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FINAL REPORT
SYSTEM DESIGN OF THE PIONEER VENUS SPACECRAFT

VOLUME 1
EXECUTIVE SUMMARY

■

By

S. D. DORFMAN

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July 1973

Prepared Under

Contract No. NAS[redacted] 2-7250

By

HUGHES AIRCRAFT COMPANY

EL SEGUNDO, CALIFORNIA

For

AMES RESEARCH CENTER

NATIONAL AERONAUTICS AND

SPACE ADMINISTRATION

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PREFACE

The Hughes Aircraft Company Pioneer Venus final report is based on study task reports prepared during performance of the "System Design Study of the Pioneer Spacecraft." These task reports were forwarded to Ames Research Center as they were completed during the nine months study phase. The significant results from these task reports, along with study results developed after task report publication dates, are reviewed in this final report to provide complete study documentation. Wherever appropriate, the task reports are cited by referencing a task number and Hughes report reference number. The task reports can be made available to the reader specifically interested in the details omitted in the final report for the sake of brevity.

This Pioneer Venus Study final report describes the following baseline configurations:

- "Thor/Delta Spacecraft Baseline" is the baseline presented at the midterm review on 26 February 1973.
- "Atlas/Centaur Spacecraft Baseline" is the baseline resulting from studies conducted since the midterm, but prior to receipt of the NASA execution phase RFP, and subsequent to decisions to launch both the multiprobe and orbiter missions in 1978 and use the Atlas/Centaur launch vehicle.
- "Atlas/Centaur Spacecraft Midterm Baseline" is the baseline presented at the 26 February 1973 review and is only used in the launch vehicle utilization trade study.

The use of the International System of Units (SI) followed by other units in parentheses implies that the principal measurements or calculations were made in units other than SI. The use of SI units alone implies that the principal measurements or calculations were made in SI units. All conversion factors were obtained or derived from NASA SP-7012 (1969).

The Hughes Aircraft Company final report consists of the following documents:

<u>Volume 1 - Executive Summary</u> - provides a summary of the major issues and decisions reached during the course of the study. A brief description of the Pioneer Venus Atlas/Centaur baseline spacecraft and probes is also presented.

Volume 2 - Science - reviews science requirements, documents the science-peculiar trade studies and describes the Hughes approach for science implementation.

Volume 3 - Systems Analysis - documents the mission, systems, operations, ground systems, and reliability analysis conducted on the Thor/Delta baseline design.

Volume 4 - Probe Bus and Orbiter Spacecraft Vehicle Studies - presents the configuration, structure, thermal control and cabling studies for the probe bus and orbiter. Thor/Delta and Atlas/Centaur baseline descriptions are also presented.

Volume 5 - Probe Vehicle Studies - presents configuration, aerodynamic and structure studies for the large and small probes pressure vessel modules and deceleration modules. Pressure vessel module thermal control and science integration are discussed. Deceleration module heat shield, parachute and separation/despin are presented. Thor/Delta and Atlas/Centaur baseline descriptions are provided.

Volume 6 - Power Subsystem Studies

Volume 7 - Communication Subsystem Studies

Volume 8 - Command/Data Handling Subsystems Studies

Volume 9 - Altitude Control/Mechanisms Subsystem Studies

Volume 10 - Propulsion/Orbit Insertion Subsystem Studies

Volumes 6 through 10 - discuss the respective subsystems for the probe bus, probes, and orbiter. Each volume presents the subsystem requirements, trade and design studies, Thor/Delta baseline descriptions, and Atlas/Centaur baseline descriptions.

Volume 11 - Launch Vehicle Utilization - provides the comparison between the Pioneer Venus spacecraft system for the two launch vehicles, Thor/Delta and Atlas/Centaur. Cost analysis data is presented also.

Volume 12 - International Cooperation - documents Hughes suggested alternatives to implement a cooperative effort with ESRO for the orbiter mission. Recommendations were formulated prior to the deletion of international cooperation.

Volume 13 - Preliminary Development Plans - provides the development and program management plans.

Volume 14 - Test Planning Trades - documents studies conducted to determine the desirable testing approach for the Thor/Delta spacecraft system. Final Atlas/Centaur test plans are presented in Volume 13.

Volume 15 - Hughes IR&D Documentation - provides Hughes internal documents generated on independent research and development money which relates to some aspects of the Pioneer Venus program. These documents are referenced within the final report and are provided for ready access by the reader.

Data Book - presents the latest Atlas/Centaur Baseline design in an informal tabular and sketch format. The informal approach is used to provide the customer with the most current design with the final report.

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1. INTRODUCTION

This Final Report for the System Design Study of the Pioneer Venus Spacecraft is submitted by the Hughes Aircraft Company in accordance with Statement of Work 2-17502, Item 2.1(17), of contract NAS2-7249. The final report is comprised of 15 volumes and an informal Baseline System Data Book. An outline for the management and development plans is presented in Volume 13; a detailed set of plans will be submitted concurrent with the Hughes proposal in response to NASA RFP No. 2-21976.

The NASA Ames Research Center Pioneer Venus Project objective is to conduct scientific investigations of the planet Venus using spin stabilized spacecraft. The defined approach to accomplish this goal is to implement a multiprobe spacecraft mission and an orbiter spacecraft mission. Candidate launch vehicles for the Pioneer Venus missions were the Thor/Delta and Atlas/Centaur.

The multiprobe spacecraft consists of a probe bus, one large probe, and three small probes. The probes are designed to survive to the surface of Venus, and to make in situ measurements of the Venusian atmosphere; the probe bus enters the atmosphere and makes scientific measurements until it burns out. The orbiter mission uses a spacecraft designed to orbit Venus for 225 days with an orbit period of about 24 hours (h). The probe bus and orbiter designs are to use a common spacecraft bus.

The Pioneer Venus Systems Design Study started 2 October 1972. The objectives were to "provide sufficient analysis, trade-offs or alternative designs and operational concepts, test results, and design studies to fully define mission parameters, requirements, constraints and the optimum system and subsystem design for the 1976/77 multiple-probe mission, and the 1978 orbiter mission." On 7 November the contract was modified so that both the Atlas/Centaur and the Thor/Delta launch vehicle were to be considered. It was required to ascertain whether significant cost savings could be obtained through use of the extra weight capability provided by the Atlas/Centaur. In addition, it was desired that international cooperation with the ESRO on the orbiter mission be investigated. It was desired that "the design shall be conducted to a depth that will provide sufficient analysis, test data, system and subsystem design trade-offs that will lead to the single optimum spacecraft, probes, and mission design." It was emphasized in the Statement of Work that low cost and innovative techniques for attaining low cost were of paramount importance in the Systems Design Study. In addition to the required analysis and trade studies, and the recommended configuration, preliminary development plans were also requested.

A preliminary science definition was provided Hughes at the outset of the study and continually updated as Ames acquired information on the candidate science instruments. A final science baseline was provided Hughes on 13 April 1973.

The Hughes study approach consisted of defining a number of detailed study tasks based upon the Ames statement of work. These tasks, listed in Appendix A of this study, were informally transmitted to Ames as they were completed and are incorporated in this final report as references. During the first half of the study emphasis was on trade-off analyses and tests in critical areas. The results of these studies and tests were used to generate preliminary baseline configurations which were presented to Ames at midterm, 26-27 February 1973. In addition, a programmatic baseline was generated and ROM costs were presented for both the Atlas/Centaur and Thor/Delta configurations. During this time period study effort focused on the Thor/Delta configuration since it posed the more challenging technical problems.

NASA utilized the information presented at the midterm of 26-27 February to reach a final decision on the launch vehicle and the mission date. In mid-April NASA selected the Atlas/Centaur for the launch vehicle and changed the multiprobe mission launch date from the December 1976/ January 1977 period to August 1978; the orbiter mission was unchanged from the May/June 1978 dates. After the April 1973 directive the design study activity focused on the detailed design and development plans for the Atlas/ Centaur launch vehicle configuration with the updated science payload and the revised mission set.

The system design final report documents all studies performed during the Pioneer Venus Systems Design Study. Much of the analysis has been directed toward investigation of the Thor/Delta baseline design with the 1976-1977 multiprobe mission launch date. This information is included in the final report for completeness. A large bulk of this data is applicable to the Atlas/Centaur and is so referenced in the final report. The general format of the final report is presentation of requirements, trade-off and design analyses directed toward meeting these requirements, selection of components and subsystems, and baseline descriptions of recommended Thor/Delta and Atlas/Centaur configurations. Time did not allow the final report to reflect the science and program requirements included in the NASA Pioneer Venus RFP 2-21976 received 14 June 1973. The Hughes proposal will address the June RFP.

This executive summary provides the highlights of the results of the Pioneer Venus Systems Design Study. A discussion of the science and mission goals are discussed in Section 2. Section 3 contains a summary of the study approach, the results of the Atlas/Centaur versus Thor/Delta trade, a highlight of the major trade results, a description of the recommended Atlas/Centaur configuration, a summary of the low cost development plans, and the results of the international cooperation study. Section 4 provides the highlights of test results.

2. SCIENCE GOALS

The Space Science Board and the Pioneer Venus Science Steering Group recommended the exploration of Venus to determine the:

- Number, thickness, and composition of cloud layers
- Nature of the circulation
- Reason for the high surface temperature
- Reason for the lack of water
- Reason for the remarkable stability of the CO_2 atmosphere
- Interaction of the solar wind with the planet
- Distribution of mass and magnetic field strength
- Seismic activity

The goals were to be achieved using a low cost spin stabilized Pioneer type spacecraft with an entry and an orbiter mission.

The multiprobe mission consists of five elements: one large probe with about 27 kg of science payload, three identical small probe each containing about 2 kg of science payload, and the probe bus carrying about 12 kg of science payload for upper atmosphere measurements.

The large probe provides a thorough sounding of the entire atmosphere. Since the critical question is the energy balance, a day side landing for the first mission is desired and should be at least 20 deg from the terminator. The clouds are the principal determinant in the heat balance and their composition, density, and layering must therefore be determined. Table 2-1 lists the instrumentation most recently under consideration for the large probe, and includes the principal requirements imposed by them on the probe.*

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*Science payload data is obtained from NASA Ames letter of 13 April 1973.

TABLE 2-1. LARGE PROBE NOMINAL PAYLOAD

Instrument	Mass, kg	Volume, cm^3	Power, W (Average)	Data, bps	Special Requirements
Temperature gauge	0.3	100	0.5	1	Sensor outside of pressure vessel
Pressure gauge	0.4	115	0.5	1	Port in pressure vessel
Accelerometers	1.25	665	2.3 plus 8 mw/g	3	Locate at c.g.; requires data storage during high speed entry
Neutral mass spectrometer	9.07	9,830	12	40	Inlet system in lower hemisphere ≈ 45 deg from spin axis; requires 12 squibs.
Solar radiometer	2.25	1,600	4	7/4	7 bps above 44 km, window or light pipes 90 deg from spin axis, up and down viewing
Cloud particle size analyzer	3.65	3,275	20	24	Requires window external mirror with undisturbed flow past mirror
IR flux radiometer	2.25	1,600	3	3	Single down looking window
Gas chromatograph	3.6	4,100	6	11	Three measured cycles, 20 min each, entry port in lower hemisphere
Hygrometer	0.5	320	0.25	0.5	Entry port
Wind/altitude radar	4.0	8,195	40	2	40 km to surface, 25 x 25 cm antenna mounted on outside base of pressure vessel requires spin rate and vertical spin orientation
Transponder			2	0	The receiver is not required for the probe mission. The transmitter is, of course, necessary for data transmissions.
	27.2	2,980	90.55	70.5/67.5	

TABLE 2-2. SMALL PROBE NOMINAL PAYLOAD

Instrument	Mass, kg	Volume, cm^2	Power, W	Data, bits/ sample	Special Requirements
Temperature	0.3	100	0.5	8/7	Must be in free stream, deployed sensor
Pressure	0.4	115	0.5	8/7	Must be at stagnation point (port in heat shield)
Accelerometer	0.18	33	1.0	8/7	At c.g., single axis data storage during blackout
IR flux radiometer	0.5	328	1.0	32	Looks up and down, deployed sensor
Nephelometer	0.45	524	1.0	43/10	Window near equator to look out perpendicular to descent axis
Stable oscillator	0.34	131	0.25		One way doppler tracking. Stability 1×10^9.
	2.2	1,231	4.25	-	

The purpose of the identical small probes is to measure atmospheric properties in widely separated regions of the planet to determine the differences in local environment, and thus provide information concerning circulation. The weight and volume constraints of the small probes limit the number of experiments. Table 2-2 summarizes the proposed small probe experiments. Winds, an essential element of the circulation system, are estimated using a stable oscillator (one part in 10^9) in the transmitter as the key element in the doppler/DLBI (doubly differenced, very long baseline interferometry) experiment to measure the vertical velocity and the transverse or lateral velocity. This experiment will also be performed with the large probe using the two-way transponder link, and the results can be compared with the large probe wind altitude radar measurements.

TABLE 2-3. PROBE BUS NOMINAL PAYLOAD

Instrument	Mass, kg	Volume, cm^2	Power, W	Data, bps	Special Requirements
Neutral mass spectrometer	5.5	8,195	12	360	Inlet System ±10 deg to velocity vector
Ion mass spectrometer	1.6	2,459	2.5	450	Inlet system ±10 deg to velocity vector, positive surfaces insulated
Electron temperature probe	1.0	1,500	3.0	70	Perpendicular velocity with m^2 ground plane Positive surfaces insulated
Ultraviolet spectrometer	2.7	2,295	1.5	12/500	20 deg from spin axis to view planet and scan limb at E-4 days and E-1 h
Retarding potential analyzer	1.2	1,967	2.5	260	Near to velocity vector and positive surfaces insulated
	12.0	16,420	21.5	1,152/ 1,640	

The probe bus itself is also used to make in situ measurements when entering the atmosphere. Since it is destroyed near 100 km altitude or higher, these experiments are selected for upper atmosphere measurements. Table 2-3 summarizes these experiments and their requirements on the bus. The bus enters the atmosphere with the spin axis oriented back to earth to maximize antenna gain (and hence bit rate). The spin axis must also be aligned with the velocity vector to facilitate pointing of the mass spectrometers. The ultraviolet spectrometer will scan the disc at 4 days out and continued through entry to make limb measurements until destruction of the bus.

The orbiter mission is used to make atmospheric measurements and to make long-term observations of Venus from an elliptical orbit. Table 2-4 lists the experiments currently considered for the orbiter. The orbiter contains experiments which will also be carried on the multiprobe mission since many of the same parameters of the ionosphere and upper atmosphere are to be measured. Solar wind interactions with the planet and the magnetic fields are also measured. The dual frequency S and X band occultation experiment will provide atmosphere data as a function of latitude, altitude, and season. The radar altimeter will map the surface characteristics over high latitude regions with relatively high resolution, and the resolution of the equatorial regions will be greatly improved as compared with present earth-based radar observations.

TABLE 2-4. ORBITER NOMINAL PAYLOAD

Instrument	Mass, kg	Volume, cm^3	Power, W	Bit Rate, bps	Special Requirements
Magnetometer	3.5	2,870 (Elec)	4	3 cruise 6 bow-shock	0.5 γ demagnetized
Solar wind analyzer	5.0	5,507	5	3	Must see sun each revolution
Electron temperature probe	1.4	1,935	2.5	24	Insulate positive surfaces
Neutral mass spectrometer	5.4	8,195	12	100	Inlet system ≤ ±10 deg* to velocity vector over primary measurement region near periapsis
Ion mass spectrometer	1.5	3,275	2	100	(same as above)
Ultraviolet spectrometer	5.5	6,556	6	34	Scan Planet and limb
Infrared radiometer	5.5	6,556	6	100	Must view planet
S and X band occultation	2.7	-	14.4	-	Must compensate for 10 deg refractive bending
Radar altimeter	9.0	2,547	15	50	Must work to local vertical from periapsis to ±1000 km
	39.5	50,760	89.5	414/443	

*This requirement is derived from Ames RFP 2-21976. Earlier payload requirements indicated ±30 to ±15 deg were adequate.

3. SYSTEMS DESIGN APPROACH

The scientific objectives of the Pioneer Venus program are to explore the planet Venus with a probe spacecraft and an orbiter spacecraft to be launched in 1978. It is of primary importance that the scientific objectives be met in a low cost program. This was emphasized in the System Design Study Statement of Work which states, "the study shall emphasize the use of existing and proven hardware and technology, including implementation concepts that will minimize overall costs while achieving overall mission objectives."

In order to attain these objectives, cost was given a primary role in all trade studies conducted as part of the system design study. As a means of acquiring the necessary data a rough order of magnitude (ROM) cost was developed in preparation for the midterm presentation. This ROM exercise identified Thor/Delta and Atlas/Centaur technical baselines and a common, and conservative, program baseline. A combination of "grass roots" and ROM estimates from subcontractors was used to build up cost information that was used in subsequent trade studies. This activity had the salutory advantage of introducing cost factors into the early stages of the design.

The results of the Atlas/Centaur versus Thor/Delta trades are discussed in subsection 3.1. The following paragraphs discuss the low cost approach that evolved during the system design study and was utilized in generating the final report baseline. The approach to low cost can be divided into the following major categories.

Use of existing equipment. Emphasis was placed on the use of existing equipment. In the case of the Hughes study team this consists principally of spacecraft hardware derived from Hughes communication satellite programs, e. g., Intelsat IV and the Canadian Domestic Satellite (Telesat) and science instrument integration hardware — both data and power — from the OSO program. In the case of the Hughes subcontractor, General Electric, this equipment was primarily derived from classified military programs. In the event identical existing equipment could not be used, a derivation of existing equipment was usually appropriate. Only in the case of the pressure vessel module, which has to withstand the high pressure and high temperature of Venus, is totally new hardware necessary. In this case, test programs were conducted to validate design concepts. In order to assure that all available applicable equipment was utilized, the industry was surveyed for existing equipment. In particular, it seemed appropriate to use hardware from the Pioneer, Mariner, and Viking programs and contractors from those programs were solicited for cost and technical data.

Commonality. Commonality between the probe bus and orbiter, large probe and small probe, and buses and probes was a major aspect of the low cost approach, and a major objective was to maximize these commonalities. The best example of the success attained was the communication transponder, which is utilized in all configurations and is essentially identical for the large probe, probe bus, and orbiter. Similar commonalities were achieved in many other areas.

Minimum quantity of hardware. In order to reduce program costs, the development plan focused on developing the minimum quantity of hardware. For example, the deletion of a separate prototype vehicle has been incorporated as part of the preliminary development plan. This type of saving was introduced only if there was no reduction in the confidence of meeting program objectives.

Minimum subsystem/unit testing. Use of existing equipment and commonality opens up the possibility of substantially reducing the amount of unit and subsystem testing. This approach was incorporated in the preliminary development plan in areas where it would not compromise program objectives.

Use program time to save cost. The present Pioneer Venus plans are for a launch approximately 5 years from the date of this final report. The preliminary development plans are based upon attaining cost savings through proper utilization of the time. This includes: 1) development of good interface specifications prior to the start of engineering, 2) development of an engineering baseline prior to the start of fabrication and assembly of hardware, and 3) allowance of extra time during the spacecraft assembly and test phase to uncover any problems that may derive from all of the cost saving techniques that have been incorporated in the program.

"Experimental shop". A program incorporating the "experimental shop" concept was part of the low cost strategy. This approach would minimize documentation, delays due to serial approvals and internal communication, and, in general, streamline program operation. The basic approach which materialized is based upon the management approach taken in the Hughes commercial communication satellite programs, as modified by the unique characteristics of a scientific spacecraft program.

In concert, the above approaches yielded a substantial cost saving benefit. The quantification of the savings will be in the 15 August submission. In subsections 3.2 through 3.6, the technical characteristics which exemplify use of existing equipment and commonality are discussed. The last four items in the above paragraphs are discussed in the development plan, which will be submitted on 15 August and is summarized briefly in subsection 3.7.

3.1 RESULTS OF ATLAS/CENTAUR UTILIZATION STUDY

In order to determine the cost savings that would result from the utilization of the Atlas/Centaur, a study which compared Atlas/Centaur and Thor/Delta was conducted. A technical and programmatic baseline was generated for both the Atlas/Centaur and Thor/Delta launched spacecraft configurations. A parallel rough order of magnitude estimation process generated ROM costs for both configurations. In this process, relative cost was emphasized. The absolute magnitude of cost was deliberately made conservative in the absence of any Ames Research Center work statement or specification for the execution phase of the program. In this way, emphasis was properly placed on differences between the Atlas/Centaur and Thor/Delta configurations, since the absolute costs are being developed.

Prior to the ROM cost estimate, several cost fact-finding sessions were held with the project team members and subsystem managers responsible for the development of the project hardware to determine cost sensitive factors. The outcome of these reviews resulted in establishing low cost guidelines for the development of the Atlas/Centaur spacecraft configuration. The project personnel were directed to be innovative in their use of proven hardware, so that the study would result in a reliable system design at the lowest overall cost. The technical details of the Thor/Delta and Atlas/Centaur launch spacecraft configurations can be found in Volume II of the final report.

The cost savings accrued by utilizing an Atlas/Centaur launch vehicle are summarized in Table 3-1 and were attained through evaluation of the ROM cost estimates and detailed iterations with the subsystem areas on designs and costs. The dollars saved are shown according to subsystem or area of responsibility. The percentages tabulated are those percentages of the cost savings for that particular area or subsystem as compared to the Thor/Delta higher weight mission cost estimates. Propulsion, thermal control, and structure subsystems represent cost increases for the Atlas/Centaur spacecraft configuration and are related by the dollars subtracted from the cost savings.

As can be seen from the summation, a total cost savings for a two mission set utilizing an Atlas/Centaur launch vehicle is $10,650,000. A large savings occurred because the Atlas/Centaur probe design allowed a larger volume, thus providing ease of assembly and integration of pressure vessels. The larger vehicle also allowed commonality in the cone angles of the probe aeroshells, reducing aerodynamic testing and associated costs. About 50 percent of the deceleration module cost reduction is due to the elimination of the use of beryllium in the Atlas/Centaur probes configuration.

Deletion of the structural and thermal test models is deemed feasible due to the high margins of safety designed into the Atlas/Centaur spacecraft because of the added weight capability. Hence, a significant cost savings is represented. It should be noted that a nonflight prototype spacecraft will be required if thermal and structural test models are deleted.

TABLE 3-1. ATLAS/CENTAUR COST SAVINGS SUMMARY

Element		$K – Saved	Percent* Saved	Reason
1)	Structures	900	15.8	• Test and analysis • Aluminum versus beryllium
2)	Deceleration module	2,390	12.4	• Testing • Aerodynamics • Aluminum versus beryllium
3)	Communications subsystems	230	4.2	• Viking transponder • Reduced efficiency
4)	Command and data handling	820	9.0	• Product design • Assembly and test
5)	Power subsystems	620	12.4	• Nickel-cadmium battery • Boost voltage
6)	Program management	1,780	9.6	• Integration and test • Materials
7)	Test models	1,500	100.0	• Structural • Thermal
8)	Magnetic cleanliness	1,430	68.0	• Parts and materials • Magnetic controls management
9)	Mechanisms	300	5.2	• Telesat BAPTA • Bicone deployment
10)	Miscellaneous (deductions)	1,560	--	• System test • Risk pool
11)	Miscellaneous (additions)	(880)	--	• Propulsion • Thermal control • Structures, etc.
	Total saved	$10,650		

*Percent of that element of cost, e.g., percent of power subsystem cost.

Conservative and simplifying assumptions can be made in establishing the margins of safety. Hence, the stress and dynamic analysis effort can be reduced for the probe bus, pressure vessels, and orbiter spacecraft due to the larger structural weight allocations. This savings is represented in item 1 of Table 3-1, along with the savings attained through the deletion of beryllium in the spacecraft structure.

Off-the-shelf selection of hardware is represented as a cost saving factor. Subsystems that use off-the-shelf hardware include communications, power, structure, propulsion, thermal control, attitude control, and command and data handling. The rf subsystem will use an unmodified Viking transponder and lower efficiency/higher power amplifiers to reduce costs. The power subsystem will incorporate an OSO discharge regulator and Telesat nickel cadmium (N_i-C_d) batteries on the probe bus and orbiter. In addition, boost circuitry will not be required in the power system.

The command and data handling subsystems will utilize OSO derived hardware. With the larger volume available, productizing, assembly and unit testing of this equipment is considered to be less costly. This cost saving is represented by item 4 in the table.

Although magnetic cleanliness shows a cost savings of $1,430,000, this should not be construed as deletion of the magnetic cleanliness program. In the Atlas/Centaur configuration, a 4.4 m (14.5 ft) boom is used instead of the 1.07 m (3.5 ft) boom on the Thor/Delta design. This is interpreted as having the magnetometer sensor deployed out far enough so that the stringent controls on subsystem hardware can be reduced or eliminated. There will still be a $500,000 to $600,000 cost to assure that the appropriate magnetic level is obtained at the sensor. The large boom and its associated testing represent added costs.

In the mechanism area, the deletion of a deployment mechanism for the bicone antenna on the Atlas/Centaur probe bus represents a cost reduction. The Thor/Delta spacecraft required such a deployment mechanism for the bicone antenna because of insufficient clearance and depth with the existing spacecraft attaching adapter. In the case of the orbiter, a modified Telesat bearing and power transfer assembly for the mechanically despun antenna and the use of titanium instead of beryllium in its design indicate a cost savings for the Atlas/Centaur configuration.

3.2 SYSTEM LEVEL TRADES

Several system level trades have led to decisions that have a significant impact on the overall vehicle configurations:

- Spacecraft spin axis orientation
- Electronically or mechanically despun antenna
- Type I or Type II transit trajectory

TABLE 3-2. ORBITER SCIENCE COVERAGE COMPARISON

Experiments	Spin Axis ⊥ Ecliptic (Baseline)		Spin Axis Toward Earth	
	Percent Mission Data Coverage	Performance	Percent Mission Data Coverage	Performance
Velocity oriented	100	Measurement between periapsis and 4000 km less than 21.5 degree sampling angle for 56°S periapsis (less than 13 degrees for 26°N periapsis)	100	Measurement between periapsis and 4000 km up to 80 degrees sampling angle for 56°S periapsis (up to 30 degrees for 26°N periapsis)
Sun oriented	100	Ideal, scan in ecliptic plane	100	Measurement angle as high as 80 degrees towards end of mission
Earth pointed (rf occultation)	When applicable	Earth tracking due to Venus atmosphere refraction easily accommodated without need for attitude maneuver	When applicable	Attitude maneuver for earth tracking necessary. Deleterious effect on planet pointed experiments
Planet oriented				
Radar altimeter	100	Coverage of all southern latitudes achievable by ±45 degree elevation boresight freedom	Variable, depending on elevation boresight freedom	35 percent mission coverage, for ±45 degree elevation boresight
Other experiments (IR UV)	100	Measurement of all southern latitudes taken within ±3 percent minimum distance	Partial	Measurement distances greatly increased for most of available opportunities.

Spacecraft Spin Axis Orientation

NASA-Ames defined one of the first design issues by constraining the spacecraft to be spin stabilized rather than a three-axis controlled. The next issue is spin axis orientation. Spin axis orientation during the cruise phase of the earth-to-Venus transit trajectory and, for the orbiter, during the post-insertion orbiting phase is of crucial importance. Subsystem design complexity is a function of orientation, as are science sensor pointing requirements.

There are three basic approaches for the orbiter spin: spin axis perpendicular to the ecliptic, spin axis in the ecliptic always pointing to earth, and spin axis perpendicular to the orbit plane. The last choice is obviously an orbiter only selection; the first two are also candidates for the earth-to-Venus cruise phase. Spin axis perpendicular to the orbit plane has advantages principally because the planet oriented sensors are provided with excellent coverage throughout, but this orientation requires a two degree of freedom despun antenna, or periods without communications coverage. Both alternatives were unacceptable.

Spin axis in the plane of the ecliptic directed toward earth offers a simple high gain antenna, but requires a two degree of freedom platform for experiment coverage, even for the simplest payload. If science instruments are mounted in a fixed position with respect to the vehicle, the associated solar angle variation complicates solar power and thermal design. This orientation necessitates use of periodic attitude maneuvers to maintain science coverage throughout the mission, thus consuming propellant.

Spin axis perpendicular to the ecliptic requires a despun antenna but eliminates the need for an experiment platform, if an electronically phased array altimeter is used and if the neutral mass spectrometer is a closed source type. Spin axis perpendicular to the ecliptic also provides simpler power and thermal subsystems than does the earth pointing orientation. Thus, complex gimbaling schemes are avoided for all experiments.

The primary objective of the Pioneer Venus mission is, of course, science data return. Trade studies resulted in the conclusion that the orbiter yields much better science data return with the spin axis perpendicular to the ecliptic, with fixed position instruments, than with the spin axis in the ecliptic. A summary of the relative performance of the various types of orbiter experiments clearly shows the superiority of this approach (Table 3-2).

The trade study results indicate either orientation to be satisfactory for the probe bus. Insofar as the orbiter spacecraft is concerned, from the viewpoint of system complexity and weight, the despun high gain antenna required in the case of spin axis perpendicular to the ecliptic is a better alternative than the heavier, more complex power and thermal control subsystem design required in the case of spin axis directed towards earth.

The decision has been made, then, to adopt spin axis orientation perpendicular to the ecliptic as a baseline for the orbiter. The same orientation is adopted for the probe bus to preserve commonality with the orbiter for low cost.

Electronically or Mechanically Despun Antenna

With the spin axis perpendicular to the ecliptic, the orbiter high gain antenna selection is between an electronically despun antenna (EDA) or a mechanically despun antenna (MDA). A stack of three bicone antennas has also been considered, but it requires excessive transmitter power.

The orbiter configuration, as will be discussed later, can accommodate either choice, but the MDA is selected on the basis of successful flight experience on many Hughes spacecraft (recently Intelsat IV and Telesat) and lower development cost. The MDA approach is also flexible should a decision be made later to include wide refractive angular variation capability for the dual frequency S/X band radio occultation experiment. This can be provided by incorporation of an elevation drive.

Type I Versus Type II Transit Trajectory

A third system level decision is the choice of Type I versus Type II earth to Venus transit trajectory. In the case of the probe bus, no significant advantage can be found for the Type II trajectory. Since the Type I trajectory results in a considerable spacecraft launch weight advantage for the selected launch years, it was adopted for the probe mission.

