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FINAL REPORT
SYSTEM DESIGN OF THE PIONEER VENUS SPACECRAFT

VOLUME 11
LAUNCH VEHICLE UTILIZATION

By

R. J. VARGA

ET AL.

July 1973

Prepared Under

Contract No. [redacted] NAS 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.

CONTENTS

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1. SUMMARY

This section presents a summary of the spacecraft descriptions; the probe bus, large probe, small probe, and orbiter. The highlights of the designs of the Atlas/Centaur spacecraft as compared to the corresponding Thor/Delta spacecraft designs are contained herein.

Figure 1-1 is a comparison of the two Atlas/Centaur spacecraft for reference. The major differences are the replacement of the probes on the forward end of the probe bus with the mechanically despun antenna of the orbiter and the replacement of the bicone antenna on the aft end with the orbit insertion motor. Figure 1-2 compares the cross sections of the large and small probes. Minimum spacing between boxes is 1.3 cm (0.5 in.) in both probes. The major features of each probe are described in Section 4.

Table 1-1 is a comparison of the Thor/Delta and Atlas/Centaur designs for the probe bus and orbiter. The usable spacecraft mass for the Atlas/Centaur is roughly twice that for the Thor/Delta if the Type I trajectory is assumed. It is somewhat less for the Type II trajectory in the designated launch years. This additional mass capability leads to cost savings in many areas which will be described in this report.

1.1 PROBE BUS

The probe bus is a spin stabilized vehicle using sun and star sensors for attitude reference. It is a basic bus providing a platform for the science experiments and the probes. It is designed to accommodate the experiment payload of 12.6 kg (27.7 lb) and one large and three small probes weighing a total of 368 kg (811 lb). Its design employs maximum use of already developed hardware and commonality with the orbiter design to reduce total program costs.

Characteristics and Performance of the Probe Bus

Table 1-2 provides a description of the major characteristics and performance of the Atlas/Centaur probe bus design compared to the Thor/Delta. This discussion will highlight those items providing cost savings for the Atlas/Centaur design. The usable spacecraft mass is about double that of the Thor/Delta version allowing the use of liberal allowances in mass and volume for the spacecraft hardware and still maintaining a 153 kg (337 lb) contingency. The Atlas/Centaur design employs aluminum throughout its design, whereas

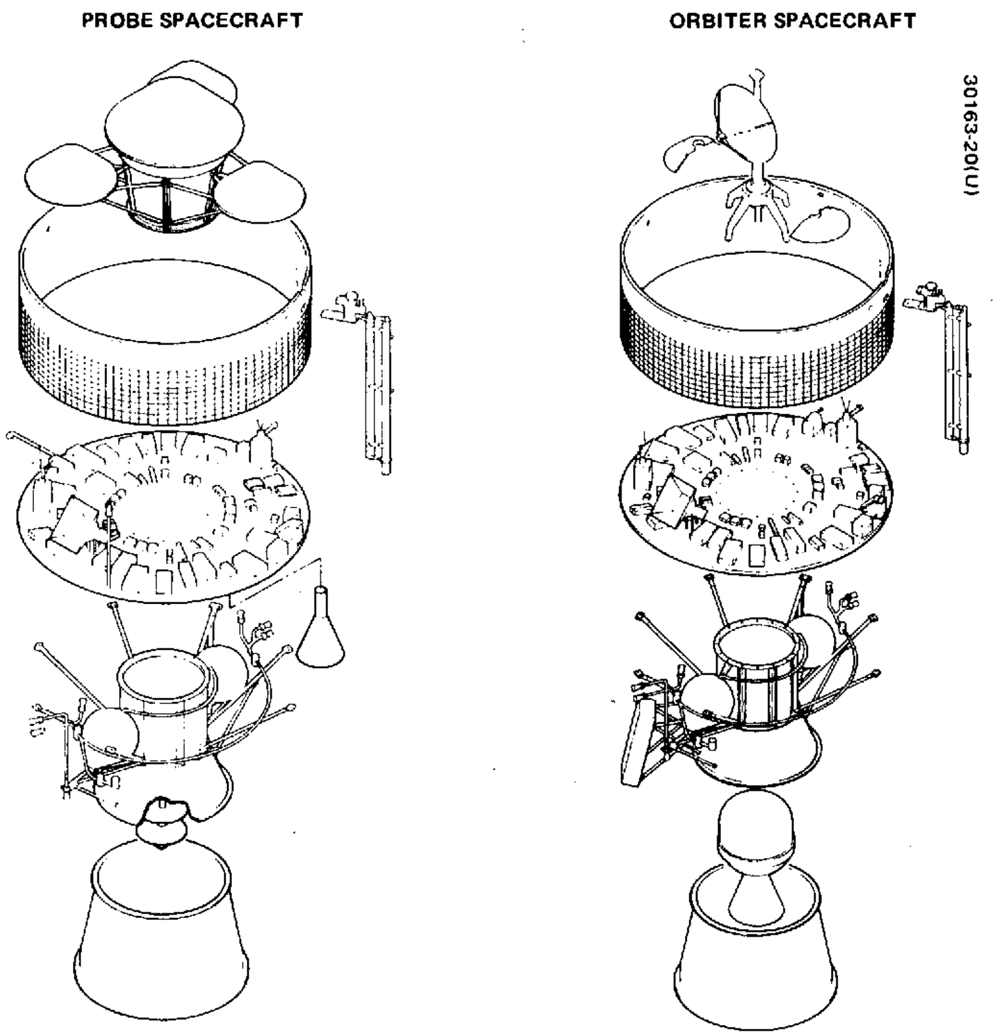
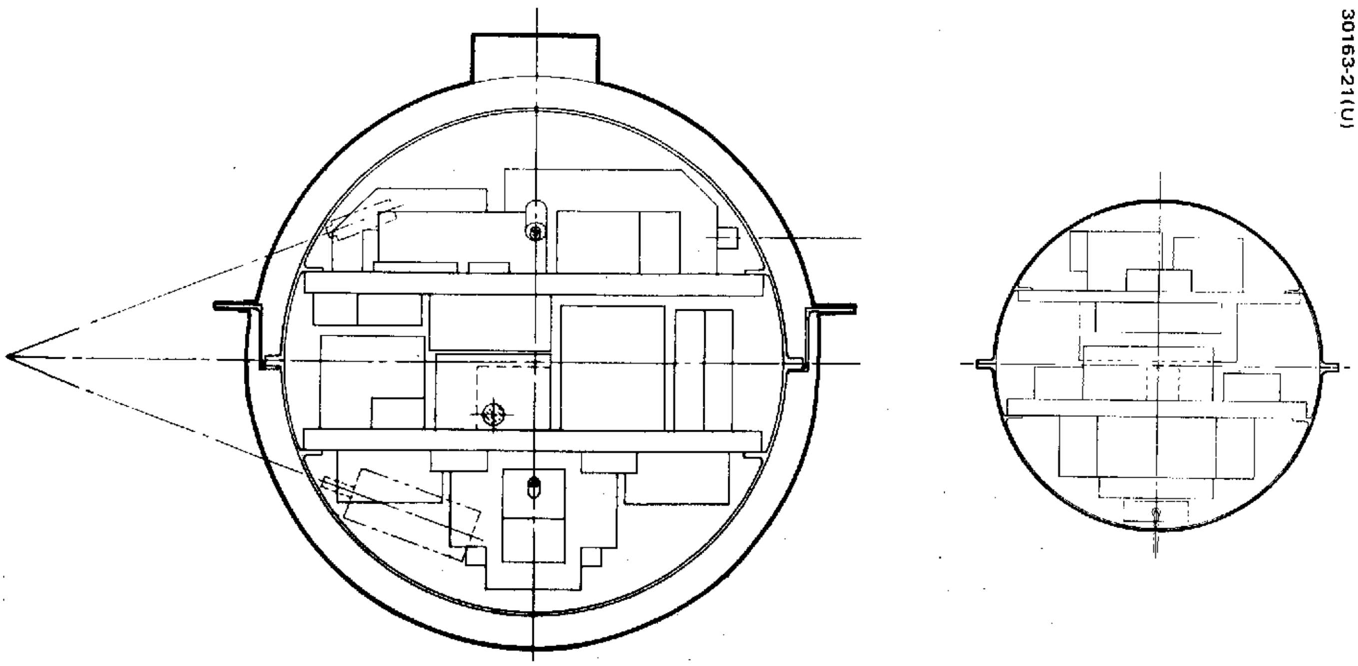


FIGURE 1-1. ATLAS/CENTAUR CONFIGURATIONS



LARGE PROBE **SMALL PROBE**

FIGURE 1-2. ATLAS/CENTAUR LARGE AND SMALL PRESSURE VESSEL MODULE CROSS SECTIONS

TABLE 1-1. MASS COMPARISON OF THOR/DELTA AND ATLAS CENTAUR

Item/Subsystem	Probe Spacecraft						Orbiter Spacecraft					
	Thor/Delta		Atlas/Centaur		Difference		Thor/Delta		Atlas/Centaur		Difference	
	kg	lb	kg	lb	kg	lb	kg	lb	kg	lb	kg	lb
RF	8.9	19.5	10.0	22.1	+1.1	+2.6	8.5	18.8	9.7	21.4	+1.2	+2.6
Antenna	3.1	6.8	3.4	7.4	+0.3	+0.6	2.5	5.6	3.5	7.6	+1.0	+2.0
Data handling	5.8	12.8	7.7	17.0	+1.9	+4.2	9.9	21.8	13.2	29.0	+3.3	+7.2
Command	7.0	15.5	12.3	27.2	+5.3	+11.7	6.6	14.5	11.0	24.2	+4.4	+9.7
Attitude control, mechanism	11.2	24.6	14.7	32.3	+3.5	+7.7	18.1	40.0	26.6	58.6	+8.5	+18.6
Structure	36.5	80.4	96.4	212.5	+59.9	+132.1	32.5	71.7	83.3	183.7	+50.8	+112.0
Power	16.0	35.3	27.2	60.0	+11.2	+24.7	22.4	49.4	29.1	64.2	+6.7	+14.8
Cabling	4.7	10.3	9.1	20.0	+4.4	+9.7	6.8	14.9	13.6	30.0	+6.8	+15.1
Thermal control	10.3	22.6	19.0	41.9	+8.7	+19.3	11.3	24.9	20.9	46.0	+9.6	+21.1
Propulsion (dry)	9.1	20.0	10.4	22.9	+1.3	+2.9	9.8	21.7	11.2	24.6	+1.4	+2.9
Orbit insertion motor case	-	-	-	-	-	-	10.1	22.3	26.7	58.8	+16.6	+36.5
Bus total	112.6	247.8	210.2	463.3	+97.6	+215.5	138.5	305.6	248.8	548.1	+110.3	+242.5
Large probe	114.6	252.6	198.7	438.0	+84.1	+185.4	-	-	-	-	-	-
Small probe	101.9	224.7	169.1	372.9	+67.2	+148.7	-	-	-	-	-	-
Spacecraft subtotal	329.1	725.1	578.0	1274.2	+248.9	+549.1	138.5	305.6	248.8	548.1	+110.3	+242.5
Contingency	22.6	50.3	153.0	337.6	+130.0	+287.3	10.1	22.1	91.7	202.5	+81.6	+180.2
Experiments (bus only)	11.6	25.6	12.6	27.7	+1.0	+2.1	31.1	68.6	35.0	77.2	+3.9	+8.6
Spacecraft total (dry)	363.3	801.0	743.6	1639.5	+380.3	+838.5	179.7	396.3	375.5	827.8	+195.8	+431.5
Propellant	20.7	45.7	15.9	35.1	-4.8	-10.6	24.3	53.6	28.3	62.4	+4.0	+8.8
Pressurant	0.1	0.1	0.1	0.1	+0.0	+0.0	0.1	0.1	0.2	0.4	+0.1	+0.3
Orbit insertion expenditure	-	-	-	-	-	-	88.7	195.6	326.9	720.7	+238.2	+525.1
Spacecraft total (wet)	384.1	846.8	759.6	1674.7	+375.5	+827.9	292.8	645.6	730.9	1611.3	+438.1	+965.7
Spacecraft adapter	13.2	29.0	31.3	69.0	+18.1	+40.0	13.2	29.0	31.3	69.0	+18.1	+40.0
T/M and C-band	8.4	18.5	-	-	-8.4	-18.5	8.4	18.5	-	-	-8.4	-18.5
Launch vehicle payload	405.7	894.3	790.9	1743.7	+385.2	+849.4	314.4	693.1	762.2	1680.3	+447.8	+987.2

TABLE 1-2. PROBE BUS CHARACTERISTICS AND PERFORMANCE

	Thor/Delta Spacecraft	Atlas/Centaur Spacecraft
General		
Overall length, cm (in)	241.3 (95)	294.6 (116)
Diameter, cm (in)	213 (84)	254 (100)
Thrust tube diameter, cm (in)	60.1 Be (24)	76.2 Al (30)
Usable spacecraft mass - dry, kg (lb)	363 (801)	730 (1,610)
Science, kg (lb)	11.6 (25.6)	12.6 (27.7)
Probes mass, kg (lb)	227.1 (477.3)	367.8 (811)
Contingency, kg (lb)	22.6 (50.3)	153 (337.6)
Equipment shelf area, m^2 (ft^2)	2.88 (31)	4.27 (46)
Louvers	6	10
Power		
Solar panel area, m^2 (ft^2)	3.07 (33)	3.48 (37.5)
Peak power, W	145	170
Science, W	15.9	24.5
A-hr	(Ag-Zn) 13.6	(Ni-Cad) 7 (2)
Boost regulator	Yes	No
Radio		
Transmitter/receiver	S-band	S-band
RF power, W	1, 5, 10	1, 5, 10
PA efficiency	28	21
Two-way doppler tracking	X	X
3 dB bicone	X	X
Two omnidirectional antennas	X	X
18 dB endfire	X	X
Command		
Modulation	PSK/PM	PSK/PM
Bit rate, bps	1, 2, 4	1, 2, 4
Word size, bits	36	36
Data handling		
Data rates, bps	8-2048	8-2048
Convolutional encoding	X	X
Modulation	PCM/PSK/PM	PCM/PSK/PM
Attitude control/mechanisms		
Spin axis ecliptic	X	X
Spin rates, rpm	15-100	15-57
Star sensor	S.S.	S.S.
Sun sensors	S.S.	S.S.
Spin axis determination acc., deg	0.9	0.9
Magnetometer boom length, m (ft)	1.07 (3.5)	4.4 (14.5)
Bicone antenna deployment	Yes	No
Propulsion		
Blowdown hydrazine	X	X
Six 22.2 N (5 lb) thrusters	X	X
Tanks, cm^3 (in^3)	18,026 (1,100)(2)	37,690 (2,300)(2)

beryllium is used by the Thor/Delta version in selected areas. The larger diameter of the Atlas/Centaur allows usage of a larger equipment shelf. This in turn promotes easier hardware integration and use of more thermal louvers as needed for thermal control margin. The active thermal control by use of the louvers is simplified by maintaining the spin axis normal to the sun-line during the cruise phase except for relatively short periods during maneuvers. Large thermal loads are placed near louvers and heaters used as necessary for temperature control.

The power subsystem has 3.48 m^2 (37.5 ft^2) of solar panel which produces 170 W near Venus and 90 W near earth. Thus, additional power is provided over the Thor/Delta design to enable use of less efficient spacecraft hardware (hence lower cost). The additional weight capability of the Atlas/Centaur allows use of nickel cadmium batteries based on an existing design. These batteries provide a design that is common to the orbiter and their higher output voltage eliminates the need for battery boost on discharge. Electrical interconnections between the bus/probes allow charging of the probe batteries, telemetry monitoring, and other necessary functions.

The RF subsystem operates at S-band frequencies with two-way doppler capability for navigation purposes. RF power outputs are 1, 5 or 10 W depending on the link requirements. A lower efficiency power amplifier is used on the Atlas/Centaur design to save cost in the development phase. The 10 W mode will be used on bus entry along with the 18 dB horn for maximum data capability. Omni antennas provide 4π steradian command coverage for the entire mission. The 3 dB gain bicone is used to transmit telemetry during cruise.

