LP	Atmosphere Structure/LAS	A. Seiff/ARC	5.1 (2.3)	90 (1475)	29 (187)	4.9	26
LP	Solar Radiometer LSFR	M. Tomasko/Ariz.	4.5 (2.0)	100 (1640)	25 (161)	4.0	21
LP	Infrared Radiometer/LIR	R. Boese/ARC	5.8 (2.6)	165 (2700)	30 (196)	5.5	16
LP	Cloud Particle Size Spectrometer/LCPS	R. Knollenberg/PMS	9.6 (4.4)	260 (4260)	50 (323)	20.0	52
LP	Nephelometer/LN	B. Ragent/ARC & J. Blamont/CNES	2.8 (1.3)	80 (1310)	18 (116)	2.4	38
		TOTAL	64.6 (29.3)	1930 (31625)	394 (2544)	92.8	227.5
SP	Atmosphere Structure/SAS	A. Seiff/ARC	2.7 (1.2)	70 (1150)	22 (142)	3.6	11/5.25
SP	Nephelometer/SN	B. Ragent/ARC & J. Blamont/CNES	2.9 (1.3)	80 (1310)	18 (118)	2.4	20/7.25
SP	Net Flux Radiometer/SNFR	V. Suomi/WISC.	2.2 (1.0)	40 (650)	18 (118)	3.8	6/1.5
		TOTAL	7.7 (3.5)	190 (3110)	58 (378)	9.8	37/14
r	Neutral Mass Spectrometer/BNMS	U. von Zahn/Bonn	15.0 (6.8)	580 (9505)	37 (239)	5.0	512
	Ion Mass Spectrometer/BIMS	H. Taylor/GSFC	6.4 (2.9)	270 (4425)	33 (212)	1.5	224
		TOTAL	21.4 (9.7)	850 (13930)	70 (451)	6.5	736

MIT(DLBI): C. C. Counselman, MIT(DLBI); T. Croft, Stanford University(MPRO); A. Kliore, JPL(MWIN); and R. Woo, JPL(MTUR).

The bus, large probe and small probe spacecraft will accept information from the scientific instruments in serial digital, analog and bilevel digital form, convert the analog and bilevel data to serial digital form and arrange all information in appropriate formats (major telemetry frames, composed of 64 minor frames for the bus and 16 minor frames for the probes) for time-multiplexed transmission to Earth. Each minor frame is composed of a series of 64 eight-bit words (512 bits per minor frame).

The data handling system for the bus provides for minor frames containing frame synchronization, subcommutated data, high rate scientific instrument data and spacecraft data. The words in a frame are arranged into preprogrammed formats, selectable by ground command and independent of bit rate and mode of operation. A single entry format containing high rate scientific instrument data is provided on the bus. In addition, a low rate subcommutator channel contains some additional scientific data. Eleven real-time (no data storage is provided on the bus) data transmission rates between 8 and 2048 bps are available on entry. The high data rate allocation for each of the two scientific instruments is shown in Table V, assuming a total spacecraft rate of 1024 bps.

The data handling system for the large probe provides for two high rate data mode formats: entry and descent. A storage capacity of 3072 bits is provided by a data storage memory. Data will be stored during RF blackout. Following the blackout period, the stored data will be read out of the memory and telemetered in the descent format. When data are being stored, storage and real-time transmission will be at 128 bps. When not storing data, i.e. during the descent format phase, real-time transmission will occur at 256 bps. Allocation of this bit rate among the seven large probe instruments is shown in Table V. Only the LAS and LN employ the entry format at 72 bps and 4 bps, respectively. Two low rate subcommutator channels also provide additional data for the LAS, LN, LCPS and LSFR.

The data handling system for each small probe provides for three high rate data mode formats: upper descent, lower descent and entry. As for the large probe, a storage capacity of 3072 bits is provided by a data storage memory. Following the blackout period, the stored data will be read out of the memory and telemetered in the upper descent format. When data are being stored, storage and real-time transmission will be at 64 bps. When not storing data, real-time transmission will occur at 64 bps initially (upper descent format) changing to 16 bps at 30 km altitude (lower descent format). Allocation of the 64 bps and 16 bps among the three small probe instruments is shown in Table V. Only the SAS and SN employ the entry format at 36 bps and 2 bps, respectively. Two low rate subcommutator channels also provide additional data for all three instruments.

4. Pioneer Venus Science Steering Group

The PVSSG is composed of the Principal Investigators listed in Tables II and V, plus G. Pettengill, MIT (Radioscience Team and Radar Team leader), C. C. Counselman, MIT (DLBI PI) and a group of interdisciplinary scientists or theorists. The latter group was chosen in response to the same Announcements of Opportunity that led to selection of the Orbiter and Multiprobe payloads, and represents NASA's first attempt to include, in planetary mission definition, development, execution and data interpretation, scientists who do not have prime responsibility for an experiment. The successful results to date support this approach for future missions. The interdisciplinary scientists on Pioneer Venus are T. Donahue, Michigan University; D. Hunten, Kitt Peak National Observatory; J. Pollack, Ames Research Center; A. Nagy, Michigan University; N. Spencer, Goddard Space Flight Center; S. Bauer, Goddard Space Flight Center; R. Goody, Harvard University; G. Schubert, UCLA; H. Masursky, USGS; G. McGill, Massachusetts University. T. Donahue serves as Chairman of the PVSSG and D. Hunten, L. Colin (Project Scientist) and R. Fellows (Program Scientist) serve as co-chairmen.