In the case of the orbiter mission, the selection affects not only the launch weight and the useful payload weight in orbit but also the quality of science data collection. A Type I trajectory for the 1978 launch opportunity results in significantly greater useful in-orbit payload weight than is possible with a Type II trajectory.

The Type II trajectory gives a near mid-latitude periapsis for the 24 hr orbit around Venus, which is more desirable from the science coverage point-of-view than the near equatorial periapsis location offered by the Type I trajectory selection.

Table 3-3 shows the Type I and Type II comparison in terms of science return and "dry" orbited weight. The two possible Type II periapsis locations, north and south, give comparable science coverage. The south periapsis case is tentatively chosen as baseline in this report, but the issue is by no means conclusive and should be reviewed.

TABLE 3-3. TYPE I AND II ORBITS - SCIENCE CONSIDERATIONS

Orbit Type	Possible Periapsis Locations: Nominal Ecliptic Latitude	Possible Periapsis Locations: Nominal Longitude From Subsolar Point	Science Considerations	Dry Orbited Mass, kg
Type I	4° to 13°N	-46° to -39° (28 to 32 days from evening terminator)	Good for in situ measurements of upper atmosphere and solar wind interactions Fair for planetary mapping	390
Type II	21° to 31°N	-63° to -54° (17 to 23 days from terminator)	Good for in situ measurements Good for planetary mapping	310
Type II	55° to 45°S	-63° to -54° (17 to 23 days from terminator)	Fair for in situ measurements Best for planetary mapping	300

The Type I trajectory is easily accommodated in the baseline design by changing the orbit insertion motor to allow for launch weight and insertion ΔV differences. Such a decision would have to be made at the outset of the execution phase. Changes between the two possible periapsis locations for Type II could be made late in the program.

3.3 BASELINE MISSION PROFILES

Multiprobe Mission

The baseline multiprobe mission profile is as follows. The 10-day launch window for the Type I trajectory in 1978 is from 9 to 22 August inclusive. The daily window is required to be 30 minutes. The transit trajectory profile up to and including the targeting of the probe bus for entry into the Venusian atmosphere is shown in Figure 3-1. A total of three trajectory correction maneuvers (TCM) may be needed in the worst

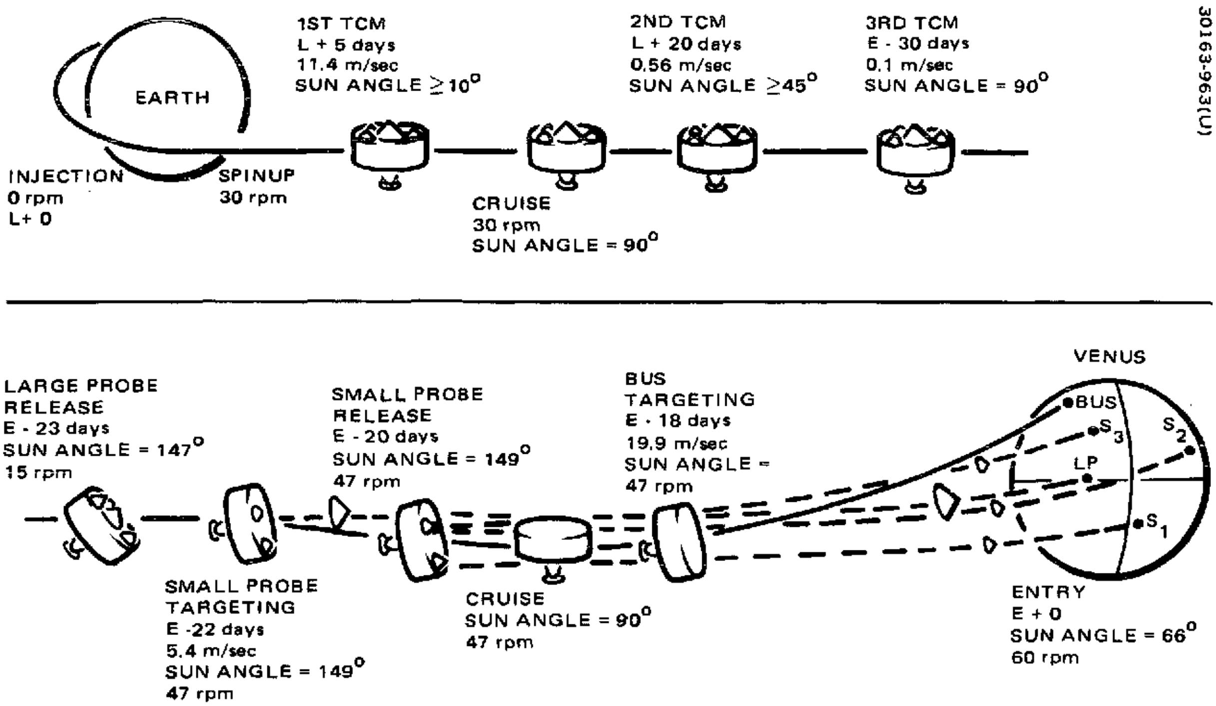


FIGURE 3-1. MULTIPROBE MISSION

case where launch injection errors are at a maximum. The approximate dates of these maneuvers are shown. Figure 3-1 shows that all TCMs are performed without attitude maneuvers, the necessary ΔV's being obtained by vector summing of the ΔV's provided by the axial and radial thrusters. This is made possible because of the small ΔV's required as a result of the accurate injection expected for the Atlas/Centaur vehicle. In the unlikely event that a large ΔV is needed for the first TCM, for instance, the alternate mode for fuel economy of first orienting the spin axis before firing the axial jets can still be employed. Attitude reference during cruise will be provided by sun and star sensors. Probe bus to earth communications links are provided through a bicone antenna at the aft end of the bus as well as two omni antennas.

At approximately 23 days prior to encounter (E-23), the spacecraft is oriented and spun down to 15 rpm for large probe release. Shortly thereafter, the bus is retargeted and spun up to 47 rpm for small probe release. The bus is then returned to the cruise attitude for 2 days, after which it is targeted for entry into the Venusian atmosphere, lagging slightly (1.5 hr) behind the probes in order that its role in the doppler/DLBI experiment may be realized during the entire descent phase of the probes. During the remainder of the mission the probe bus will continue to transmit science and housekeeping data through its end fire antenna back to earth, until rapid atmospheric heating causes acute temperature rise and the subsequent demise of the probe bus. This condition is expected to occur at approximately 117 km.

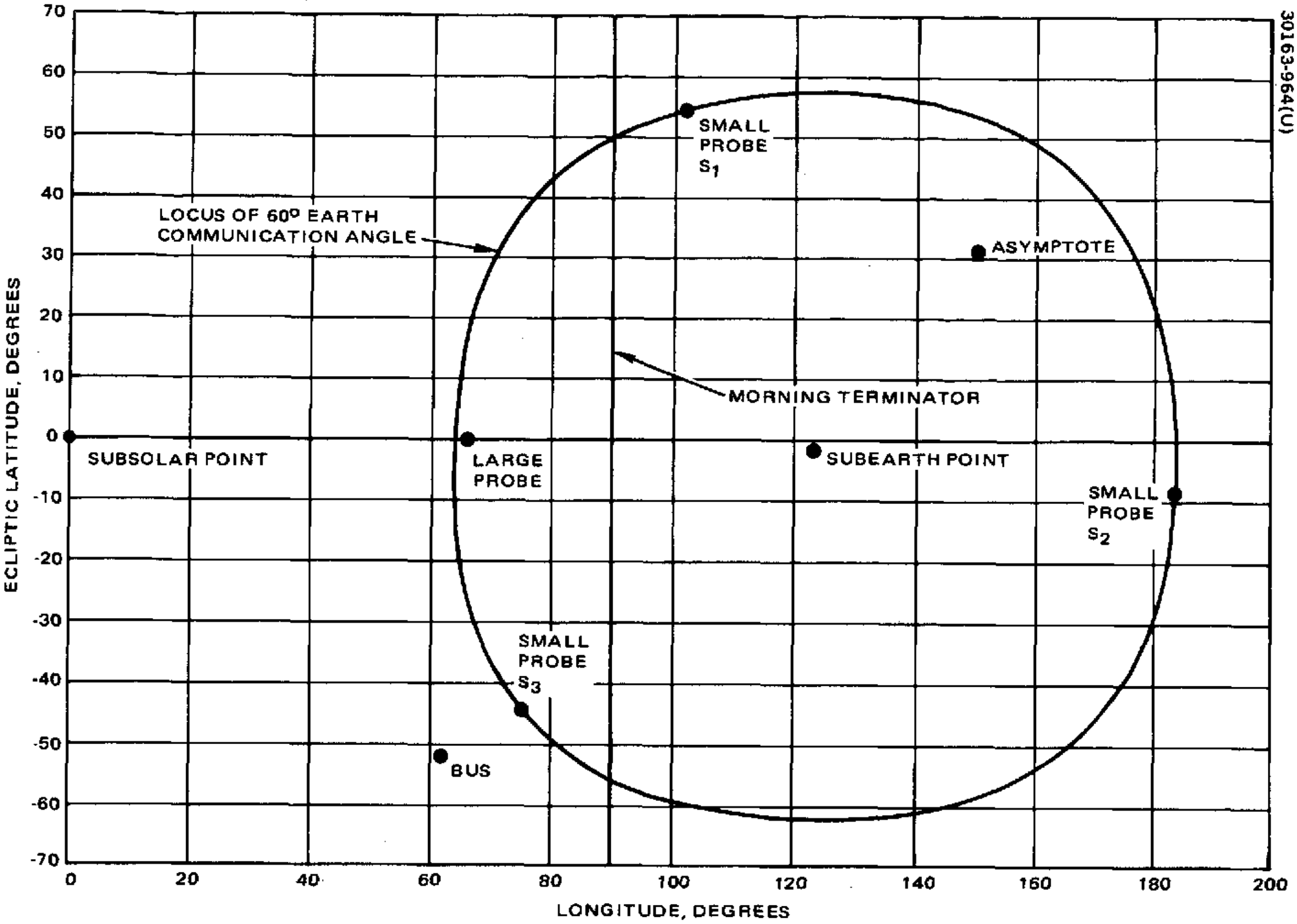


FIGURE 3-2. NOMINAL 1978 PROBE TARGETING

The nominal targeting of the probes and the probe bus is shown in Figure 3-2. The large probe will enter in the sunlit side about 25 deg away from the morning terminator and on the equator. The small probes will be dispersed along a 60 deg earth communication angle contour, beyond which the path attenuation in the Venusian atmosphere becomes excessive. Flexibility in small probe targeting exists in that they can be targeted to any position within the communication angle limit. However, the relative angular separation between the three probes is constrained to be approximately equal due to the constraint for simultaneous release of the small probes.

The large probe enters the Venusian atmosphere and is slowed down by aerodynamic braking to subsonic speeds. At approximately 67 km above the Venusian surface, a parachute extracts the pressure vessel from the aeroshell and the descent phase begins. The parachute jettison occurs at 40 km rather than at a higher altitude so that sampling through the cloud layers is maximized. The total descent time is roughly 75 min. The descent profile is shown in Figure 3-3.

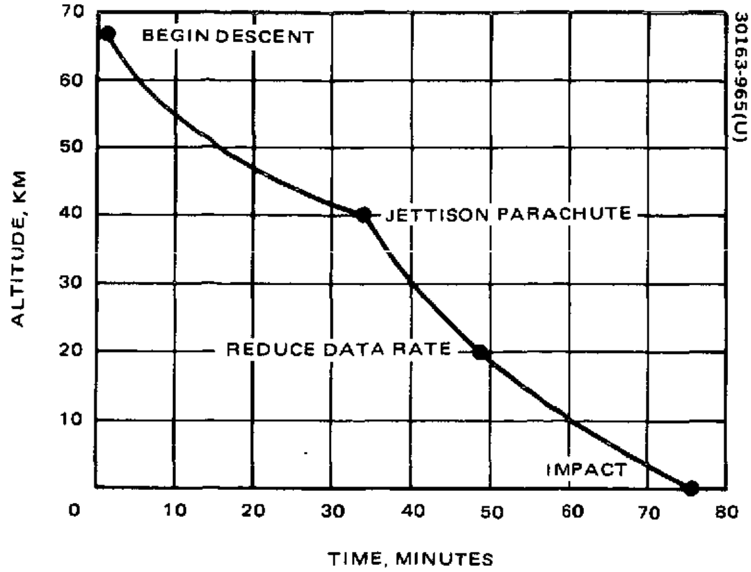


FIGURE 3-3. LARGE PROBE DESCENT PROFILE

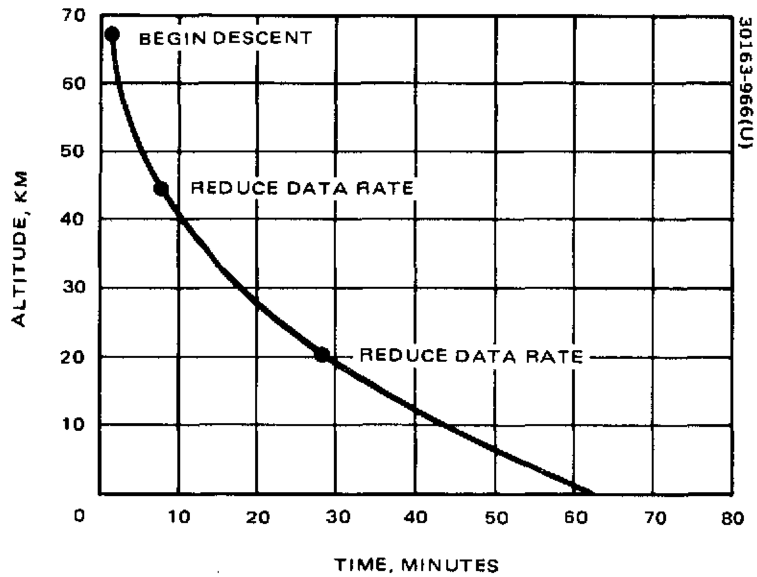


FIGURE 3-4. SMALL PROBE DESCENT PROFILE

The small probes differ in that they do not incorporate a parachute. After the high g aerodynamic deceleration phase, the small probe descends with pressure vessel and aeroshell attached all the way to the surface. The descent profile is shown in Figure 3-4. Total descent time is about 62 min. All three small probes and the large probe land within an interval of about 20 min.

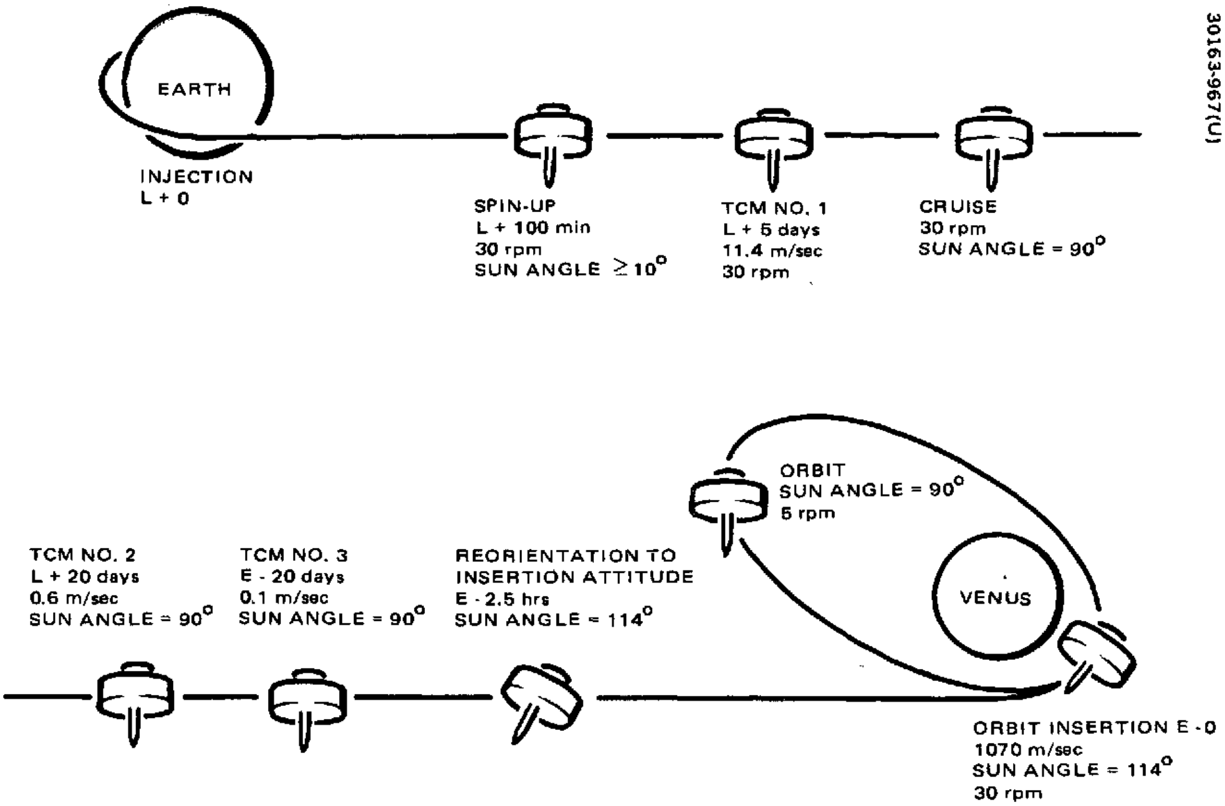


FIGURE 3-5. ORBITER MISSION

Orbiter Mission

The orbiter mission profile is as follows. The 14 day launch window for the specified Type II trajectory in 1978 is from 22 May to 4 June inclusive, with a 30 min daily window. Although launched two and a half months earlier than the multiprobe spacecraft, the orbiter spacecraft arrives at Venus on 3 December 1978, as compared with 9 December 1978 for the multiprobe spacecraft. The difference in flight time is due to the different choice in trajectories as mentioned previously. The 6 day separation in arrival time is, however, more than adequate from an operational viewpoint.

The transit trajectory profile up to and including the orbit insertion maneuvers is shown in Figure 3-5. It is similar in almost all respects to the multiprobe mission transit trajectory profile. The cruise attitude is perpendicular to the ecliptic. All TCM's may be performed in that attitude without reorientation. If the maneuver ΔV required is excessive, the TCM may be performed subsequent to an attitude maneuver. A total of three TCM's are envisaged. Two and one-half hours prior to encounter with Venus, the spacecraft is aligned with the spin axis pointed in the proper orbit insertion retro direction.

The aiming point of 400 km for orbit insertion allows for dispersion above the ultimate 200 km altitude required for periapsis. The solid orbit

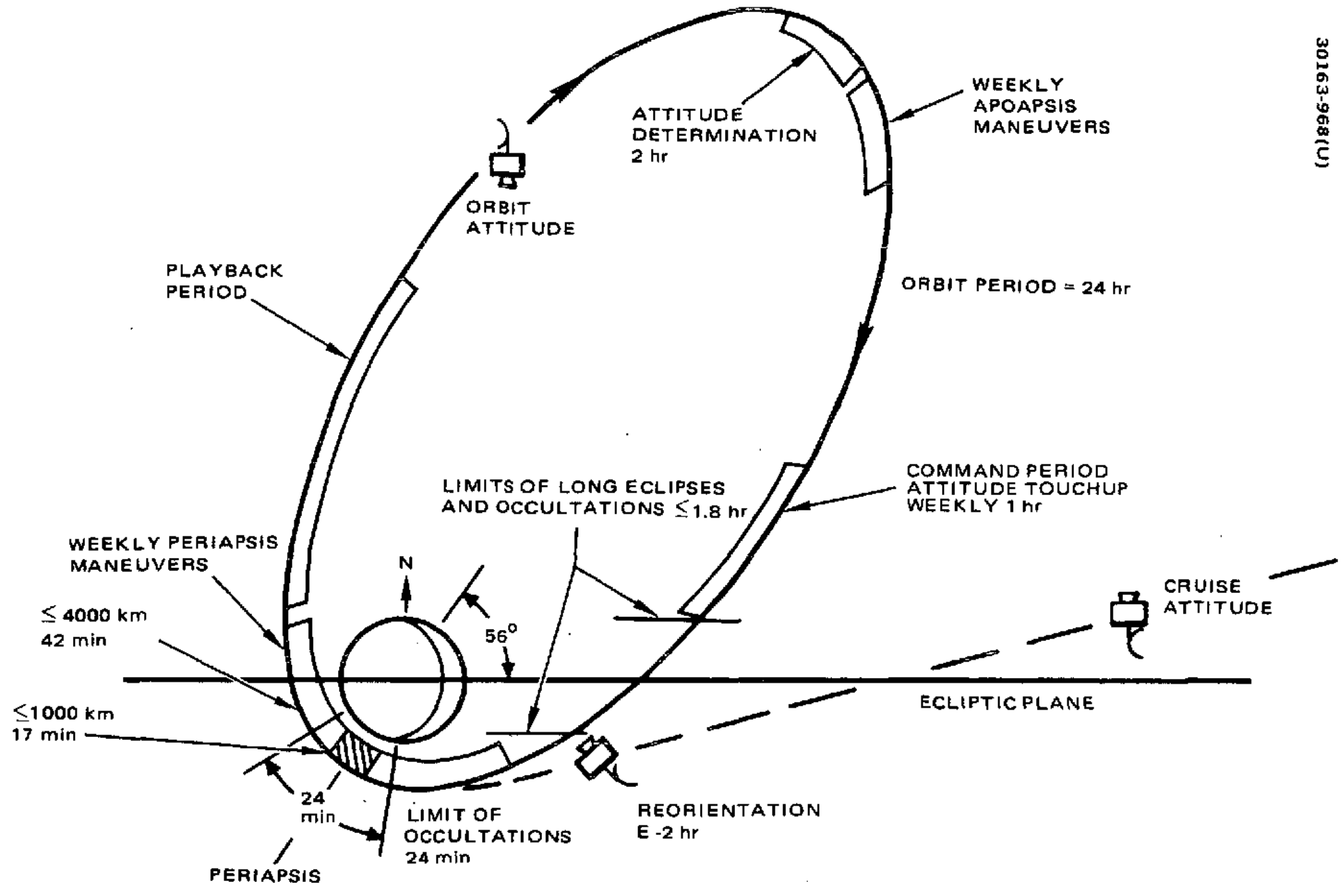


FIGURE 3-6. ORBITAL OPERATIONS

insertion motor places the orbiter spacecraft into a nominally 24 hr polar orbit around Venus. Studies by JPL have indicated that a full orbit of DSN tracking subsequent to the orbit insertion maneuver yields a very precise determination of the orbit. Thus, orbit trimming maneuvers are assumed to take place during the second orbit around Venus. The spacecraft and its subsystems should therefore be in their fully operational condition for science data gathering and transmission no later than the third orbit around Venus.

The nominal orbital operations are depicted in Figure 3-6. Because of solar and atmospheric perturbations on the orbit, weekly apoapsis and periapsis maneuvers are correspondingly required to maintain the orbital parameters within satisfactory limits. Also, weekly attitude touchups are needed to correct the spin axis precession due to various disturbance torques. Whereas some science experiments are operating during the entire 24 hr orbit, most of the data gathering activity takes place in the vicinity of the periapsis. The sequence for all periapsis operations and subsequent playback of data obtained are executed by virtue of a command memory whose contents can be updated at will from earth. If no update is considered necessary for a given orbit, then the sequence during this orbit will be identical in every respect to that of the previous orbit.

3.4 PROBE BUS/ORBITER

The probe bus and orbiter spacecraft have been designed to have maximum commonality and also make use of existing equipment. Both configurations will be described in this section.

Significant trades at the spacecraft level are the selection of the orbit insertion motor, provisions for the RF occultation experiment, magnetometer accommodation, equipment installation, and type of power bus. Each trade is briefly discussed in the following paragraphs, along with a description of the baseline probe bus and orbiter spacecraft.

Major Issues

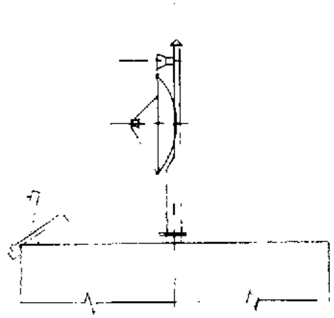
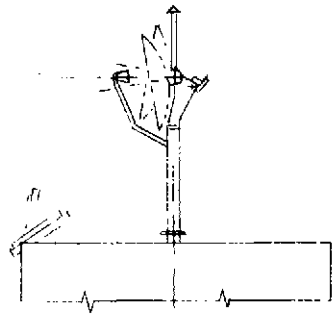
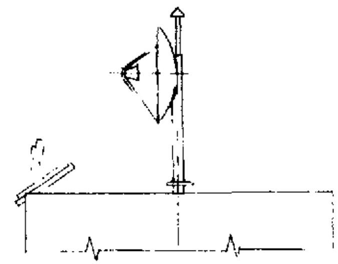
Orbit Insertion Motor

Orbit insertion ΔV of the orbiter can be provided by liquid or solid propulsion. Monopropellant liquid systems (i.e., hydrazine) suitable for attitude control because of their low specific impulse create a severe weight penalty even for the Atlas/Centaur launch vehicle. Bipropellant liquid systems are weight competitive but are more complex, less reliable, and usually cost more than comparable solid propellant systems. A stretched version of the existing TEM 521 motor meets the orbit insertion requirements. This TEM motor, together with a proven hydrazine system using existing thruster and tanks, provides a low cost, weight efficient solution to the overall needs of orbit insertion and attitude control.

Provision for RF Occultation Experiment

The requirement for the S/X band dual frequency radio occultation experiment introduces a special trade issue in the spacecraft design. Beam refraction angles of ±10 deg (required) and ±20 deg (desired) need to be accommodated in the design. The mechanical design antenna (MDA) selected with its azimuth degree of freedom opens up several simple options for implementing this experiment. Option A is to add a fixed 20 deg beamwidth X band horn and a 3 W X band TWT transmitter. The ±10 deg requirement is met with the antenna pattern. Option B uses the same (82.5 cm) reflector for both S and X band, but a dual S/X band feed, a 200 mW X band solid state transmitter, and an elevation gimbal for the reflector. The elevation gimbal provides the antenna motion to accommodate the ±20 deg goal. Option C would add a defocused X band feed to the fixed 82.5 cm (32.5 in.) antenna, a 3 W, X band TWT transmitter, together with spacecraft precession to accommodate the ±10 deg refraction angle. Option A, using the separate X band horn antenna, has been selected as baseline for the following reason: lowest cost, weight, and complexity; best experiment/spacecraft interface/ and high reliability and simple mission operational procedures. Should the beam refraction angle requirement be increased to ±20 deg, the elevation gimbaled antenna can be readily added. The dual frequency occultation trades are summarized in Table 3-4.

TABLE 3-4. DUAL FREQUENCY OCCULTATION TRADES

Configuration (All antenna assemblies are despun in azimuth)		 A	 B	 C
Design approach		Fixed S band HGA, Separate X band horn 3 W X band transmitter	Moving HGA reflector Fixed S/X feed 0.2 W X band transmitter	Moving spacecraft Fixed antenna with S/X feed 3 W X band transmitter
Increases to add occultation experiment	Cost K$	750	1020	770
	Mass, kg (lb)	5.26 (11.6)	8.40 (18.5)	7.67 (16.9) for 40 days
Reliability		Best	Must accommodate elevation drive failure	Additional thruster pulses required
Science		Adequate (to ±10 deg), pointing accuracy <1 deg. Best interface (separate X band)	Best boresight to earth, ±20 deg, pointing accuracy <1 deg	Worst attitude uncertainty, pointing accuracy <2.5 deg. Impacts radar altimeter usage
Mission operations		Best	Some additional operations	Worst – requires daily attitude control. Uses 1.9 kg (4.2 lb) propellant in 40 days at ±10 deg

Magnetometer Accommodation

Magnetometer accommodation in the orbiter science payload is a critical issue because of the high cost of designing magnetically clean spacecraft equipment. Spacecraft induced magnetic field at the sensor is reduced as the sensor boom length is increased. Spacecraft magnetic cleanliness cost was traded against boom development costs, in the areas of deployment, stowage, mounting, methods of articulation, and effect on spacecraft dynamics. A 4.4 m, three link boom was selected because of lowest program cost, best science performance, and adequate reliability.

Equipment Installation

The selection of spin axis perpendicular to the ecliptic (except maneuver phases) allows for simple equipment installation because the thermal control design is easy to implement. An open equipment shelf arrangement has been selected rather than a compartmentized approach. The open shelf is structurally lighter, more accessible, and less constraining in achievement of mass balance, lower harness weight, and desired power dissipation distribution for thermal control. On the probe bus, the aft end of the spacecraft is selected for thermal louver radiation as the four probes limit use of the forward side. The aft side is also used on the orbiter to maintain commonality. Use of louvers on the aft side and placement of equipment on the forward side also facilitates access, experiment integration, and thermal blanket design and placement. The shelf commonality is further increased by maintaining identical installations of those subsystems and/or experiments that are common to both spacecraft.

Power Bus

An unregulated power subsystem has been selected over a regulated one for both the probe bus and orbiter. While a larger solar panel is required for this configuration and individual subsystem and experiment regulators are required, the unregulated approach eliminates the heavy costly central regulator and is simpler and more reliable. Moreover, since experiments and subsystem units have their own regulators, they achieve better precision in regulation, better EMI isolation, and voltages more suitable to their own needs.

Spacecraft Baseline Design

The following is a combined description of probe bus and orbiter baseline configuration and subsystems. The description of the probes will follow in the next subsection.