Real time ground command capability is available using PSK/PM modulation at a 1, 2 or 4 bps bit rate with a 36 bit command word size. This is supplemented during the launch phase by onboard command storage. Nine telemetry formats will be used to provide capability for the varying experiment complements at various phases of the mission. Convolution encoding is employed and PCM/PSK/PM modulation. Data rates vary from 8 to 2048 bps as necessary to support mission needs. Less program cost risk is incurred by the more conservative design approach used. The Thor/Delta design uses 10 new LSI elements whereas no new LSIs are required for the Atlas/Centaur.

The spacecraft is spin stabilized with the spin axis aligned normal to the ecliptic except for short maneuver periods. Inertial attitude will be determined by solid-state sun and star sensors. Spin axis attitude will be determined to an accuracy of 0.9 deg. Spin rates will be variable between 15 and 57 rpm, depending on the mission event. The Atlas/Centaur design employs a magnetometer boom that positions the sensor 4.4 m (14.5 ft) from the solar panel surface as compared to 1.07 m (3.5 ft) for the Thor/Delta. This additional distance greatly reduces spacecraft magnetic cleanliness requirements and reduces program costs significantly. In addition, the bicone antenna on the Thor/Delta design requires a deployment mechanism whereas none is required for the Atlas/Centaur.

Six hydrazine thrusters (developed for Intelsat IV) are used for mid-course maneuvers, spin rate changes, and precession control. A total of 16 kg (35 lb) of hydrazine is provided for velocity changes totaling 36 m/sec, spin rates changes of 114 rpm, and precession control totaling 496 deg. The larger sizes of the tanks and tubing makes the Atlas/Centaur spacecraft cost slightly greater than the Thor/Delta.

Characteristics and Performance of the Large Probe

Table 1-3 is a comparison of the key characteristics and performance of the Atlas/Centaur and Thor/Delta designs. The aeroshell cone angle is 45 deg on the Atlas/Centaur design. This reflects a common aerodynamic design with the small probe and reduces total aerodynamic testing. In addition, the material for the Atlas/Centaur version aeroshell is aluminum with that of the Thor/Delta being beryllium, a considerable cost saving. Thicker phenolic nylon is used on the Atlas/Centaur version heatshield to minimize development testing.

The 61.0 cm (24.0 in.) internal diameter of the pressure vessel promotes easier unit integration and greater accessibility. A minimum box separation of 1.3 cm (0.5 in.) is maintained as compared to 0.65 cm (0.25 in.) for the Thor/Delta design. The pressure vessels of both spacecraft are made from titanium monocoque. A more conservative knockdown factor (ratio of expected pressure vessel strength to theoretical) is used on the Atlas/Centaur spacecraft to minimize development testing. For the same reason, thicker insulation is used for the pressure vessel.

In the power subsystem, more volume is available for packaging the batteries, thus easing the battery packaging problems. In the RF area, use of a lower efficiency power amplifier reduces design and testing required and results in a cost saving. In the command and data handling area, the primary cost saving results from the use of more volume for packaging and less packaging constraints. Use of more standard packaging techniques is possible. A single data rate will supply all mission data handling needs in the Atlas/Centaur design.

Characteristics and Performance of the Small Probe

Table 1-4 is a comparison of the small probe designs of the Atlas/Centaur and Thor/Delta configuration. It is similar to Table 1-3 and is largely self-explanatory. The overall design objectives of the small probe are similar to the large, the major ones being as follows:

1) Use of commonality between the large and small probes to the maximum possible in the aerodynamic design.

2) Use of non-exotic material as much as feasible.

TABLE 1-3. LARGE PROBE CHARACTERISTICS AND PERFORMANCE

	Thor/Delta Spacecraft	Atlas/Centaur Spacecraft
General		
Aeroshell cone, deg.	55	45
Phenolic nylon heat shield	X	X
Deceleration module base diameter, cm (in.)	117.0 (46)	123.2 (48.5)
Deceleration module structure	Be	Al
Main parachute diameter, m (ft)	3.5 (11.5)	4.6 (15.0)
Pressure vessel i.d., cm (in.)	54.6 (21.5)	61.0 (24)
Total mass, kg (lb)	114.6 (252.6)	199 (438)
Pressure vessel mass, kg (lb)	75.9 (167.5)	111 (245)
Science payload mass, kg (lb)	22.5 (49.6)	25.6 (56.5)
Volume available, cm^3 (in^3)	84,180 (5,198)	118,544 (7,234)
Science payload volume, cm^3 (in^3)	22,860 (1,395)	24,547 (1,498)
Minimum box separation, cm (in)	0.65 (0.25)	1.3 (0.5)
Power		
Peak power, W	212	245
Science payload, W	49.6	55.3
A-hr (Ag-Zn)	19.2	30
Radio		
Transmitter	S-band	S-band
RF power, W (6.6 W module)	2	2
PA efficiency, percent	28	21
Two-way doppler tracking	X	X
4.8 dB equiangular spiral	X	X
Command		
No ground commands	X	X
96 command register	X	X
20-sec accuracy clock	X	X
Data handling		
Convolutional encoding	X	X
Modulation	PCM/PSK/PM	PCM/PSK/PM
Data rates, bps	276/184	184
Semiconductor memory, bits	4096	4096
A/D resolution, bits	10	10

TABLE 1-4. SMALL PROBE CHARACTERISTICS AND PERFORMANCE

	Thor/Delta Spacecraft		Atlas/Centaur Spacecraft	
General				
Aeroshell cone, deg	45		45	
Phenolic nylon heat shield	X		X	
Deceleration module base diameter, cm (in)	61.0	(24)	72.6	(28.6)
Deceleration module structure	Be		Ti	
Pressure vessel, i.d., cm (in)	32.5	(12.8)	38.1	(15)
Total mass, kg (lb) - 3	101.9	(224.7)	169.1	(372.9)
Pressure vessel mass, kg (lb) - 3	70.8	(156)	96.6	(213)
Science payload mass, kg (lb)	2.2	(4.9)	2.9	(6.3)
Volume available, cm^3 (in^3)	16,748	(1,022)	28,939	(1,766)
Science payload volume, cm^3 (in^3)	966.8	(59)	116.3	(71)
Minimum box separation, cm (in)	0.25	(0.1)	1.3	(0.5)
Power				
Peak power, W	57		75	
Science payload, W	4.3		7.5	
A-hr (Ag-Zn)	10		12	
Radio				
Transmitter	S-band		S-band	
RF power, W	6.6		6.6	
PA efficiency, percent	28		21	
One-way doppler tracking	X		X	
2.5 dB loop vee antenna	X		X	
Command				
No ground commands	X		X	
64 command register	X		X	
20-sec accuracy clock	X		X	
Data handling				
Convolutional encoding	X		X	
Modulation	PCM/PSK/PM		PCM/PSK/PM	
Data rates, bps	16		16	
Semiconductor memory, bits	512		512	
A/D resolution, bits	10		10	

TABLE 1-5. ORBITER CHARACTERISTICS AND PERFORMANCE

	Thor/Delta Spacecraft	Atlas/Centaur Spacecraft
General		
Overall length, cm (in)	277 (109)	330 (130)
Diameter, cm (in)	213 (84)	254 (100)
Usable spacecraft mass - dry, kg (lb)	179.6 (396.3)	375.5 (827.8)
Science, kg (lb)	31.1 (68.6)	35.0 (77.2)
Mass contingency, kg (lb)	10.0 (22.1)	91.9 (202.5)
Thrust tube diameter, cm (in)	60.96 Be (24)	76.2 Al (30)
Equipment shelf area, m^2 (ft^2)	2.88 (31)	4.27 (46)
Louvers	8	12
Power		
Solar panel area, m^2 (ft^2)	3.76 (40.5)	3.99 (43)
Peak power, W	179	193
Science, W	48.5	51.5
A-hr	(Ni-Cad) 10.0	(Ni-Cad) 7 (2)
Boost regulator	Yes	No
Radio		
Transmitter/receiver	S-band	S-band
RF power, W	1, 5, 10	1, 5, 10
PA efficiency, percent	28	21
Two-way doppler tracking	X	X
23.5 dB MDA	X	X
Two omnidirectional antennas	X	X
Command		
Modulation	PSK/PM	PSK/PM
Bit rate, bps	1, 2, 4	1, 2, 4
Word size, bits	36	36
Data handling		
Data rates, bps	8-2048	8-2048
Memory	Core	Core
Convolutional encoding	X	X
Modulation	PCM/PSK/PM	PCM/PSK/PM
Attitude control/mechanisms		
Spin axis ecliptic	X	X
Spin rates, rpm	5-100	5-25
Star sensor	S.S.	S.S.
Sun sensors	S.S.	S.S.
Spin axis determination acc., deg	0.9	0.9
Magnetometer boom length, m (ft)	1.07 (3.5)	4.4 (14.5)
Experiment pointing acc., deg	1	1
High gain antenna pointing acc., deg	3	3
BAPTA	Be	Ti (Telesat mod.)
Propulsion		
Blowdown hydrazine	X	X
Seven 22.2 N (5 lb) thrusters	X	X
Tanks, cm^3 (in^3)	18,025 (1,100)(2)	37,690 (2,300)(2)
Insertion motor	TE-M-521	TE-M-616

3) Use of thicker insulation for thermal margins.

4) Use of larger volumes for easier integration and accessibility.

5) Use of greater structural margins to minimize testing.

6) Use of commonality in the subsystem hardware with the large probe and buses.

1.2 ORBITER

The orbiter basic design goals remain unchanged from the probe bus. These emphasize science payload accomodation, commonality between the two missions, maximum use of existing hardware, and use of the additional weight and power capability of the Atlas/Centaur approach to arrive at a low cost/reliable design.

Table 1-5 provides a description of the major characteristics and performance of the Atlas/Centaur orbiter design compared to the Thor/Delta. It is similar to Table 1-2 previously presented on the probe bus. Much of the discussion of the probe bus is relevant due to the commonality of the two spacecraft and will not be repeated here. Only those major differences will be noted.

The major changes from the probe design include substitution of a high gain, mechanically despun antenna for the bicone and endfire antennas. A memory is required in the data handling subsystem for occultation and eclipse data storage. An additional two louvers are required for thermal control of the spacecraft because of greater electrical power. Larger solar panels are incorporated to generate this power. In the propulsion area, an orbit insertion motor is required along with one additional hydrazine thruster for orbital maneuvers.

Spin rates for the orbiter mission will be variable between 5 and 25 rpm, depending on the mission event. The solid propellant motor is the TE-M-616 used on the Canadian Technology Satellite with an ISP of 289.9 sec, and giving a ΔV of 1706.9 m/sec at a spacecraft mass of 724 kg (1596 lb).

A total of 28.3 kg (63 lb) of hydrazine has been provided for velocity changes of 130 m/sec, spin rate changes of 45 rpm, and precession control of 130 m/sec. An additonal axial thruster is required over the probe bus design for satisfaction of these requirements.

1.3 COST SAVINGS SUMMARY

Several cost fact-finding sessions were held with the project team members and major subsystem managers responsible for the development of the project hardware. Cost estimates were obtained for the Atlas/Centaur and Thor/Delta spacecraft configurations based on the designs developed at

TABLE 1-6. 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		

the study midterm. In-depth subsystem cost trade details are provided in Volume 12, Cost Analysis of the final report. The cost savings accrued by utilizing an Atlas/Centaur launch vehicle are provided herein. Table 1-6 summarizes the cost savings (and pertinent increases) attained through evaluation of the cost estimates and many detailed iterations with the subsystem areas on their designs and cost. 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 mission cost estimates. Propulsion, thermal control, and structure subsystems represent cost increases 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 was attributed to larger volume/ease of assembly and integration of pressure vessels and commonality in the cone angles of the probe aeroshells and hence, reduced aerodynamic testing. In addition, about 50 percent of the deceleration module cost reduction is due to the elimination of the use of beryllium.

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 by this deletion. It should be noted that a non-flight prototype spacecraft will be required if thermal and structural test models are deleted.

Conservative and simplifying assumptions can be made in establishing the margins of safety and 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 1-6, 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 which use off-the-shelf hardware include the communications, power, structure, propulsion, thermal control, attitude control, and command and data handling subsystems. The RF subsystem will use an unmodified Viking transponder and lower efficiency/higher-power 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. No boost voltage circuitry will be required in the power system also.

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

Although magnetic cleanliness shows a cost savings of $1,430, this should not be construed as a deletion of the magnetic cleanliness program. In the Atlas/Centaur configuration a 4.4 m (14.5 ft) boom is utilized instead of a 1.07 m (3.5 ft) boom on the Thor/Delta design. This is to be interpreted as

TABLE 1-7. ATLAS/CENTAUR COST SAVINGS COMPARED TO THOR/DELTA

Discipline	Existing Hardware/ Proven Designs	Commonality	Overdesign	Increase Power/ Weight/Volume	Installation and Access	Test Considerations	Other
Systems					Integration assembly and simplified	System test operations reduced	
Structure/ harness	HS-339 basic bus technology. Less exotic materials		Structural margin increased F.S. of 2	24 versus 28 gauge wire	247.7 cm (97.5 in) diameter shelf	Delete STM and TBM. Reduce static testing and development	Less stress and dynamic analysis
Communica-tions				PA efficiency 21 percent (28) percent Heavier filters			
Data/ command				Probe packaging simplified Less use of LSI			
Controls	BAPTA Telesat modification (titanium)			Nondeployable bicone antenna			
Probes		45 deg cone angle on aeroshells	Structural margin pressure vessel (0.4 versus 0.7) = K 4.6 m (15 ft) parachute		0.5 in spacing between boxes	Deceleration module Pressure vessel Reduce specimen testing	Reduce aerody-namic, heatshield, and structural testing
Electrical power	Telesat batteries OSO discharge regulator and no boost voltage circuitry	Ni-Cd on probe bus and orbiter					
Experiments			4.4 meter (14.5 foot) boom for mag. clean		Experiment integration simplified		

having the magnetometer sensor deployed out far enough such that the stringent controls on subsystem hardware can be reduced or eliminated. There will still be a $500 to $600K cost to assure that the appropriate magnetic level is obtained at the sensor. The larger boom and its associated testing represent added costs.

In the mechanism area, the deletion of a deployment mechanism for the bicone antenna on the probe bus represents a cost reduction. For the orbiter, a modified Telesat bearing and power transfer assembly for the mechanical despun antenna and the use of titanium instead of beryllium in its design indicate a cost savings.

More detailed cost information on the subsystems is presented in Section 6.2 of this report. Table 1-7, however, provides in matrix form the cost factors influencing the cost savings attained in the Atlas/Centaur spacecraft design. Across the top of this table are the influencing cost factors. In the far left column are the affected subsystems. Entries are made in the appropriate columns where pertinent cost savings were obtained.

Some potential cost savings approaches were rejected after study showed that they were not cost effective. Three of these potential cost savings approaches were derived from the Lockheed low cost effects studies; however, these approaches were not consistent with a mission of only two spacecraft or the philosophy of refurbishment or repair in space. The typical rejected potential cost saving approaches are shown in Table 1-8. The approaches are self-explanatory as are the reasons for rejection.

TABLE 1-8. TYPICAL REJECTED POTENTIAL COST SAVING APPROACHES (INCLUDING LOCKHEED PROPOSALS)

Approach	Reason Rejected
Use of low reliability components	Test cost increase offsets parts reduction
Fixed antenna (orbiter)	Could not meet requirements without excessive thermal, power, and TWT problems
OSO PMT star sensor	Solid state cost effective
Eliminate parachute on probe	Alternate separation techniques not cost effective
Cold gas propulsion	Ineffective for trajectory correction maneuvers
Less efficient (cheaper) solar cells	Labor overhead problem
Centralized computer	Software costs/hardware uncertainty

2. INTRODUCTION

The purpose of this study was to perform a Systems Design of the Pioneer Venus spacecraft utilizing at Atlas/Centaur launch vehicle with the prime objective of saving costs. With relatively unconstrainted weight, volume, and power capability as compared to the Thor/Delta spacecraft design, the main thrust of the Atlas/Centaur launched spacecraft activity was to provide sufficient analysis, tradeoffs of alternate designs and studies to fully define mission parameters, requirements, constraints and the optimum low cost system and subsystem design for the 1977 multiprobe mission and the 1978 orbiter mission.

Details of the Thor/Delta design descriptions are found in other volumes of the Pioneer Venus final report. This volume contentrates on the differences of the Atlas/Centaur configuration as compared to the Thor/Delta configuration as of the midterm review on February 26 and 27, 1973. Cost savings in the design of the Pioneer Venus spacecraft launched from Atlas/ Centaur are provided herein as well as a discussion of the cost factors influencing the selection of the subsystems for this design.