The PVSSG is structured into Working Groups which address key scientific questions associated with these Venusian missions. Each of the above-named PVSSG members plus the radio scientists serve on at most two of the Working Groups. The Working Groups (Chairmen) are listed below: Atmospheric Composition and Structure (J. Hoffman), Cloud Composition and Structure (R. Knollenberg), Atmosphere Dynamics (G. Schubert), Thermal Balance (V. Suomi), Ionosphere and Solar Wind (S. Bauer), Surface and Interior (H. Masursky). The next six papers are the contributions to this issue of each Working Group.

5. Orbiter Spacecraft

The Orbiter spacecraft, shown in Figure 7, provides a spin-stabilized platform for the twelve scientific instruments selected for this mission. A circular equipment shelf having an area of approximately 50 ft^2 (4.37 m^2) is mounted on the forward end of a thrust tube. Twelve equally spaced struts support the shelf perimeter from the lower end of the thrust tube. The cylindrical solar array is attached to the equipment shelf with 24 brackets. All of the scientific instruments and the spacecraft electronic units are mounted directly to the top side of the shelf as shown in Figure 7. The OMAG sensor is mounted on a three link 15.5 ft (4.72 m) boom which is deployed from the spacecraft cylinder. The Orbiter spacecraft includes a 43 in (109 cm) diameter mechanically despun S/X band parabolic reflector antenna. A sleeve dipole antenna serves as a backup to the high gain antenna. A forward omni antenna and an aft omni antenna complete the spacecraft antenna array. The S/X-band dual-frequency radio occultation experiment is accommodated with a 750 mW X band transmitter and a high gain antenna elevation positioner. A quadripod structure at the upper end of the thrust tube supports a Bearing and Power Transfer Assembly (BAPTA) which mechanically despins the high gain antenna reflector, S/X band dual feed, and the forward omni and sleeve dipole antennas. The high gain antenna mast is attached to the despun flange of the bearing assembly to elevate the high gain antenna line of sight above the solar array with adequate clearances above the ORAD instrument which is located on the equipment shelf. The radar antenna is located at the outer edge of the spacecraft above the forward plane of the solar array to allow a radar beam pointing of 140° from the spin axis, and to maintain beam clearance of spacecraft structure.

Fifteen thermal louver modules, attached to the lower surface of the shelf, control thermal radiation out of the aft cavity. Large energy dissipators such as the RF power amplifiers are located directly over several of these louvers.

Beneath the equipment shelf, supported from the thrust tube, are two conispheric hydrazine propellant tanks 12.8 in (33 cm) in diameter. These tanks store the propellant necessary to activate the three axial and four radial thrusters which effect commanded changes in spacecraft attitude, velocity and spin rate. The thrusters are attached through support structures to the equipment shelf and provide redundancy in trajectory and attitude control. A star sensor, mounted on the shelf at a nominal angle of 56° to the spin axis, and sun sensors provide the required attitude references.

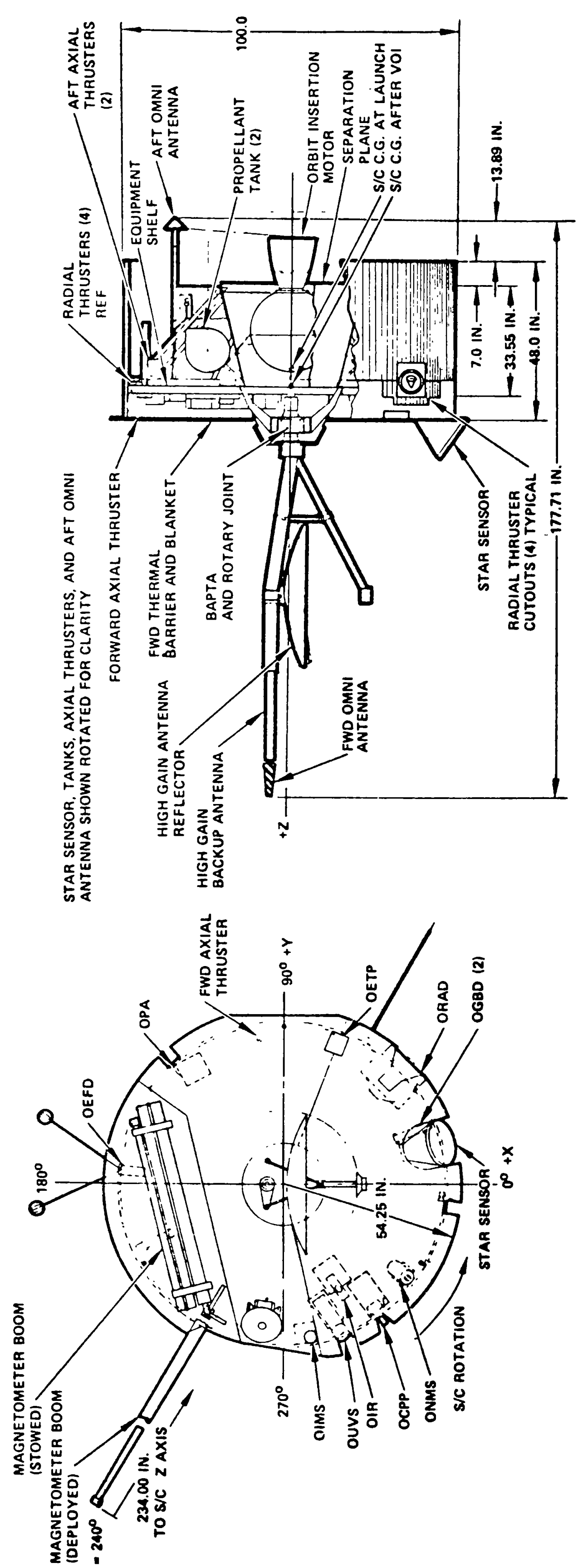

Fig. 7. Orbiter configuration.