Spacecraft Configuration

The configurations of the spin stabilized probe bus and orbiter are shown in Figure 3-7, and an exploded view is shown in Figure 3-8. The

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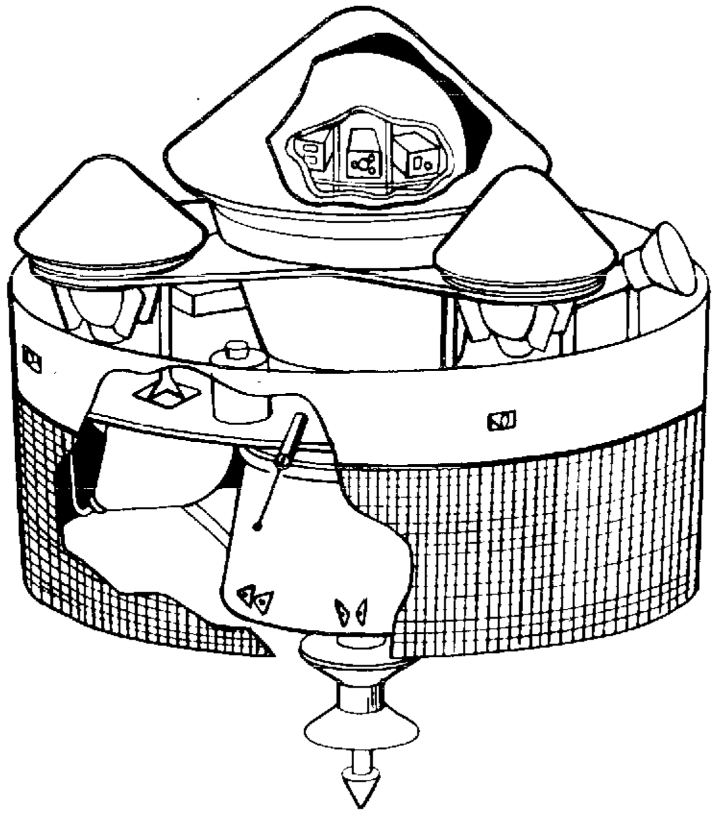
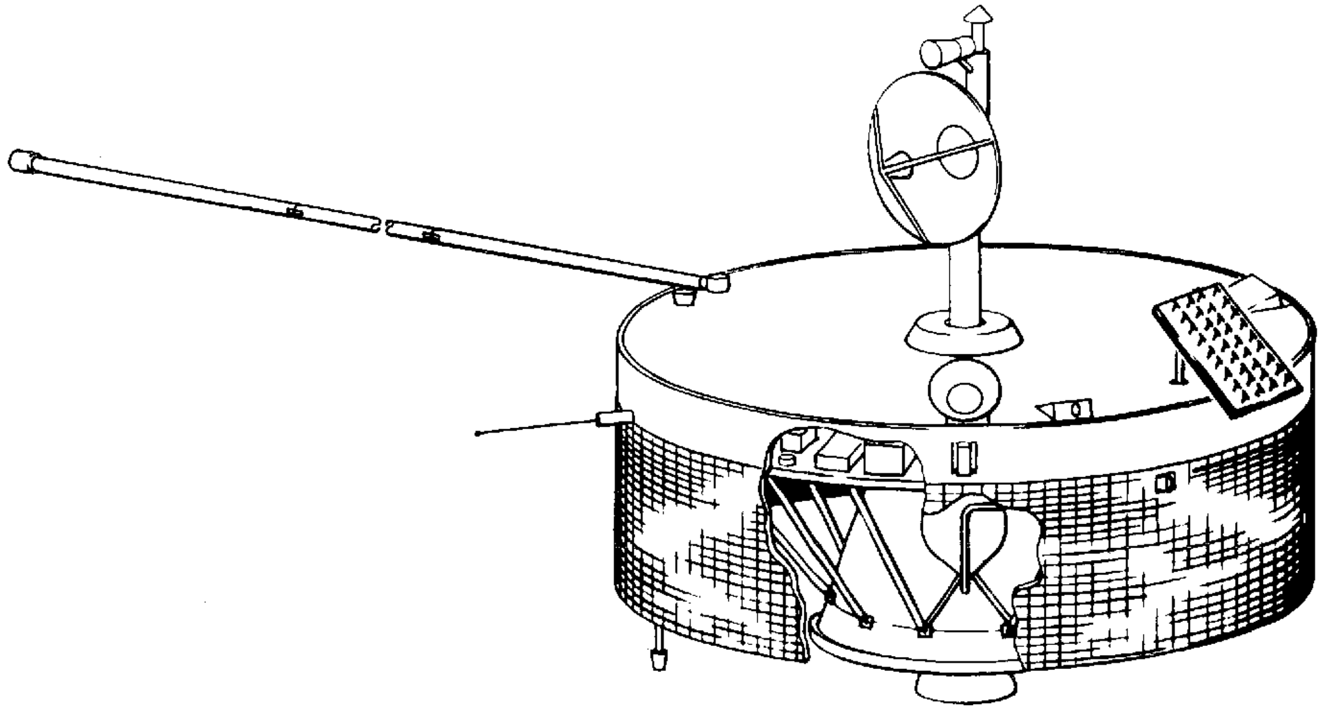


PROBE BUS



ORBITER

FIGURE 3-7. SPACECRAFT CONFIGURATIONS

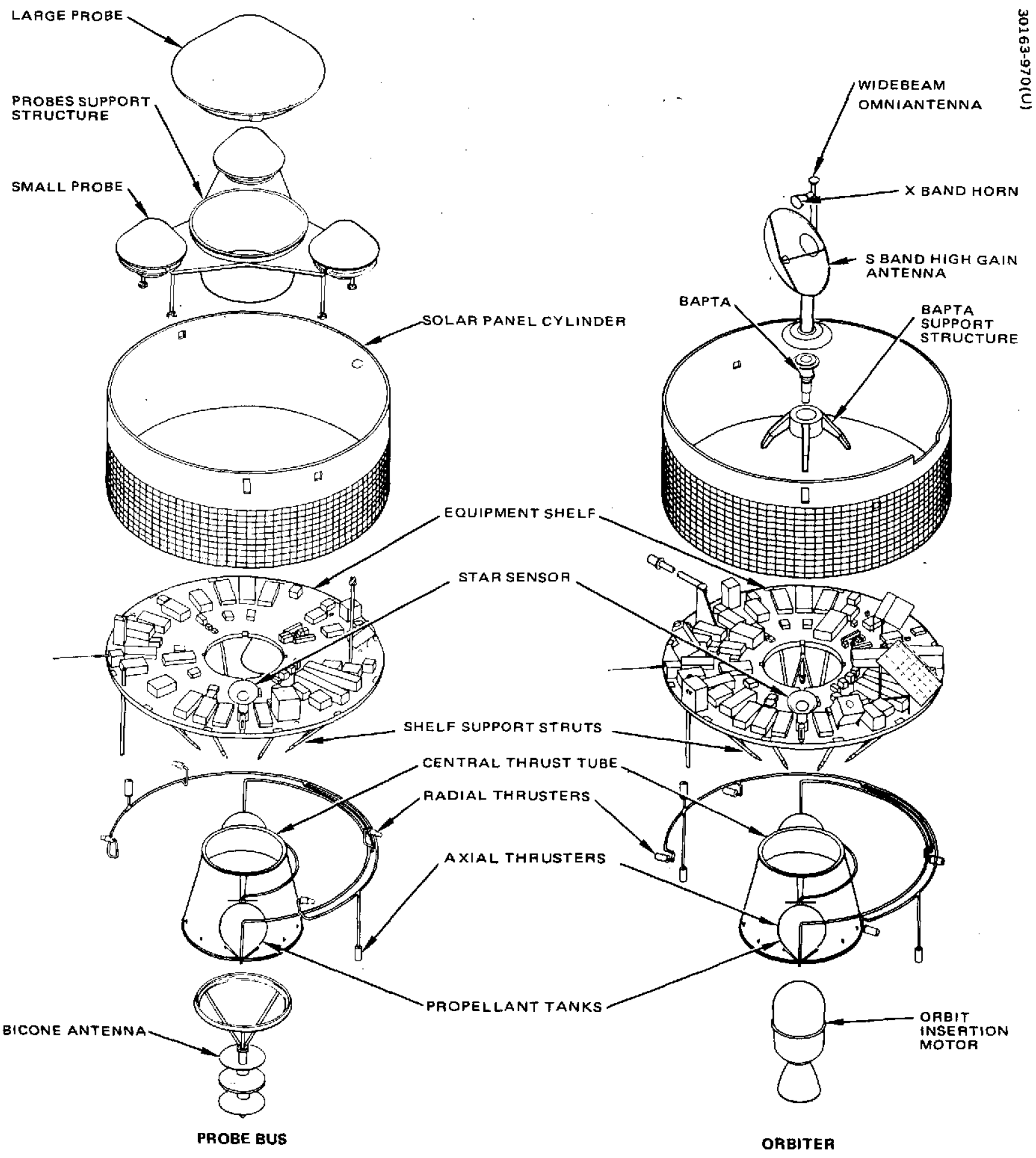


FIGURE 3-8. SPACECRAFT CONFIGURATIONS EXPLODED VIEW

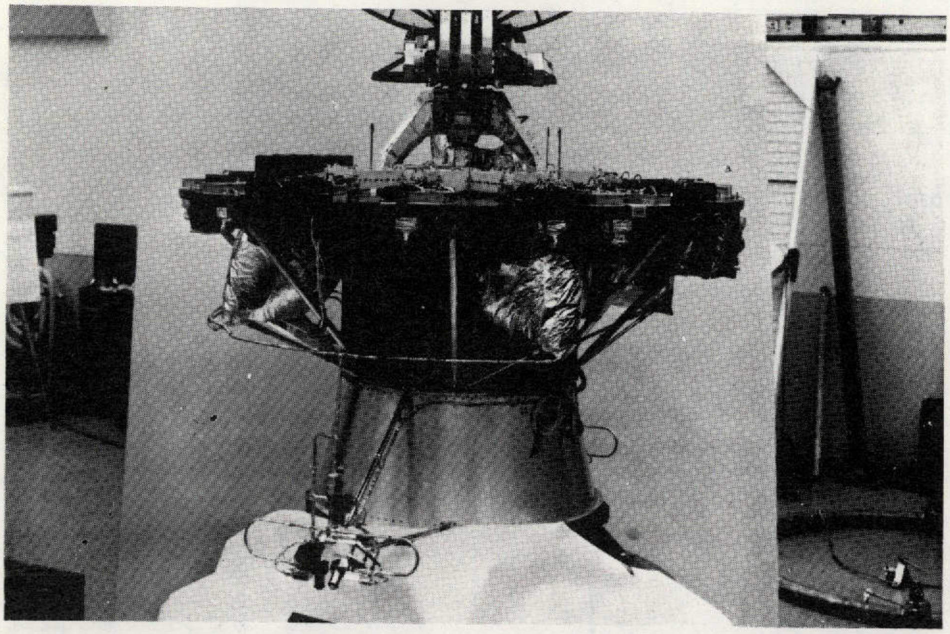


FIGURE 3-9. TELESAT STRUCTURE CONFIGURATION
(PHOTO 30163-971)

configurations are derived from Hughes built flight-proven communications satellites, particularly Telesat-Anik (Figure 3-9), and feature a high degree of subsystem hardware commonality between probe bus and orbiter. Both spacecraft are designed for operation with spin axis perpendicular to the ecliptic, except for maneuvers and special periods such as trajectory correction, probe release, bus entry, and orbit insertion.

The primary structural elements common to the probe bus and orbiter are the central conical aluminum thrust tube, aluminum honeycomb sandwich equipment shelf, 12 shelf support struts, and the 254 cm (100 in.) diameter solar panel cylinder made of fiberglass face sheet and aluminum honeycomb core. On the probe bus, a central conical support structure with associated secondary structure attaches to the upper end of the thrust tube to support the large probe and three small probes. The large probe is separated axially using springs; the small probes are simultaneously spin-separated to minimize targeting errors due to induced nutation. On the orbiter, an existing quadrapod structure (from Telesat) attaches to the forward end of the thrust tube and supports the bearing and power transfer assembly (BAPTA) used to mechanically despin the mast mounted, 82.5 cm (32.5 in.) diameter S band high gain antenna, X band occultation horn, and the forward omni antenna. The Intelsat IV launch vehicle/spacecraft attach fitting is used with both spacecraft.

The hydrazine propulsion tanks and feed are supported on the thrust tube. Four radial and two axial thrusters on the probe bus and four radial and three axial thrusters on the orbiter are attached through support structures to the equipment shelf. The thrusters provide redundancy in trajectory correction and attitude control. A solid state star sensor, mounted on the shelf at an angle of 58 deg to the spin axis, and sun sensors provide the required attitude references.

On the orbiter, the case-stretched TEM-521 solid propellant orbit insertion motor is mounted on the motor attach ring internal to the thrust tube. The bicone antenna is mounted inside the thrust tube on the probe bus.

Most probe bus and orbiter experiments are installed on the forward side of the platform. Velocity oriented experiments are positioned for bus entry conditions and Venus orbit periapsis latitude. The three link, 4.4 m (14.5 ft) magnetometer boom is stowed just above the solar cylinder on the orbiter and is centrifugally deployed, after initial spacecraft spinup, in a plane perpendicular to the spin axis. The electronically steered radar altimeter antenna is positioned with its pointing angle at 34 deg to the spin axis to provide required radar beam pointing at the baseline orbit periapsis latitude of 56 deg. Other latitudes are accommodated by varying the fixed angle of the radar antenna installation. The high gain antenna mast length is sufficient to allow repositioning for a periapsis latitude of 13 deg without rf interference.

Spacecraft subsystem components are mounted on the forward side of the platform. Identical shelf installations are planned for equipment common to probe bus and orbiter. Ten louver modules are mounted on the aft side of the probe bus shelf for primary thermal control by radiation out of the aft spacecraft cavity; twelve are used on the orbiter. The body of the spacecraft is enclosed on all external surfaces (except the outer solar panel cylinder) with multilayer aluminized Kapton blankets. In the aft cavity, the blankets are placed over the solar panel cylinder, the outer thrust tube and tanks, and the aft shelf with cutouts at each louver module.

The spacecraft mass summary is shown in Table 3-5 for the baseline missions, Type I, 1978 for multiprobe and Type II, 56°S, 1978 for orbiter. The experiment payloads used are those specified by NASA-ARC in April 1973 and include the recommended 15 percent experiment contingency. The orbiter experiment payload has been increased by 2.13 kg (4.7 lb) to account for the current Hughes estimate of the additional mass (over the NASA-ARC allowance) required to implement the dual frequency occultation experiment.

Thermal Control

The thermal control subsystem must provide a proper thermal environment in the spacecraft from earth orbit, through transit, and into Venus orbit. Solar intensity changes by a factor of two but the basic design solution is simplified because of spacecraft orientation perpendicular to the ecliptic. Constant attitude with respect to the sun allows for the top and bottom ends of the spacecraft always being available for radiation to dark space. Overall requirements for the mission are conventional thermal limits of about 4° to 50°C. Heaters are used to keep specific equipments within allowable temperature ranges and to circumvent special problems associated with specific maneuvers.

TABLE 3-5. SPACECRAFT MASS SUMMARY

Item	Multiprobe, kg (lb)	Orbiter kg (lb)
Bus (dry)	191.5 (422.1)	317.59 (479.7)
Large probe*	245.1 (540.4)	------ -----
Small probe (3)**	190.8 (420.6)	------ -----
Spacecraft subtotal	627.4 (1383.1)	217.59 (479.7)
Contingency	149.6 (329.8)	25.31 (55.8)
Experiments	13.7 (30.3)	47.54 (104.8)
Spacecraft total (dry)	790.7 (1743.2)	290.44 (640.3)
Propellant and measurement	22.4 (49.4)	26.96 (59.4)
Orbit insertion motor expendables	----- -----	143.33 (316.0)
Spacecraft total (wet)	813.1 (1792.6)	460.70 (1015.7)
Spacecraft attach fitting	31.3 (69.0)	31.30 (69.0)
Launch vehicle payload	844.4 (1861.6)	492.00 (1084.7)

*Includes 31.6 kg (69.7 lb) for experiments

**Includes 2.6 kg (5.8 lb) for experiments (for one probe)

The thermal control subsystem is made up of three basic elements: 1) thermal insulation blankets made of sheets of aluminized Kapton, 2) louvers to keep the internal temperature of the spacecraft constant by radiating excess heat away from the baseplate of the platform, and 3) electric heaters to maintain the temperature of specific items. The blankets of superinsulation keep the heat generated by the operating components within the spacecraft and prevent it from radiating away into space. Thus, there are layers over both ends of the spacecraft, except for the louvered areas, which must have radiating fields of view. The injection motor on the orbiter and the probes on the probe bus are thermally isolated and are essentially independent of the spacecraft so that the spacecraft can operate effectively when the probes are separated.

While the insulating blankets serve to suitably maintain the internal temperature by conserving the heat generated internally, louvers are necessary to radiate away some of this heat, particularly when the spacecraft reaches Venus. A louver module consists of four bimetallically actuated blades. Ten louver modules are mounted radially on the aft side of the probe bus equipment shelf; twelve modules are used on the orbiter.
High temperature components such as power amplifiers and the converter units are mounted directly above the louvers. Since the louvers are mounted to the bottom of the platform, this part must be kept out of sunlight. However, at orbit injection the aft end must be exposed to sunlight. The exposure angle depends upon the periapsis location. For a 56 deg latitude periapsis, components will begin to reach their maximum allowable levels after about 3.5 to 4 hrs.

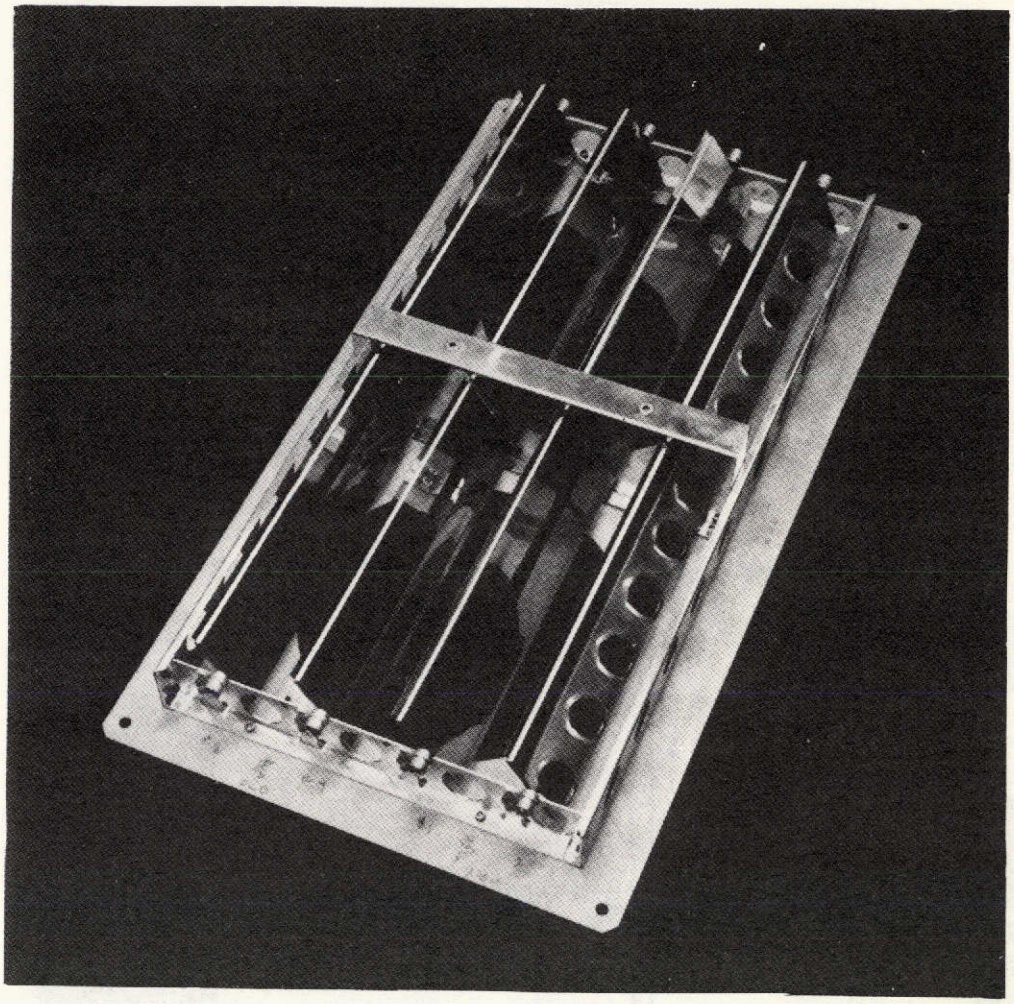


FIGURE 3-10. THERMAL LOUVER MODULE
(PHOTO A24594

Insulation and thermal coating techniques have been used extensively, beginning with Surveyor, and currently on a number of commercial satellites such as Intelsat IV and Telesat. A louver module similar to those to be used on Pioneer Venus is shown in Figure 3-10 and has been used on a military satellite.

Sufficient insulation and thermal coupling to the rest of the payload could be provided at the beginning of the mission. However, excessive heating would appear in the Venus orbit. The solution provided here is insulation sufficient for the Venus orbit and the use of heaters during the early portion of the mission. The heaters, added where necessary, maintain the temperatures of the propulsion lines, tanks, and thrusters and are also used to maintain the probe temperatures at the beginning of the multiprobe mission.

The probe mission presents a special problem because it is desirable for the probe temperatures be relatively low at separation time. The

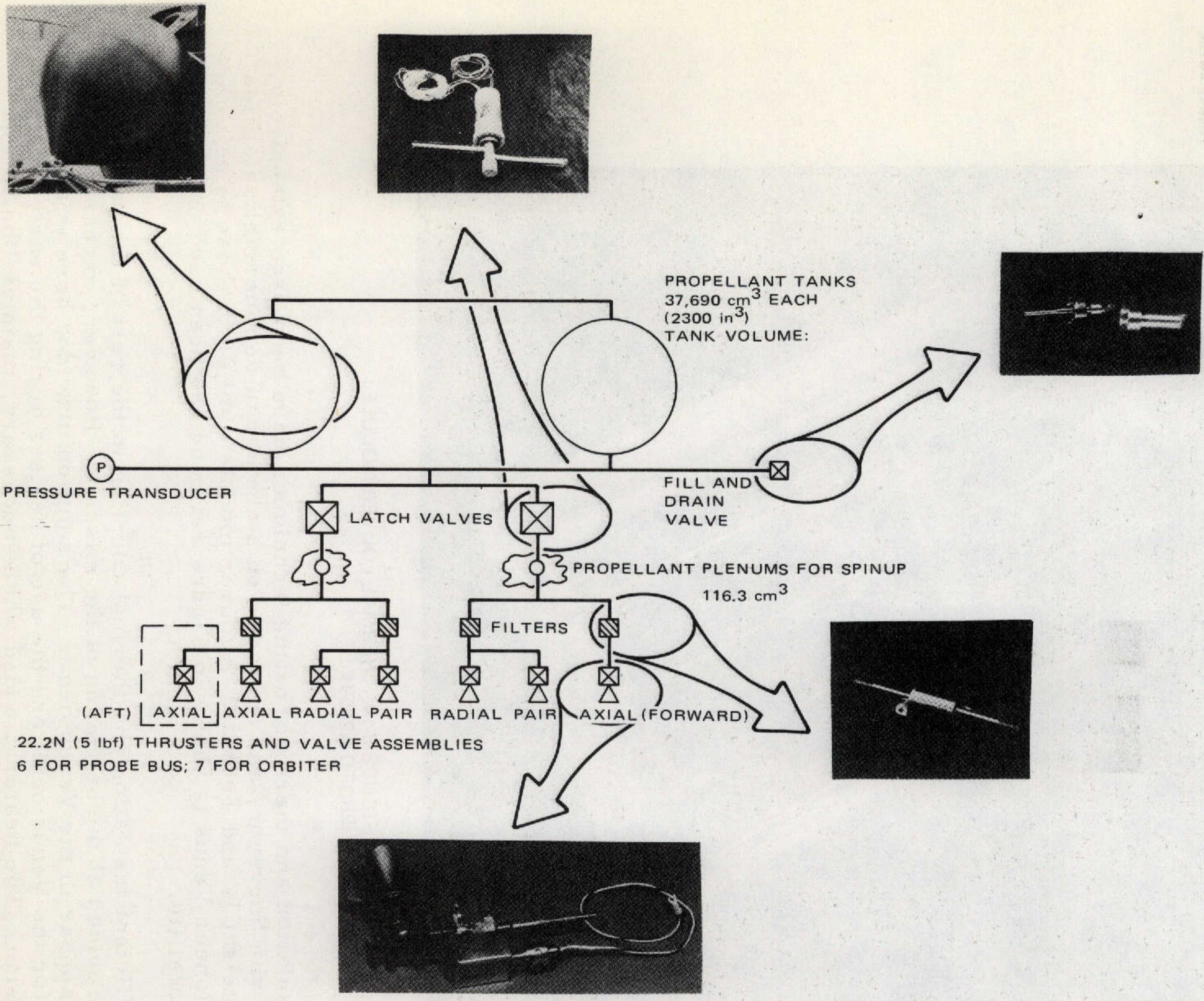


FIGURE 3-11. PROPULSION SUBSYSTEM

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probes view the sun so that they must be coated to reduce the absorption of solar radiation. Because of the coating, probe temperatures could be excessively low at the beginning of the mission, near earth. Thus, probe temperatures must be maintained by heaters during the initial phase of the transit.

Propulsion/Orbit Insertion

The spacecraft propulsion subsystem, shown in Figure 3-11, must provide propulsion for midcourse corrections, attitude orientation maneuvers, spin speed control, and orbit injection.

The propulsion subsystem uses monopropellant liquid hydrazine, operates in a pressure blowdown mode, and stores propellant and pressurant together in a common tank to provide low mass, reliable design. Two tanks are located 180 deg apart about the spacecraft spin axis. The tank size selected accommodates the propellant required for either the orbiter or probe spacecraft, thus providing common hardware for the two configurations. The propellant/pressurant interface and propellant orientation within the tanks is controlled by the centrifugal force associated with spacecraft rotation. This represents a low cost, low mass, high reliability design.

Interconnecting manifolds equalize the pressure between tanks and distribute the propellant uniformly from each tank through appropriate filters to the thrusters. Thrusters are arranged in two groups, for redundancy, with one axial and a pair of radial thrusters in each group for the probe spacecraft configuration and an additional aft mounted axial thruster for the orbiter configuration. A bistable latch valve in the manifold feeding each thruster group permits isolation of that group in the event of propellant valve leakage. A pressure transducer in the liquid manifold provides a telemetered reading of subsystem internal pressure. Propellant flow is controlled by electrical signal to the torque motor operated, dual-seat valve supplied with each thruster. The propellant decomposes in the thruster and the hot gases exhaust through the nozzle to produce the required thrust. Propellant and pressurant are loaded through a single fill and drain valve, a concept proved successful on the Intelsat IV, Telesat, and Basic Bus programs.

The propulsion subsystem is basically an all-welded system fabricated from 6Al-4V titanium (a technique developed on Intelsat IV and used successfully on all subsequent satellites). Weld surfaces with some stainless steel components are accomplished with titanium to stainless steel diffusion bonded, coextruded transition joints. Thus, the only mechanical joints in the system are the redundantly sealed fill and drain valve and propellant control valves for each thruster. All components have been flight proven on Intelsat IV or Telesat or qualified for flight on Intelsat IV or Marisat. The subsystem uses passive thermal control by placing components and interconnecting manifolds under insulation blankets. Heaters are required on each thruster valve as well as on each tank to maintain the respective temperatures above the allowable minimums.

The baseline orbit insertion motor for the Atlas/Centaur, Type II trajectory mission consists of a Thiokol Model TE-M-521 motor modified for the required propellant load. The propellant and expended inert mass required to place the spacecraft into Venus orbit is 143.4 kg (316.0 lb) requiring a lengthening of the cylindrical portion of the titanium motor case by 13.0 cm (5.1 in.), with corresponding modification to the internal motor insulation and nozzle design to accommodate the increased mass flow. Additionally, the high density TP-H-3135 propellant replaces the lower energy TP-H-3135 propellant to provide a performance advantage.

Attitude Control

The attitude control subsystem must be capable of moving the spin axis of the spacecraft in any spatial direction so that midcourse corrections in any direction can be made, if required. In addition, spacecraft spin speeds from a few rpm to 71 rpm must be controlled by the subsystem.

The orbiter spacecraft and the probe bus have identical attitude control subsystems. A block diagram of the attitude control and mechanisms subsystem (ACMS) is shown in Figure 3-12. Attitude is determined to 0.9 deg and controlled to 1.2 deg.

Attitude determination is provided by data from a group of three sun sensors and a star sensor using a silicon detector and a 5.2 cm (2 in.) telescope. The star sensor is sensitive to +1 magnitude (silicon) stars. There are 25 stars brighter than +1 magnitude, 17 in the southern hemisphere and 8 in the northern hemisphere, so that either hemisphere contains sufficient stars for attitude determination. The star sensor has a fan beam slit of 1 x 25 deg and its mast is about 2.5 kg (including a sun shield). Both sun and star sensor configurations contain redundant detectors and electronics. The sun sensors are identical to those currently operating on Hughes satellites.

The attitude data processor provides a phase locked loop which, together with the sensor system, generates the inertial reference and provides body angular references for the jet firings, despun antenna, and experiments. The command system supplies the firing impulses through the command memory using the angular data from the attitude processor to solenoid drivers. There are six thrusters on the probe bus and seven on the orbiter. These assure redundancy in all firings: axial and radial for velocity corrections, spin up and spin down, and axial pulsing for orientation control. The angular data supplied to the experiments are referenced to the sun or to a selected star if the sun is eclipsed.