2.1 GUIDELINES

This study emphasized the low cost approach and did not use the increased capabilities of the Atlas/Centaur launch vehicle to enhance or modify the mission or spacecraft capabilities compared to the Thor/Delta launch vehicle spacecraft design. This was a basic guideline established in the Ames Research Center statement of work. The project personnel were directed to be innovative so that the study would result in a reliable system design at the lowest overall cost. The study was to minimize development and use existing or proven hardware with low cost as the primary design criteria, not low weight or restricted volume. Table 2-1 is a list of low cost guidelines that was initially prepared and reviewed with the project team at the beginning of the study. Further iteration on these guidelines developed the cost savings factors as discussed in Section 6.2 and were fundamental in achieving the cost savings that was obtained.

The spacecraft configuration development effort started by evaluating the applicability of the Intelsat IV, Telesat (HS-333) and the Domestic Satellite (HS-339) with minimal modification considered to the Pioneer Venus mission requirements. The probe designs were initiated with increases of 30 percent

TABLE 2-1. LOW COST GUIDELINES

Set and meet firm cost targets
Increase hardware mass allowance
Increase hardware power allowance
Increase volume of packages
Identify high cost items and try alternates
Solid cost basis
Use proven technology off-the-shelf hardware and commonality
Employ hardware update (versus new hardware)
Use of minimum mandatory design and performance requirements
Consider requirements as goals and tradeoff features for cost
Use low cost materials/components
Use high structural safety factors and reduce parts/components stress levels
Minimumize development/qualification testing (higher allowable risk)
Reduce precision tolerances for fabrication and assembly
Simplify/modularize hardware (for accessibility and repair)
Simplify contract/document requirements
Specify only minimum requirements for purchased equipment
Simplify configuration management (traceability)
Eliminate complex deployable mechanisms for fixed installations where possible and hence minimize testing

in mass and volume over the Thor/Delta configuration and with an increase of 20 percent in power and footprint area on the shelves. Preliminary sizing in the probes started with 2.5cm (1 in) separation between equipment boxes for ease of assembly and integration in the pressure vessels. Atlas/Centaur science was provided with increases in size, mass, and power as directed by Ames Research Center. The orbiter design was to utilize one of the existing orbit insertion motor alternates to eliminate any development or qualification costs.

The outputs of this study culminated in some 20 launch vehicle (LV) tasks reports submitted under separate covers and included the following:

1) Evaluation of the capabilities of the Atlas/Centaur launch vehicle in relation to the Pioneer Venus mission requirements.

2) Definition of the key system parameters for the Atlas/Centaur launched 1977 probe spacecraft and the 1978 orbiter spacecraft.

3) Description of the baseline system design which satisfies the mission requirements and is consistent with the aforementioned considerations relating to cost, mass, volume, and the use of existing equipment.

4) Preparation of design descriptions for the subsystems and subsystem elements of the baseline design.

5) Evaluation of performance capabilities of the baseline system versus reliability, cost, testing and overall mission suitability.

6) Cost comparisions of Thor/Delta and Atlas/Centaur spacecraft design and programs with considerations of cost saving alternatives.

3. MISSION ANALYSIS/REQUIREMENTS

The purpose of this subsection of the report is to summarize the mission requirements imposed by the science instruments and to show the highlights of those requirements identified in the mission analysis process.

The science requirements are summarized in subsection 3.1. This is a presentation of the probe bus, probes and orbiter main objectives, and the requirements imposed by the mission experiments. Details of the experiment accommodation are contained in subsection 5.7. The parameters pertinent to the mission trajectories are covered in summary form in subsection 3.2. The capability of the Atlas/Centaur launch vehicle to perform its assigned function is shown herein. Further details of this analysis may be found in Reference 3-1.

Graphic displays of some of the key times in the probe bus and orbiter mission sequences are contained in subsection 3.3, with further details in Reference 3-2. And finally, the rationale behind the probe descent profiles optimization and selection is contained in subsection 3.4. Required data rates, power, parachute size, deployment altitude, jettison altitude, etc., are discussed in summary fashion.

3.1 SCIENCE REQUIREMENTS

The objective of the Pioneer Venus Program is to conduct scientific investigations of the planet Venus and its environment. The program originally included two missions: an atomospheric entry multipleprobe mission in the 1976/77 launch opportunity, and an orbiter mission in the 1978 opportunity. Data in this volume are presented based on the orginal 1976/77 and 1978 missions.

Figure 3-1 describes the atmosphere regions of Venus showing the temperature variation with altitude. The turbopause divides the atmosphere into upper and lower portions at about 130 Km above the mean surface. These portions are further subdivided into the exosphere, thermosphere, stratosphere, and troposphere to describe the areas of scientific interest in the following paragraphs.

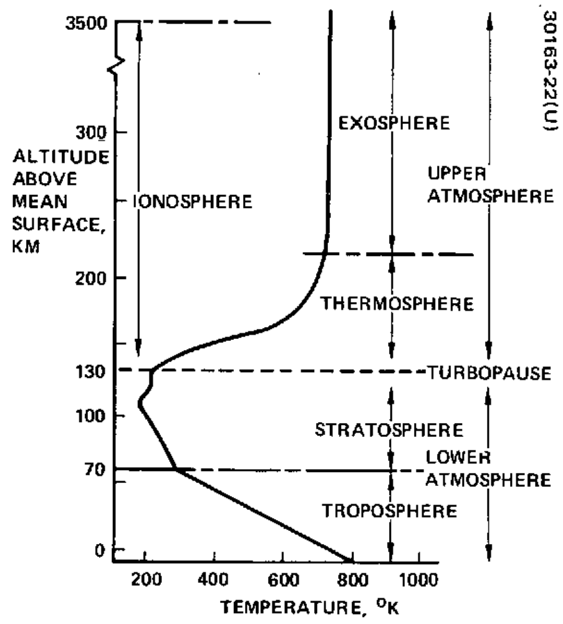


FIGURE 3-1. SCIENCE OBJECTIVES FOR MULTIPROBE MISSION

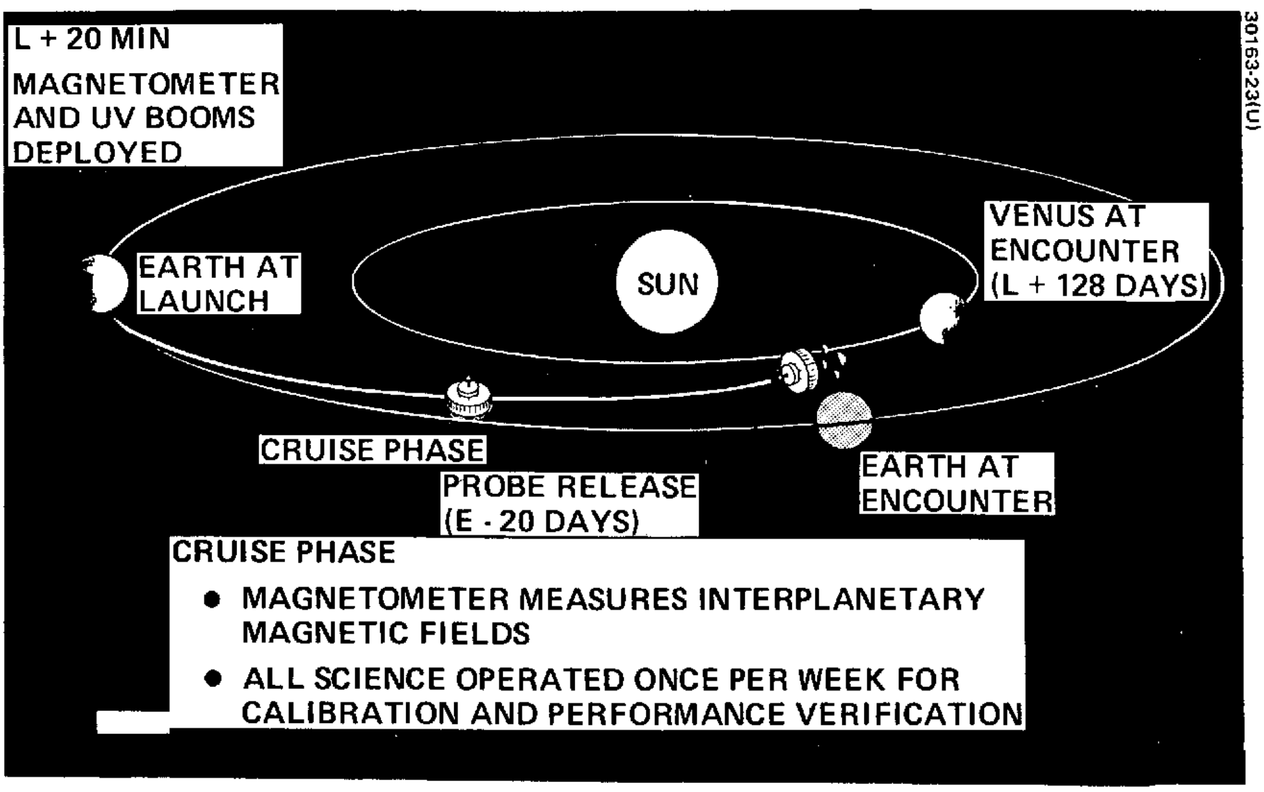


FIGURE 3-2. MULTIPROBE MISSION

Multiprobe Mission

The scientific objectives of the multiprobe mission are to investigate the dense regions of the Venus atmosphere and are summarized in Table 3-1. The nature and composition of the clouds will be investigated using the large probe for measurement of the troposhere from an altitude of about 70 km to the surface. The composition and structure of the atmosphere will be investigated via the probe bus measurements and the large probe. The bus will obtain measurements of the stratosphere and troposphere. Data on the general circulation pattern of the atmosphere will be obtained by spacing the small probe entry at points widely separated from the large probe entry.

Figure 3-2 is a summary of the multiprobe mission for reference purposes. It will be discussed in more detail in subsection 3.3 of this report. The cruise from earth to Venus lasts about 128 days as shown. The magnetometer and ultraviolet (uv) booms are deployed shortly after launch. Interplanetary magnetic field measurements will be made. All science is operated once per week during the cruise for calibration and performance verification. At 20 days prior to encounter, the four probes are targeted and released from the bus to continue their flight to Venus. At about 128 days after launch, the encounter phase begins and data is obtained from the bus and four probes over a 1 to 2 hour period, terminating with impact of all probes on the planet's surface and destruction of the bus by burnup in the lower atmosphere.

The implementation of the science objectives of the probe bus are shown in Table 3-2. There are five experiments carried onboard with various objectives as shown. The data samples required per minute are shown along with the capability. In all cases, the capability of the bus, exceeds the minimum requirements. The altitude range of operation of the various experiments is shown. Figure 3-3 is a plot of bus measurement requirements per scale height for the five experiments.

To ensure that the proper data measurements are obtained, a number of targeting constraints must be observed. These are summarized in Table 3-3. The probe bus must have a shallow entry for maximum observation time before burnup. In addition, the entry must be delayed by 1.5h until all probes have reached the surface. This is necessary for the proper operation of the DLBI experiment as indicated by Reference 3-3.

The large probe must enter on the day side, not closer than 20 deg to the terminator so that the sun is above the horizon. The small probes must be widely separated to give greatest coverage in latitude and longitude but must remain within a 60-deg earth communication angle limit. The timing must be such to ensure entry within 30 min of each other for accomplishment of the DLBI experiment. Figure 3-4 is a graphic display of the selected impact points of the probes and bus. A review of this figure verifies satisfaction of all of the targeting constraints.

Tables 3-4 and 3-5 summarize the entry and descent measurements for the large and small probes respectively. A listing of all experiments, their measurement objective, and the samples per minute are shown. The

TABLE 3-1. SCIENCE OBJECTIVES FOR MULTIPROBE MISSION

- Nature and composition of clouds

  Large probe measurement of troposphere (70 km to surface)

- Composition and structure of atmosphere

  Probe bus measurements of upper atmosphere (to ~130 km)

  Large probe measurements of stratosphere and troposphere

- General circulation pattern of atmosphere

  Small probe entry at points widely separated from large probe entry.

TABLE 3-2. PROBE BUS ENTRY MEASUREMENTS

Experiment	Objective	Samples/Min		Altitude Range
		Required	Capability Provided	
Neutral mass spectrometer	Measure number density of selected constituents of upper atmosphere: H, He, O, CO, N_2, Ar, CO_2	2	20	1000 km to bus burnup
Ion mass spectrometer	Measure number density of selected thermal ions in upper atmosphere: H^+, D^+, He^+, O^+, CO^+, NO^+, $O_2{}^+$, $CO_2{}^+$	2	20	5000 km to bus burnup
Langmuir probe	Measure temperature and number density of ionosphere thermal electrons	60	600	5000 km to bus burnup
UV Fluorescence	Measure amount of O and CO in upper atmosphere (backup to neutral mass spectrometer)	20	200	1000 km to bus burnup
Magnetometer	Measure magnetic profile through solar wind bow shock, sheath, plasmapause, and ionosphere	20	200	Cruise to bus burnup

TABLE 3-3. MULTIPROBE TARGETING REQUIREMENTS

Probe bus
Shallow entry (γ = -12 deg) for maximum observation time before burn up
Entry delayed (1.5 h) until all probes have reached surface (for DLBI experiment)
Large probe
Enter on dayside not closer than 20 deg to terminator, so sun is not near horizon
Small probe
Widely separated to give greatest coverage in latitude and longitude within acceptable earth communication angle limits (60 deg)
Minimum acceptable spread in latitude, 0 to ±30 deg; desirable 0 to ±60 deg
Minimum spread in longitude, 90 deg; desirable 120 deg
Enter within 30 min of each other (for DLBI experiment)

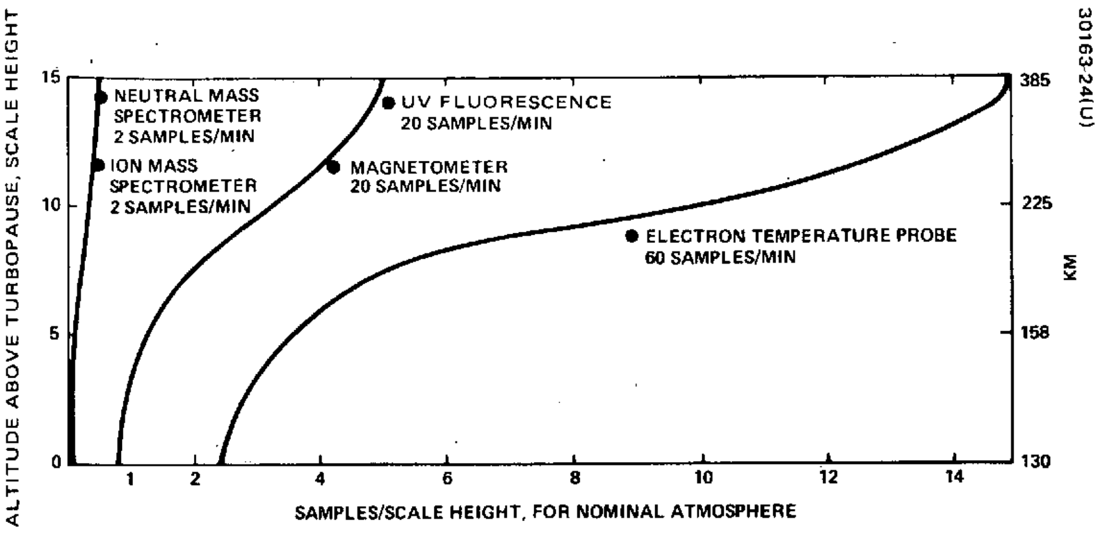


FIGURE 3-3. PROBE BUS MEASUREMENT REQUIREMENTS

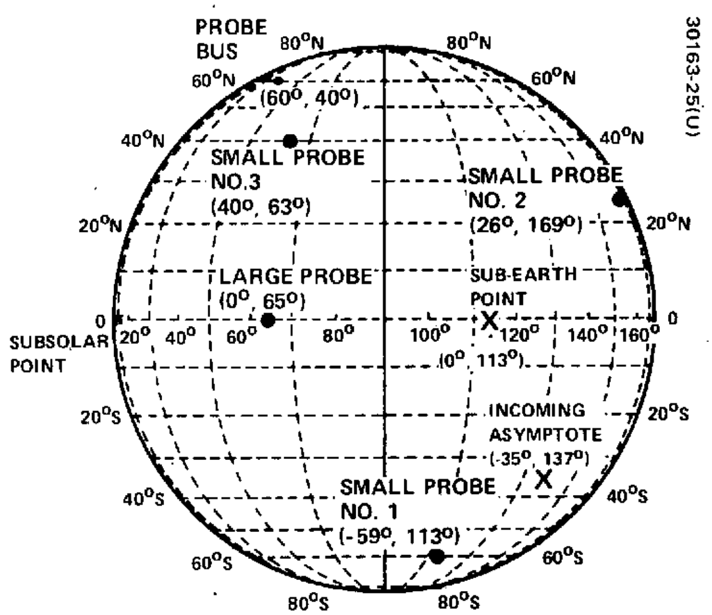


FIGURE 3-4. MULTIPROBE TARGETING

TABLE 3-4. LARGE PROBE ENTRY AND DESCENT MEASUREMENT REQUIREMENTS

Experiment	Measurement Objective	Samples Per Min.	Samples/Scale Height								
			1	2	3	4	5	6	7	8	Blackout Data
Temperature	Temperature profile to impact	6	150	66	36	15	30	18	-	-	
Pressure	Pressure profile to impact	3	75	33	18	7.5	15	9	-	-	
Accelerometers	Deceleration history and atmospheric turbulence	3	75	33	18	7.5	15	9	30*	5*	(Stored) 25**
Neutral mass spectrometer	Atmospheric and cloud constituents	1/6	4.2	1.8	1	0.4	0.8	0.5	-	-	
Cloud particle size analyzer	Cloud particle size	16	400	176	96	40	80	48	-	-	
Solar flux	Deposition of solar energy in atmosphere	2.4	60	26	14	6	12	7	-	-	
Planetary flux	IR radiation from planet	2.4	60	26	14	6	12	7	-	-	
Aureole	Extinction of sunlight and particle size	24	-	-	144	60	120	72	-	-	
Nephelometer	Presence of clouds	24	600	264	144	60	120	72	-	-	
Shock layer radiometer	Radiation from heated atmosphere during entry	-	-	-	-	-	-	-	-	-	(Stored) 25
Hygrometer	Water vapor content	6	-	-	-	15	30	18	-	-	
Transponder	Wind velocity	Continuous									