The orbit insertion motor is a solid propellant motor, attached to the thrust tube and is centrally located below the equipment shelf with its nozzle extending into the spacecraft attach fitting to the launch vehicle.

The power system provides 28 V DC ±10 percent electrical power to the scientific instruments and spacecraft subsystems. Seven resistive shunt limiters clamp the maximum panel voltage at 30.8 V DC. When the bus voltage drops below 27.8 V DC the batteries start to share the load through discharge controllers. Small solar boost charge arrays recharge the batteries at a constant current rate. The overload/undervoltage switch protects the main or essential power bus from a high current overload or an undervoltage condition by automatically tripping off the scientific instruments, switched loads, and transmitter buses. The power interface unit provides on/off switching of propulsion heaters. The solar array, consists of 7.4 m^2 of 2×2 cm cells and provides, when normal to the sunline, 226 W near Earth and 312 W at Venus. The two 7.5 A-hr Ni–Cad batteries will provide 252 W-hr at 60 percent depth of discharge. Power is provided to the instruments from the science power bus through redundant fuses located in the power interface unit. On/off power switching, in lieu of centralized switching in the power interface unit, is performed in the individual instruments for growth flexibility.

The communications subsystem provides a command uplink, in any spacecraft attitude, through two S band transponders connected to two omni antennas. The receivers are frequency addressed and automatically reversed by the central decoders if no command is received for 36 hours. The two receiver outputs are cross-strapped to redundant exciters; the active exciter is selected by ground command. The transponder provides a coherent phase-locked VCO carrier for two-way lock or a local auxiliary carrier for one-way lock. The downlink is provided by an S band transmitter with redundant power amplifiers operating through one of two omnis, a high gain antenna, or a sleeve dipole. Any of these antennas may be selected by ground command. One omni antenna is permanently connected to one of the two receivers and one of three despun antennas is connected to the other receiver by a series of transfer switches through the dual frequency rotary joint. Pulse commands to these switches are provided through the BAPTA sliprings and brushes. Any one of the four antennas can be selected for the downlink by ground command. An X band transmitter provides a signal that is coherent with the S band transmitter in the ratio 11/3. Its output is routed through the rotary joint to the despun antenna. The X band signal has no modulation capability. Dual frequency S/X band RF occultation is provided by an antenna that is steered in azimuth and elevation. The system provides capability for operation to refractive angles of 17°. The beamwidths are 2.2° at X band, 7.6° at S band.

The command subsystem accepts the receiver baseband signal. The command demodulators convert this signal to NRZ and apply it to cross-strapped command processors. Commands are either stored for later execution or executed immediately. All Orbiter units receive command from redundant command output modules. The command subsystem accepts a PCM/FSK/PM data stream at 4 bps. The command word consists of 48 bits including 13 bits for synchronization in order to satisfy the requirement for less than 10^{-9} probability of false command execution. A total of 192 pulse commands and 12 magnitude commands are available. The command memory can store up to 128 commands (redundantly) for later execution. The pyrotechnic control units provide redundant initiator firing current to perform magnetometer boom deployment, orbit insertion motor ignition, and scientific instrument mechanism deployment. The orbit insertion motor igniter has a motor actuated safe and arm device. This device is armed from a blockhouse power supply just prior to liftoff.

The data handling subsystem telemetry processor sequentially samples each telemetry measurement by means of an instruction word to the PCM encoder which addresses a data module to read out the selected channel. The interrogated channel can be either analog, serial digital, or binary discrete information. The PCM encoder ships the encoded measurement to the telemetry processor where it is frame formatted, convolutionally encoded, and used to biphase modulate a subcarrier. This subcarrier then phase modulates the downlink carrier. The telemetry processors and PCM encoders are cross-strapped and fully redundant. Critical telemetry measurements are assigned data channels on two different data modules. The data handling subsystem can accept up to 256 channels of analog, serial digital, or bilevel discrete data. Analog data will be digitized into eight-bit words. The data input is multiplexed and formatted into frames of 64 eight-bit words; of the 64 words, three are required for sync and ID and three are subcommutated for housekeeping data. The output of the subsystem is an 8 to 2048 bps PCM/PSK convolutionally encoded data stream biphase modulated on a 16 384 Hz subcarrier. A data storage capability of 1.048×10^6 bits is available (12.5% reserved for spacecraft engineering overhead) to effect data return during occulted periapsis passes early in the mission and to match the 24 hr per day collection to the desirable 4 hr per day transmission period. Stored apoapsis data (729 kilobits) and periapsis data (987 kilobits) are played back at a minimum of 170 bps (maximum range) before and after a periapsis pass satisfying the desirable four hour DSN daily usage limitation.