The despun antenna bearing and power transfer assembly (Figure 3-13) supports the main despun 82.5 cm (32.5 in.) parabola, the X band horn, and an omni antenna. The mechanical assembly is the same type of system as is being used on the Canadian Telesat satellites. Redundant brushless motors and electronics units provide the reliability necessary for the mission. Seven sliprings transfer power: two for antenna switching, three for housekeeping data, and two for growth. This assembly is guaranteed on

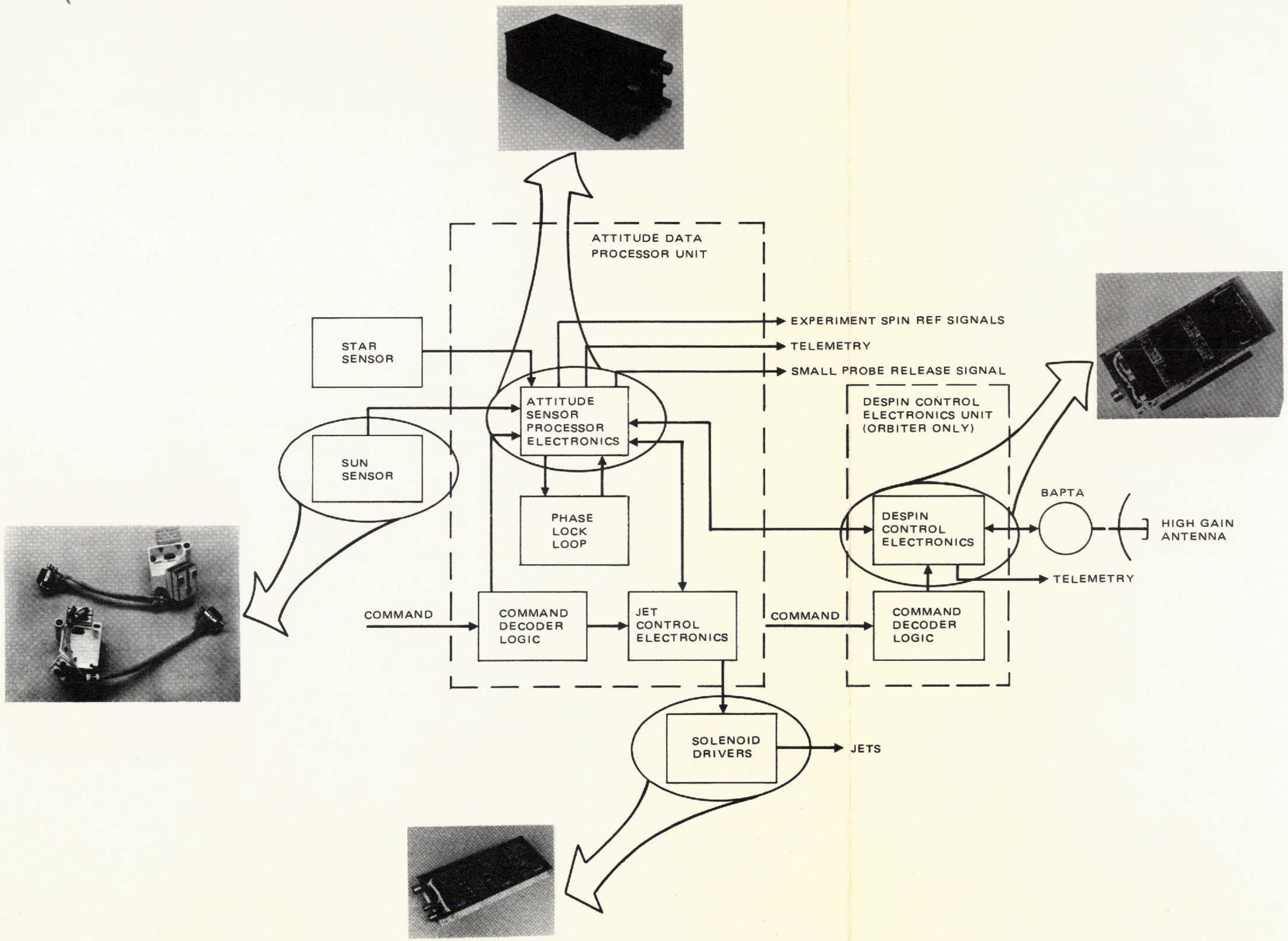

FIGURE 3-12. ACMS FUNCTIONAL BLOCK DIAGRAM

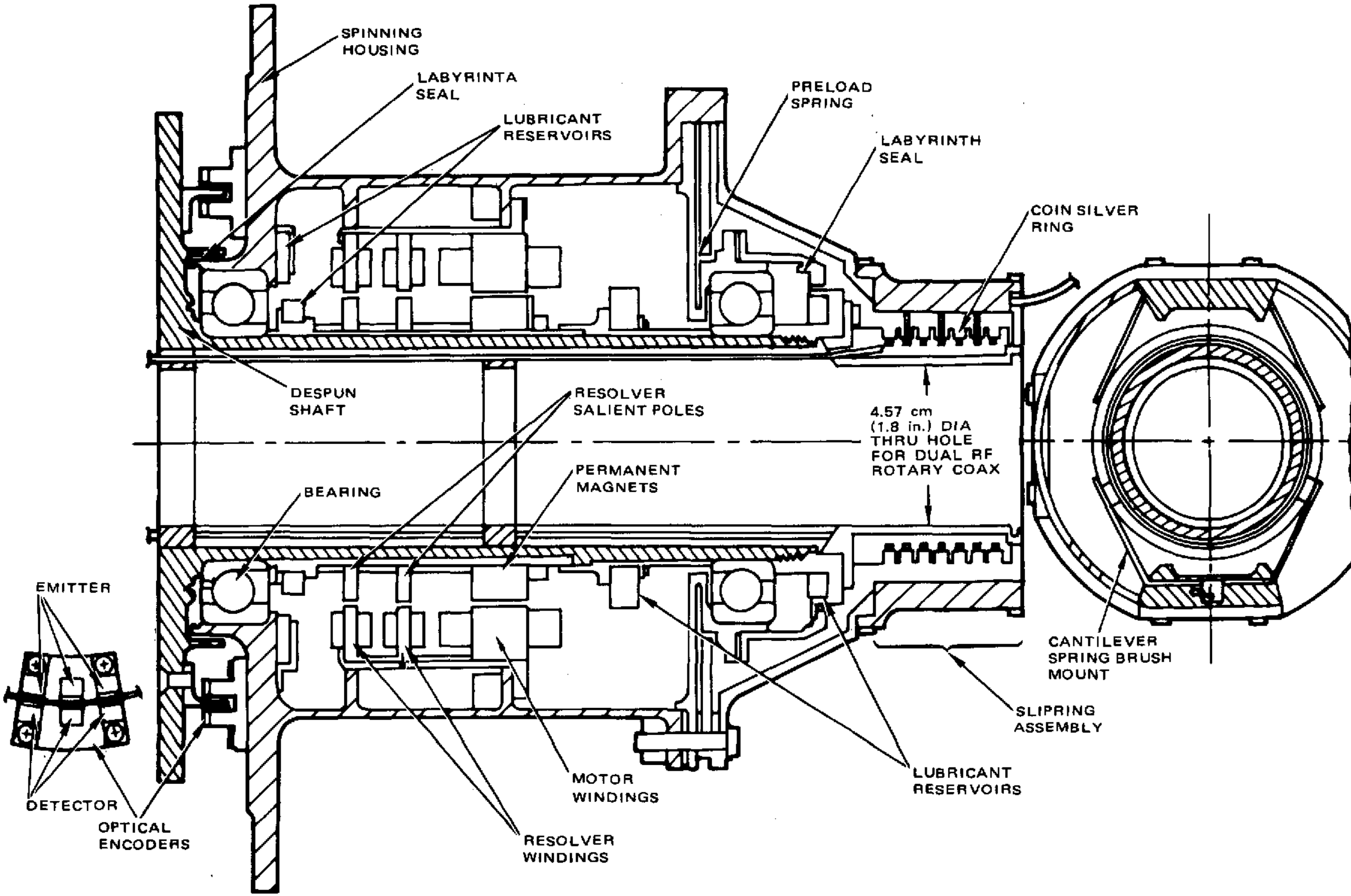

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FIGURE 3-13. BAPTA ASSEMBLY CROSS SECTION

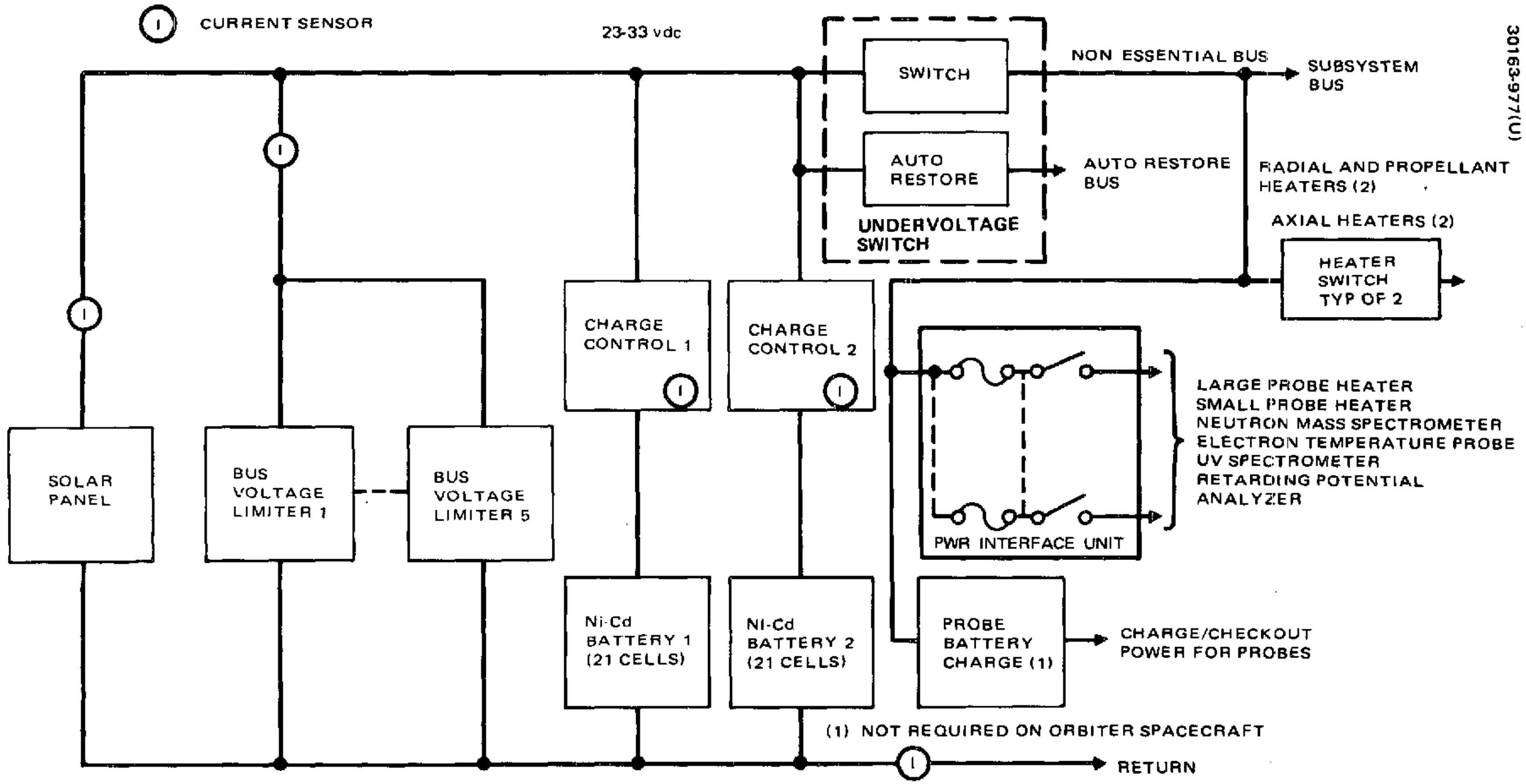

FIGURE 3-14. ATLAS/CENTAUR PROBE SPACECRAFT POWER SUBSYSTEM

Hughes commercial satellites for 7 yr. The despin electronics unit controls the antenna angular position with respect to the selected inertial reference by using data from the shaft angle encoder and the data processor assembly. The angle with respect to the sun or star pip is commanded from earth and the step size is 0.4 deg. The rate of change in the angle to earth varies each day, but near the end of the mission, it is also about 0.4 deg/day. A single 8.9 N (2 lb) mercury ring damper of the type used on many spacecraft provides nutation damping on both spacecraft.

Electrical Power

The spacecraft electric power subsystem for both the probe bus and orbiter is similar to provide commonality although the power requirements of the probe bus are substantially less. For the probe bus, the power system must supply stored power during the launch sequence and solar power in the transit phase, as well as possibly some requirements for stored power during 3σ midcourse corrections. Power must be supplied during probe separation and up through entry of the probe bus into the Venus atmosphere. In the orbiter, power must also be supplied during the orbit insertion sequence and in orbit, in particular during solar eclipse.

The probe bus and orbiter spacecraft power subsystem block diagrams are shown in Figures 3-14 and 3-15. The solar array is sized in case of the orbiter mission to supply 193 W at the end of mission life and 170 W for the probe bus at Venus. The major source of degradation is solar proton event and a large but not maximum solar proton event has been assumed to occur on the first day of the mission. The result of this assumption is to reduce the array output by about 5 percent. Two ohm/cm n/p cells have been selected, 0.20 mm thick and with a 0.15 mm coverglass. The array assembly techniques and general configuration is the same as that used in the current Canadian Telesat design.

A nickel cadmium, 7 A-hr battery has been selected on the basis of the orbiter's eclipse cycle. The short daily eclipse period that occurs early in the mission and the long eclipse at the end of the mission require nickel cadmium. A depth of discharge of approximately 70 percent is allowed. Two 21 cell battery packs have been assumed both for reliability and for efficiency of design in an unregulated system. A charge control, using constant current has been selected, thus making use of equipment currently in existence for the OSO spacecraft. Before the battery reaches a specified level, it charges at whatever level the array supplies over and above the spacecraft load. The battery discharge control assures that the bus will not run below the allowable 24 V. Five shunts provide limiting on the probe bus and six provide limiting on the orbiter. Each handles 66 W and, like the other equipment, is based on the current OSO design. On the probe bus, a battery charger is used to maintain the probe silver zinc batteries during transit if required.

Communication

The communication subsystem block diagram for the spacecraft is shown in Figures 3-16 and 3-17. As can be seen, the basic components for each of the systems are identical. The basic elements consist of a transponder (receiver and exciter), both from the Viking program and both in redundant configurations.

In the orbiter, the exciters drive two solid state power amplifiers in parallel, which in turn supply a combined output power of approximately 17 W. This output is then filtered and delivered to the antennas, either the high gain or the omni antennas. The power amplifiers are presently under development for a current Hughes program and will be available at the end of the year. The omni antennas are also from past Hughes programs. The high gain parabolic antenna is essentially the same unit as is currently flown of the Canadian Telesat program. This general configuration is straightforward and almost all of the components are fully developed.

Command/Data Handling

The spacecraft command subsystem must be capable of receiving and executing commands in real time and must be able to accept and store commands for later execution.

The command subsystem is shown in Figure 3-18. Each of two demodulators is attached to one of the two receivers and then fed into a redundant central decoder. Decoded commands are then distributed to remote decoders which support specific sets of spacecraft experiments and equipment. Another demodulator function is selecting antennas and providing receiver reverse switching. The command subsystem also provides the pyrotechnic control units which supply power to fire the squibs. Key design parameters are shown in Table 3-6.

TABLE 3-6. COMMAND SUBSYSTEM DESIGN PARAMETER SUMMARY

Modulation	Phase shift keying	
Command word size	36 bits	
Command rates	4 bps	
Command numbers	Probe	Orbiter
Pulse commands	158	167
Magnitude commands	8	7
False command probability	1 in 10^9	
Storage capacity (both missions)	85 words of 24 bits each	
Timing accuracy	±0.5 sec within 1 h*	

*Adequate for orbit insertion. Higher accuracy, if required, is readily achievable.

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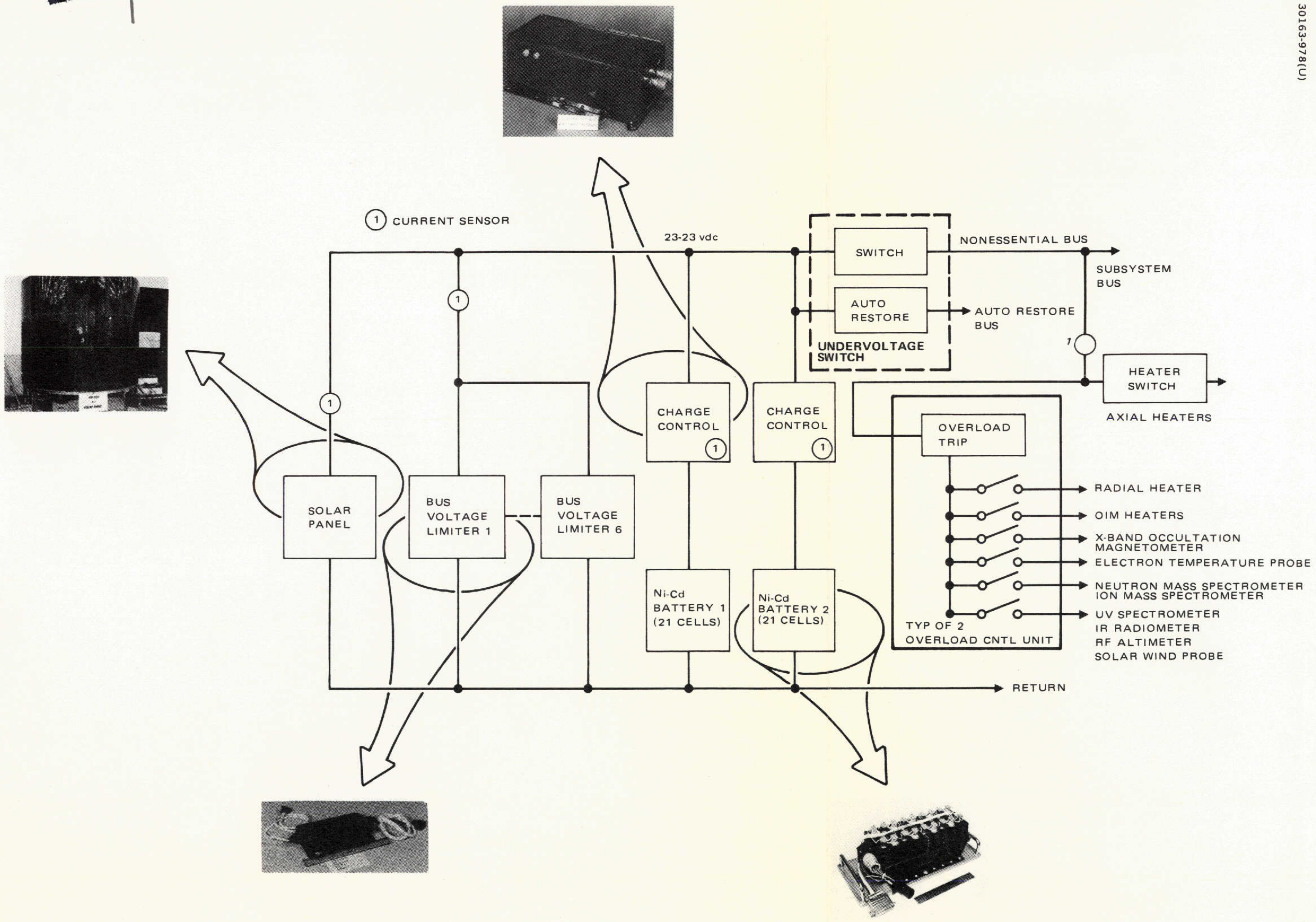


FIGURE 3-15. ATLAS/CENTAUR ORBITER POWER SUBSYSTEM

FOLDOUT FRAME 1

FOLDOUT FRAME 2

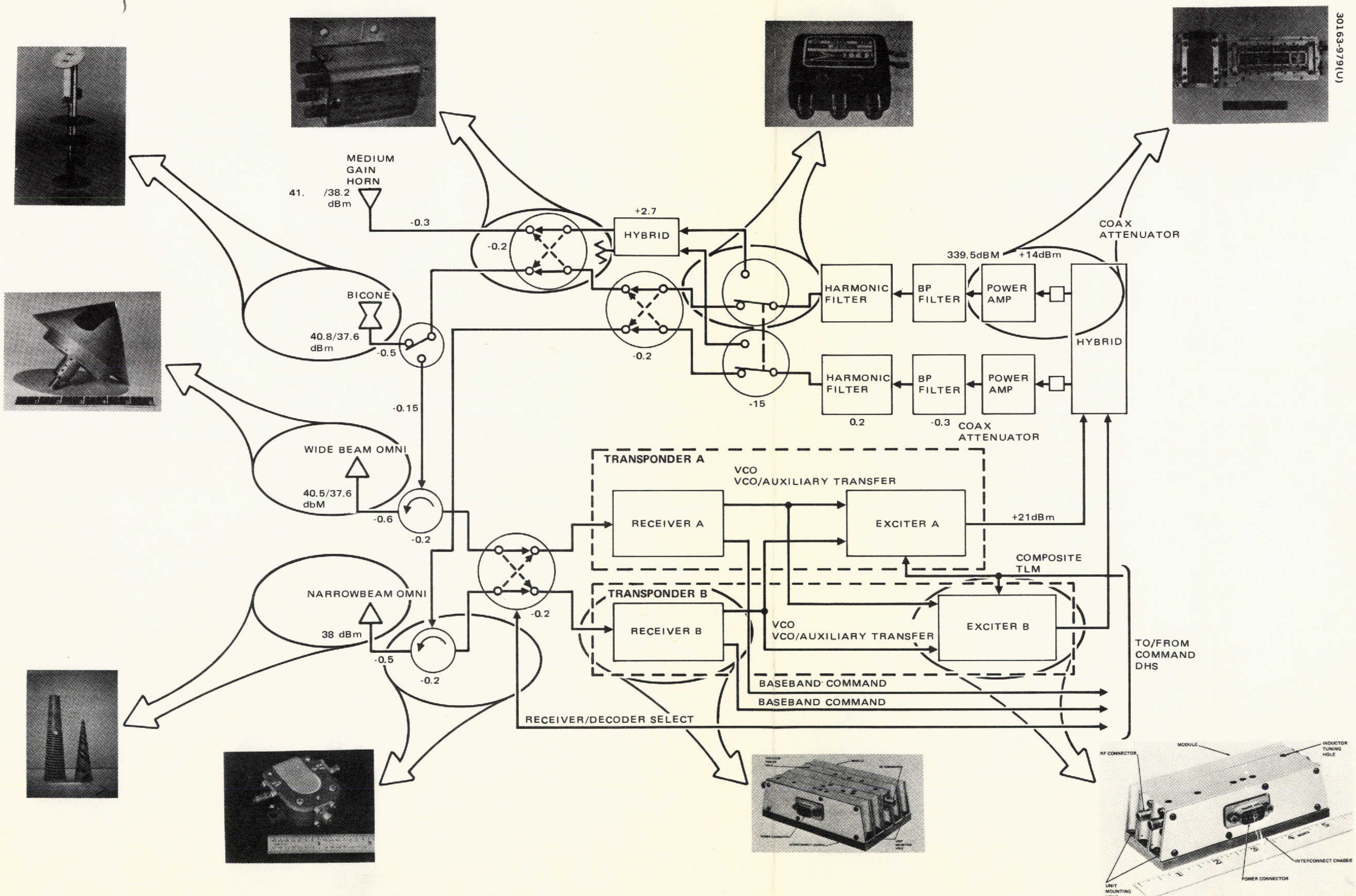


FIGURE 3-16. PROBE BUS COMMUNICATIONS SUBSYSTEM

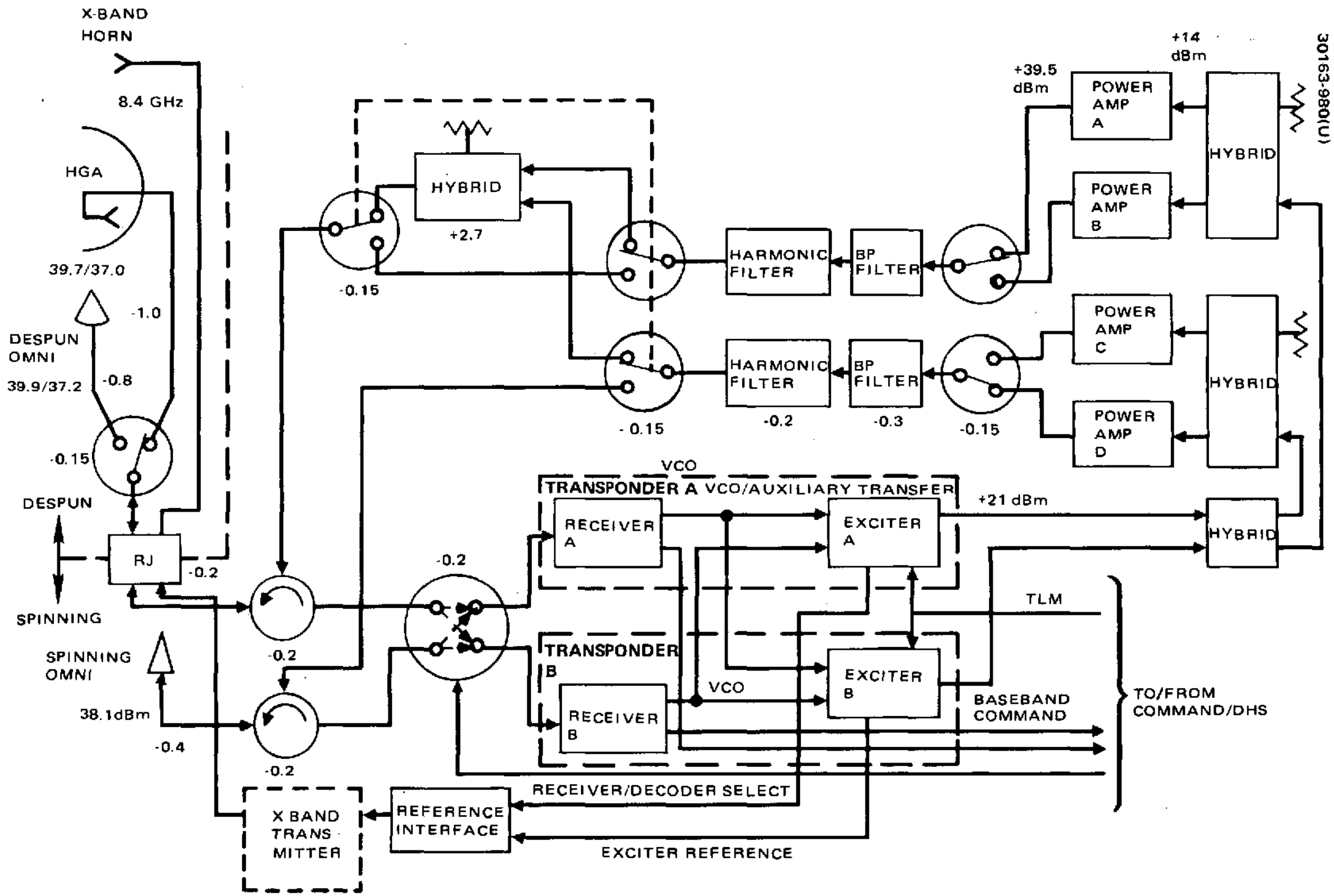


FIGURE 3-17. ORBITER COMMUNICATIONS SUBSYSTEM

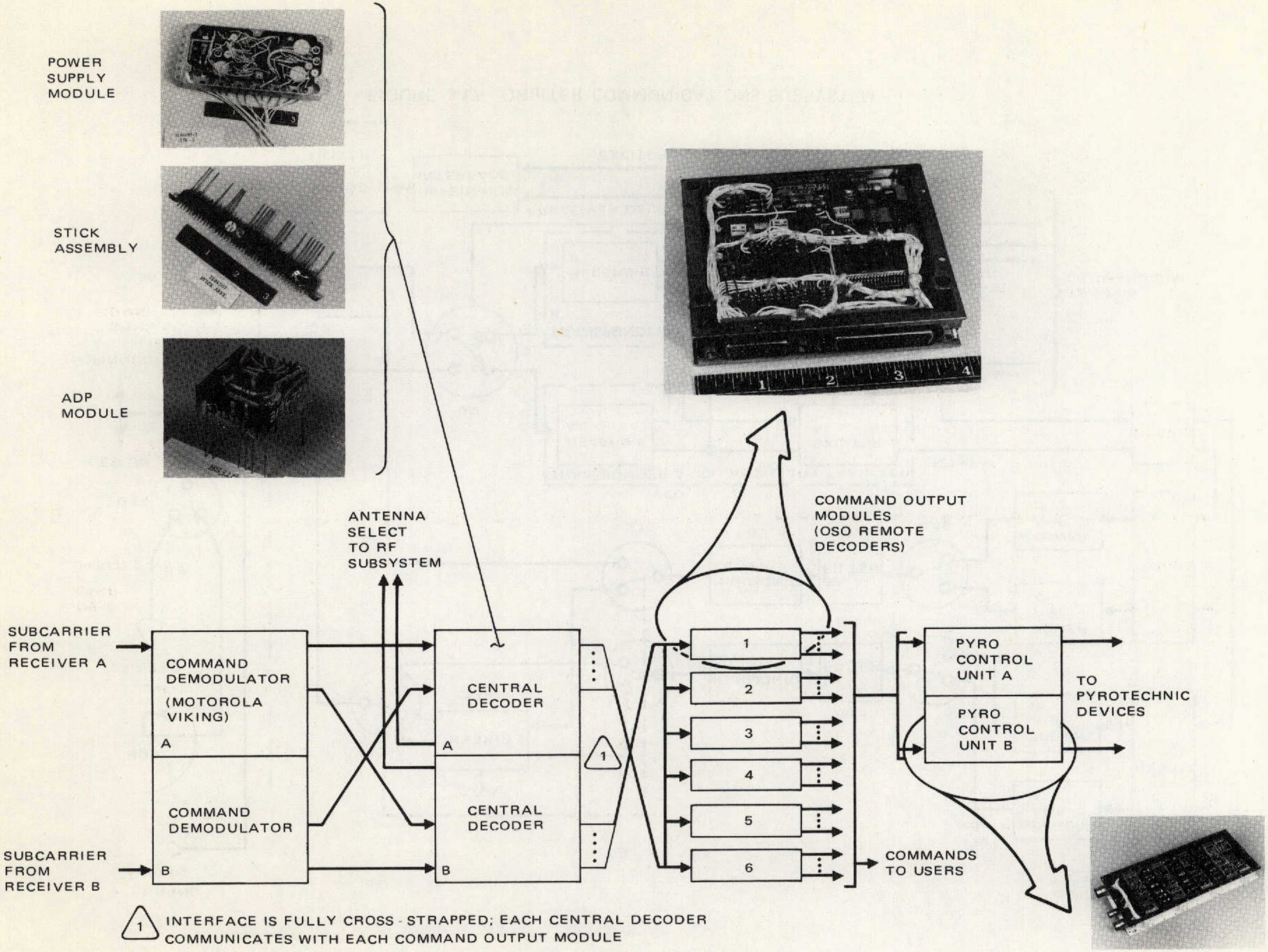


FIGURE 3-18. COMMAND SUBSYSTEM

The basic hardware is existing OSO equipment which provides a flexible, modular system well suited to meeting the Pioneer Venus mission requirements. A Motorola unit developed for the Viking program is used for the dual demodulator. The unit's central decoder is a slightly modified version of the current OSO central decoder and incorporates a new packaging design and some specific new circuits. The remote decoders are identical to the OSO units and need no modification. The pyro control units are modifications of the OSO pyro control and are 70 percent developed.