* Post blackout mode 1 sample/sec
**Blackout mode 2.5 samples/sec

TABLE 3-5. SMALL PROBE ENTRY AND DESCENT MEASUREMENT REQUIREMENTS

Experiment	Measurement Objective	Samples Per Min.	Samples/Scale Height 1	2	3	4	5	6	7	8	Blackout Data
Temperature	Temperature profile	3	114	54	30	12	6	1	--	--	
Pressure	Pressure profile	3	114	54	30	12	6	2	0.12	0.05	(Stores) 10*
Nephelometer	Presence of clouds	2	76	36	20	8	4	1	0.08	0.03	
Accelerometer	Deceleration history	3	114	54	30	12	6	2	0.12	0.05	
Magnetometer	Planetary magnetic field	2	76	36	20	8	4	1	0.08	0.03	
State oscillator	Wind velocity	-									

*Blackout mode, 1 sample/sec

TABLE 3-6. SCIENCE OBJECTIVES FOR ORBITER MISSION

Objectives	Orbit Preferences	Science Instruments
Detailed structure of upper atmosphere and ionosphere by "in-situ" techniques	Lowest possible periapsis altitude ($\leq$150 km) Highly inclined orbits (>60 deg) Low periapsis latitudes (>45 deg)	Neutral mass spectrometer Ion mass spectrometer
Interaction of solar wind with Venus ionosphere and magnetic field	Highly elliptical orbit (to 5 to 10 Venus radii) Highly inclined orbits (>60 deg) Low periapsis latitudes (>45 deg)	Magnetometer Electron temperature probe Solar wind probe
Characteristics of atmosphere and surface of Venus on planetary scale by remote sensing	Low altitude circular, highly inclined orbits (>60 deg) Midperiapsis latitudes for elliptical orbits (45 deg)	UV spectrometer IR radiometer RF occultation RF altimeter
Gravitational field harmonics from perturbations of spacecraft orbit around Venus	Free orbit (no corrections) Highly inclined orbits (>60 deg)	

samples per scale height are numbers generated for the Thor/Delta design and due to variations in probe descent rates are not exact for the Atlas/ Centaur design but are close enough to be representative.

Orbiter Mission

The scientific objectives of the 1978 orbiter mission are shown in Table 3-6. The orbit preferences to enable optimum data collecting are shown. The actual orbit selected is a compromise of the desired orbits and will be shown in subsection 3.2 of this report. Measurements of the detailed structure of the upper atmosphere and ionosphere will be obtained by the neutral and ion mass spectrometers using "in-situ techniques. The interaction of the solar wind with Venus ionosphere and magnetic field will be investigated by gathering data with the magnetometer, electron temperature probe, and solar wind probe. The characteristics of the atmosphere and surface of Venus on a planetary scale by remote sensing will be investigated with the spectrometer, radiometer, occultation, and altimeter experiments. The final major objective of the orbiter is to investigate the gravitational field harmonics from perturbations of the spacecraft orbit around Venus.

The implementation of the above science objectives of the orbiter are shown in Table 3-7. The nine orbiter experiments are listed along with their measurement objective, samples required, and the operating range. Three of the experiments operate for the entire orbit, namely, the magnetometer, spectrometer, and solar wind probe. The other experiments operate for a portion of the orbit as noted. Figure 3-5 shows a portion of the orbit around periapsis along with a summary of the orbit characteristics. The time from periapsis is shown for the critical altitudes. Operation time for the experiments operating under 5000 km is approximately 50 min and those under 1000 km is 20 min. Switching of the experiments at the appropriate times is accomplished by onboard stored commands.

3.2 MISSION/TRAJECTORY ANALYSIS

The analysis performed and summarized herein leads to a definition of the Atlas/Centaur capability to perform the mission requirements for the 1977 multiprobe and 1978 orbiter missions. Nominal mission launch dates are defined for both missions and a selection between Type I and Type II orbits for the orbiter mission is accomplished. Propulsion requirements for the transit phase and orbital phase are defined. A nominal orbit is selected to satisfy orbiter science requirements. A detail report on this work is covered in Reference 3-1.

Table 3-8 is a summary of Atlas/Centaur payload mass capability that was used in this analysis. This was based on the information contained in the Statement of Work for Pioneer Venus Mission Systems Design Study, dated March 1, 1972. Launch is accomplished from ETR with a maximum azimuth angle 108 deg. Use of the Intelsat IV adapter was assumed. Fourteen daily launch periods of 30 min each have been assumed for both probe and orbiter missions. For the launch opportunities of interest, the Type II

TABLE 3-7. ORBITER MEASUREMENT REQUIREMENTS

Experiment	Measurement Objective	Samples/ Min	
Magnetometer	Magnetic fields in solar wind/ ionosphere interaction	5	Entire orbit
Electron temperature probe	Electron temperature and density in ionosphere	60	5000 km
Neutral mass spectrometer	Neutral atmosphere composition and density	0.2	1000 km
Ion mass spectrometer	Ion composition and density of upper atmosphere	0.4	5000 km
UV spectrometer	Minor constituents of atmosphere, air glow	2	Entire orbit
IR radiometer	Thermal structure of atmosphere above cloud tops	10	5000 km
RF occultation	Dispersive absorption and scattering by cloud particles	-	Occultation
RF altimeter	Surface height variation, reflectivity, and roughness	5	1000 km
Solar wind probe	Flux and energy distribution of solar wind particles	5	Entire orbit

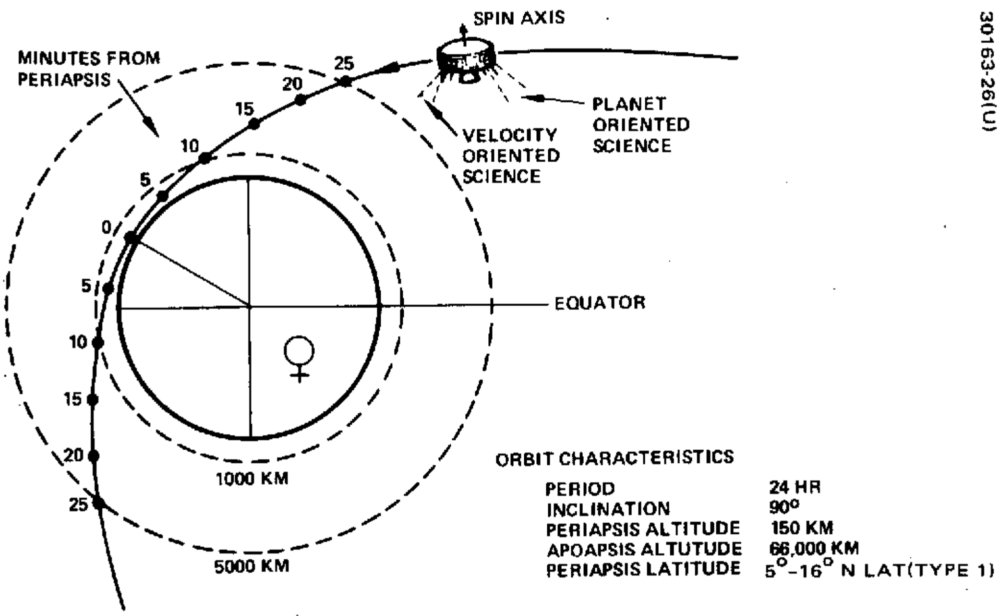


FIGURE 3-5. ORBITER MISSION

TABLE 3-8. ASSUMED ATLAS/CENTAUR PERFORMANCE CAPABILITY (DUE EAST LAUNCH)

C_3 $(km/sec)^2$	Injected Mass* kg (lb)
4.	878 (1935.)
6.	814 (1794.)
8.	752 (1658.)
10.	693 (1527.)
12.	634 (1398.)
14.	576 (1270.)
16.	519 (1144.)
18.	463 (1020.)
20.	408 (900.)
22.	356 (785.)
24.	306 (675.)

*Not including adapter and attach fitting.

requires significantly more energy than Type I with a resulting substantial reduction in allowable spacecraft mass. ΔV required for midcourse corrections is small for all cases because of the relatively accurate injection of the Atlas/Centaur as compared to Thor/Delta.

Multiprobe Mission

The probe trajectory performance optimization requires only minimization of C_3 and is independent of the launch vehicle; therefore, the baseline mission is identical to that derived for the Thor/Delta in Reference 3-4. The 14 selected launch opportunities are given in Table 3-9 (values are given at the start of each daily window; launch azimuth increase throughout the daily window is about 4 deg with correspondingly small variations in other parameters). The arrival date and the probe targeting considerations are identical to those for the Thor/Delta mission. The maximum spacecraft mass for the 14 days launch window is 759.6 kg (1674.7 lb); 16 kg (35.1 lb) of hydrazine are required for velocity and attitude control throughout the mission. Probe targeting locations and dispersions are reported in References 3-5 and 3-6.

The multiprobe mission transit geometry is shown in Figure 3-6 for a nominal 120-day transit period. The positions of the earth, Venus, and spacecraft are shown each 10 days during the transit phase. Figure 3-7 is a presentation showing the spacecraft distance to the Sun, Venus, and earth. Table 3-10 is a summary of propulsion ΔV requirements for the multiprobe mission.

Orbiter Mission

The optimization of the orbiter trajectories is described in Reference 3-7. The actual optimum trajectories obtained with the Atlas/Centaur launch booster are somewhat different from the Thor/Delta counterparts because the slope of the mass/C_3 characteristics of the launch booster (Table 3-8) are different. As a result, the Type I transit trajectory permits a substantially larger spacecraft mass than the Type II trajectory when the Atlas/Centaur launch vehicle is used. The Type I transit trajectory was originally selected as the baseline to utilize this performance advantage. The science coverage considerations related to the ecliptic latitude and longitude of the Venus orbit periapsis are identical to those discussed for the Thor/Delta in Reference 3-7. The science coverage is somewhat less desirable than that for a Type II trajectory because the periapsis latitude is near the equator. A comparison of science coverage for Type I and Type II is shown in Table 3-11.

The 14 selected launch opportunities for the baseline mission are presented in Table 3-12 (values are given at the start of each daily window; launch azimuth increases throughout the daily window are about 4 deg with correspondingly small variations in the other variables).

Mass statements are given in Table 3-13 for both the Type I and Type II trajectories. Minimum dry mass requirements for the Type II mission is 276 kg (609 lb) as presently configured, however, only 259 kg— (570.5 lb) is available. Use of the Type II would significantly reduce subsystem design flexibility and tend to increase total costs.

TABLE 3-9. 1977 PROBE NOMINAL LAUNCH WINDOW

Launch Date Mo./Day/GMT	Arrival Date Mo./Day/GMT	Parking Orbit Coast Time, Min	Atlas/Centaur Final Burns		Solar Aspect Angle, deg	Flight Time, Days	C_3 $(km/sec)^2$	Launch Azimuth, deg	Approach Asymtote		
			Latitude	Longitude					V_∞-km/sec	Ecliptic Latitude, deg	Ecliptic Longitude From Sun, deg
1/4/0604	May/17/1300	22.9	-12.9	27.9	39.5	133.3	7.754	90.0	4.421	-37.9	134.2
1/5/0555		23.0	-13.1	28.4	37.9	132.3	7.678		4.411	-37.5	134.6
1/6/0546		23.1	-13.2	28.8	36.3	131.3	7.614		4.404	-37.0	134.9
1/7/0537		23.2	-13.3	29.2	34.7	130.3	7.561		4.398	-36.6	135.3
1/8/0529		23.2	-13.4	29.4	33.0	129.3	7.520		4.394	-36.2	135.6
1/9/0521		23.2	-13.5	29.6	31.4	128.3	7.490		4.391	-35.9	135.9
1/10/0514		23.3	-13.5	29.7	29.7	127.3	7.473		4.389	-35.5	136.2
1/11/0506		23.3	-13.6	29.8	28.1	126.3	7.467		4.388	-35.2	136.4
1/12/0459		23.3	-13.6	29.8	26.4	125.3	7.474		4.388	-34.9	136.7
1/13/0452		23.3	-13.6	29.8	24.7	124.3	7.494		4.388	-34.6	136.9
1/14/0445		23.3	-13.6	29.8	23.0	128.3	7.528		4.389	-34.3	137.2
1/15/0488		23.3	-13.5	29.7	21.4	122.3	7.575		4.391	-34.1	137.4
1/16/0431		23.3	-13.5	29.6	19.7	121.3	7.636		4.393	-33.8	137.6
1/17/0424		23.3	-13.5	29.5	18.0	120.3	7.713		4.395	-33.5	137.8