The controls subsystem has one mid-range sun sensor, two extended range sun sensors, and a star sensor to sense inertial attitude and provide a spin angle reference. Redundant attitude data processor units format the sensor outputs for telemetry transmission to the Earth for attitude determination. The attitude data processor also provides sequenced thruster firing commands to the axial and radial thrusters through solenoid drivers to effect attitude, velocity, and spin rate changes. A velocity change can be accomplished in either a continuous firing or pulsed firing mode. The continuous mode uses axial thrusters and the pulsed mode uses two radial thrusters. Spin axis precession can be accomplished by pulsing axial thrusters. Spin rate can be adjusted by continuous firing of two radial thrusters. Specific velocity change maneuvers at a fixed orientation in space can be accomplished by either (1) reorienting the spin axis to the required direction and firing an axial jet in the continuous mode or (2) firing an axial jet in a continuous mode and two radial jets in a pulsed mode to form the desired velocity vector change without changing the spin axis orientation. The controls subsystem provides redundant despin control electronics units to drive one of two redundant BAPTA motors to despin and point the high gain antenna toward Earth. The despin control system functions as a closed loop, autonomously operating the system to maintain antenna pointing.

Motor torque commands are generated by the despin control electronics based upon the sun or star sensor and the BAPTA master index pulses. Ground commands control the operating mode and antenna pointing. An elevation drive mechanism is provided for the despun antenna for RF occultation pointing. The positioner control electronics converts rate commands into discrete pulses, which power the elevation stepper motor to drive a linear jackscrew actuator. The antenna beam is thereby positioned over a range of $\pm 15°$ from normal to the spin axis.

Redundant separation switches sense launch vehicle separation. These switches generate a telemetry indication and also initiate a stored command sequence that fires selected radial thrusters which provide an initial spinup to about 5 RPM.

6. Multiprobe Spacecraft

The spacecraft for the Multiprobe mission, shown in Figure 8, is composed of a bus, a large probe and three identical small probes. The bus uses the same basic structure and most of the same spacecraft units as the Orbiter. The BIMS is mounted atop a support structure above the equipment shelf to provide the required clear field-of-view. The BNMS is positioned on the shelf to achieve an unobstructed field-of-view from the spacecraft spin axis.

An inverted right conical structure with a radial secondary structure is installed at the upper end of the thrust tube to accommodate the probes. The large probe is located on the bus spin axis and is spring separated in the axial direction. The small probes are supported by hinged clamps attached to the radial structure and are released in a tangential direction.

The bus is fitted with a fixed antenna configuration consisting of an omni attached to and extending above the equipment shelf, an omni located below the shelf and a medium gain horn antenna attached to the aft side of the shelf.

The bus power subsystem is similar to that for the Orbiter spacecraft and provides 28 V DC $\pm$ 10% electrical power to the scientific instruments and spacecraft subsystems. Five dissipative shunt limiters clamp the maximum voltage at 30.8 V DC. The solar array consists of 6.9 m^2 of 2×2 cm cells and provides 214 W near Earth and 241 W at Venus.

The bus communications subsystem also is similar to that for the Orbiter. The primary differences are in the antenna arrangement. The bus command and data