The spacecraft data handling subsystem must process signals from the scientific experiments and the spacecraft housekeeping functions so that they can be transmitted back to earth. The subsystem must also provide data storage for the occulted periods as well as possible backup data modes.

Subsystem functions are to generate the addresses of the data to be sampled, to sample the data, to send timing signals to users, to convert analog data into 8 bit words where necessary, to multiplex all input data for the appropriate transmission rate, to encode and modulate this data, to format the data into serial bits for transmission, and to provide bulk data storage. The present ground data system being developed for the Viking configuration appears to be completely suitable for the Pioneer Venus subsystem.

The basic Pioneer convolutional coding scheme is to be used unless Viterbi coding is available at the time of program go-ahead. Key features of the subsystem are shown in Table 3-7.

The basic data handling subsystem (Figure 3-19) is modular and is based on the current OSO design. Seven remote multiplexers are located near the instruments which they sample. These multiplexers transmit the data to a redundant PCM encoder. The coded data are then transmitted to the central telemetry processor which can direct the data to the transmitter or into the storage unit. The central processor has the additional function of extracting data from the storage unit for transmission to the ground.

TABLE 3-7. DATA HANDLING SUBSYSTEM DESIGN FEATURES

Modulation	PCM/PSK/PM
Data rates	8 to 2048 bps in binary steps
Major data formats	16
Data storage requirements	900,000 bits*
Word size	8 bits

*One million bits storage is proposed.

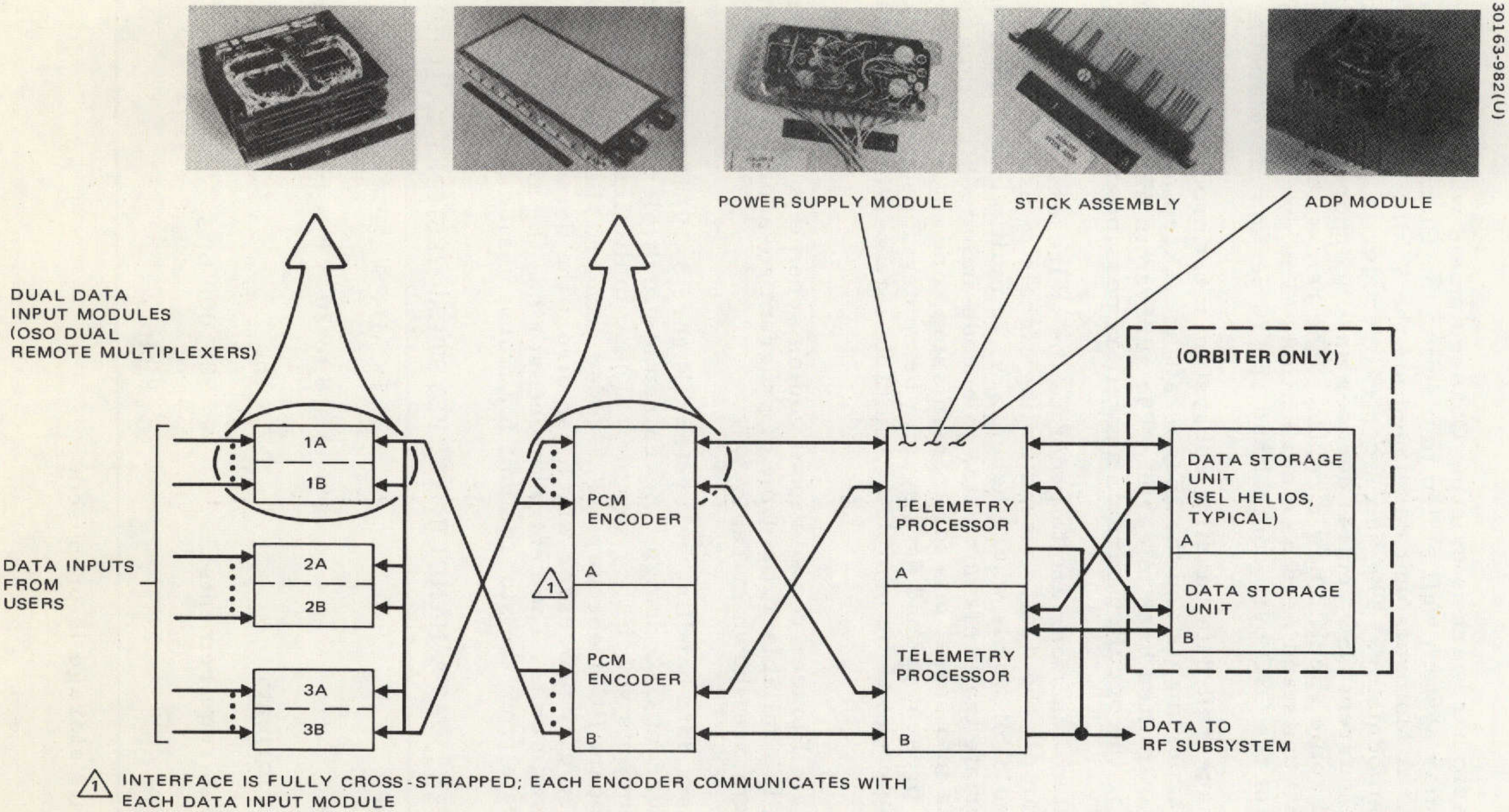


FIGURE 3-19. DATA HANDLING SUBSYSTEM

The remote multiplexers and the PCM encoder are taken "as is" from OSO. The telemetry processor is a new design but 70 percent of its circuits are from the OSO unit. The 500, 000 bit storage device developed by SEL of Germany for the Helios program offers an excellent potential for possible use as the data storage unit if the device can be proven before program go-ahead. If not, a new data storage unit, consisting of two 500,000 bit units, must be used.

Experiment Accommodation

The orbiter experiments are relatively straightforward. The high gain parabola is used for the S band occultation experiment. An 11 deg fixed S band beam appears adequate to meet the 10 deg refraction requirement. The 20 deg X band horn and 3 W transmitter are sized similarly. (If it becomes necessary to track the earth for the occultation experiment, an elevation gimbal can be added.) The radar altimeter is assumed to be a phased array which will move its beam ±45 deg so as to measure the local vertical 4000 km from periapsis. It is mounted on the forward end so that it can look conveniently at a 53 deg south periapsis with the forward end north. The infrared radiometer and ultraviolet spectrometer are also mounted so they can scan to the local vertical at periapsis.

The magnetometer is mounted on a three-piece 4.4 m boom to assure magnetic isolation from the spacecraft at minimum cost. The solar wind probe is mounted so that it spins in the plane of the ecliptic, seeing the sun once each spin. The electron temperature probe is mounted on the bare area above the solar cells and is extended 40 cm from the spacecraft surface.

Both the ion and neutral mass spectrometers are mounted so their inlet systems are parallel to the velocity vector at periapsis, which is adequate for certain types of instruments. For instruments requiring a ram velocity and a wide coverage before and after periapsis, a different approach, possibly a platform, will be examined during the proposal.

The probe bus ultraviolet spectrometer must be mounted to view the whole planet at long range and for limb scanning near the planet as well as requiring a vehicle spin rate of 60 rpm. The ultraviolet spectrometer is positioned on the equipment shelf for an unobstructed forward look angle. A mounting angle of 20 deg to the spin axis is provided as specified, but analysis indicates a mounting angle of up to 40 deg may be required, which can easily be accommodated. The 60 rpm spin rate is also provided.

The electron temperature probe and ion mass spectrometer also requires coating of the solar cells to prevent positive charge buildup on the spacecraft from degrading particle measurements by the instruments.

3.5 PROBES

Four entry probes, one large and three identical small probes, separate from the probe bus and enter the Venusian atmosphere at several locations for in situ measurements of the clouds and lower atmosphere. All probes are comprised of two major elements: 1) a deceleration module composed of an aeroshell and heat shield and, in the case of the large probe only, a parachute, and 2) a pressure vessel for housing and protecting the science instruments as well as the supporting subsystems such as power, radio, and data handling and command.

The deceleration module is separated from the large probe pressure vessel after high g entry to facilitate science instrument access to the atmosphere. The small probe deceleration module and pressure vessel remain attached throughout the descent.

Major deceleration module trades involved the aeroshell configuration and the heat shield material. The primary pressure vessel trades were those related to the vessel configuration, thermal protection, structure, subsystem integration, science instrument integration, and power bus. These are summarized in the following paragraphs.

Major Trades

Aeroshell Configuration

A blunt aeroshell aerodynamic configuration is required to reduce probe speeds, thus allowing initiation of science instruments at altitudes above the Venusian clouds. The aeroshell configuration must also maintain stability during entry into the Venusian atmsophere. A common forebody half cone angle is selected for both the large probe and the small probes in order to minimize aerodynamic testing and hence developmental costs. A 45 deg half cone angle is selected to provide the required stability and necessary speed reduction. Larger angles such as 55 deg were considered because of potential weight saving but rejected due to decreased small probe subsonic stability margin during descent through the atmosphere.

Heat Shield Material

The heat shield material protects the pressure vessel modules from heat loads during entry. The primary candidates for heat shield were phenolic nylon and carbon phenolic; both have extensive flight background and are low in cost. Phenolic nylon is the lighter of the two but carbon phenolic has a much smaller mass loss during entry. Carbon phenolic also maintains a more constant and predictable drag coefficient. Minimum mass loss and predictable drag are very important in the atmospheric reconstruction experiment. Thus, carbon phenolic is selected for the baseline.

Pressure Vessel Configuration

A major consideration in the selection of the large probe pressure vessel configuration is its aerodynamic stability. Since the structure must withstand the high Venusian atmosphere pressure with minimum weight, the basic configuration is spherical. The aerodynamic configurations considered were essentially derivatives or augmented versions of spheres, i.e., stepped sphere, ring-stabilized, flared, and fin-stabilized versions. The selected configuration is a ring-stabilized, stepped sphere with spin vanes integrated into the ring. The spin is required during descent to provide roll rate for proper operation of the single beam wind/altitude radar. The chief reasons for selection are excellent stability characteristics and ease of integration into the deceleration module. The small probe pressure vessels are not separated from the deceleration module during descent. Hence they are simply spheres, with stability augmentation provided by the deceleration module.

Thermal Protection

Large and small probe pressure vessels must protect the equipment from the high atmospheric temperatures. Two fundamental approaches are available for thermal control: 1) external (cold wall) or 2) internal (hot wall) insulation. External insulation with Min-K, a rigid, low density porous material, is the minimum weight solution. However, the internal insulation (fiberglass) scheme does not expose the insulation to the unknowns of the Venusian atmospheric environment. Its performance is already well characterized, and the developmental testing required is relatively minimal. Furthermore, installation onto the pressure vessel is simpler, and access to the equipment on the shelves of the pressure vessel will also be easier. All these advantages lead to lower overall cost and higher reliability for the internal insulation approach. Hence, the choice is made to use the internal insulation design with conventional FA fiberglass material.

Pressure Vessel Structure

The pressure vessel structural material must provide the strength to withstand the high atmospheric pressures at a low cost and with minimum weight. The structural material must also be chemically stable within the Venusian atmosphere. The competing choices in materials are steel, titanium, and beryllium. The contenders for structural configuration are monocoque, waffle, and sandwich constructions. Titanium, a leading material candidate because of its light weight, was rejected because it has autocombustion properties in a high temperature and pressure environment. Beryllium is too expensive to be considered. The most conservative and lowest cost choice is a high temperature, high strength maraging steel which is easy to fabricate. The lightest structural configuration is monocoque, which is also the lowest cost. Hence, the final selection of a steel monocoque configuration is both a low cost and a reasonably lightweight solution.

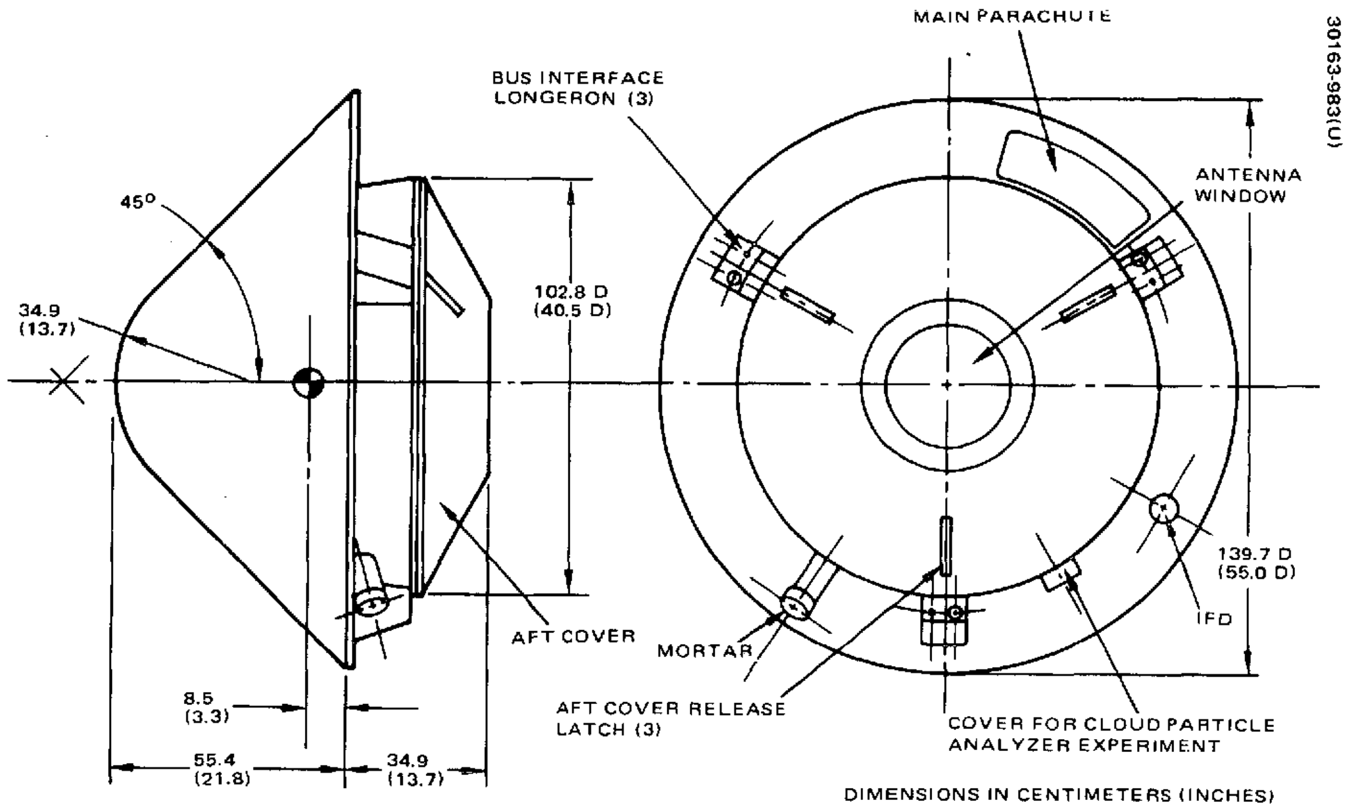


FIGURE 3-20. ATLAS/CENTAUR LARGE PROBE CONFIGURATION

$\left(\frac{M}{C_D A} = 149 \text{ kg/m}^2\right)$

Experiment Integration

Integration of the science instruments into the pressure vessel is critical because of the requirement to provide access to the environment and also to provide protection from the deceleration loads and high pressure and temperature. Critical issues relate to horizontal or vertical shelf mounting, window design, and methods of coping with possible external window contaminants/condensation. Ease of instrument accessibility leads to the shelf mounted approach. The bottom shelf is supported by flanges attached to the shell and the top shelf is supported from the bottom shelf because this gives excellent thermal isolation to the top shelf. The varied science instruments require unique structural solutions to window design based on using sapphire for visible instruments and zinc selenide for infrared instruments. Window heaters and a jettisonable outer window were selected on the basis of low cost and high reliability. The optical science instruments are shelf mounted rather than shell mounted to maintain a better thermal environment, better alignment under high g loads, and better accessibility.

Power Bus

An unregulated power supply was selected over the regulated bus because it is lower in cost, weight, and volume. Moreover, most equipment and science instruments have their own regulators for better EMI protection and special regulation. For example, the RF transmitter has its own regulator for special EMI protection.

Large Probe Baseline

A summary of the deceleration module and pressure vessel configuration/structure is contained in the following paragraphs. The selection rationale and major subsystems within the pressure vessel are also discussed.

The external configuration of the large probe is shown in Figure 3-20, and weights are shown in Table 3-8. The major elements of the large probe are shown in the exploded view of Figure 3-21. For convenience the probes are divided into a pressure module and a deceleration module. Overall responsibility for the probes is that of Hughes Aircraft Company. Hughes is also responsible for the design and development of the pressure vessel module. Hughes subcontractor, the General Electric Company, is responsible for the design and development of the deceleration module. This division of effort was based on reducing overall program cost and providing a relatively simple interface.

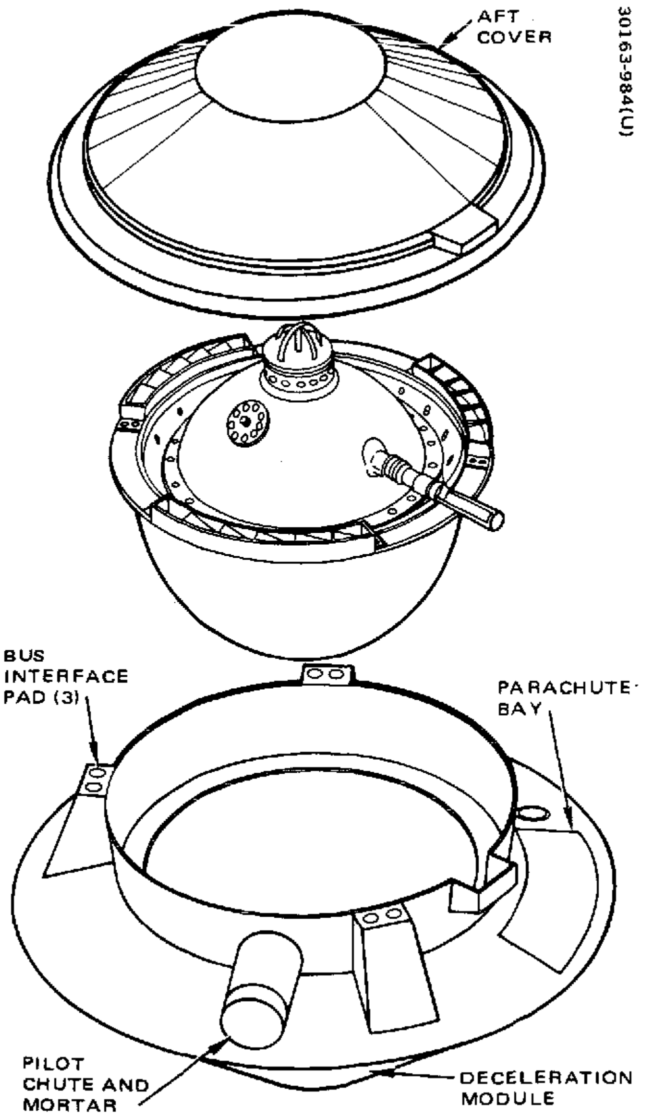


FIGURE 3-21. LARGE PROBE EXPLODED VIEW

TABLE 3-8. LARGE PROBE WEIGHT SUMMARY - ATLAS/CENTAUR MISSION

Deceleration Module		90.8 kg (200.2 lb)
Heat shield	35.5 (78.3)	
Structure	39.1 (86.3)	
Aft cover	7.5 (16.5)	
Parachute	4.7 (10.4)	
Harness	1.7 (3.8)	
Pressure vessel separation	0.7 (1.5)	
Instrumentation	0.2 (0.4)	
Ballast	1.4 (3.0)	
Pressure Vessel Module		152.5 kg (336.2 lb)
Structure	69.7 (153.6)	
Thermal control	27.2 (60.0)	
Radio	5.8 (12.7)	
Command/data handling	3.5 (7.8)	
Power	12.9 (28.5)	
Antenna	0.2 (0.5)	
Harness	1.0 (2.3)	
Instrumentation	0.5 (1.1)	
Science instruments	31.6 (69.7)	
Entry Weight		243.3 kg (536.4 lb)
Bus separation	1.8 (4.0)	
Weight on Bus		245.1 kg (540.4 lb)

The deceleration module contains the heat shield, aeroshell structure, parachute aft cover, and associated separation hardware. GE responsibility also includes the mechanical interface with the multiprobe bus, which is accomplished via three bus interface pads and associated separation spring assemblies. Electrical interface with the bus via an in-flight disconnect harness that supplies power and commands to the probe and returns telemetry data is also a GE responsibility. Separation of the pressure vessel and deceleration module/aft cover occurs during the descent phase upon command. The aft cover is removed by a pilot chute, and separation between the pressure vessel and deceleration module is accomplished via the main parachute attached to the pressure vessel. Both parachutes are stored in compartments on the deceleration module as shown and are a responsibility of GE.

The pressure vessel portion of the probe is the responsibility of Hughes. This division of effort makes Hughes responsible for all electronics — both spacecraft and probes — thereby maximizing commonality and minimizing overall cost.

The small probe division of responsibilities is similar to the large except the pressure vessel is not separated from the deceleration module which eliminates the requirement for a parachute.

The Hughes and GE team technical capabilities match those needed for the Pioneer Venus program, and the allocation of responsibilities between team members provides simple management interfaces with single point responsibility for all key functions.

Deceleration Module

The 45 deg cone angle aeroshell forebody is protected by a carbon-phenolic heat shield. This material has a very stable char and experiences much smaller mass loss than any of the alternative heat shield materials. It will also allow a relatively predictable probe drag coefficient. Both characteristics in turn reduce the uncertainties in the atmosphere reconstruction experiment. A relatively constant drag coefficient is required to reduce the uncertainties in the atmospheric reconstruction experiment. The heat shield has a 50 percent margin (0.94 cm) in thickness to avoid risk and reduce test requirements. The heat shield is bonded to the aeroshell structure using an elastomeric bond designed to permit cold soak to -73 deg, much lower than the worst near earth conditions.

The primary aeroshell structure is a ring-stiffened aluminum monocoque shell, which has skin thickness ranging from 0.16 cm (0.063 in.) at the stagnation point to 0.28 cm (0.12 in.) at the end of the skirt. The conical portion of the shell is machined as a single unit including rings, a technique that has proven cost-competitive with other fabrication methods. The shell and the rest of the structure are aluminum because the deceleration module does not have to survive below parachute deployment altitude, about 67 km. As with the heat shield, a 50 percent safety margin is used in structural design, virtually eliminating development testing.

An aluminum cover, coated with a heat resistant 3 cm silicone elastometer, protects the aft end from entry heating. An RF transparent fiberglass disk is used on the end so that the antenna can radiate back to earth before entry prior to aft cover separation.

The 4.6 m (15 ft) disc-gap-band, nylon main parachute is subsonically deployed by the 0.84 m (2.75 ft) ribbon pilot chute which in turn is deployed by firing a mortar. The pilot chute also removes the aft cover in the process. After a delay to allow the main chute to stabilize, the aeroshell is released and separates from the pressure vessel module, which now stays with the main chute until approximately 40 km altitude. The science instruments mounted on and within the pressure vessel measure the atmosphere directly from about 67 km downward.

Pressure Vessel

The pressure vessel external configuration is ring-stabilized stepped sphere whose largest diameter is 96.5 cm (38 in.). It houses all the scientific instruments and the housekeeping subsystems in a spherical volume 68.1 cm (25.8 in.) in diameter shown in the exploded view of Figure 3-22. An aerodynamic fairing radome encompasses the wind/altitude radar antenna. The pressure vessel is aerodynamically stabilized by a 6 cm ring located aft of the equator. This ring contains spin vanes for module rotation and also acts as the structural adapter to the aeroshell. This pressure vessel aerodynamic configuration geometry has good aerodynamic stability and adequate roll control for science as verified in Hughes IR&D wind tunnel tests conducted at NASA/LRC. The large probe pressure vessel equipment packaging is illustrated in Figure 3-23.

The pressure vessel shell uses a high temperature, high strength maraging steel to provide low cost structure with low weight and to avoid reactive problems with the Venusian atmosphere. The equipment is mounted on two horizontal shelves of solid aluminum to provide for the necessary heat sink capacity. The top shelf is mounted from the bottom shelf to reduce heat flow to the top shelf and to permit convenient disassembly of the pressure vessel modules which helps maintain low test costs. The pressure vessel is internally insulated with a 1.3 cm (0.5 in.) blanket of FA fiberglass mounted on a rigid support structure which holds the blanket against the shell. This design maintains temperature of all units below 52°C (125°F) up to impact except for the rf power amplifiers which are allowed to reach 60°C (140°F).

Electrical Power

A block diagram of the large probe unregulated power subsystem which furnishes all power to equipment and experiments after separation