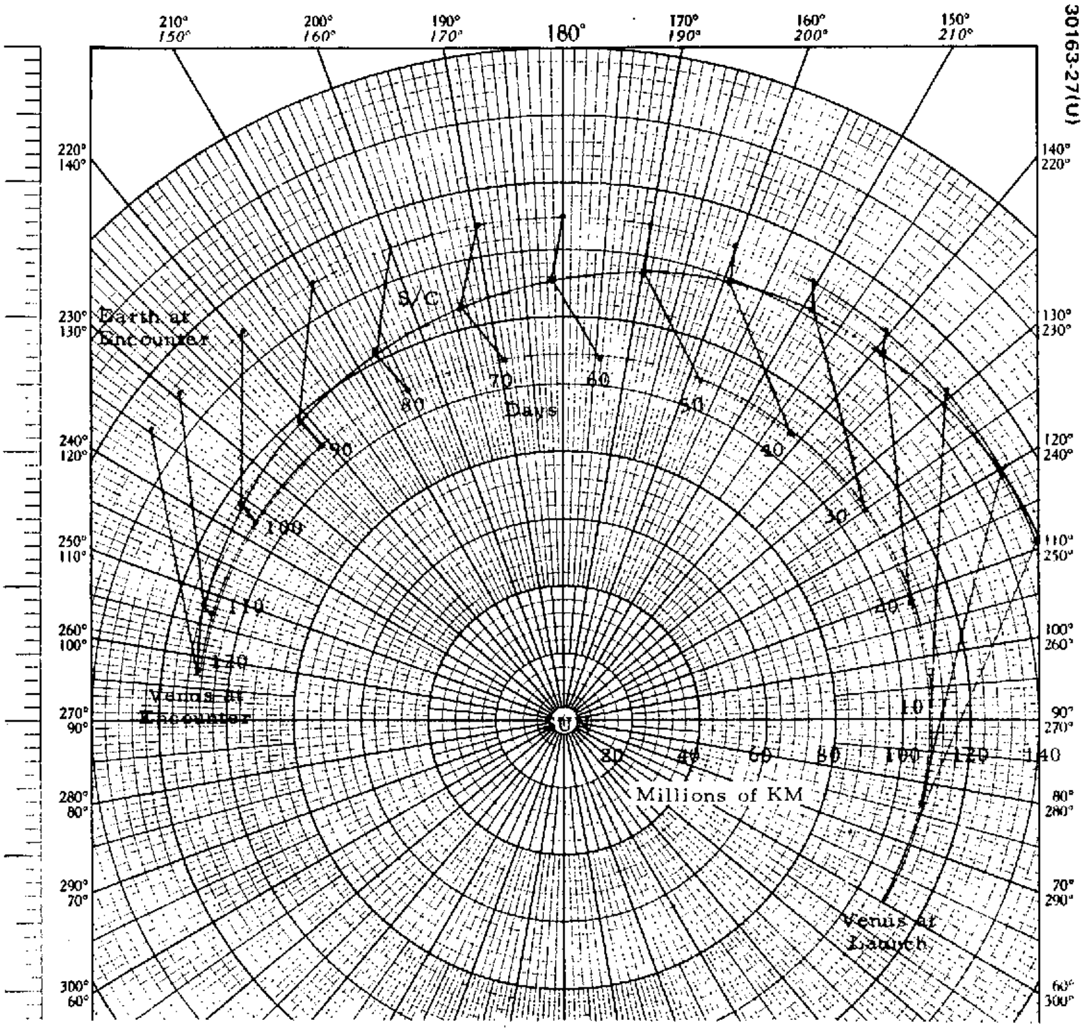


FIGURE 3-6. MULTIPROBE MISSION TRANSIT GEOMETRY

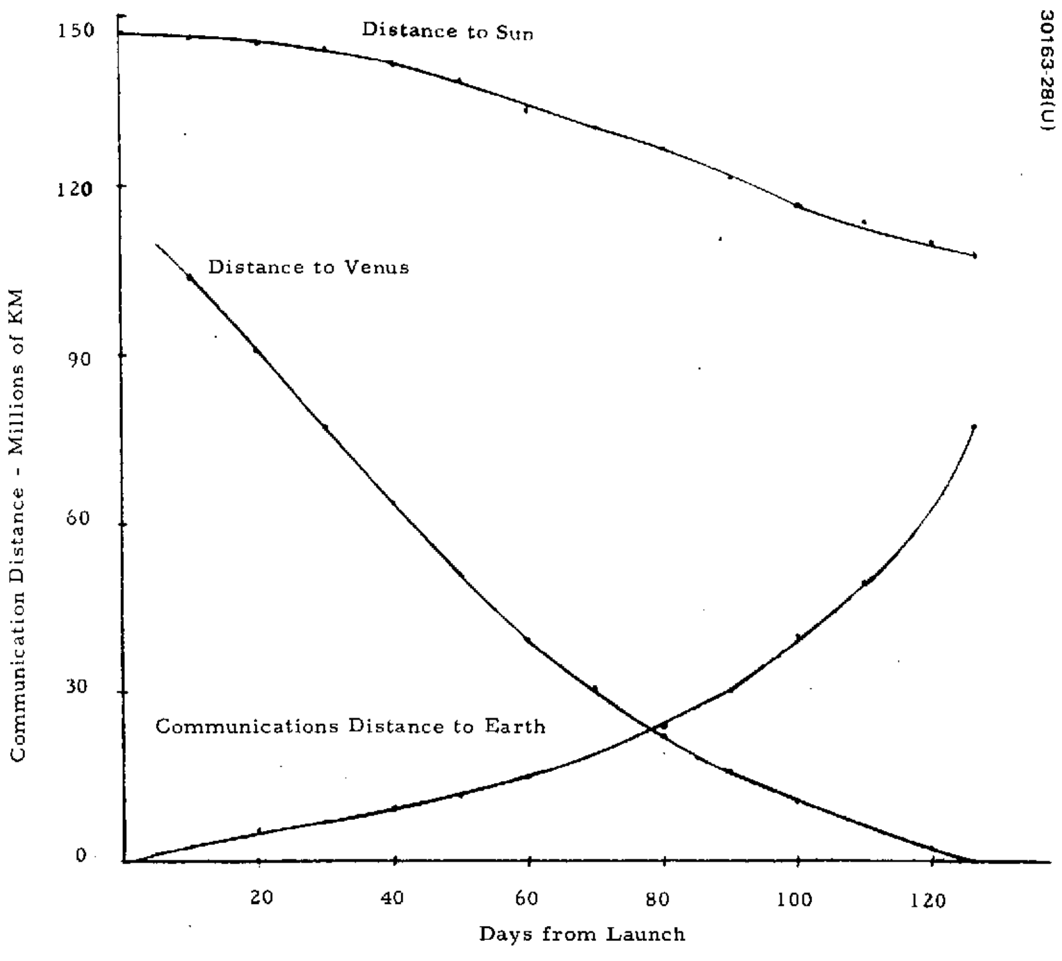


FIGURE 3-7. SPACECRAFT DISTANCE TO SUN, VENUS, AND EARTH

TABLE 3-10. PROBE ΔV REQUIREMENTS

Probe targeting	5.7 m/sec
Bus targeting	16.9 m/sec
Midcourse	12.7 m/sec
Total	34.7 m/sec

TABLE 3-11. TYPE I AND TYPE II ORBITS — SCIENCE CONSIDERATIONS

Orbit Type	Possible Periapsis Locations		Science Considerations
	Nominal Ecliptic Latitude	Nominal Longitude From Subsolar Point	
Type I	5° to 16° N	-44° to -33° (28 to 32 days from evening terminator)	Good for "in situ" measurements of upper atmosphere and solar wind interactions Fair for planetary mapping
Type II	21° to 31° N	-63° to -54° (17 to 22 days from terminator	Good for "in situ" measurements Good for planetary mapping

TABLE 3-12. ATLAS/CENTAUR ORBITER NOMINAL LAUNCH WINDOW
(Type I Trajectory)

Launch Date Mo./Day/GMT	Arrival Date Mo./Day/GMT	Parking Orbit Coast Time, Min	Atlas/Centaur Final Burn		Solar Aspect Angle, deg	Flight Time, days	C_3, $(km/sec)^2$	Launch Azimuth, deg	Approach Asymptote, V_∞-km/sec	Venus Orbit Periapsis Location	
			Latitude	Longitude						Ecliptic Latitude, deg	Ecliptic Longitude From Sun, deg
8/12/0405	12/13/1800	28.5	-21.6	53.6	24.6	123.6	8.605	90.0	5.249	9.4	322.2
8/13/0404	12/13/1800	28.2	-21.6	52.1	23.4	122.6	8.530	90.0	5.231	9.8	322.5
8/14/0402	12/13/1800	27.9	-20.8	50.7	22.2	121.6	8.469	90.0	5.215	10.2	322.7
8/15/0400	12/13/1800	27.6	-20.4	49.7	21.0	120.6	8.421	90.0	5.200	10.6	322.9
8/16/0358	12/13/1800	27.3	-20.0	48.2	19.7	119.6	8.388	90.0	5.186	11.0	323.2
8/17/0419	12/12/1800	25.7	-17.6	40.7	18.5	117.6	8.269	90.0	5.219	13.5	325.2
8/18/0415	12/12/1800	25.5	-17.3	39.9	17.1	116.6	8.275	90.0	5.213	13.9	325.4
8/19/0412	12/12/1800	25.4	-17.1	39.2	15.8	115.6	8.294	90.0	5.207	14.3	325.6
8/20/0408	12/12/1800	25.2	-16.8	38.5	14.5	114.6	8.327	90.0	5.201	14.7	325.8
8/21/0404	12/12/1800	25.1	-16.6	37.9	13.2	113.6	8.374	90.0	5.197	15.0	325.9
8/22/0418	12/11/1800	23.9	-14.7	32.5	12.0	111.6	8.417	90.0	5.270	17.5	327.9
8/23/0414	12/11/1800	23.9	-14.6	32.1	10.6	110.6	8.500	90.0	5.270	17.8	328.0
8/24/0409	12/11/1800	23.8	-14.5	31.7	9.3	109.6	8.599	90.0	5.271	18.1	328.2
8/25/0404	12/11/1800	23.8	-14.4	31.5	7.9	108.6	8.714	90.0	5.271	18.5	328.3

TABLE 3-13. ATLAS/CENTAUR

1978 Orbiter Spacecraft Mass – Kg (lb)

Effect	Type I North Periapsis	Type II North Periapsis
Total spacecraft	730.9 (1611.3)	406.0 (895.0)
Expendables prior to retrofire	6.7 (14.8)	4.9 (10.9)
Orbit insertion expendables (ΔV-km/sec)	326.9 (720.7)	125.8 (277.3)
Initial orbit mass	397.3 (875.8)	275.3 (606.9)
Expendables	21.6 (47.6)	16.3 (36.0)
Propulsion system pressurant	0.2 (0.4)	0.2 (0.4)
Dry orbited mass	375.5 (827.8)	258.8 (570.5)
Orbit insertion motor case	26.7 (58.8)	19.1 (42.1)
Useful orbit mass, kg (lb)	348.8 (769.0)	239.7 (528.4)

The transit geometry of the 1978 orbiter mission is shown in Figure 3-8 with positions of the earth, Venus, and the spacecraft shown throughout the transit phase. Figure 3-9 shows orbiter distances to the two planets and the Sun during the transit phase and Figure 3-10 shows distances to earth over the 225 day orbit period. Table 3-14 gives the characteristics of the baseline orbit selected for the 1978 orbiter mission. Table 3-15 lists the velocity requirements during the Venus orbit.

3.3 MISSION SEQUENCE

Multiprobe Mission Description

The probe spacecraft was originally scheduled for launch from AFETR during a 14-day launch window from 4 January to 17 January 1977 inclusive. Table 3-16 presents an outline of the mission sequence. A daily launch window of 30 min exists between 0424 GMT and 0604 GMT, depending upon launch date. Following launch, a thruster calibration will be performed within 24 hours. It will be followed by the first trajectory correction maneuver (TCM) at 1 to 5 days after launch, which corrects for launch injection errors. Subsequent TCM's will be performed at 20 and 50 days after launch and at 30 days prior to encounter each of which corrects for previous maneuver execution errors. At 20 days prior to encounter, the larger probe and then the small probes will be released. Following small probe release, the bus will be tracked for 2 days to yield an accurate indication of small probe trajectories. At 18 days before entry, the bus will be retargeted to its entry point and retarded in velocity so that its entry occurs after probe descent. Ten days before entry, a small ΔV maneuver, the fifth TCM, will correct for execution errors induced in the bus targeting maneuver. Approximately 3 hours before entry, science will be turned on and allowed to operate until destruction in the Venusian atmosphere. For the original mission entry occurred at 1430 GMT on 17 May 1977 after a 120 to 133 day Type I transit.

A summary of the probe bus operational aspects during the mission is shown in Table 3-17. The various modes of the spacecraft from prelaunch to entry along with duration of each is shown. The data modes and data rates are shown as well as antenna in use and transmitter power output for each phase of the mission. The science equipment in operation is noted along with vehicle spin rate and incident sun angle.

The figures that follow give a graphic display of some of the data presented in Tables 3-16 and 3-17.

Figure 3-11 is a graphic display of spin rate at important events during the mission. As can be seen, the spacecraft is spun up to 25 rpm at separation from the Atlas/Centaur booster and remains at that spin rate throughout the fourth TCM (Trajectory Correction Maneuver). After the fourth TCM, it is despun to 15 rpm in preparation for the large probe release and remains there for the rest of the mission. Entry at 50 to 60 rpm is desired by the fluorescence experiment.