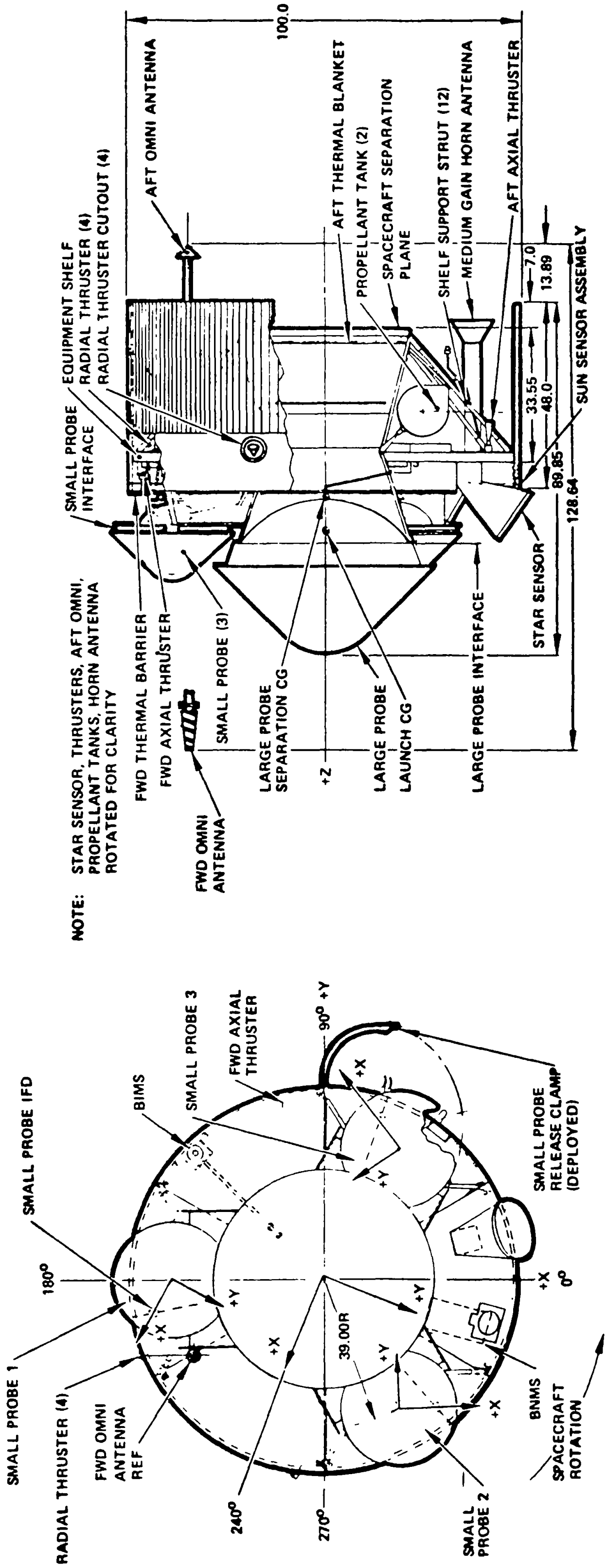

Fig. 8. Multiprobe configuration.

handling subsystems are similar to those on the Orbiter. The major difference is that the bus does not incorporate a data storage unit. The bus controls subsystem is similar to that for the Orbiter.

The large and small probes are geometrically similar vehicles (see Figures 9 and 10). Each incorporates a spherical pressure vessel which houses the scientific instruments and all spacecraft subsystems: communications, data, command, and power. Entry stabilization, and heat protection are provided by conical forebodies having base diameters of 56 in (142 cm) for the large probe and 30 in (76 cm) for the small probe. The entry configuration of both probes is a 45° half-angle cone with a spherically blunted tip whose radius is equal to half the base radius.

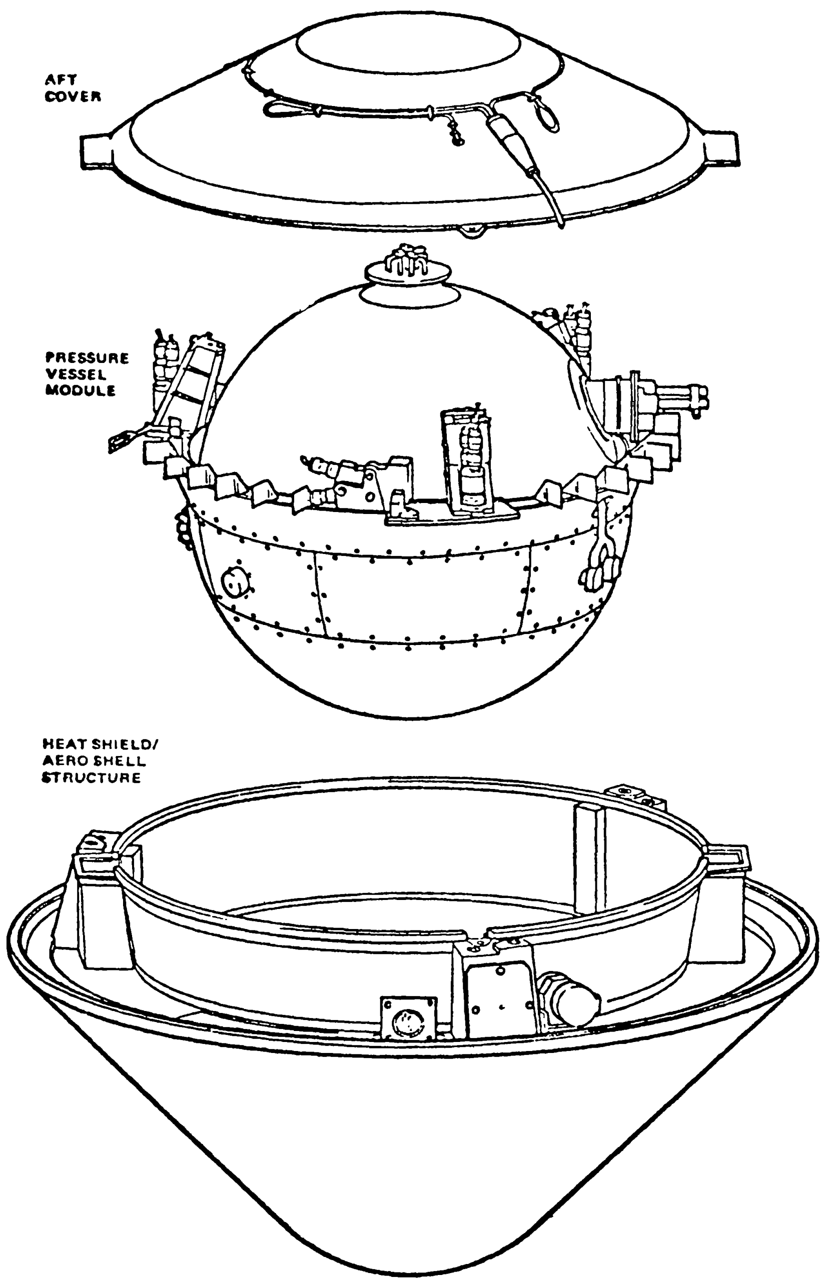

Fig. 9. Large Probe external configuration.