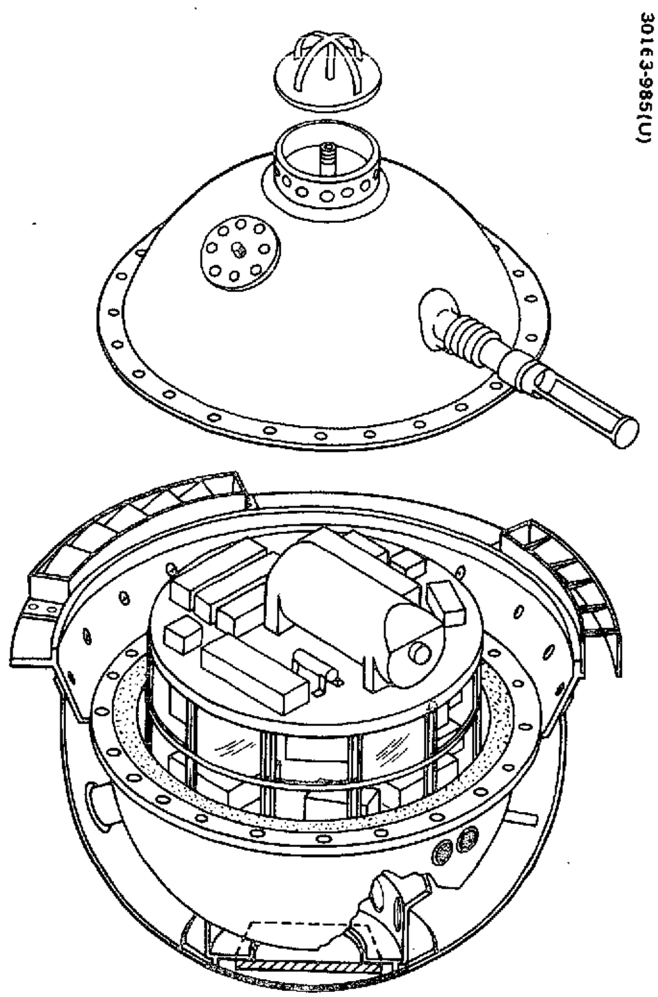


FIGURE 3-22. PRESSURE VESSEL EXPLODED VIEW

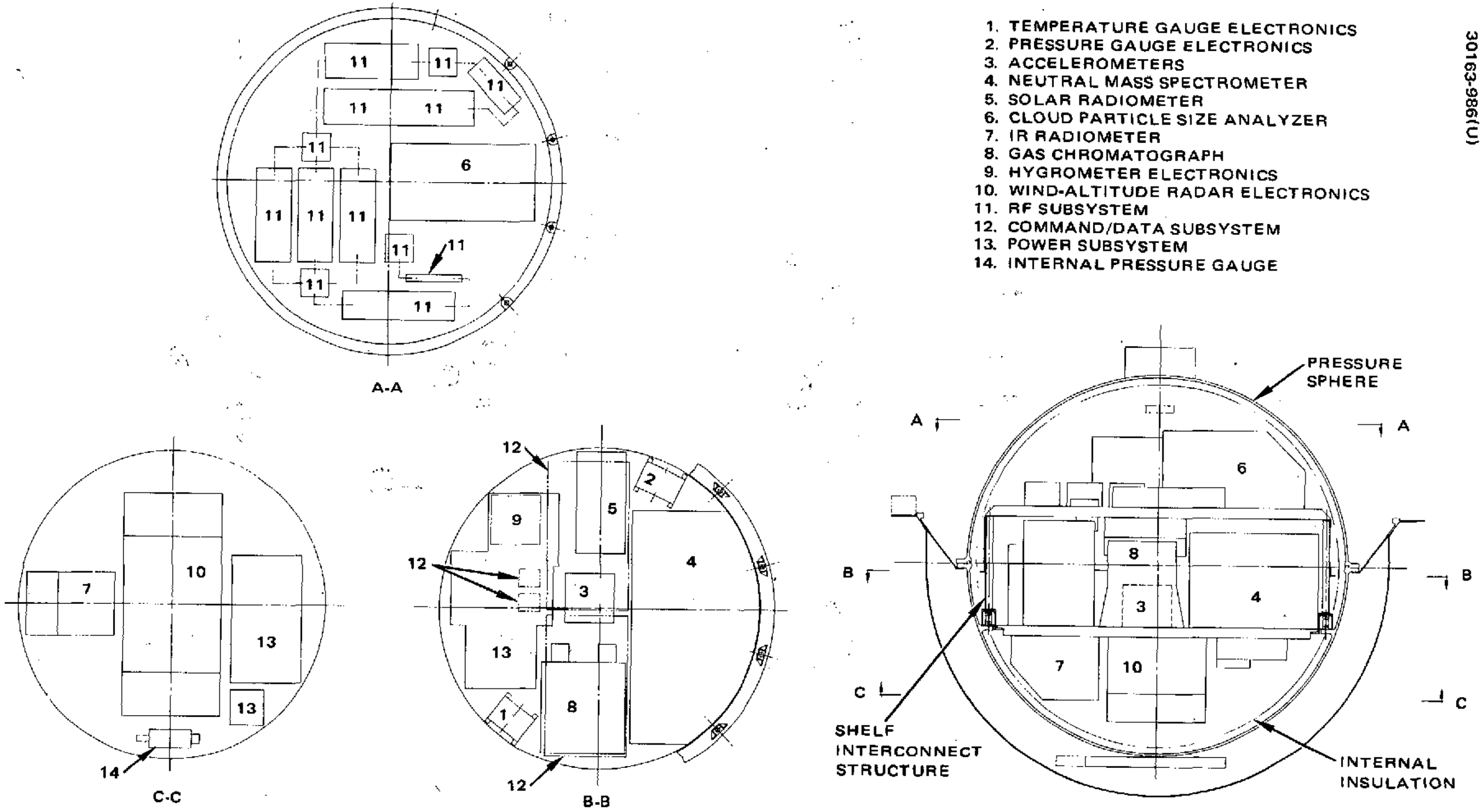


FIGURE 3-23. ATLAS/CENTAUR LARGE PROBE PRESSURE VESSEL MODULE CONFIGURATION

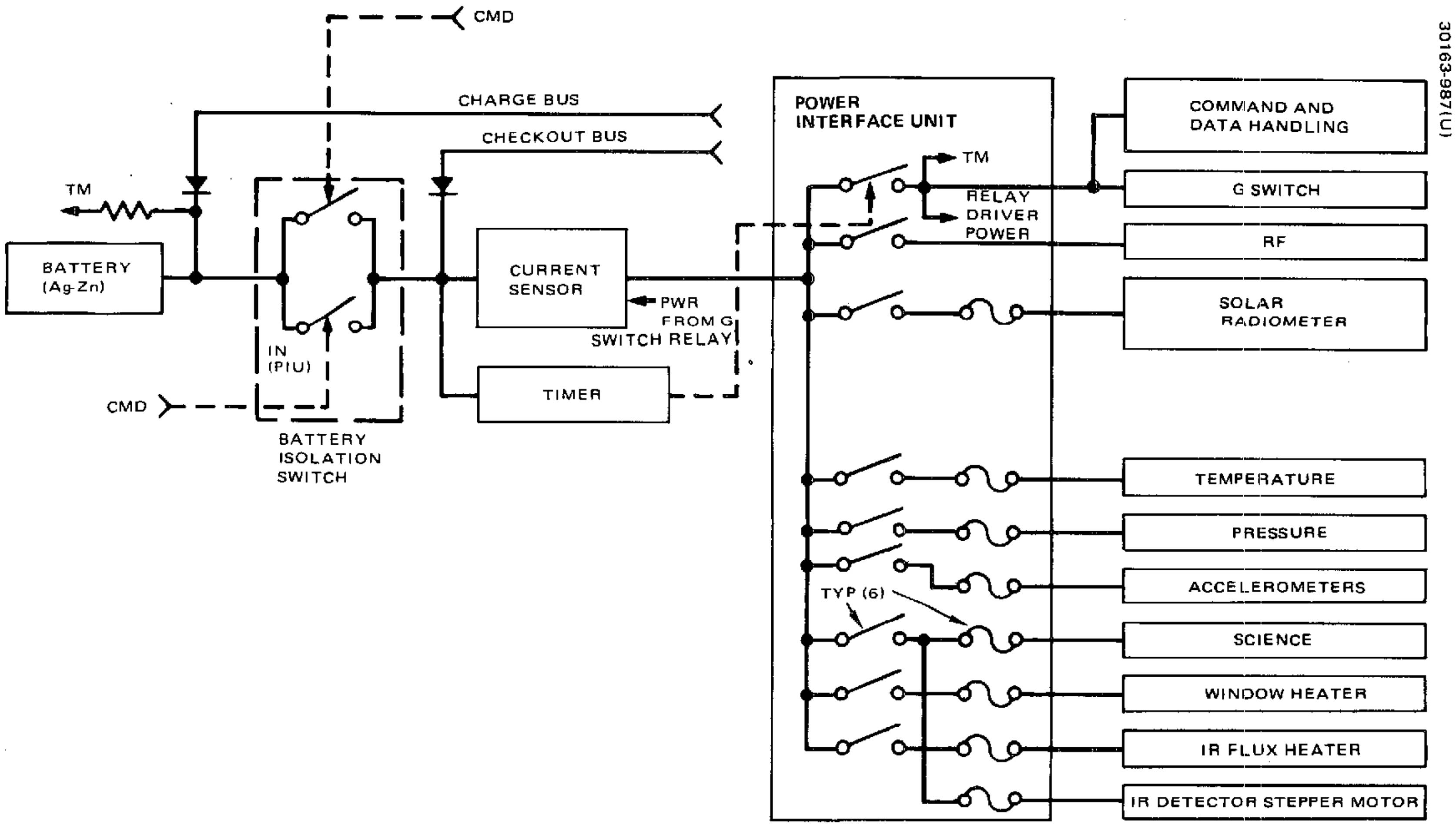


FIGURE 3-24. LARGE PROBE POWER SUBSYSTEM

from the probe bus is shown in Figure 3-24. The simple, low cost power subsystem with bus voltage (24 to 33 V) supplied by an 18-cell silver zinc battery is designed to withstand the high g environment during entry. Power switching is accomplished with the power interface unit, and a current sensor has been included for status determination.

Communication

The large probe communication subsystem (Figure 3-25) contains a transponder for doppler tracking and telemeters all data to earth after it separates from the probe bus. The transmitter contains three 9 W output solid state modules operated in parallel with 25 W of radiated power. These modules are also used in the probe bus and orbiter and are under development on a current Hughes defense satellite program. A curved turnstile antenna shown in Figure 3-26, nominally provides 0 dB at 90 deg off axis and maximum gain at angles of 60 to 65 deg off axis. The wide angle (hemispherical) coverage allows a common antenna to be used in both large and small probes and also allows disturbances such as wind gusts during the descent phase of the probes. The nominal ERP of 44 dBm allows 160 bps at high altitudes and 80 bps at low altitudes. During the study loop vee and equiangular antennas were built and tested. These antennas provided higher gain at the communication angles expected for the 1978 mission. However, they did not provide the flexibility of the baseline.

Command/Data Handling

The command system must be capable of executing commands which are stored prior to launch. The command and data handling subsystem block diagram is shown in Figure 3-27. No real time earth command is required and is therefore not implemented. A pyrotechnic control unit is shown which provides the firing pulses to all provide pyrotechnics. The large and small probe designs are very similar. The small probes require fewer multiplexer elements for gathering data, command decoding and command outputs, memory elements, and drivers for firing of pyrotechnics.

The command portion of the subsystem:

1) Accepts pulse and magnitude commands from the probe bus spacecraft for preseparation tests and for initialization of the probe's cruise timer.

2) Provides an accurate timer to initiate probe preentry sequences after a 20 to 23 day cruise period.

3) Provides pulse command outputs to probe subsystems and scientific instruments based upon predetermined (stored) sequences and real time events.

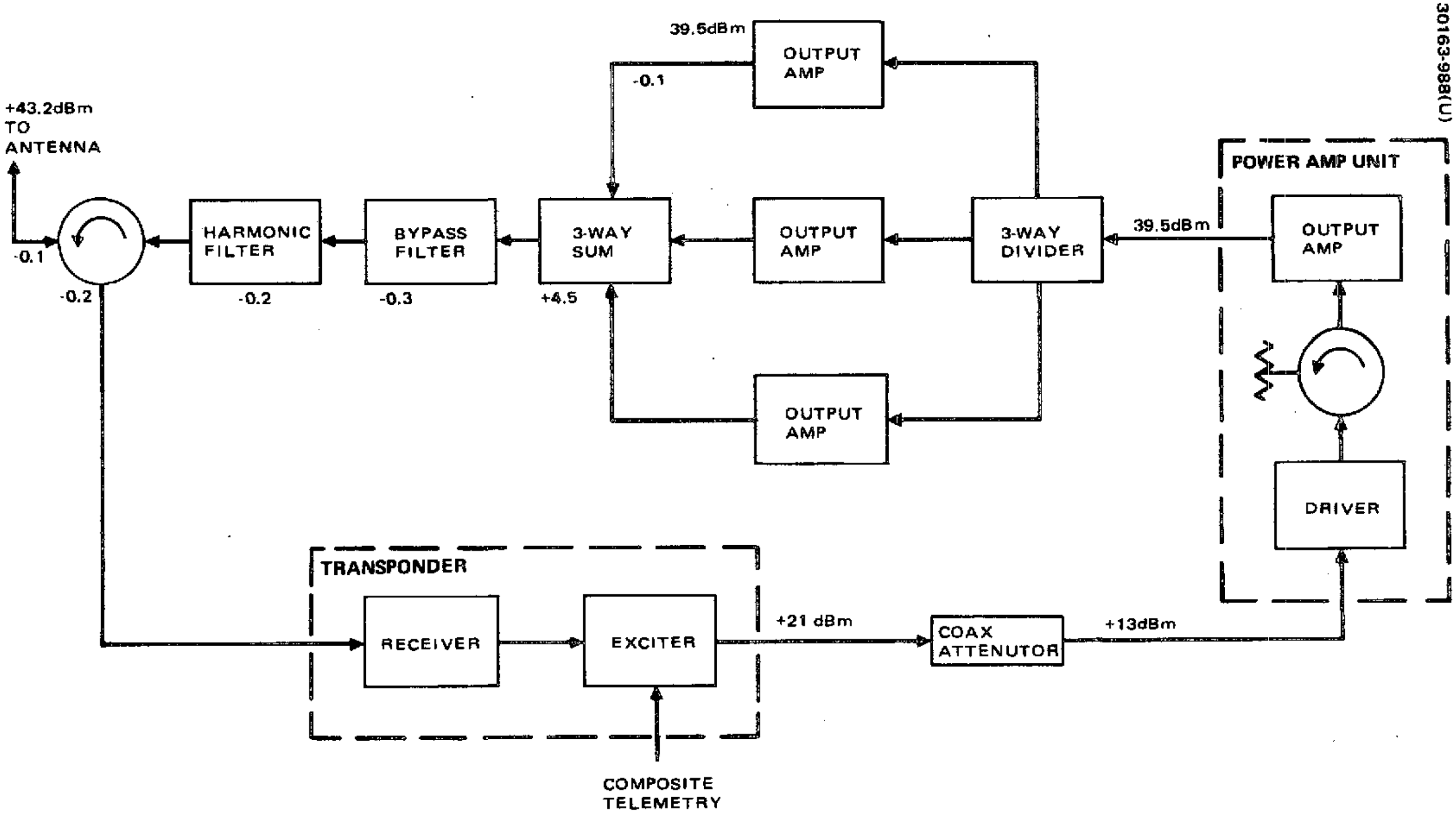


FIGURE 3-25. LARGE PROBE RF SUBSYSTEM

4) Detects two pressure and two acceleration events during descent to initiate certain command sequences.

5) Provides high current switching capability for commanded firing of pyrotechnic devices.

The data handling portion of the subsystem:

1) Accepts analog, bilevel discrete, and serial digital data from the probe subsystems and science instruments.

2) Digitizes analog data into 10 bit serial words.

3) Formats all data according to one of three stored formats.

4) Convolutionally encodes and biphase modulates the PCM data.

5) Provides for data storage during entry blackout.

Table 3-9 shows the important functional characteristics of the subsystem. The hardware design is based on existing technology using a combination of the Hughes MICAM (microelectronic assembly method) and printed circuit board packaging techniques which have been tested and shown to withstand the high g entry environment.

TABLE 3-9. COMMAND/DATA HANDLING FUNCTIONAL CHARACTERISTICS

Parameter	Characteristic	
	Large Probe	Small Probe
Command		
Cruise timer stability/resolution	1×10^{-5} (20 sec in 23 days)/2 sec	
Descent timer stability/resolution	5×10^{-5}/0.1 sec	
Command type	Stored, pulse	
Command initiation	Time or event	
Number of event inputs	8	
Number of command outputs	48	32
Data handling		
Modulation type	PCM/PSK	
Data bit rates	160 or 80 bps 20 bps (store only)	60 or 30 or 10 bps
Number of data formats	3	
Data storage capacity	2048 bits	512 bits
Word size	10 bits	

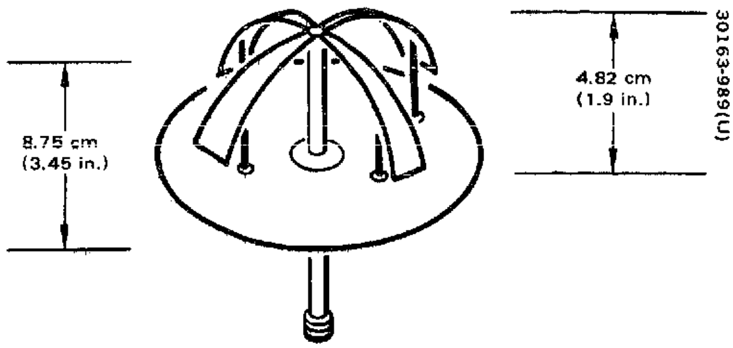
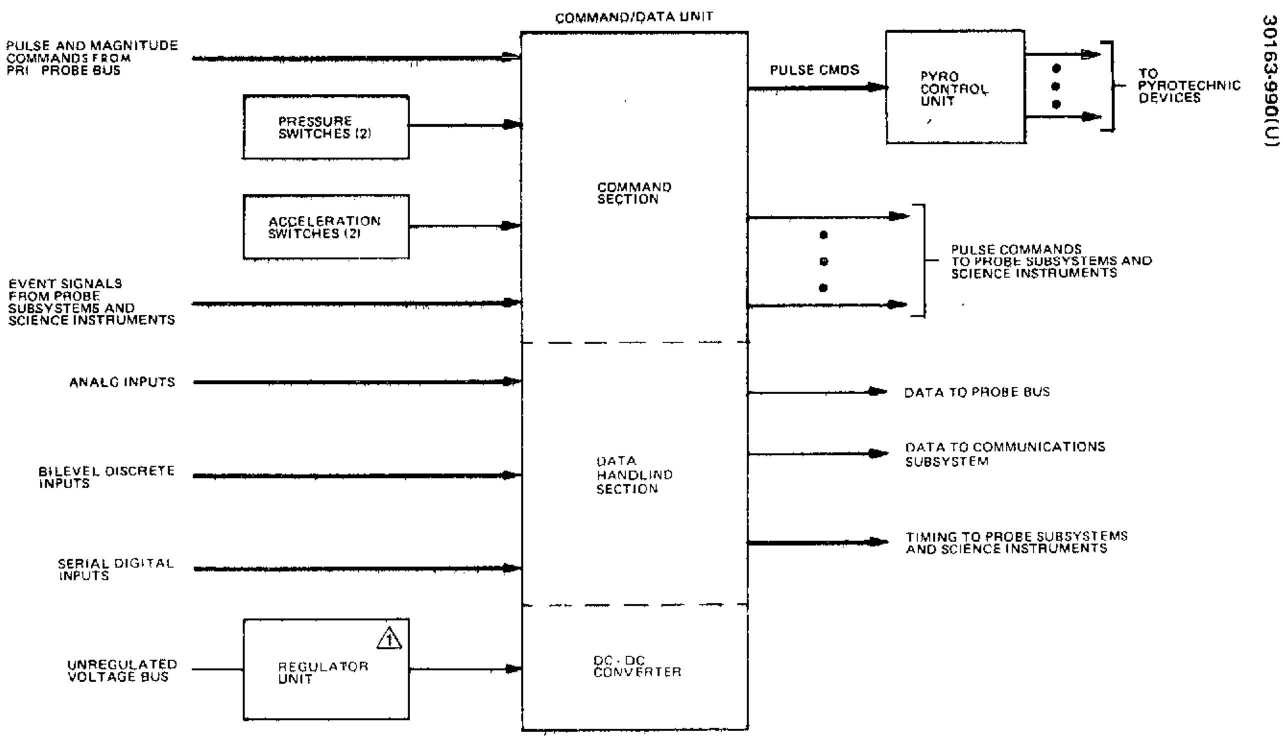


FIGURE 3-26. CURVED TURNSTILE ANTENNA MODEL



1 REGULATOR UNIT IS SEPARATE ON SMALL PROBE; THE REGULATOR IS CONTAINED WITHIN THE COMMAND/DATA UNIT ON LARGE PROBE.

FIGURE 3-27. LARGE AND SMALL PROBES COMMAND AND DATA HANDLING SUBSYSTEM

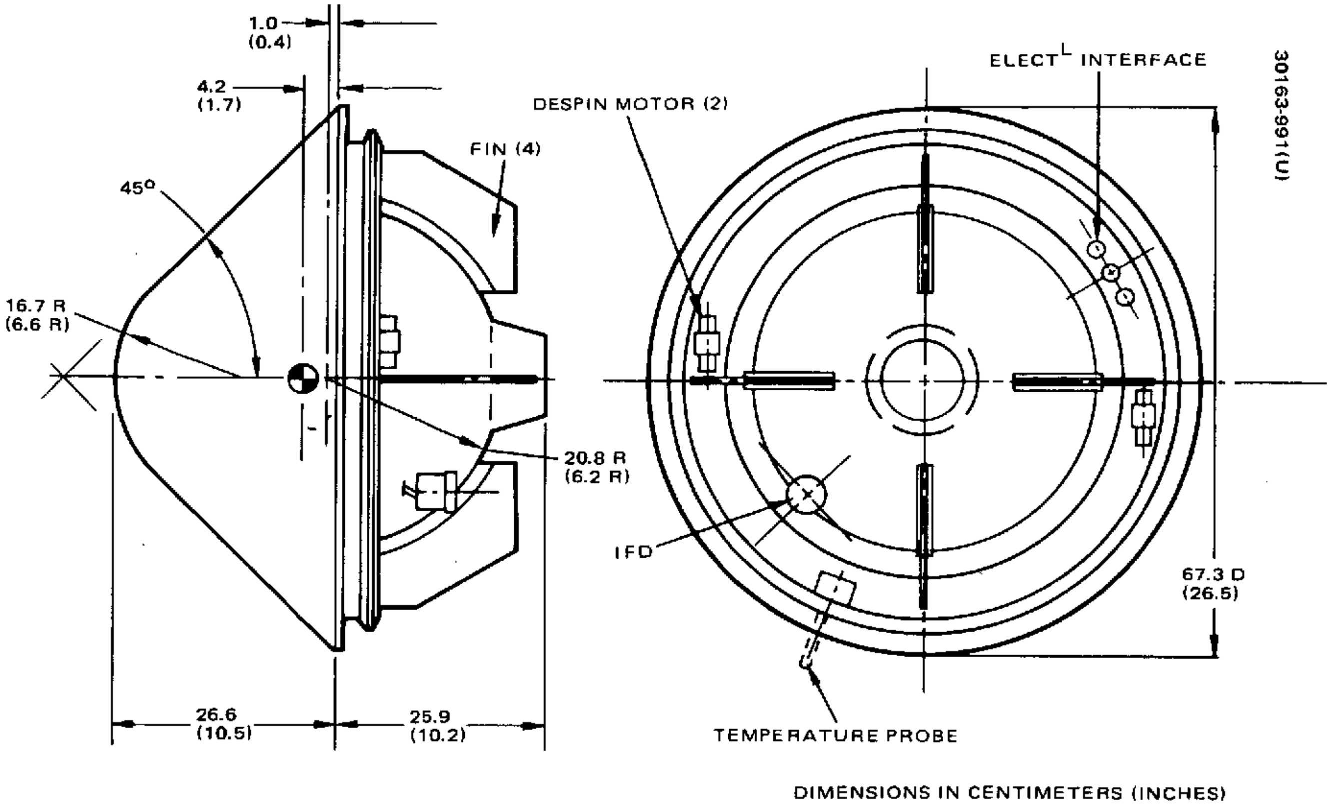


FIGURE 3-28. ATLAS/CENTAUR SMALL PROBE CONFIGURATION

Small Probes Baseline Design

The small probe external configuration is illustrated in Figure 3-28. It weighs 63.6 kg (140.2 lb) mounted on the probe bus. It has a 45 deg half-angle conical forebody with nose radius/base radius = 0.5, identical to those of the large probe. The aft configuration shows four fins which are not present in the large probe. These roll damping fins are incorporated because of a series of wind tunnel tests which showed roll induced dynamic instability at low speeds for the non-separated aeroshell and pressure vessel configuration. The fins will help to provide roll damping in the event a roll is caused by possible ablation asymmetries. The aft heat shield is applied directly to the pressure vessel.

The three small probes are each mounted on the probe bus using a clamp which attaches to the ring visible just aft of the base. The outward half of the clamp is pivoted at one end and held in place by an explosive nut/bolt at the other end. As the nut is broken, the clamp rotates 160 deg and locks in place. Centrifugal force caused by the spinning bus moves the probes away from the bus and each other. Electrical separation is provided by the same model in-flight disconnect used on the large probe.

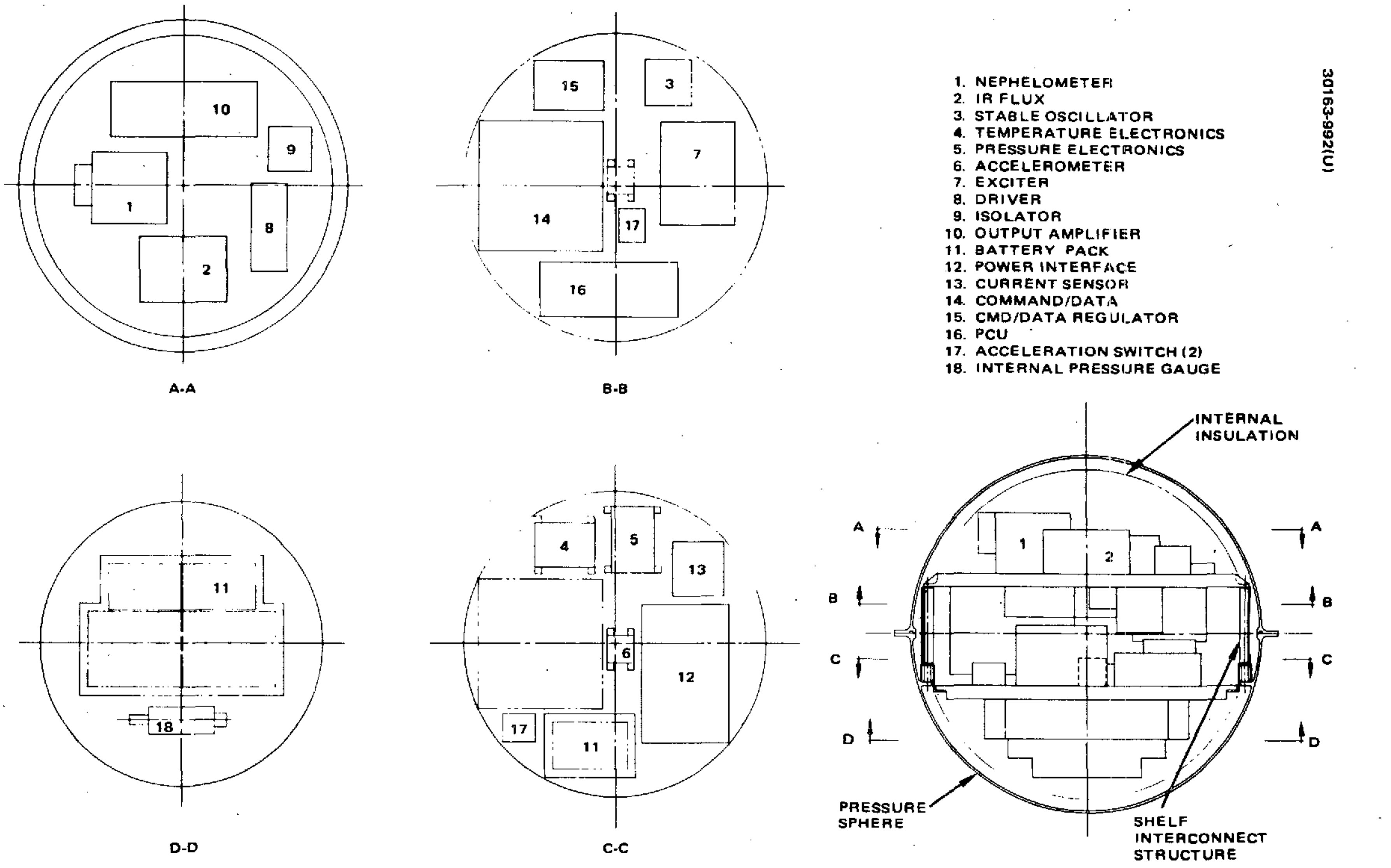


FIGURE 3-29. ATLAS/CENTAUR SMALL PROBE PRESSURE VESSEL CONFIGURATION

The small probes are initially spinning at a high rate because of the targeting procedure. Since the targeting also precludes orienting the probes for zero angle of attack at entry, it is necessary to limit spin rate so that angle of attack convergence will not be impeded. This is done by firing two small despin solid propellant rocket motors located at the base of the small probe vehicle.

The small probe heat shield material is the same as for the large probe, carbon phenolic, for identical reasons.

The aeroshell structure is of stainless steel and monocoque in construction. Steel was chosen rather than titanium because of the uncertainty about the chemical compatibility of titanium with the lower atmosphere.

The small probe pressure vessel module (Figure 3-29) is a sphere of outer diameter 41.4 cm (16.3 in.) It contains all the instrument electronics and the housekeeping subsystems. The basic shell structure is a monocoque constructed of maraging steel and is internally insulated. The pressure vessel contains two equipment mounting shelves, with the upper shelf mounted from the lower shelf rather than to the shell, as in the large probe.

The internal insulation/heat sink thermal control scheme is the same as in the large probe: a fiberglass blanket held in place against the shell by an internal retainer and solid aluminum shelves which are also heat sinks. The stacked washer concept is also used for shelf thermal isolation.

Because it is feasible with the internal insulation design and also greatly simplifies the pressure vessel/aeroshell interface, several items of equipment are mounted on the pressure vessel rather than the aeroshell. Specifically, the four roll-damping fins, the IFD, and the two swingout arms which house temperature sensor and radiometer mirror are all pressure vessel mounted.

The small probe carries 2.6 kg (5.7 lb) of instruments in a 38.4 kg (84.7 lb) pressure vessel module and has a ballistic coefficient of 169 kg/m^2 (34.6 lb/ft^2). A weight breakdown by subsystem is given in Table 3-10.

The small probe power subsystem shown in Figure 3-30 is identical to that of the large probe except for power handling capability and the number of switches required for loads.

The small probe rf subsystem, shown in Figure 3-31, uses the same output modules for the transmitters and the same antennas as the large probe.

Finally, as was mentioned previously, the command/data handling subsystem of the small probe is identical to that of the large probe except that fewer elements are needed by data handling and command outputs, as well as less memory and devices for pyrotechnics.

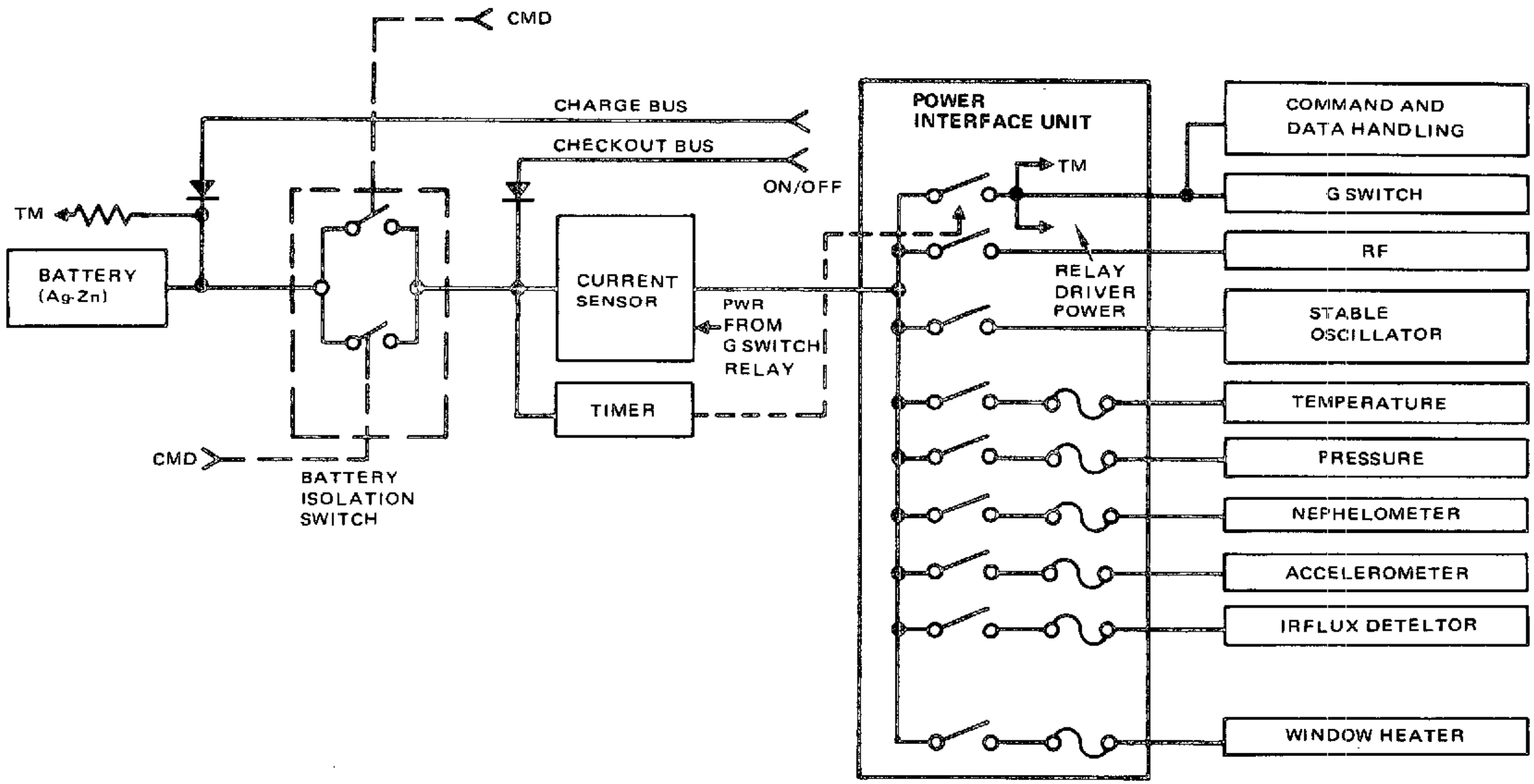


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FIGURE 3-30. SMALL PROBE POWER SUBSYSTEM

TABLE 3-10. SMALL PROBE WEIGHT SUMMARY – ATLAS/CENTAUR MISSION

Deceleration Module		23.8 kg (52.5 lb)
Heat shield	10.0 (22.1)	
Structure	13.2 (29.2)	
Harness	0.2 (0.4)	
Ballast	0.4 (0.8)	
Pressure Vessel Module		38.4 kg (84.7 lb)
Structure	15.9 (35.0)	
Thermal control	9.8 (21.5)	
Radio	1.6 (3.6)	
Command/data handling	2.7 (5.9)	
Power	5.2 (11.4)	
Antenna	0.2 (0.5)	
Harness	0.3 (0.6)	
Instrumentation	0.2 (0.5)	
Science instruments	2.6 (5.7)	
Entry Weight		62.2 kg (137.2 lb)
Separation/despin	1.1 (3.0)	
Weight on Bus		63.6 kg (140.2 lb)

Probe Experiment Accommodation

The key instrument integration problems for the large probe are the wind altitude radar planar array antenna to be located at the pressure vessel nose; the mass spectrometer inlet system, with a large number of squibs required for opening and shutting valves; and the cloud particle analyzer mirror mount which requires an alignment to 1 mr throughout the descent. Wind altitude radar accommodation is solved by locating the planar array antenna outside the forward end of the large probe pressure vessel covered with a ceramic foam radome. The radome provides the required clean aerodynamic surface to maintain stability during descent. The mass spectrometer inlet system is 45 degrees from the probe axis and protrudes beyond the aerodynamic fairing to ensure that the boundary layer does not influence the instrument. An allowance of 12 squib actuated events has been made to handle the valve operations. The cloud particle analyzer uses an integral mirror/instrument mounting approach to meet the alignment requirements. A jettisonable transparent heater cover is provided to minimize possible window contaminations.