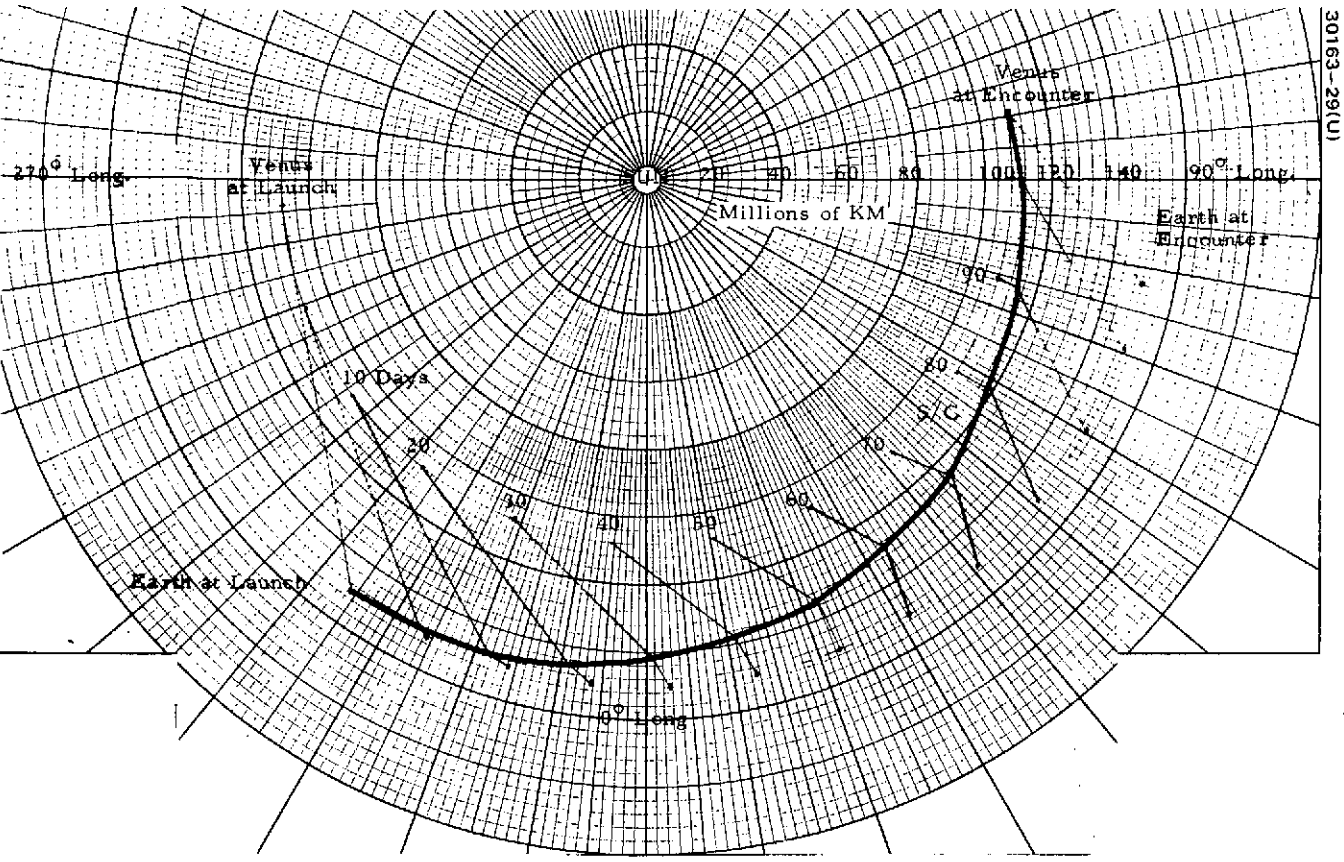


FIGURE 3-8. ATLAS/CENTAUR ORBITER TYPE I TRANSIT GEOMETRY

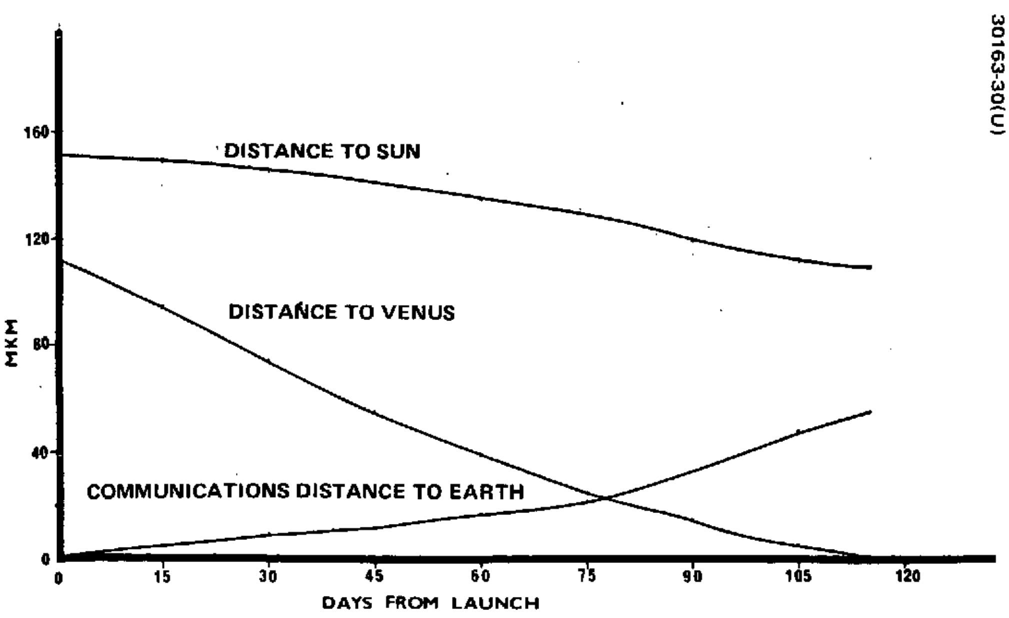


FIGURE 3-9. ATLAS/CENTAUR ORBITER DISTANCES TO SUN, VENUS, AND EARTH

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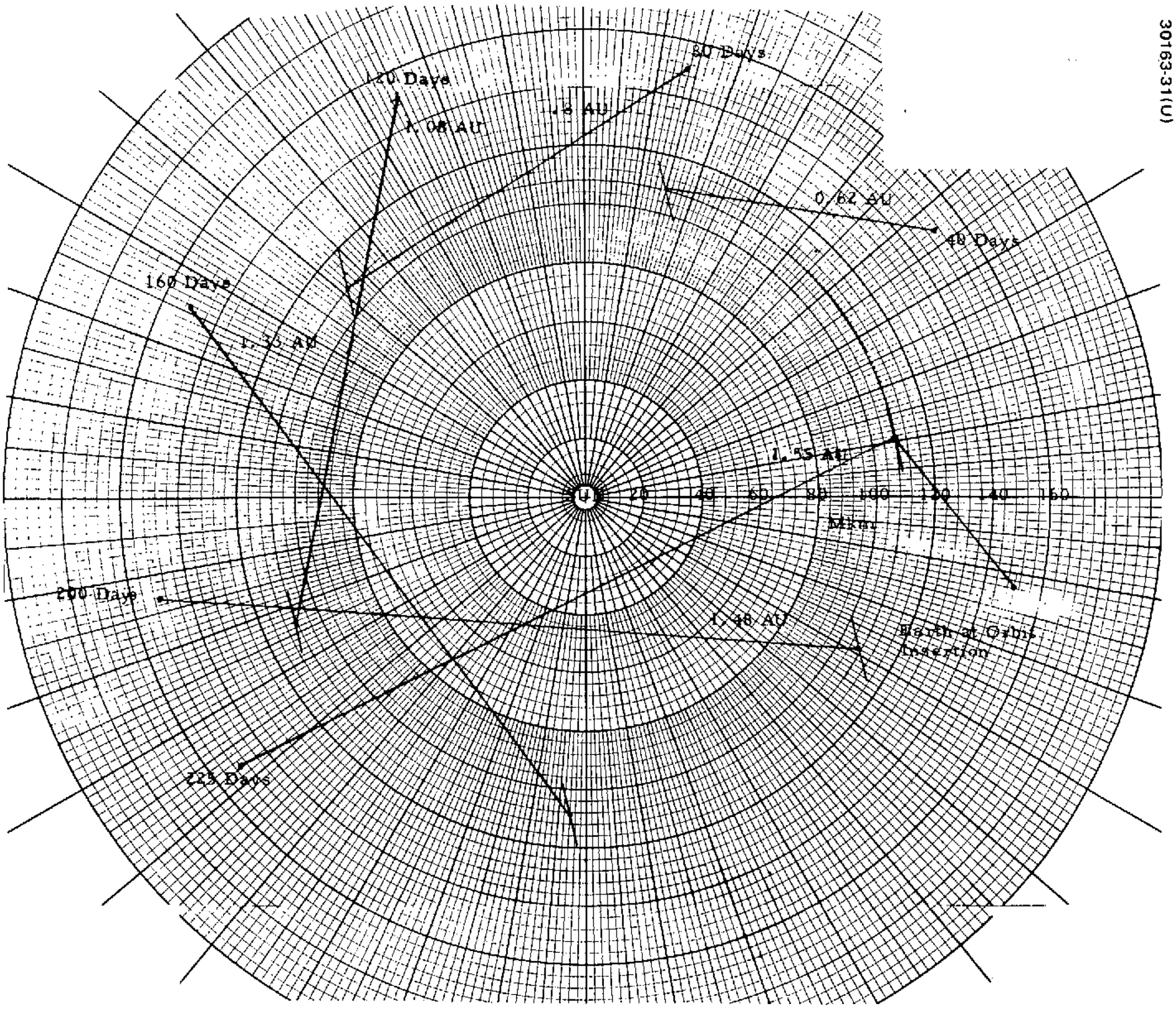

FIGURE 3-10. ORBIT PHASE GEOMETRY

TABLE 3-14. 1978 ORBITER NOMINAL VENUS ORBIT

24 h period
150 km altitude periapsis
Initial periapsis latitude 5 deg to 16 deg N
longitude -44 deg to -33 deg from subsolar
Initial orbit inclination 90 deg to ecliptic
Maximum eclipse duration 4 h

TABLE 3-15. ATLAS/CENTAUR ORBITER VELOCITY REQUIREMENTS

Effect	Type I North Periapsis m/sec	Type II North Periapsis m/sec
Correct initial orbit to 24 h period with 200 km periapsis altitude	35	26
Change periapsis by 200 km (at constant orbit period)	13	13
Atmospheric drag compensation (weekly corrections to minimum altitude)	40	25
Solar perturbations and period control	30	63
In-orbit total	118	127
Midcourse	12	12

TABLE 3-16. PROBE MISSION SEQUENCE OUTLINE
ATLAS/CENTAUR

Date	Time		Event	ΔV Size, m/sec
4-17 Jan 1977	0424-0604 GMT	L + 0	Launch	
		L + 1 day	Jet calibration	
		L + 1-5 days	1st TCM	11.4
		L + 20 days	2nd TCM	0.56
		L + 50 days	3rd TCM	0.10
		E - 30 days	4th TCM	0.042
27 Apr 1977	1430 GMT	E - 20 days	Large probe	
			Release	0.1 axial
	1906 GMT		Small probe	
			Release	5.7 lateral
		E - 18 days	Bus retargeting	16.9
		E - 10 days	5th TCM	1.0
17 May 1977	1300 GMT	E - 0	Probe entry	
	1430 GMT		Bus entry	

TABLE 3-17. PROBE BUS MISSION SUMMARY

Mode	Duration	Data Mode	Max. Data Rate Capability, bps	Antenna	Transmitter Power, W	Science	Spin Rate, rpm	Sun Angle, Deg.
Prelaunch	5 min	Engr	128	Omnis	1	None	0	
Launch to separation	27.3 min	Engr	128	Omnis	1	None	0	0-39.5
Separation to acquisition	2.1 h	Engr	128	Omnis	1	None	25	90
Acquisition to TCM 1	1-5 days	Cruise	2048	Bicone	1	None	25	90
Jet calibration	4.1 h	TCM	2048	Bicone	1	None	25	90
TCM 1	2.05 h	TCM	2048	Omnis	1	None	25	10
Science calibration	15 min	Entry	16 to 2048	Bicone	1/5	All	25	90
TCM 2	47.2 min	TCM	1024	Omnis	1	None	25	90
TCM 3	53.1 min	TCM	256	Bicone	5	None	25	90
TCM 4	33 min	TCM	256	Bicone	5/10	None	25	90
Interplanetary cruise	108 days	Cruise	8 to 2048	Bicone	1, 5, 10	Magnetometer	25	90
Pre-separation probe test	61 min	Engr	256	Bicone	10	Probes	25	90
Large probe separation	2.75 h	TCM/Engr	8	Omnis	10	None	15	134.8
Small probe separation	5.8 h	TCM/Engr	8	Omnis	10	None	57	143.1
Separation targeting cruise	2 days	Cruise	16	Bicone	10	Magnetometer	57	90
Bus targeting	3.05 h	TCM	64	Endfire	5	None	57	38
Final cruise	18 days	Cruise	64	Endfire	5	Magnetometer	57	50-66.1
TCM 5	24.8 min	TCM	64	Endfire	10	None	57	90
Entry	2.5 h	Entry	128	Endfire	10	All	57	113.9

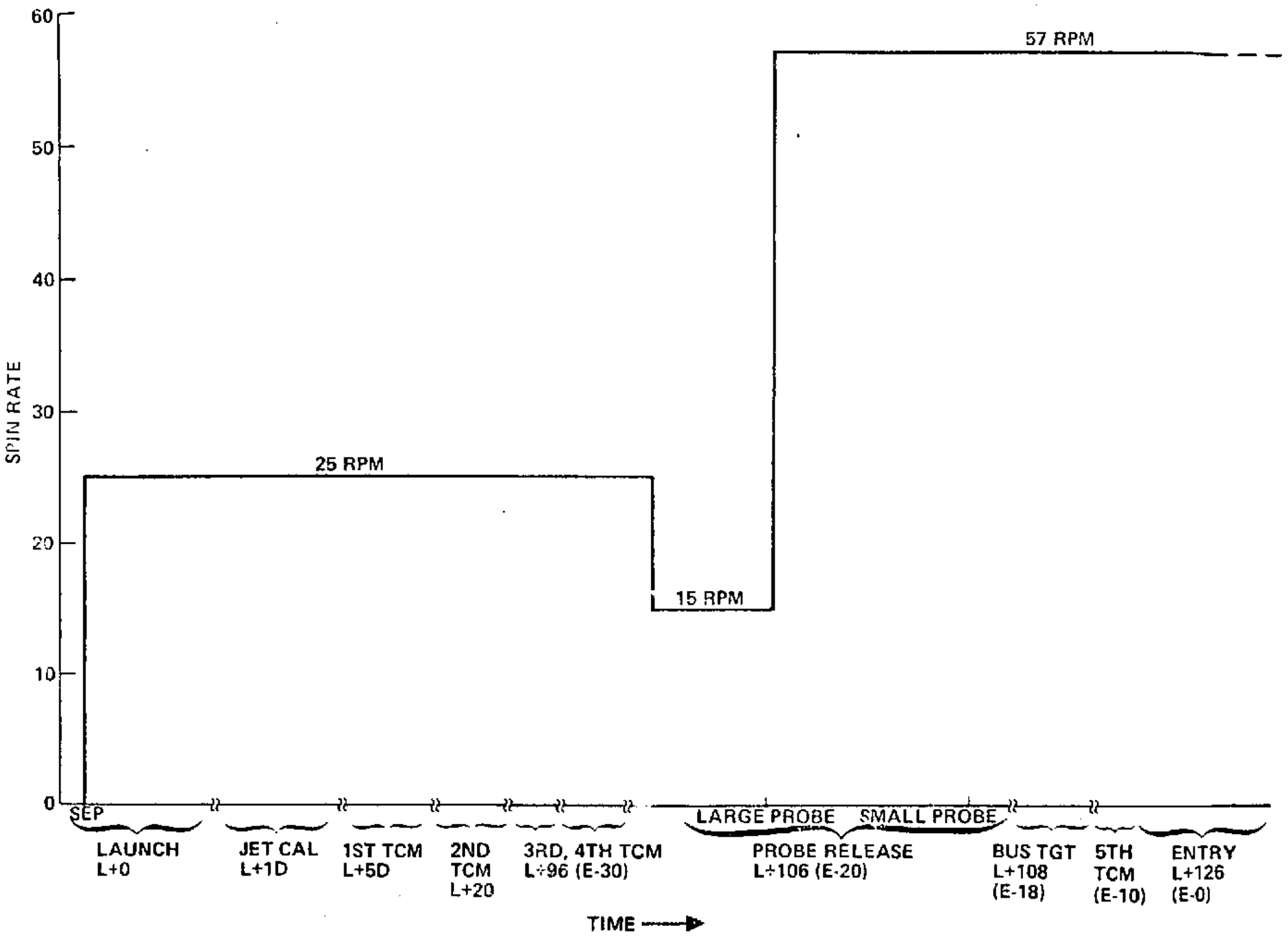

FIGURE 3-11. ATLAS/CENTAUR PROBE SPIN RATE MISSION HISTORY

Figure 3-12 is a display of sun incidence angle at key points in the mission. The solar panel power is assumed to be near zero during the launch phase (about 100 min) and due to the sun angle range at the first TCM, it is assumed to be near zero for 50 min. Other times in the mission that have reduced solar panel power due to sun angle is at probe release and during bus retargeting. At other times in the mission, the sun angle is near 90 deg and solar panel power is high and adequate to supply mission needs with no battery discharge.

Figure 3-13 shows the communication angle from earth at key phases of the mission. On the right hand axis is shown the antenna that is used at the various communication angles encountered. Finally, Figure 3-14 shows the entry timing of the probe bus and four probes. Impact and entry times relative to the large probe are shown.

Orbiter Mission Description

The orbiter spacecraft was originally scheduled to be launched from AFETR during a 14 day launch window from 12 to 25 August 1978 inclusive. Table 3-18 presents an outline of the transit mission sequence. A daily launch window of 30 min exists between 0358 and 0419 GMT, depending upon launch date. Following launch, a thruster calibration will be performed within 24 h. It will be followed by the first TCM at 1 to 5 days after launch which corrects for launch injection errors. Subsequent TCM's will be performed at 20 to 50 days after launch and 20 days prior to encounter. The spacecraft will be reoriented to its orbit insertion attitude at one day prior to encounter. For the original mission orbit insertion was scheduled to occur at 1719 GMT on 11 to 13 December 1978 after a 108 to 123 day Type I transit. It occurs out of view of the earth and is followed by an orbit period adjustment maneuver at the first periapsis and a periapsis altitude adjustment (to 150 km altitude) at the second apoapsis.

A summary of the spacecraft operational aspects during the orbiter mission is shown in Table 3-19. The various modes of the spacecraft from prelaunch to orbit insertion is shown. The data modes and rates are shown as well as antenna usage and transmitter power output for each phase of the mission. The science equipment is turned on for calibration after TCM and again for operation during the long cruise phase. Vehicle spin rate and incident sun angle is noted for the key points in the mission.

The orbiter mission will utilize the 26 m network throught the mission except during TCM's, pre-insertion reorientation and orbit insertion when the 64 m net will be employed. All transit maneuvers, orbit insertion, and periapses will be timed to occur during Goldstone view periods. The sun incidence angle at key points in the mission is shown in Figure 3-15.

The design of the MDA constrains its use to periods when the spacecraft is spinning between 5 and 30 rpm. It will remain despun throughout the mission except during orbit insertion, and during the long eclipses which occur during the on-orbit phase. It will not be possible to utilize the MDA when it is not despun or pointed at earth so the omnis will be used during these periods.