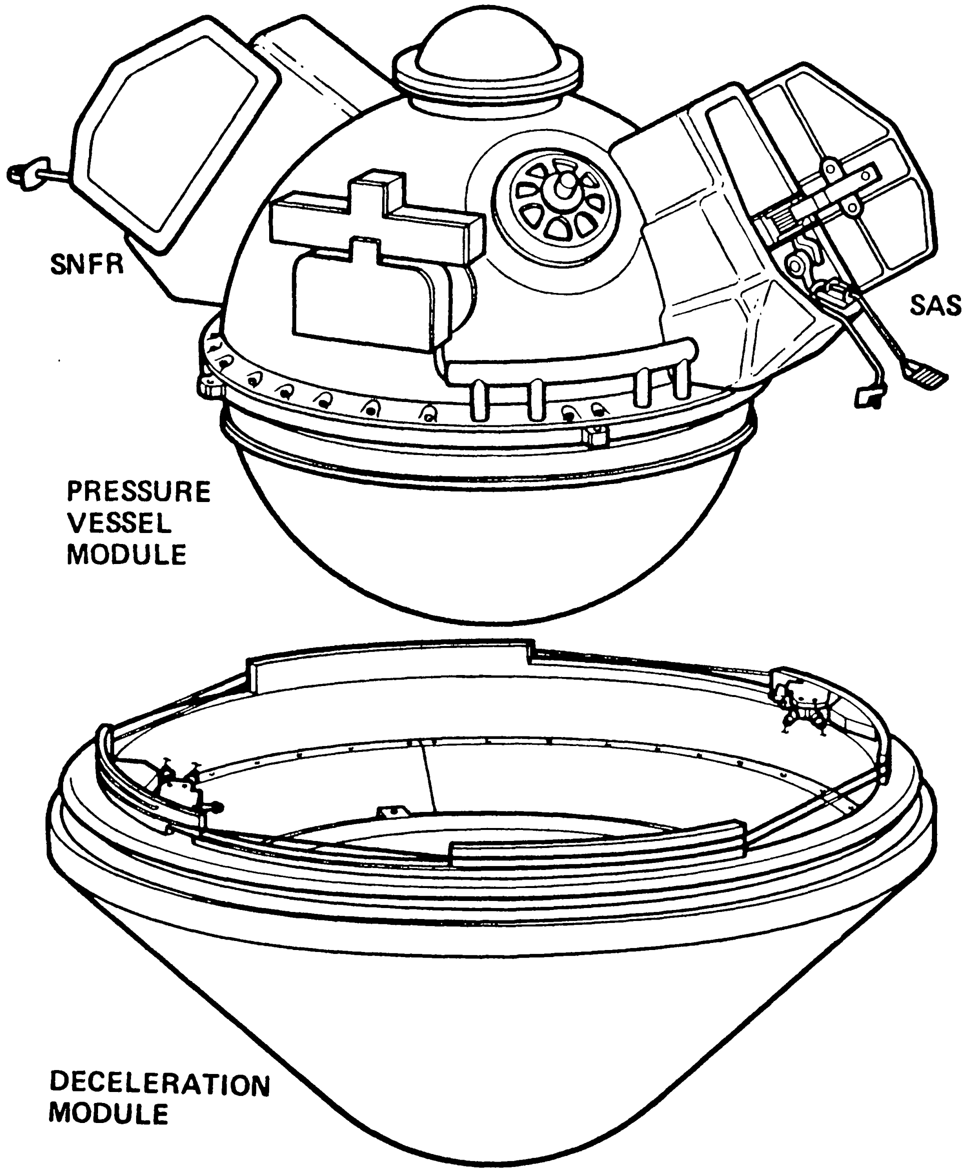

Fig. 10. Small Probe external configuration.

Thermal protection during entry is provided by a carbon phenolic forebody heat shield and a low density elastomeric coating over all after body surfaces. The large probe aeroshell is a semi-monocoque structure with integrally machined stiffening rings. Because it is jettisoned at parachute deployment it does not have to withstand the 900°F temperature of the lower atmosphere and consequently is fabricated of aluminum. The small probe aeroshell, on the other hand, remains with the pressure vessel to the surface and is therefore made of titanium. It is also a pure monocoque structure. The main parachute is extracted from its compartment in the aeroshell by a 2.5 ft (76 cm) diameter mortar deployed pilot chute.

A cross-sectional view of the large probe pressure vessel module is shown in Figure 11. The scientific instruments and probe subsystems are mounted on two