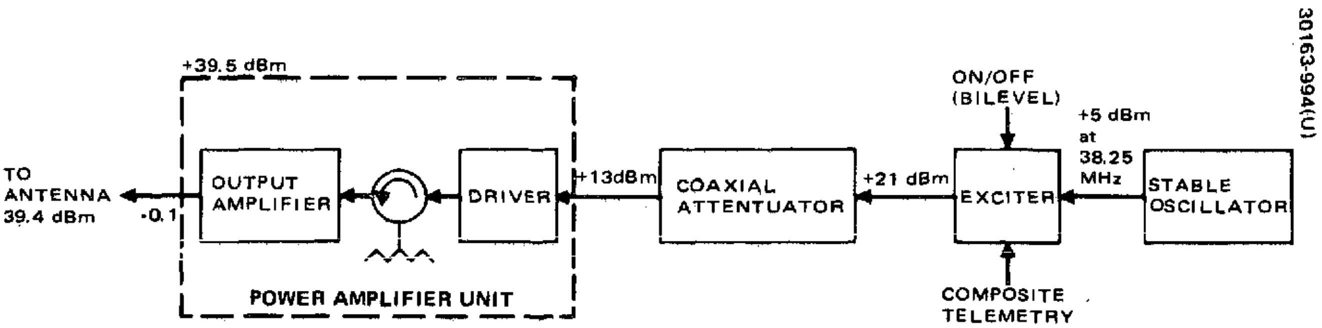

FIGURE 3-31. SMALL PROBE RF SUBSYSTEM

For the small probe, a deployable pressure sensor is required for temperature and IR flux measurements and a pressure port is required at the stagnation point because the small probe heat shield remains with the vehicle throughout the descent. The temperature sensor is mounted on a deployable boom stowed behind the aft deceleration module step for protection during entry. After entry, a squib activated device permits the sensor arms to swing out clear of the heat shield step. The pressure sensor inlet is protected during entry by a jettisonable teflon plug which is removed by a squib activated spring driven piston to provide access to the environment during descent.

3.6 SYSTEM SUMMARY

The preceding sections briefly described the critical design issues addressed during the study. A brief description of the baseline design is also presented. The detailed discussions are contained in the remaining volumes. Table 3-11 provides a tabular summary of the Hughes Pioneer Venus system features, selection rationale, and hardware derivation.

3.7 PRELIMINARY DEVELOPMENT PLAN

A Preliminary Development Plan has been generated as part of the Systems Design Study. Submission of this plan has been delayed till 15 August in order to properly reflect any changes due to the NASA ARC Pioneer Venus RFP. This section briefly summarizes the characteristics of the Preliminary Development Plan as of the date of the submission of the final report.

The principal objective has been to provide a plan that meets the low cost objectives of the Pioneer Venus project without compromising the performance requirements nor the integrity of the spacecraft. Project management and supervisory personnel have been reduced by collocation of the project office and responsible engineering activities and by centralizing probe, probe bus, and orbiter responsibilities whenever possible. The schedule is structured such that all engineering development is virtually complete prior

TABLE 3-11. HUGHES PIONEER VENUS SYSTEM SUMMARY

Feature	Selection	Rationale	Hardware Derivation
Multiprobe Mission			
Transit Trajectory	Type I	Greater launch vehicle payload	
Probe Separation	E-23 deg, E-20 deg	Lowest weight solution	
Bus Entry Angle	γ_E = -12.5 deg	Approach dispersions, maximize time between 150 ≥ alt ≥ 115 km	
Probe Bus			
Weight	844.4 kg	Launch vehicle payload capability	
Length	287 cm	Launch vehicle fairing	
Diameter	254 cm	Launch vehicle fairing	
Payload	• 1 large probe • 3 small probes • 5 instruments (14 kg)	Mission requirements	
Radio			
Power amplifier	2-9W solid state modules	Entry downlink requirements, Commonality with probes	Military satellite program
Antennas	3 dBi bicone	Cruise data return	Military satellite program
	2 omnidirectional antennas	4π command and telemetry coverage	Surveyor and HS 350*
	18 dBi medium gain horn	Entry attitude, downlink requirements	Intelsat IV
Tracking	2 way Doppler transponder	DSN compatibility, existing hardware	Viking
Power			
Solar panel	4.76 m^2	139 watts for cruise	
	2 ohm-cm n/p cells	Existing technology	Telesat design
Battery	2-5 A-hr Ni-Cd batteries	Orbiter commonality	Telesat design
Power distribution	24 to 28 Vdc	Low cost, elimination of regulator	OSO
Propulsion			
Type	Hydrazine blowdown	Reliability, existing technology	
Jet configuration	5-22.2N jets 2 axial 4 radial	Spin axis precession, spin rate control, ΔV maneuvers, back-up modes	Intelsat IV
Tankage	2-41.2 cm diameter tanks	Mission requirements	Marisat
Attitude Control			
Spin axis orientation	Normal to ecliptic	Power, thermal simplicity, orbiter commonality	
Attitude sensors	Solid state star sensor	Provides required attitude determination accuracy and spin reference, proven approach	New
	Silicon detector sun sensor	Existing hardware	Telesat
Thermal Control			
Approach	Passive blankets, louvers, heaters	Standard approach with consideration of mission geometry and variation in solar flux	Military satellite

*Hughes satellite program.

Table 3-11 (Continued)

Feature	Selection	Rationale	Hardware Derivation
Data Handling			
Multiplexing	Decentralized	Design flexibility	OSO
Encoding	Convolutional	DSN compatibility	OSO and OSO derived
Formatting, modulation	PCM/PSK/PM	Design flexibility	
Data rates	8-2048 bps	Instrument requirements	
Command			
Command distribution	Decentralized modular	Design flexibility	OSO
Command storage	64 command semiconductor memory	Orbiter commonality	New
Command processor	36 bit format, 4 bps	DSN compatibility, simplicity	OSO derived
Demodulator	PCM/PSK/PM	DSN compatibility, existing hardware	Viking
Large Probe Mission			
Entry angle	-30 deg	Desired impact site	
Initial operating altitude	67 km	Data return above cloud tops	
Parachute separation altitude	40 km	Maximize science return, 50 ≥ alt ≥ 40 km	
Small Probe Mission			
Entry angles	-20 deg ≥ γ_E ≥ -90 deg	Targeting flexibility	
Initial operating altitude	67 km	Data return above cloud tops	
Large Probe			
Height	90.3 cm	Provide W/C_DA for parachute deployment, pressure vessel/mortar accommodation	
Diameter	139.7 cm		
Weight	245.1 kg		
Payload	10 instruments 31.3 kg	Science requirements	
Deceleration Module			
Aeroshell configuration	45 deg blunt cone	Similarity to small probe	PAET
Heat shield material	Carbon phenolic	Entry characteristics predictability	Mark 12; RV to 1B, 3B; LAR
Structure	Aluminum monocoque	Weight/cost considerations	
Parachute diameter	4.6 m	Safe aeroshell/pressure vessel separation	
Parachute configuration	Disc gap band, nylon	Existing parachute vendor technology	Many GE programs
W/C_DA	149 kg/m^2	Subsonic parachute deployment	
Pressure Vessel			
Diameter	68.1 cm I.D.	Required volume	
Aerodynamic configuration	Ring-stabilized sphere	Aerodynamic tests in spin tunnel	
Material	Maraging steel	Suitability to Venus environment	
Thermal configuration	Internal insulation	Ease of fabrication	

Table 3-11 (Continued)

Feature	Selection	Rationale	Hardware Derivation
Radio			
Power amplifier	3-9W solid state modules	Commonality of module size, surface data rate (80 bps)	Military satellite
Antenna	0 dBi hemispheric pattern	Targeting flexibility, insensitivity to winds	
Tracking	2-way Doppler transponder	DSN interface, mission requirements	Viking
Command			
Command storage	96 command register	Mission sequence	New
Command initiation	Clock, inertia switches, pressure switches	Mission sequence, reliability	New
Data Handling			
Data processing	Centralized 160/80 bps data rates	Weight, volumetric efficiency	New
	Convolutional encoding	DSN compatibility	
	PCM/PSK/PM	Instrument requirements	
Power distribution	24 to 32 VDC	Simplicity, weight	
Battery	8 amp-hour Ag - Zm	Weight	
Small Probe			
Height	52.5 cm	Minimize weight, stability margin	
Diameter	67.3 cm		
Weight	63.69 kg		
Payload	2.6 kg		
Deceleration Module			
W/C_DA	169 kg/m^2	Subsonic at initial operating altitude	
Aeroshell configuration	45 deg blunt cone	Transonic stability	
Heat shield material	Carbon phenolic	Entry characteristics predictability	Mark 12; RV to 1B, 3B; LAR
Structure	Stainless steel	Weight, cost	
Pressure Vessel			
Diameter	40.6 cm I. D.	Required volume	
Material	Maraging steel	Suitability to Venus environment	
Thermal configuration	Internal insulation	Ease of fabrication	
Radio			
Power amplifier	1-9W solid state module	Commonality of module size, surface data rate (10 bps)	Military satellite
Antenna	0 dBi hemisphere pattern	Targeting flexibility, insensitivity to winds	
Tracking	1-way Doppler stable oscillator	DSN interfaces, mission requirements	
Command			
Command storage	64 command register	Mission sequence	New
Command initiation	Clock, inertia switches, pressure switches	Mission sequence, reliability	

Table 3-11 (Continued)

Feature	Selection	Rationale	Hardware Derivation
Data Handling			
Data processing	Centralized 60/30/10 bps data rates	Weight, volumetric efficiency	New
	Convolutional encoding	DSN compatibility, instrument requirements	
	PCM/PSK/PM		
Power			
Distribution	24 to 32 VDC	Simplicity, weight	
Battery	24 amp-hour Ag - Zm	Weight	
Orbiter Mission			
Transit trajectory	Type II	Allow more favorable periapsis latitude	
Periapsis latitude	South (56 deg S)	Radar altimeter coverage	
Orbit period	24 h	Compromise between spacecraft mass and science operations, coverage with same ground station	
Inclination	90 deg	High latitude planetary coverage	
Periapsis altitude	150 km	Atmospheric sampling versus aerodynamic drag	
Orbiter			
Weight	492 kg	Launch vehicle capability	
Length	356 cm	Launch vehicle fairing	
Diameter	254 cm		
Payload (experiments)	46 kg		
Radio			
Power amplifier	2-9W solid state modules		Military satellite
Antenna	23.5 dBi mechanically despun antenna	Hughes technology	Intelsat IV
Tracking	2 omnis	All attitude telemetry and command coverage	Surveyor and HS-350
	2-way Doppler	Existing hardware	Viking
Power			
Solar panel	6 m^2	157 watts cruise power at L +50 days	Telesat
	2 ohm-cm n/p cells	Existing technology	
Battery	269 W-hr Ni-Cd	3.1 hr eclipse, number of charge/discharge cycles	Telesat
Power distribution	24 to 33 Vdc	Low cost by elimination of regulator	OSO
Propulsion			
Type	Hydrazine blowdown	Simplicity, reliability, existing hardware	
Jets	7-22.2N jets 3-axial 4 radial	Spin axis precession, spin rate central, ΔV maneuvers, back-up mode	Intelsat IV
Tankage	2-2.67 kg tanks 2 41 cm tanks	Mission requirements	Marisat
Orbit insertion motor	Solid motor, stretched TEM-521	Cost, developed, low risk Venus orbit selected	SKYNET I, NATO I, IMP II, J

Table 3-11 (Continued)

Feature	Selection	Rationale	Hardware Derivation
Attitude Control			
Spin axis orientation	Normal to ecliptic	Power, thermal simplicity, instrument coverage from polar orbit	
Attitude sensors	Solid state star sensor	Provides required attitude determination accuracy and spin reference, proven approach	New
	Silicon detector sun sensors	Provides required attitude determination accuracy and spin reference, proven approach	Telesat
Despin bearing	BAPTA	Reliability, existing technology	Telesat
Thermal Control			
Approach	Passive blankets, louvers,	Standard approach with consideration of mission geometry, eclipses and variation of solar flux	Military satellite
Data Handling			
Data storage	10^6 bits magnetic core	Storage of periapsis data, occultation	New
Multiplexing	Decentralized modular	Design flexibility, standard Hughes approach	OSO
Encoding, formatting, modulation	Convolutional PCM/PSK/PM	DSN compatibility, design flexibility	OSO and OSO derived
Data rates	8-2048 bps	Instrument requirements	
Command			
Command distribution	Decentralized modular	Design flexibility. Standard Hughes approach	OSO
Command storage	64 commands, semi-conductor memory	Periapsis command requirements – occultation	New
Command processor	36 bit format 8 bps	DSN compatibility, mission flexibility requirement	OSO derived
Demodulator	PCM/PSK/PM	Existing hardware	Viking

to start of manufacturing, which will permit an orderly and efficient buildup of flight hardware. Emphasis has been placed on fixing interfaces early so that hardware might be developed in an orderly manner. Elaborate management tools such as PERT and Performance Analysis Reporting have been deleted in favor of less costly and schedule control procedures. Engineering documentation has been reduced whenever possible.

Major cost savings have been achieved through a reduction in testing. Particularly important is the incorporation of protoflight testing of probes, and probe bus, and orbiter spacecraft. This is by far the largest single cost saving item since a complete set of hardware and all associated testing is eliminated by this protoflight concept.

In addition, selective testing of flight hardware at the unit level has resulted in sizable savings. Finally, development testing has been reduced through the use of existing designs and commonality of designs between probes and spacecraft. This close scrutiny of the test program has been quite productive in reducing costs, yet none of these reductions has been made without evidence from past programs at Hughes that the deletion of a test will not compromise the integrity of the spacecraft.

Numerous measures have been taken to reduce the quantity of hard-wave, and hence program cost. Particularly noteworthy is the use of the qualification units for spares, which eliminates an additional set of flight hardware fabricated specifically for this purpose. Other examples are a common set of test equipment for engineering development testing and flight hardware testing and use of flight structures for certain system level development testing.

The master phasing schedule is shown in Figure 3-32. Key characteristics of this schedule are:

1) A Phase I activity that focuses on determining detailed interfaces, conducting key technology tests, and culminating with a conceptual design review which updates the baseline configuration proposed on 15 August.

2) An engineering phase at the outset of Phase II which completes all development tests, engineering models, breadboards, spacecraft structural and thermal tests, and detailed designs and drawings and culminates with a final design review to be completed at the end of 17 months.

3) Early electrical and mechanical integration tests of design verification experiments.

4) A flight equipment fabrication, assembly, and test phase which is started after completion of design reviews in the 18th month and has delivery of first flight units 24 months after Phase II start.

5) Assembly, integration, and test of the probes starting in month 25, the probe bus in month 29, and the orbiter in month 28. This early start on spacecraft testing allows ample time to uncover and fix any problems that may materialize.

This plan attempts to distribute the funding requirements relatively evenly over the duration of the project. An alternate plan would have the flight hardware fabrication start earlier and allow even more time for spacecraft testing. This plan would have a large percentage of funding required in fiscal year 1976 and was considered probably undesirable from NASA's viewpoint.

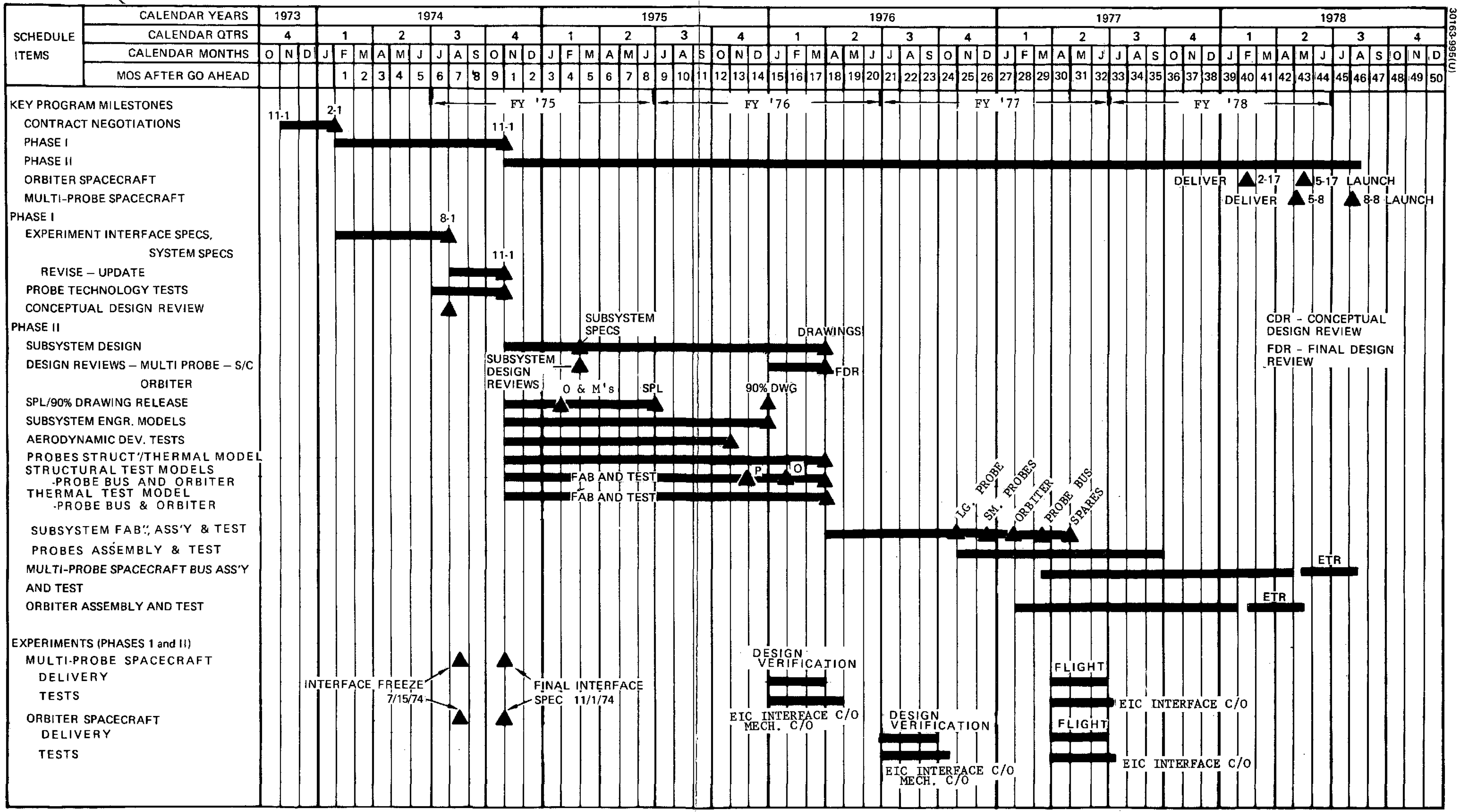

FIGURE 3-32. PIONEER VENUS PRELIMINARY MASTER PHASING SCHEDULE

3.8 INTERNATIONAL COOPERATION

The Systems Design Study work statement requested that "an analysis will be conducted to establish the most effective method of interfacing the proposed European Space Research Organization (ESRO) participation with NASA for the orbiter mission." At the midterm the results of this study were offered. The criteria utilized for the recommended alternate was: 1) reduced U.S. costs, 2) a manageable interface, and 3) reasonable European content. The conclusion of the study was that it was feasible to construct such a program and that it could save a substantial amount of money for the NASA.

Of the numerous options considered, the one that appeared to have the most merit was as follows. ESRO would have overall responsibility for the orbiter spacecraft. This would include responsibility for any thermal structural and prototype testing. In addition, ESRO would supply orbiter unique subsystems. These would include at least the despun antenna, orbit insertion motor, data storage, and orbiter unique power structure and thermal elements. NASA would supply probe bus drawings, specifications, and finally the common subsystems. NASA would also supply probe bus structural, thermal, and prototype models for subsequent ESRO modification. In addition, NASA would supply technical support. An important feature of the recommended program was an early start of the ESRO program in order that the ESRO contractor work with the NASA contractor during formative stages of the spacecraft program in order to ensure commonality. Preliminary cost study at midterm indicated a cost saving of approximately 18 percent of spacecraft costs could be realized through this plan of international cooperation.

Subsequent to the midterm review, plans for international cooperation were discontinued by NASA and no further work was done on this subject.

4. TESTS CONDUCTED DURING STUDY PHASE

To insure validity of certain conclusions of the Pioneer Venus study, selective testing in key areas was required. Most of the tests were concerned with probes technology, since it is the major area of technical advancement for this program. Several of the tests summarized below were part of an ongoing internal research and development program conducted at Hughes for the advancement of planetary entry technology. They are presented here, since the results are directly applicable to the study.

4.1 THERMAL TESTS – AFT CAVITY SOLAR INTERREFLECTION

One of the important temperature control devices used for the spacecraft bus is the thermal louver assembly. In the proposed application four sets of louvers are mounted in the aft cavity of the bus. In normal flight attitudes the spin axis of the bus is perpendicular to the sun line and the louver receives no direct solar radiation. However, there will be periods (during trajectory correction maneuvers and orbit insertion motor firing) when the aft end of the spinning spacecraft receives direct solar illumination, and the louvers may cyclically receive this direct illumination as well as reflected illumination from the surrounding cavity. The geometry of the cavity is far too complicated to model analytically and expect accurate results. Hence a one-third scale model of the cavity geometry representative of the Pioneer Venus orbiter spacecraft was tested to provide estimates of the solar loading at the thermal control louver area. (The effectiveness of louvers themselves was not part of this test.)

These spin-averaged intensities resulting from the test are plotted against elevation angle in Figure 4-1, along with a plot of sin ϵ as a comparison. This latter curve corresponds to the direct solar load if there were not interreflections or shading in the cavity. The trend of the data indicates that cavity shadowing and interreflection tend to offset each other; i.e., the measured solar loads approach those of an unobstructed plane surface. The overall results of this test program indicate that significant solar loading will be experienced by the louvers even at moderate sun angles, especially at Venus where the solar intensity is nearly doubled. Therefore, the duration of the spacecraft in this attitude has to be restricted.

TABLE 4-1. PRESSURE VESSEL STRUCTURAL TESTS SUMMARY OF STRENGTHS

Spec No.	Description	Sphericity Condition	Failure Pressure, N/cm^2	Failure Mode	Knock Down Factor, K
T-1	Ti monocoque hemi. - no window	Near perfect*	2320	Buckled near flange	0.78
T-2	Ti monocoque hemi. - no window	Near perfect*	2320	Buckled near flange	0.78
TW-1	Ti monocoque hemi. - with window	Near perfect*	2340	Buckled near window and flange simultaneously	0.79
TW-2	Ti monocoque hemi. - with window	Near perfect*	2330	Window and flange buckles simulatneous	0.79
T-2	Ti monocoque hemi. - no window	Imperfect**	1250	Buckled at imperfections near flange	0.42
S-1	Ti monocoque sphere - 1 window	Imperfect***	1340	Buckled at imperfections near window	0.45
T-WA-1	Ti waffle hemi. - no window	Perfect	1810 local 1940 complete	Local: larger panels buckle Complete: vertical ribs collapse at buckled panels	0.40 of equivalent weight monocoque
B-1	Be monocoque hemi. - no window	NA	1030 yield	Yields uniformly throughout	NA
B-2	Be monocoque hemi. - no window	NA	1030 yield 1680 fracture	Yield throughout and then shatters	NA
BW-1	Be monocoque hemi. - with window	NA	1030 yield	Yields throughout	NA
BW-2	Be monocoque hemi. - with window	NA	1030 yield 1840 fracture	Yields throughout then shatters	NA

*Sphericity accurate to within 3 percent of shell thickness

**Waves on surface near flange 0.028 cm double amplitude

***Waves on surface near window 0.036 cm double amplitude

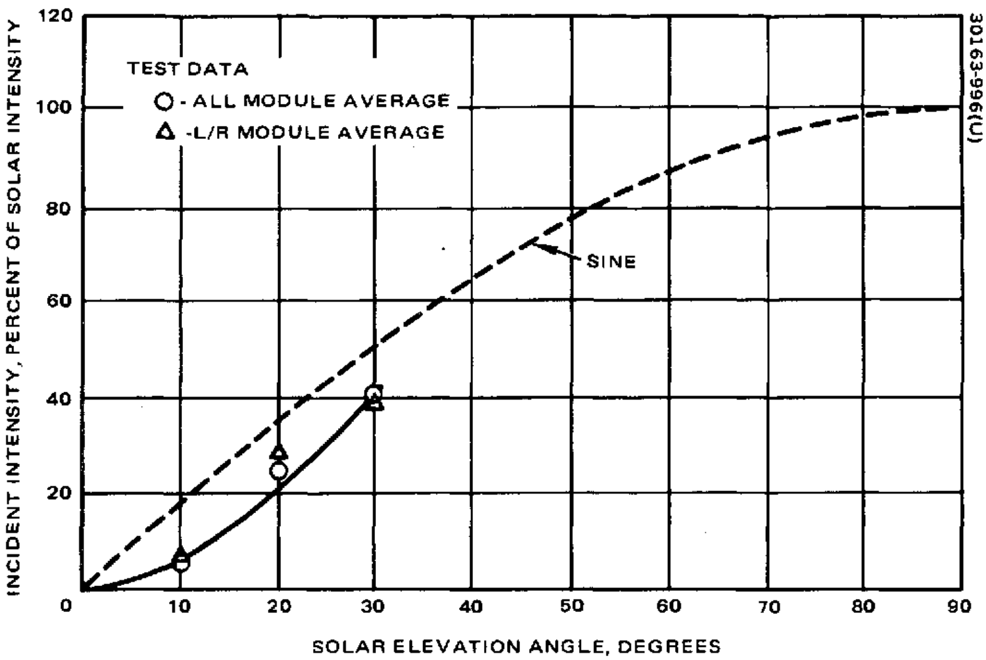

FIGURE 4-1. SPIN AVERAGED MEASURED LOUVER MODULE SOLAR LOADS

4.2 PRESSURE VESSEL STRUCTURAL TESTS

Because the extreme pressure near the Venusian surface presents a serious structural design challenge for the Pioneer Venus probes, a series of room temperature strength tests were performed on 22.9 cm (9 in.) diameter models of hemispherical and spherical shells representing the probe pressure vessel.

The test series investigated the effect of flanges, window penetrations, and geometrical imperfections on the ultimate strengths. Titanium shell specimens of R/t = 67 which failed by elastic instability were included, as well as beryllium shells of R/t = 38 whose mode of failure was yielding and fracture. A rib stiffened titanium shell was also investigated. Buckling failures at pressures as high as 79 percent of the theoretical pressure for a perfect sphere were achieved in accurately fabricated titanium specimens. Failures as low as 40 percent of the classical value occurred in the presence of small imperfections. Beryllium exhibited unexpectedly high plastic deformation prior to fracture. The integrity of the window design as well as of two different types of high pressure seals was proven.

The results of these tests shown in Table 4-1 provided the essential information needed to design the full scale pressure vessels for the large and small probes.

Subsequent to the above, a series of pressure vessel main seal tests were conducted under pressures and temperatures simulating the atmosphere at the surface of Venus. A seal with a V-shaped cross section successfully withstood these rigorous terminal conditions as well as a simulated space cruise condition.

TABLE 4-2. LARGE PROBE PRESSURE VESSEL AERODYNAMIC CHARACTERISTICS

Configuration	Characteristics
C_{D_o}	0.58
C_{N_α}	1.60
Cm_α	-0.041
Cm_q	-0.052
$Cm_{\dot{\alpha}}$	0
C_{ℓ_o}	0.0025
C_{ℓ_p}	-0.10
ℓ ref. m (ft)	0.965 (3.167)
S ref, m^2 (slug-ft^2)	0.731 (7.877)
Ixx, Kg-m^2 (slug-ft^2)	7.43 (5.48)
Iyy, Kg-m^2 (slug-ft^2)	7.31 (5.39)
Izz, Kg-m^2 (slug-ft^2)	7.31 (5.39)
m, Kg (slug)	152.8 (10.47)

4.3 SMALL PROBE SUBSONIC TESTS

A series of tests were conducted in the NASA Langley vertical spin tunnel and the NASA/Ames water tank to evaluate qualitatively the subsonic terminal fall stability for the small probe. Terminal stability is critical for the small probe because of the long time (approximately 1 h) required to reach the planet surface after achieving terminal conditions, and to the 10 deg angle of attack limitation for communication. These tests examined configuration variables such as nose cone angle, center of gravity locations, and the use of roll damping fins and their effects on the observed motion.