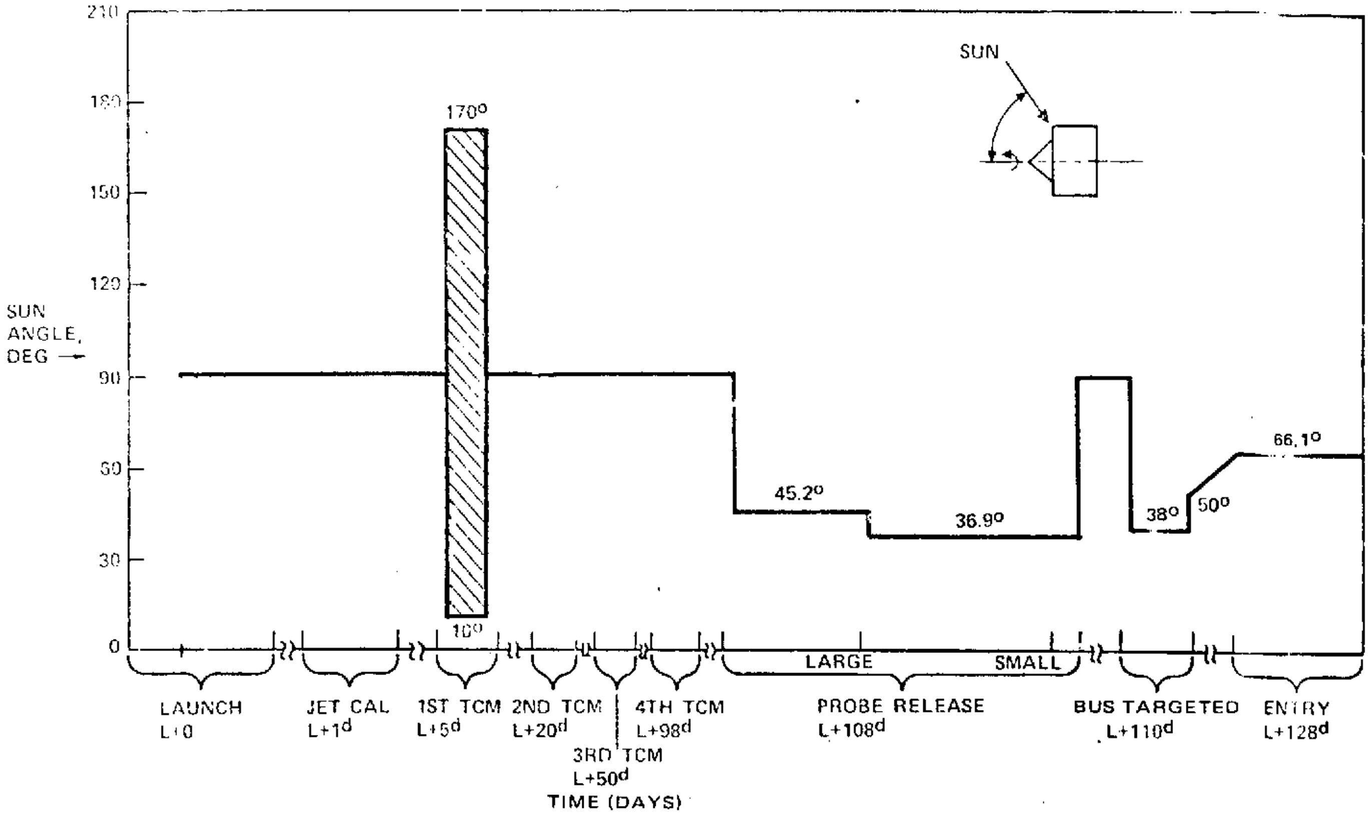

FIGURE 3-12. PROBE BUS SUN ANGLE

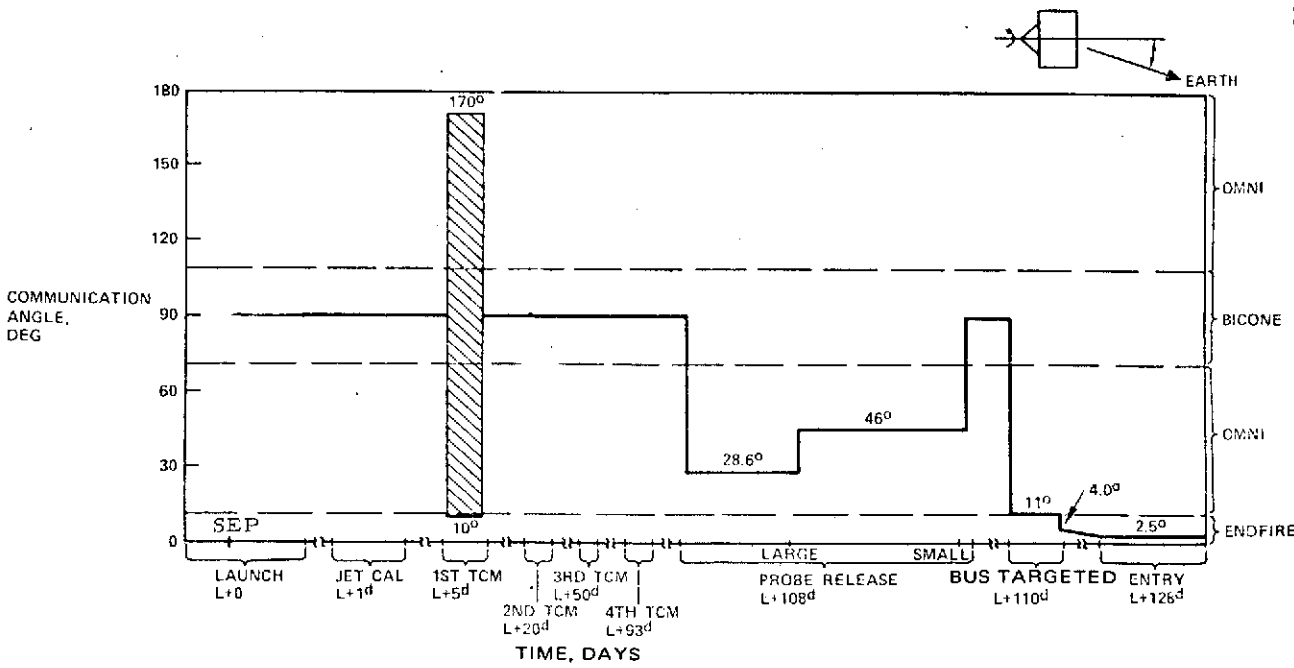

FIGURE 3-13. PROBE BUS COMMUNICATION ANGLE HISTORY

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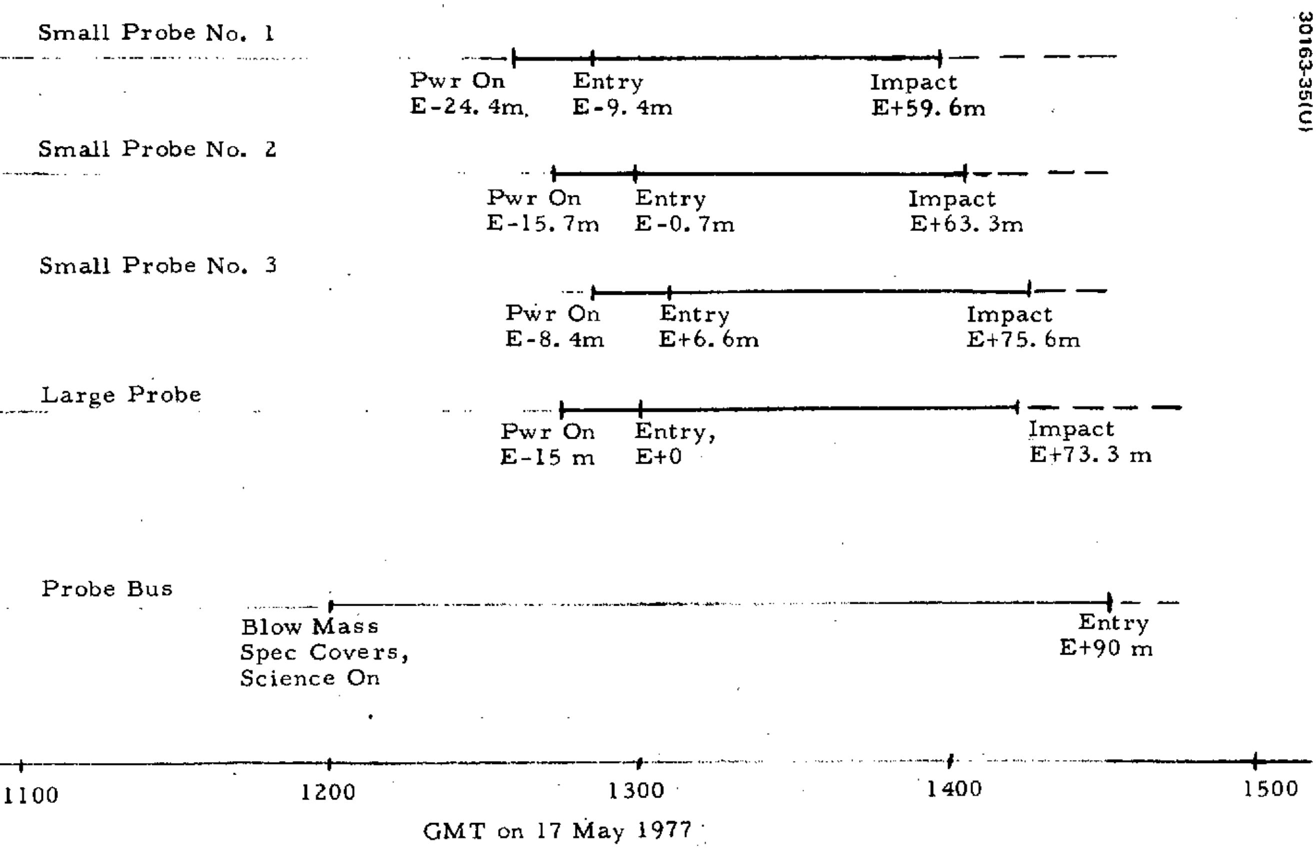


FIGURE 3-14. MULTIPROBE MISSION ENTRY TIMING

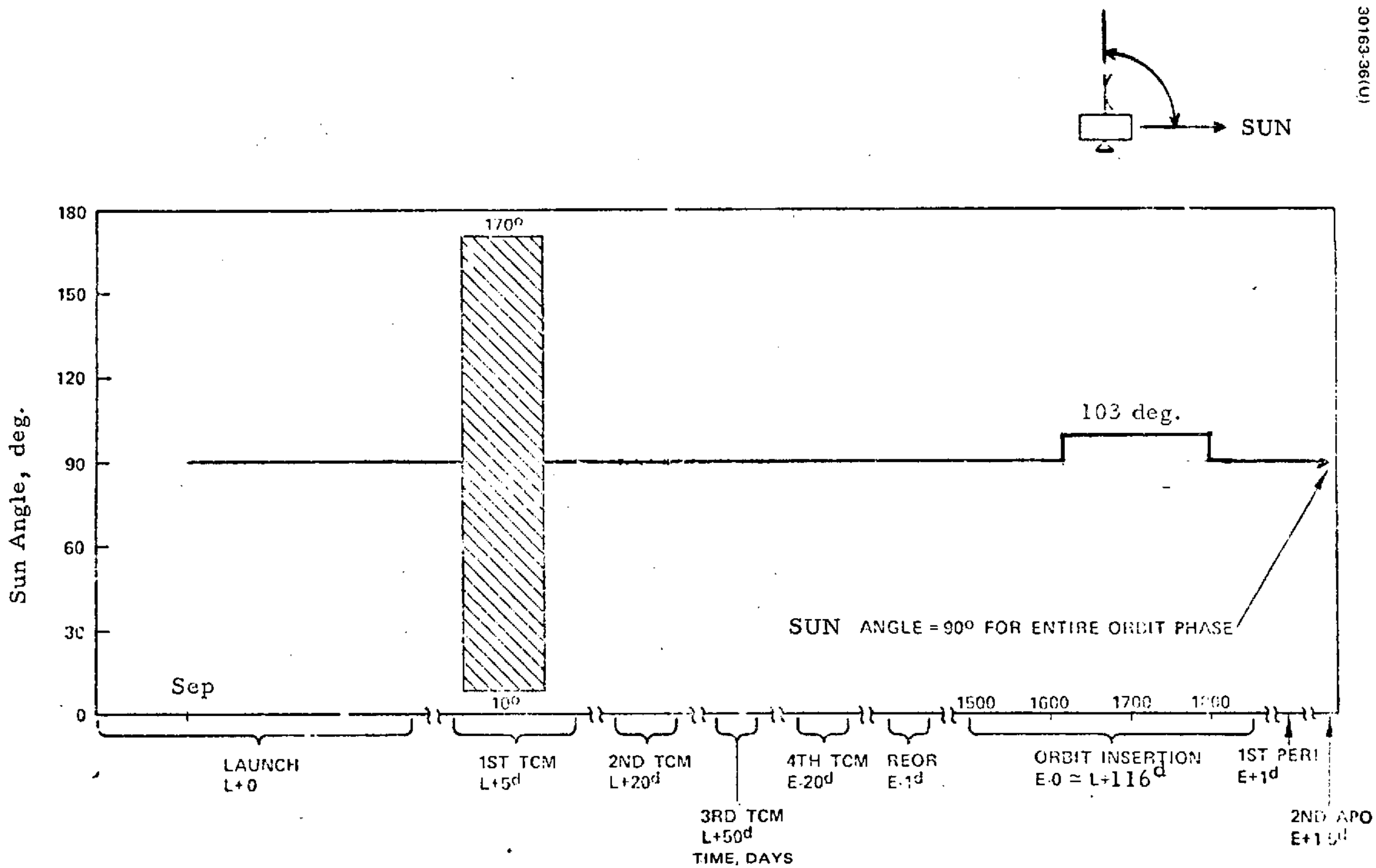


FIGURE 3-15. ATLAS/CENTAUR ORBITER SUN ANGLE HISTORY

TABLE 3-18. ORBITER TRANSIT SEQUENCE OUTLINE

Date	Time		Event	ΔV Size, m/sec
12-25 Aug 1978	0358-0419 GMT	L + 0	Launch	
		L + 1 day	Jet calibration	
		L + 1-5 days	1st TCM	11.4
		L + 20 days	2nd TCM	0.56
		L + 50 days	3rd TCM	0.1
		E - 20 days	4th TCM	0.042
		E - 1 day	Orient to insertion attitude	
11-13 Dec 1978	1719 GMT	E - 0	Orbit insertion	1708
	First periapsis	E + 1 day	Adjust orbit period to 24 hours	35
	Second apoapsis	E + 1.5 days	Lower periapsis to "safe" altitude (200 km)	

TABLE 3-19. ORBITER MISSION SUMMARY – TRANSIT PHASE

Mode	Duration	Data Mode	Data Rate	Antenna	Transmitter Power, W	Science	Spin Rate	Sun Angle Deg.
Prelaunch	5 min	Engr	128	Omnis	1	None	0	--
Launch to separation	27.3 min	Engr	128	Omnis	1	None	0	--
Separation to sun angle	2.2 h	Engr	128	Omnis	1	None	25	90
Acquisition to TCM 1	1 to 5 days	Cruise	32 to 128	Omnis	1	None	25	90
Jet calibration	4.1 h	Engr	32 to 128	Omnis	1	None	25	90 ± 5
TCM 1	2.75 h	TCM	2048	Omnis	1	None	25	90 ± 80
Science calibration	15 min	Periapsis	2048	MDA	1/5	All	25	90
TCM 2	46 min	TCM	256	Omnis	1	None	25	90
TCM 3	49 min	TCM	2048	MDA	1	None	25	90
TCM 4	33.1 min	TCM	2048	MDA	5	None	25	90
Interplanetary cruise	116 to 123 days	Cruise	2048	MDA	5	Cruise	25	90
Pre-insertion reorientation	3.2 h	TCM	2048	Omnis	10	None	25	103
Orbit insertion	5.5 h	Engr	8	Forward omni	10	None	15	103

The forward omni will be used at orbit insertion and will constantly view the earth during the reorientation preceding orbit insertion. Figure 3-16 is a history of communication angles throughout the transit phase.

The initial maneuver insertion is a spinup to 25 rpm. All maneuvers until orbit insertion will be at this spin rate. Figure 3-17 is a history of the spin rate throughout the transit phase with major maneuvers shown. Maneuvers will be executed using the thruster combination that minimizes fuel required and/or execution errors. The first midcourse will employ axial thrusters if possible or radial thrusters if the required ΔV is within 10 deg of the sun line. During the orbit phase, orbit keeping maneuvers will employ axial thrusters only and will be executed where the orbit becomes tangent to the ecliptic normal. Both maneuvers will be executed during the same orbit with the periapsis maneuver (to correct orbit period) executed first. Orbit trim maneuvers will be performed typically once per week.

Attitude and spin rate determinations and touchup will be performed daily during the first 50 days of the transit phase, weekly thereafter until orbit insertion, and daily during the orbit phase. Additionally, they will be performed as required near maneuver events.