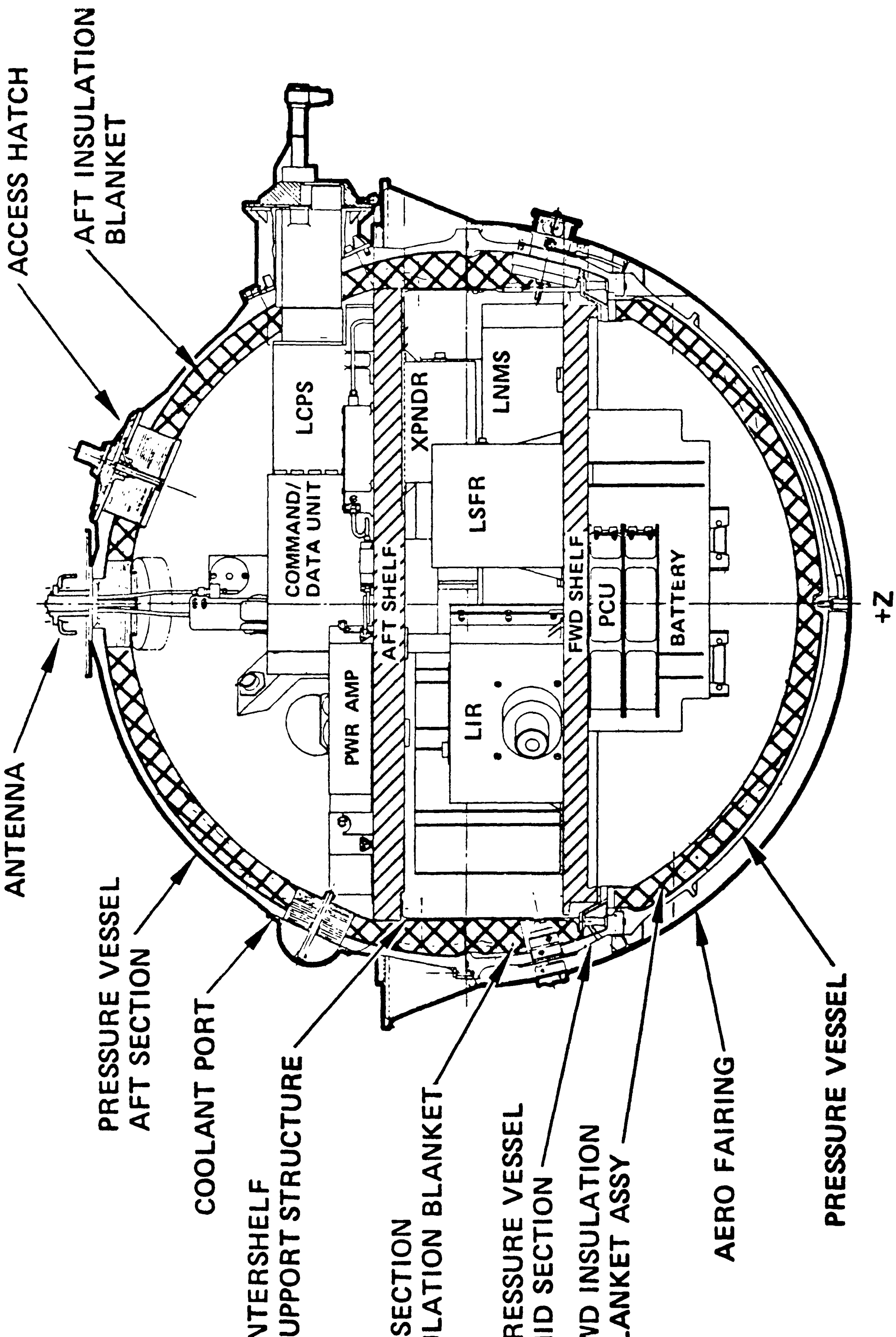

Fig. 11. Large probe pressure vessel internal arrangement.

shelves inside an internally-insulated, spherical pressure vessel structure. A flow separation ring is installed aft of the vehicle center of gravity to provide aerodynamic stability and to serve as the attach structure between the pressure vessel and the aeroshell. It also carries the small vanes used to provide vehicle roll during descent. Drag plates in three sectors of the ring tailor descent time to meet science requirements. The large probe pressure vessel structure is a 28.8 in (73.2 cm) diameter, spherical titanium shell of thin wall thickness. Equipment inside the pressure vessel module is protected thermally by a 1 in (2.5 cm) thick aluminized kapton blanket. The rest of the thermal control system consists of two beryllium equipment shelves which serve as heat sinks. Fifteen penetrations of the pressure vessel are required for windows, inlets, feedthroughs, etc.

A cross-sectional view of the small probe pressure vessel module is shown in Figure 12. This module is similar in concept to that for the large probe. The scientific instruments and probe subsystems are mounted on two beryllium heat sink shelves inside an internally-insulated (kapton), spherical titanium pressure vessel structure (18″ ID). Eight penetrations of the pressure vessel are required on the small probe. Scientific accommodation in the probes is one of the major engineering challenges of this program. All of the instruments require access to the atmosphere, either to view it or sample it directly, and yet all must be protected from the adverse effects of the atmosphere.

The probe communication subsystems provide direct, DSN-compatible downlinks and employ solid state transmitters and hemispherical coverage antennas. The large probe subsystem has the capability for a 256 bps data stream using four 10 W solid state output amplifiers which deliver 44.5 dBm at the antenna. It has a transponder which provides transmission at 2.3 GHz and reception at 2.1 GHz. The receiver is included for two-way doppler tracking only and does not receive commands. The small probe communication subsystem is similar to that for the large probe in that it employs an identical antenna and power amplifier, but differs in that it uses only one output amplifier and has no receiver. Nominal RF power is 29.7 dBm at the antenna which will support a data rate of 64 bps from entry to 30 km altitude and 16 bps below that to impact. One-way doppler tracking is provided using a stable oscillator as reference frequency source.