During initial tests, predicted spin induced dynamic instabilities were observed. At that time large fins were attached to the models and were noted to be extremely effective in damping the initial roll rate and hence preventing the instability. Additional tests were completed with the object of evaluating the effect of various fin designs on the roll damping. Both solid and fabric drogue devices were tested at various axial locations behind the model with, and without, high initial spin rates. In addition, flow characteristics of the boundary layer were evaluated using smoke and tufts.

The principal result from these tests was to recommend a 45 deg half cone angle for the baseline configuration. This was preferred over the lighter 55 deg configuration because of its better stability in the high subsonic flight regime.

4.4 LARGE PROBE PRESSURE VESSEL MODULE — WATER TANK AND SPIN TUNNEL TESTS

The pressure vessel, after separation from its deceleration module, becomes the principle scientific instrument platform and must provide good stability and a well defined descent motion. The proposed pressure vessel configuration is a sphere, modified to provide pitch and yaw stability and a predictable spin rate. Models of several configurations were built and tested in the Ames Research Center water tank facility. These tests indicated that a number of these configurations had the required stability and developed the predicted spin rate.

The best performing models from the water tank test, plus a new configuration with variations which include steps, flares, discs and fins, were selected for tests in the NASA Langley spin tunnel. Models were fabricated of fiberglass and balsa, with adjustable weight and c.g. The dynamic behavior was recorded on film and analyzed. All of the configurations tested were found to be stable with the nominal c.g. location and one of these has been selected as the baseline. It meets all the requirements for drag, static and dynamic stability, and roll damping. Various protuberances simulating the typical scientific instruments were found to have negligible effect on the dynamic performance. Results of the tests have been covered into aerodynamic coefficients and presented in Table 4-2 for the Atlas/Centaur configuration.

TABLE 4-3. PLASMA ARC SCREENING TEST MATRIX

Facility	Test Gas	Flow Condition	Convective Heat Transfer		Radiative Heat Transfer		Local Pressure		Number of Models			
			w/m^2	Btu/ft^2-sec	w/m^2	Btu/ft^2-sec	N/m^2	atm	PN	CP	TEF	ESM
G. E. hyperthermal arc	Air	Laminar splash	5.1×10^7	4,500	0	0	$3\text{-}4 \times 10^5$	3 to 4	1	1	1	1
	CO_2	Laminar splash	5.0×10^7	4,400	0	0	$3\text{-}4 \times 10^5$	3 to 4	1	1	1	1
	CO_2	Laminar splash	1.6×10^7	1,400	0	0	$3\text{-}4 \times 10^5$	3 to 4	1	1	1	1
	Air	Turbulent wedge	2.4×10^7	2,100	0	0	$1\text{-}2 \times 10^5$	1 to 2	1	1	1	1
	CO_2	Turbulent wedge	2.5×10^7	2,200	0	0	$1\text{-}2 \times 10^5$	1 to 2	1	1	1	1
NASA Ames AEHS	Air	Laminar splash	1.1×10^7	1,000	0	0	10^5	1	2	2	2	2
	Air	Laminar splash	1.1×10^7	1,000	1.7×10^7	1,500	10^5	1	2	2	2	1
	Air	Laminar splash	1.8×10^7	1,600	0	0	10^5	1	1	1	1	0
	Air	Laminar splash	1.8×10^7	1,600	1.1×10^7 to 2.3×10^7	1,000 to 2,000	10^5	1	4	4	4	3
								Total	14	14	14	11

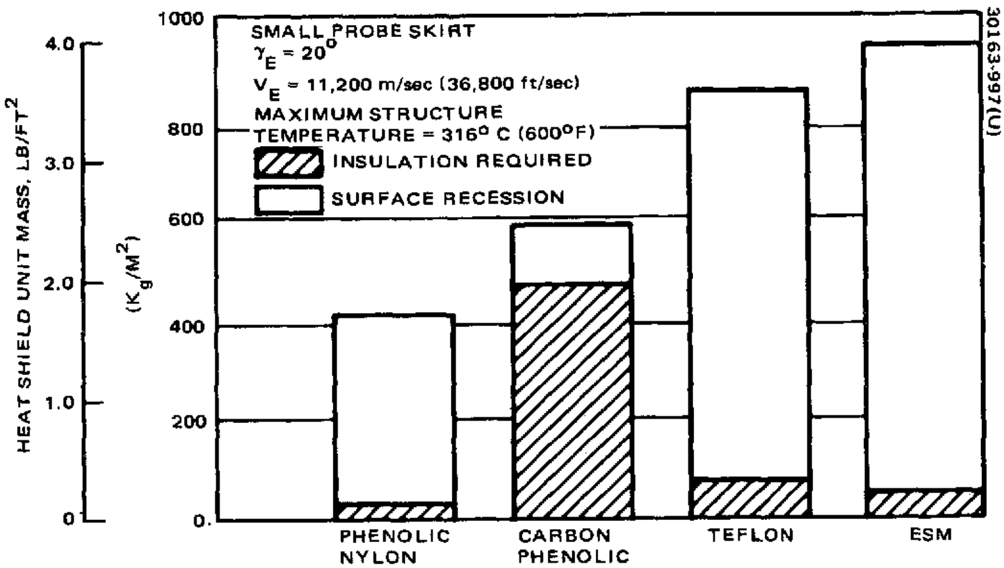

FIGURE 4-2. COMPARISON OF SURFACE RECESSION AND INSULATION REQUIREMENTS FOR VARIOUS CANDIDATE MATERIALS

4.5 HEAT SHIELD ABLATION TEST PROGRAM

As a verification of the recession rates computed for the four candidate heat shield ablation materials, a series of tests was conducted in the plasma arc ablation facilities at GE-RESD and NASA Ames. The environmental and heating parameters of the test program are summarized in the test matrix shown in Table 4-3.

Overall, the results of the test program indicated that the heat shield requirements defined in the system trade study are adequate as illustrated in the bar chart Figure 4-2.

4.6 PROBE WINDOW INTEGRATION TESTS

The provision of windows for the probe experiments presents several difficult design problems which are best solved by actually testing representative window assemblies in a simulated Venusian atmosphere. A series of tests were conducted for the purpose of determining window heating efficiency of selected heaters and to verify the use of a metallic O-ring window seal under various atmospheric pressure and temperature conditions. The test specimen consisted of a 19 mm (0.75 in.) diameter 3.5 mm (0.125 in.) thick sapphire window, a washer configuration heating element, insulating washer, clamp ring, and a base plate. One side of the sapphire window was exposed to a CO_2 environment, while the other was sealed from the CO_2 by a metal O-ring in a clamp ring assembly.

During the tests the assembly survived a combined environment of CO_2 gas pressure of 88.4 to 95.2 atm and gas temperatures of 482°to 504°C (900° to 940°F) for 20 min during which time the window heater was operating at three power levels up to 65 W. At the 65 W level, the surface of the sapphire window exposed to the CO_2 gas reached a temperature of 593°C (1100°F). The response of the window surface temperature to a 20 W heater step input was less than 2 min for an external condition 88.4 atm and 493°C (920°F) of CO_2.

A separate test series was conducted to determine the suitability of four candidate infrared transparency windows. Two types of infrared transparent ZnSe windows were found to degrade from 30 to 60 percent when exposed for 1 min to concentrated sulfuric acid and fluosulfonic acid. On the other hand, a sapphire window (the baseline selection for windows operating at visible wavelengths) suffered no degradation when exposed to these atmospheres for times comparable to the probe's descent time.

The above data and the overall structural integrity of the assembly confirmed the various design features and fabrication method chosen for the baseline.

4.7 HIGH G TESTS

The Pioneer Venus mission requires that the electronic components in the probes function during and/or after exposure to entry loads of 650 g. Typical flight quality components were subjected to high g environment in a centrifuge. The overall results of the centrifuge tests shown in Table 4-4 substantiate the fact that Hughes Aircraft Company's methods of electronic component packaging will allow electronic components to function during and after exposure to the probe high g environments.

TABLE 4-4. CENTRIFUGE TEST RESULTS

Test Unit	Component Assembly Technique Employed	Test Level, g	Test Results
Countdown chain	MICAM	804	Output signals strong and sharp during and after tests
Preregulator	Cordwood	508	Checked out during five of six tests and after all tests
Cross strap converter	Point-to-point	512	No frequency change or conversion loss after test
Quartz oscillator (Hewlett-Packard off-the-shelf item)	Printed circuit	697	Frequency change after test. Fastener sheared off during test

4.8 PROBE ANTENNA TESTS

Tests were conducted on a Hughes IR&D program to evaluate two antennas that were also early candidates for the probe antennas. An equiangular spiral and loop-vee antenna model were fabricated and extensively tested. Tables 4-5 and 4-6 present a comparison of the design and performance parameters of the two antennas. The equiangular spiral antenna meets the design goals but the loop-vee antenna fell short. Reevaluation of the loop-vee analysis indicates the design goals were too optimistic.

TABLE 4-5. EQUIANGULAR SPIRAL ANTENNA CHARACTERISTICS

	Design Parameter	Test Values
Frequency, GHz	2.1 to 2.3	2.1 to 2.3
Coverage angle, deg	45 ± 10	45 ± 10
Gain, dB		
At ± 35 deg	4.1	4.4 avg
At ± 45 deg	5	5.1 avg
At ± 55 deg	4.1	4.4 avg

TABLE 4-6. LOOP-VEE ANTENNA CHARACTERISTICS

	Design Parameter	Test Values
Frequency, GHz	2.3 ± 0.05	2.295 ± 0.05
Coverage angle, deg	60 ± 10	60 ± 10
Gain, dB		
At ± 50 deg	3.6	2.8 avg
At ± 60 deg	4.5	3.1 avg
At ± 70 deg	3.6	3.0 avg

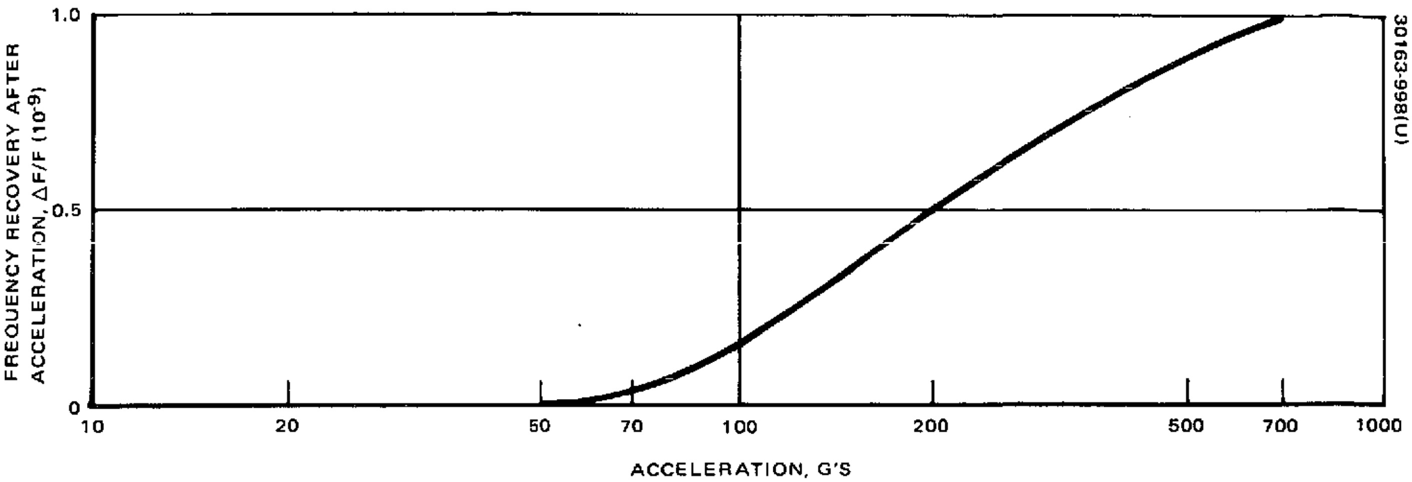

FIGURE 4-3. STABLE OSCILLATOR TEST RESULTS

4.9 STABLE OSCILLATOR

A stable oscillator has been tested to demonstrate that the required frequency stability is not degraded due to a high g deceleration pulse as anticipated for the small probe environment. A test model of the 38.2 MHz oscillator was built (single oven type with a Dewar flack package) for a study of the effects of acceleration. The oscillator was subjected to 700 g of acceleration by a centrifuge machine. Results indicate that there was no apparent frequency change immediately after the deceleration and that there were no deleterious effects to the crystal after it was subjected to a high g force. The test crystal was a fifth overtone AT cut type housed in a TO-8 package. The housing of the proposed crystal will be a "C" size holder, which is more rigid than the holder of the test crystal. This "C" size packaged crystal has passed vibration and shock tests in previous programs and is best suited for the requirements of the stable oscillator design.

The centrifuge test results shown in Figure 4-3 demonstrate that the oscillation frequency was retained to 1 part in 10^{-9} after being subjected to an acceleration pulse as high as 700 g.

4.10 EXTERNAL INSULATION TEST

A series of tests were run with the objective of measuring the transient response of two candidate insulation materials in a simulated Venusian descent environment. Thermal tests were needed early in the program because of uncertainty in external insulation performance in the descent environment. Insulation thickness is critical because it directly influences aeroshell size and, hence, has a major effect on overall probe weight.

The two materials tested were Min-K TE 1400 and FA fiberglass, which were applied separately to the outside of a 17.8 cm diameter solid aluminum sphere and exposed to a typical large probe temperature/pressure profile. The Min-K insulation was 3.2 cm (1.25 in.) thick, and the fiberglass was the same thickness with a density of 0.16 g/cc (10 lb/ft^3). The Min-K was machined into two hemispheres from blocks which were bonded together, and then fitted over the sphere with a step joint in the seam. While the Min-K was carefully machined, it apparently did not fit tightly enough around the sphere because of int instrumentation leads. The density and thickness of the external fiberglass configuration were also difficult to control.

The test specimens were each tested in a CO_2 pressure chamber. Starting at room temperature, the gas was bled into the chamber and heated, reaching a nominal temperature of 768°K at 93.7 atm pressure, simulating a descent from 68 km in 72.4 min.

The Min-K sphere temperature reproducibility was excellent, but the temperatures were much higher than expected. The average temperature rise was 48°C (87°F), while analysis predicted only an 18°C (32°F) rise. Inspection after the test showed a discoloration pattern on the inner surface of the Min-K around the seam, indicating the possibility that it was partially open and that a convective heat short developed through the insulation. The fiberglass sphere temperature rose an average of 60°C (108°F), but the fiberglass density for the test was only about 20 percent of the Min-K and, hence, a greater rise was expected.

The lack of success in correlating the analytical data with the empirical data and the apparent sensitivity of performance to seam construction makes the use of Min-K risky. Depending on the instrument complement, the large probe pressure vessel would have as many as 35 separate pieces of insulation material. The difficulty in fitting this many pieces of Min-K into a leak-tight configuration was a strong factor in the eventual selection of fiberglass internal insulation for the Atlas/Centaur application.

4.11 PROBE MOCK-UPS

One each large and small probe mock-up is currently being fabricated for delivery to NASA ARC. These mock-ups represent the final report probe design for the Atlas/Centaur launch vehicle. The Hughes mock-up of the large probe for the midterm (February 1973) Thor/Delta is shown in Figure 4-4. This mock-up is representative of the mock-up to be delivered by 1 August 1973.

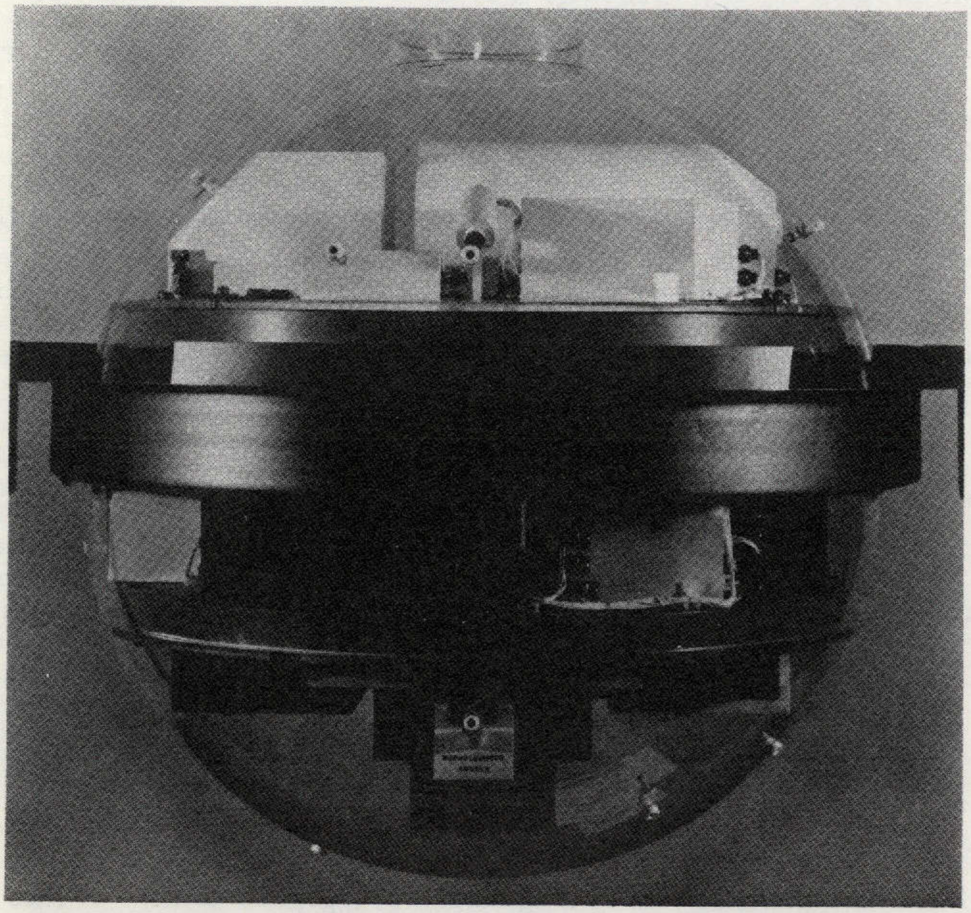

FIGURE 4-4. THOR/DELTA LARGE PROBE MOCKUP
(PHOTO 73-18409)

APPENDIX. TASK SUMMARY

Task No.	Task	Hughes Reference No.
EX	EXPERIMENT INTEGRATION	
EX1	Payload tradeoff analysis	HS-507-0022-158
EX2	Payload design integration	HS-507-0022-42
EX8	Wind drift radar/altimeter study	HS-507-0022-159
EX10	Externally mounting upper atmosphere sensors	HS-507-0022-116
EX11	External sensors alignment/stability study	HS-507-0022-146
EX12	Spin axis orientation/science requirements	HS-507-0022-63
EX15	Magnetometer studies	HS-507-0022-71
MS	MISSION ANALYSIS	
MS1	Baseline sequence of functional events	HS-507-0022-85
MS2	Nominal mission profile (now part of MS1)	HS-507-0022-85
MS3	Mission launch dates	HS-507-0022-103
MS4	Nominal orbital elements	HS-507-0022-151
MS5	Nominal probe target locations	HS-507-0022-97
MS8	Midcourse guidance requirements	HS-507-0022-120
MS19	Probe trajectory dispersions	Final report data
MS22	Probe communications	HS-507-0022-0350
MS23	Sequencer implementation and atmospheric sensitivity	HS-507-0022-138

Task Summary (continued)

Task No.	Task	Hughes Reference No.
MS24	Reduced payload analysis	HS-507-0022-83 HS-507-0022-117
MS25	1978 Orbiter transit Trajectory Selection	HS-507-0022-131
MS26	Probe mission sequences	HS-507-0022-134
PW	ELECTRICAL POWER SUBSYSTEM	
PW1	Power subsystem requirements and performance	HS-507-0022-137
PW2	Regulated versus unregulated bus, power subsystem studies	HS-507-0022-44
PW3	Losses and required margins	HS-507-0022-143
PW4	Subsystem functional design	Final report data
PW5	Power electronics design	HS-507-0022-155
PW6	Battery selection and operating characteristics	HS-507-0022-69
PW7	Solar panel analysis, design and performance	HS-507-0022-93
PW8	High-g battery study	HS-507-0022-92
CM	COMMUNICATIONS	
CM1	Bus enhanced navigation	Final report data
CM2	Probe coherent two-way doppler tracking	HS-507-0022-119
CM3	Probe modulation/signal design	HS-507-0022-95
CM4	Probe coding scheme	

Task Summary (continued)

Task No.	Task	Hughes Reference No.
CM5	Link Analysis	HS-507-0022-96
CM6	Probe antennas	HS-507-0022-102
CM7	Design criteria for high-g and high temperature operation of probe electronics	HS-507-0022-129
CM10	Solid state power amplifier modules	HS-507-0022-79
CM11	Micromin receiver usage	HS-507-0022-77
CM12	Bus antenna trades	HS-507-0022-104
CM13	RF subsystem functional design	Final report data
CM14	Atmospheric effects/multipath	HS-507-0022-118
CM16	Communications subsystem components performance study	HS-507-0022-78
CM17	Bus and orbiter antenna tradeoffs	HS-507-0022-33
CM18	Antenna subsystem functional design	Final report data
CM19	Orbiter radio science impact on communications subsystems design	HS-507-0022-147
CC	COMMAND/CONTROL	
CC1	Preliminary command list preparation	HS-507-0022-73
CC2	Analyze command storage requirements	Final report data
CC3	Command modulation techniques and message formats	Final report data
CC6	Command interfaces with experiments	HS-507-0022-148
CC7	Probe stored sequence requirements	HS-507-0022-362

Task Summary (continued)

Task No.	Task	Hughes Reference No.
CC8	Analyze prevention of inadvertent irreversible command execution	HS-507-0022-68
CC9	Command subsystem functional design	Final report data
DH	DATA HANDLING	
DH1	Data storage requirements	Final report data
DH2	Use of central programmable processor	Final report data
DH3	Multiplexer and analog/digital converter requirements	Final report data
DH4	Evaluate digital interface designs	HS-507-0022-149
DH5	Probe data storage	HS-507-0022-38
DH6	Probe data rate	HS-507-0022-37
DH7	Probe data handling frame optimization	Final report data
DH8	Data handling list preparation	Final report data
DH9	Bus data handling format requirements	Final report data
DH12	Data handling subsystem functional design	Final report data
TH	THERMAL CONTROL	
TH1	Candidate thermal design comparison	Final report data
TH2	Spacecraft temperature distribution	Final report data
TH3	Probe preentry thermal design trades	HS-507-0022-54
TH4	Aft cavity solar interreflection test evaluation	HS-507-0022-75

Task Summary (continued)

Task No.	Task	Hughes Reference No.
TH5	Rocket exhaust plume impingement study	HS-507-0022-130
TH6	Thermal control subsystem functional design	Final report data
PP	PROPULSION	
PP1	Pressure supply system and pressurant	HS-507-0022-82
PP2	Propellant management technique	HS-507-0022-31
PP3	Propellant tradeoff	HS-507-0022-43
PP4	Thruster arrangement evaluation	Final report data
PP5	Propellant utilization effect on center of mass	HS-507-0022-76
PP6	Operational flexibility	Final report data
PP7	Power and thermal insulation requirements	Final report data
PP8	Latch valve arrangement optimization	HS-507-0022-122
PP9	Thruster operating requirements	HS-507-0022-127
PP10	Orbit insertion motor design selection	HS-507-0022-89
PP11	Propulsion subsystem functional design	Final report data
AC	ATTITUDE CONTROL	
AC1	Attitude dynamics analysis	HS-507-0022-115
AC2	Attitude/ΔV error analysis	HS-507-0022-114
AC4	Attitude reference and sensor tradeoff	HS-507-0022-157

Task Summary (continued)

Task No.	Task	Hughes Reference No.
AC5	Orbiter MDA dynamics/control system analysis	HS-507-0022-141
AC6	Development of attitude control system operational backup modes	Final report data
AC7	Attitude control system functional design	Final report data
AC8	Attitude control system electronics design study	HS-507-0022-140
AC9	Mechanical and pyrotechnic devices design study	HS-507-0022-132
PB	PROBES	
PB2	Pressure vessel insulation selection study	HS-507-0022-41
PB3	Insulation evaluation	Final report data
PB4	Pressure vessel thermal analysis	HS-507-0022-156
PB5	Non-pressure protected payload study	HS-507-0022-128
PB6	Probe trajectory studies	HS-507-0022-58
PB7	Aerodynamic heating analysis	HS-507-0022-70 HS-507-0022-106
PB8	Heat shield material trade study	
PB9	Heat shield screening tests	Final report data
PB10	Small probe subsonic stability	HS-507-0022-124
PB11	Pressure vessel subsonic stability	HS-507-0022-142

Task Summary (continued

Task No.	Task	Hughes Reference No.
PB12	Heat shield design study	HS-507-0022-136
PB13	Reflective heat shield design study	HS-507-0022-123
PB14	Descent study optimization 4/6 Rev A. 0022-30A	HS-507-0022-30 HS-507-0022-30A
PB15	Engineering experiments study	HS-507-0022-107
PB16a	Large probe conceptual design	Final report data
PB16b	Large probe conceptual design — deceleration module	Final report data
PB17a	Small probe conceptual design	Final report data
PB17b	Small probe conceptual design — deceleration module	Final report data
PB18	Probes aerodynamic configuration selection	Final report data
PB19	Entry vehicle dynamic stability analysis	HS-507-0022-125
PB20	Parachute dynamic stability analysis	HS-507-0022-126
PB21	Pressure vessel dynamic stability analysis	HS-507-0022-133
PB22	Probes/bus separation analysis	HS-507-0022-108
PB24	Despin subsystem design study	HS-507-0022-109
PB25	Aeroshell structural design study	HS-507-0022-110
PB26	Parachute design study	HS-507-0022-135
PB27	Pressure vessel structural design	HS-507-0022-121
PB28	Pressure vessel structural test evaluation	HS-507-0022-84

Task Summary (continued)

Task No.	Task	Hughes Reference No.
PB29	Experiments/structure interaction design study	HS-507-0022-154
PB30	Small probe integrated sturcture study	HS-507-0022-111
PB32	Deceleration module atmospheric sensitivity	HS-507-0022-112
PB38	Pressure vessel configuration screening test evaluation	HS-507-0022-56
PB39	Experiment window design	HS-507-0022-113
VI	VEHICLE INTEGRATION	
V14	Dynamic loads analysis	HS-507-0022-88
V15	Stress analysis	Final report data
V16	Preliminary configuration and interface drawings	HS-507-0022-101
V17	Mass property and weight statements	Final report data
EP	ELECTRONIC PACKAGING/HARNESS	
EP1	MICAM construction under high-g	HS-507-0022-90
EP2	Other foaming materials	HS-507-0022-90
EP3	Use of microminiature connectors	HS-507-0022-80
EP4	High temperature, high-g, high pressure parts, materials, and processes evaluation and selection	HS-507-0022-72
TP	TEST PLANNING	
TP1	Methods and sequence of test and launch operations	HS-507-0022-86

Task Summary (continued)

Task No.	Task	Hughes Reference No.
TP2	Tradeoffs on environmental levels	HS-507-0022-74
TP3	Program test hardware cost analysis	HS-507-0022-94
TP4	STV testing philosophy	HS-507-0022-64
TP5	Airplane versus balloon testing of parachutes	HS-507-0022-105
RE	RELIABILITY	
RE1	System reliability definition	HS-507-0022-153
CA	COST ANALYSIS	Proposal, Final Report and Plans data
SP	SPECIFICATIONS*	
SP3	Spacecraft/DSN interface definition	HS-507-0300-2-1
SP5	Spacecraft/launch vehicle interface definition	HS-507-0300-1-1
SP9	GSE design specification	HS-507-0022-145
SP19	Project specification requirements	HS-507-0022-50
SP21	Spacecraft block diagrams	HS-507-0022-91
SP22	Spacecraft master indices	HS-507-0022-32
SP25	Large probe functional requirements	HS-507-0300-4
SP26	Small probe functional requirements	HS-507-0300-5
SP27	Orbiter spacecraft functional requirements	HS-507-0300-6
SP28	Multiprobe bus functional requirements	HS-507-0300-3
SP29	Pioneer Venus baseline definition	HS-507-0022-66

*Final specifications and plans will be provided concurrently with the proposal in response to RFP No. 2-21976.

Task Summary (continued)

Task No.	Task	Hughes Reference No.
PL	PROJECT DEVELOPMENT PLAN*	
IC	INTERNATIONAL COOPERATION (NASA/ESRO)	Final report data
LV	LAUNCH VEHICLE UTILIZATION STUDY MISSION DEFINITION	
LV1	Launch vehicle capabilities	HS-507A-0022-7
LV2	Mission Analysis	HS-507A-0022-10
LV3	Nominal mission profiles	HS-507A-0022-11
	SYSTEM DESIGN	
LV4A	Large probe functional requirements	HS-507A-0022-1
LV4B	Small probe functional requirements	HS-507A-0022-2
LV4C	Orbiter functional requirements	HS-507A-0022-3
LV4D	Probe bus functional requirements	HS-507A-0022-4
LV5	System description	HS-507A-0022-6
	SPACECRAFT	
LV7	Configuration/Structures	HS-507A-0022-9
LV8	Power functional design	Final report data
LV9	Comm. functional design	Final report data
LV10	Command/Control functional design	Final report data
LV11	Data handling functional design	Final report data
LV12	Thermal control functional design	Final report data

*Final specifications and plans will be provided concurrently with the proposal in response to RFP No. 2-21976.

Task Summary (continued)

Task No.	Task	Hughes Reference No.
LV13	Propulsion functional design	Final report data
LV14	Attitude control functional design	Final report data
	PROBE DESIGN	
LV15	Probe integration	Final report data
LV16	Pressure vessel module design	Final report data
LV17	Deceleration module design	Final report data
	EXPERIMENT INTEGRATION	
LV18	Experiment integration	Final report data
	TEST PLANNING	
LV19	Cost reduction in test and operations	HS-507A-0022-8
	RELIABILITY	
LV20	Reliability and risk analysis	HS-507A-0022-152
	COST ANALYSIS	
LV21	Cost analysis tradeoffs using Atlas/ Centaur launch vehicle	Final report data
	MIDTERM REPORT	HS-507-0422