The probe command/data subsystems consist of a command unit, a pyrotechnic control unit, a data handling unit, and the sensors necessary to service the command unit. The units are identical for all probes. The command portion provides 64 discrete commands for spacecraft and scientific instruments. It contains the cruise timer (the only operating probe unit during the 24 day period between bus separation and entry), an entry sequence programmer, and a command decoder. Commands are initiated by a clock generator or a g-switch. A pressure switch provides backup for the timer at large probe parachute jettison. The pyrotechnic control unit is made up of twelve squib drivers that provide 4.5 A to each initiator. The data handling unit accepts 36 analog, 12 digital, and 24 bilevel channels from science instruments and probe subsystems. The unit also

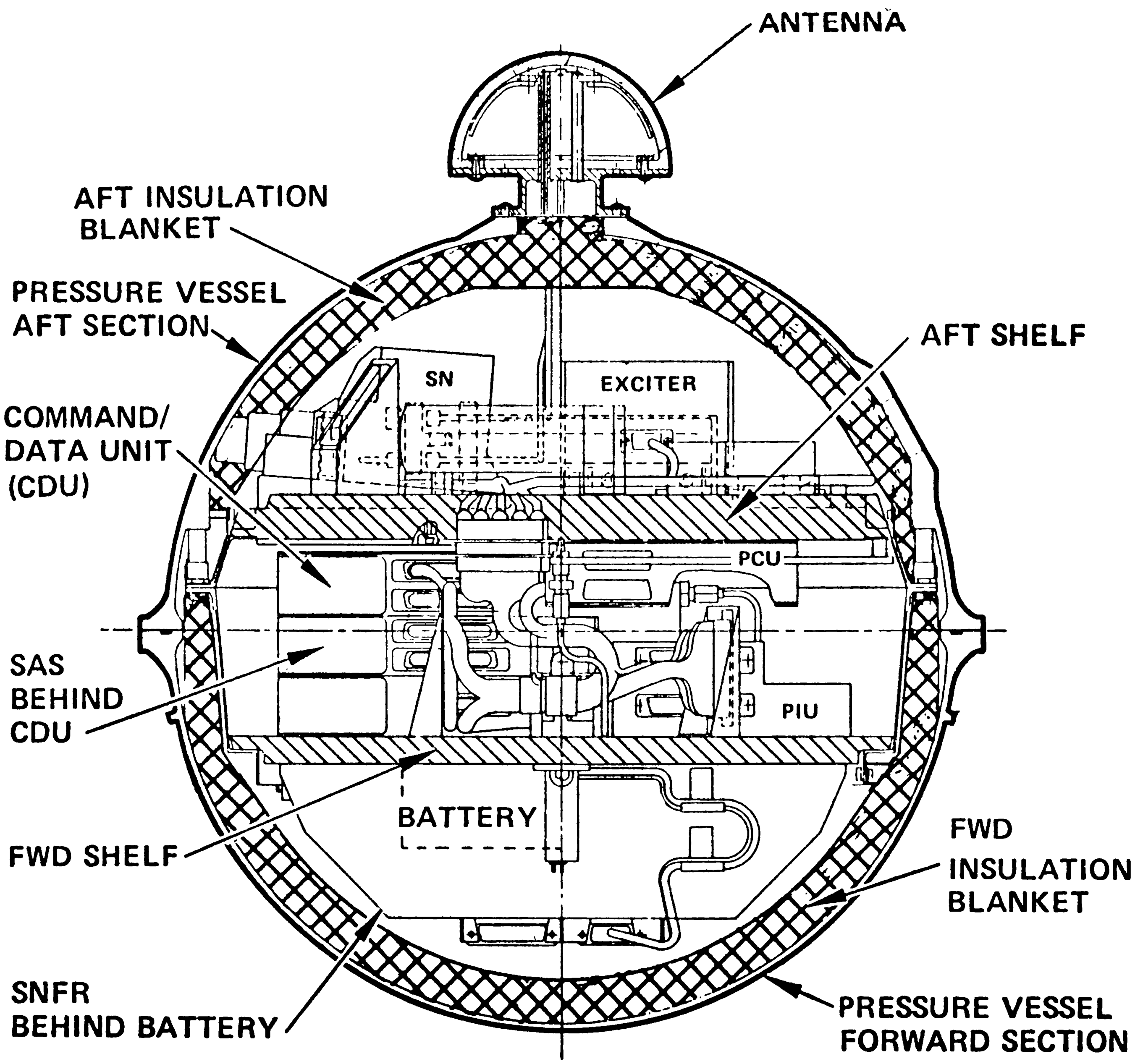

Fig. 12. Small probe pressure vessel internal arrangement.

provides 3072 bits storage capability for entry data which is later transmitted during descent.

The power subsystems in all probes use silver–zinc batteries as basic energy sources and provide a nominal bus voltage of 28 V. The subsystems consist of a battery, a power interface unit, and a current sensor. The large probe battery provides 40 A-hr and the small probe 11 A-hr. Power interface units contain all subsystem electronics as well as fuses and power switching relays for all scientific instruments and probe subsystems. Power for probe checkout and heating is provided by the bus prior to probe separation. During this time, the batteries are open-circuited by switches contained within the power interface units.

7. Concluding Remarks

We have briefly summarized the major elements of the Pioneer Venus program and the design and development status as of July 1976. Naturally, some of the quantitative data given may change somewhat as the spacecraft and instrument hardware proceed in their fabrication and testing leading to the launch periods two years hence.