The magnetometer, spectrometer, and solar wind probe will be turned on immediately after the first midcourse and operated continuously except during transit phase maneuvers and apoapsis eclipses. Operation of the science equipment at key orbital mission phases is noted on Table 3-20. The duration of each orbital phase is shown as well as altitude, data modes, and rates. A graphic display of orbital operations is shown in Figure 3-18. Most of the key modes listed on Table 3-20 are shown.

3.4 DESCENT PROFILE/OPERATIONS

Science data gathering requirements imposed on the probes are interpreted as data points in a given altitude range, or bits of data per kilometer. Therefore, the required data rate in bits per second varies with velocity. A number of parameters that affect velocity (and thus data rate) can be traded off to arrive at an optimum descent profile. For the large probe, the major parameters are pressure vessel aerodynamic configuration (area and drag coefficient), parachute size, deployment altitude, and jettison altitude. For the small probe, this reduces to only the aerodynamic configuration. In the Thor/Delta spacecraft configuration, the prime driving function for these tradeoffs is low weight whereas in the Atlas/Centaur spacecraft configuration, low cost is of greater importance.

Large Probe

Table 3-21 gives a summary of the design rationale for the large probe descent profile. The major changes from the Thor/Delta approach are given along with reasons for the change. The parachute diameter for the Thor/Delta is selected to give a minimum weight design. The 4.57 m (15 ft) Atlas/Centaur chute provides increased margin for aeroshell separation. This increased margin results in decreased development risk.

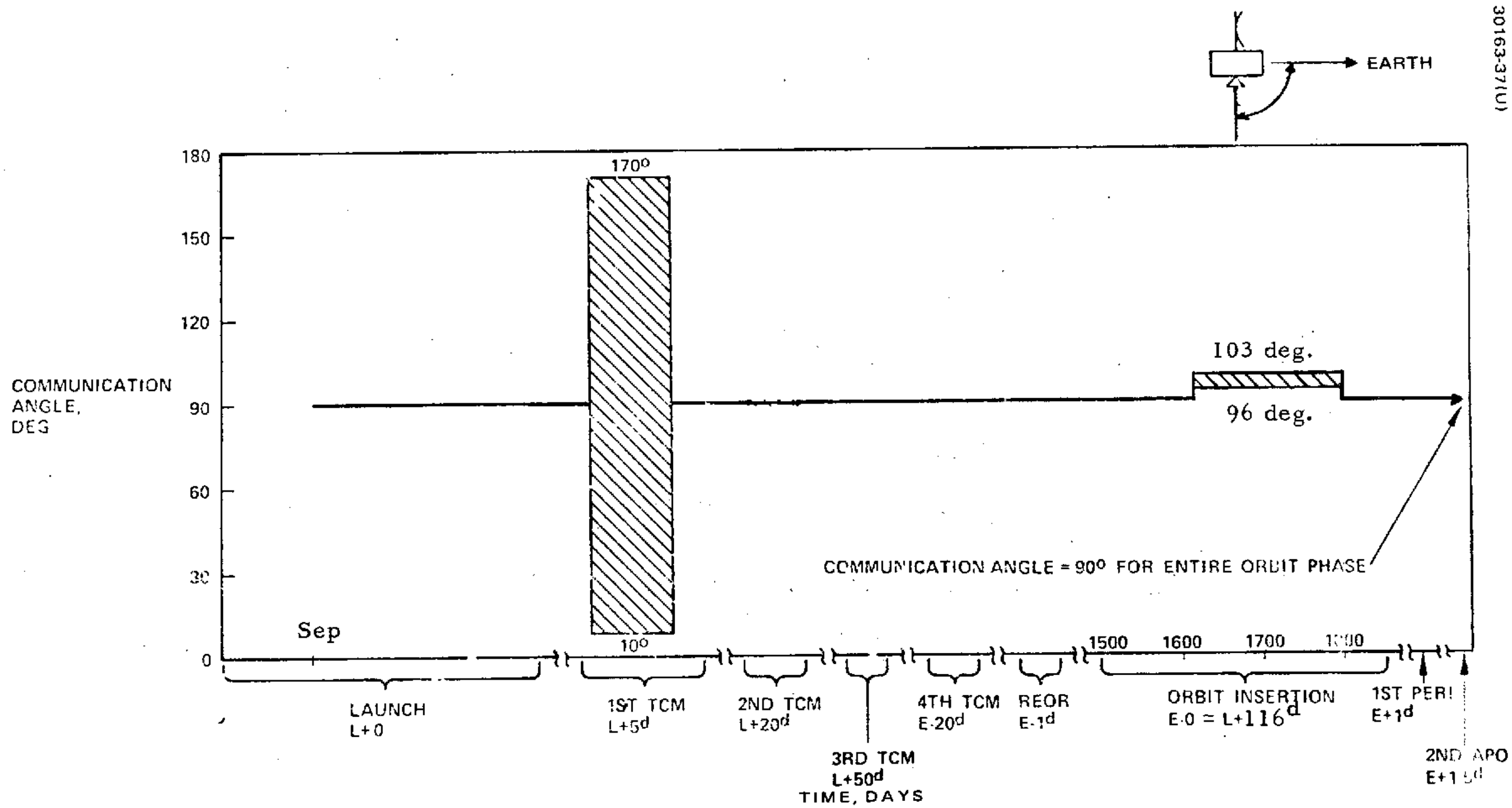


FIGURE 3-16. ATLAS/CENTAUR ORBITER COMMUNICATION ANGLE HISTORY

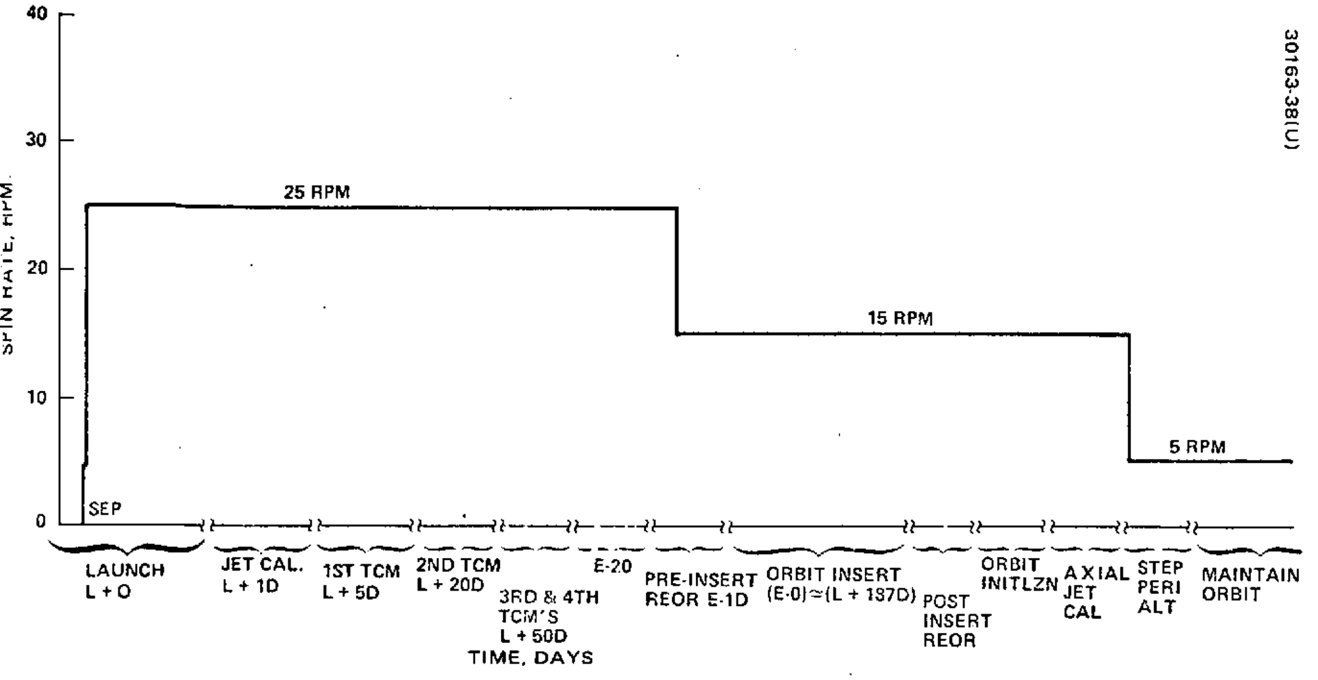


FIGURE 3-17. ATLAS/CENTAUR ORBITER SPIN RATE HISTORY

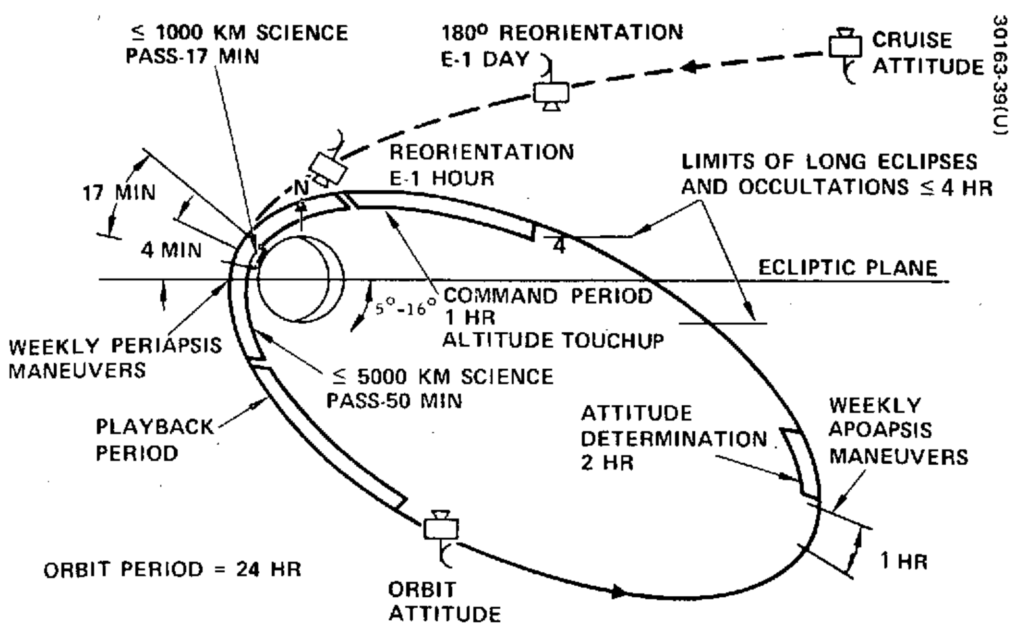


FIGURE 3-18. ORBIT OPERATIONS

TABLE 3-20. ORBITER MISSION SUMMARY – ORBIT PHASE

Mode	Duration Per Orbit	Attitude, km	Data Mode	Data Rate (Transmit), bps	Data Rate (Record), bps	Science On
Apoapsis	23 h	>5,000	Apoapsis	64 to 1024	0	Mag., uv spec., solar wind
Periapsis 1	33.2 min	1000 to 5000	Periapsis	64 to 1024	0	Apoapsis and ETP, ion spec., ir radar
Periapsis 2	8.4 min	400 to 1000	Periapsis	64 to 1024	5	All (radar altimeter recorded)
Periapsis 3	8.4 min	150 to 400	Periapsis	64 to 1024	300	All (radar altimeter recorded)
Playback	51 min	>5,000	Playback	64 to 1024	0	Apoapsis
Periapsis eclipse	≤30 min	Same as periapsis 1, 2, and 3				
Apoapsis eclipse	≤4 h	>10,000		0		Solar wind
Periapsis occultation 1	≤8.4 min	400 to 1000	Periapsis	0	133	All (rf occultation)
Periapsis occultation 2	≤8.4 min	150 to 400	Periapsis	0	428	All (rf occultation)
Orbit period keeping	1 min	1,350	Periapsis	64 to 1024	0	Periapsis 1
Periapsis altitude keeping	1 min	66,000	TCM	64 to 1024	0	Apoapsis
Attitude keeping	2 h	>10,000	TCM	64 to 1024	0	Apoapsis

TABLE 3-21. LARGE PROBE DESCENT PROFILE DESIGN RATIONALE

	Thor/Delta	Atlas Centaur	Rationale For Change
Chute diameter, D_o m(ft)	3.49 (11.45)	4.57 (15)	Increased margin for aeroshell separation
Chute jettison altitude, km	55	40	Increased sampling of lower cloud layers
Aerodynamic configuration	Perforated ring	Perforated ring	Unchanged - lowest cost approach

TABLE 3-22. LARGE PROBE DESCENT PROFILE

Profile Summary	Thor/Delta	Atlas/Centaur
Ballistic coefficient (kg/m^2)		
On parachute	16.3	13.4
Off parachute	363.3	417.0
Descent time, min	54.2	73.3
Time on chute, min	9.6	36.3
Jettison altitude, km	55.0	40.0
Design velocity, m/sec	72.6	36.9
Transmitter rf power, w	12.0	12.0
Data rate, bps	276/184	184.0
Margin at design velocity, bps	10.0	8.0

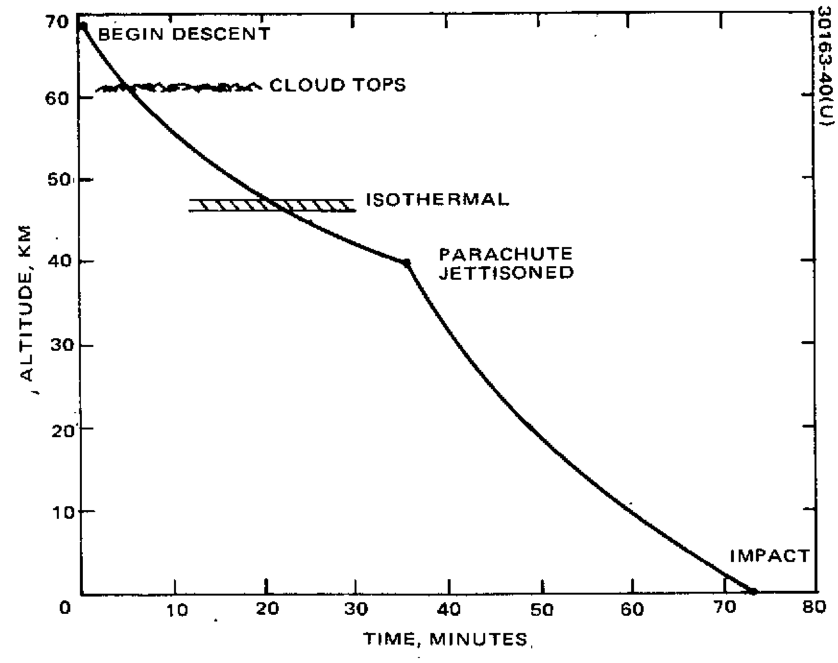


FIGURE 3-19. ATLAS/CENTAUR LARGE PROBE DESCENT PROFILE

The chute jettison altitude of 55 km for the Thor/Delta selected to minimize system weight can be lowered to 40 km for the Atlas/Centaur design. This altitude reduction results in a slower velocity and lower data rate requirement. Only one data rate was originally required for the entire descent, whereas the Thor/Delta required two rates--one for the initial high velocity phase, and a lower one for below 20 km operation. In addition, the science return is improved for the 40 to 55 km lower cloud altitudes.

The pressure vessel basic design remains unchanged between the Thor/Delta and Atlas/Centaur configurations. It consists of a perforated ring and appears to be the lowest cost approach of those considered.

A summary of the large probe descent profile is shown on Table 3-22. The longer descent time of the Atlas/Centaur results in a larger power requirement and thus larger batteries. The Atlas/Centaur configuration can meet the science requirement with a single data rate of 184. In both the Thor/Delta and Atlas/Centaur cases, 12 w transmitter output at the antenna supports an adequate data rate at design velocity. Figure 3-19 is a graphic display of the descent altitude as a function of time. The major points in the profile such as cloud top region, isothermal region and parachute jettison altitude are noted for reference.

Table 3-23 summarizes the large probe descent sequence starting with separation from the bus and ending with impact on the surface of Venus. Separation occurs at 20 days prior to the encounter with Venus. At this time, a 10 min checkout by Mission Control is accomplished to verify probe performance subsequent to separation. After the checkout, all equipment is turned off except for the data subsystem timer to conserve power for the long cruise phase. For the next 18 days, the probe operates on battery power in the powered down mode. Two days prior to encounter, the planetary flux heater is turned on by the data timer and thus initiates the Pre-Entry I phase. In the time period E-15 to E-5 min, various equipments are turned on and data rates/formats selected as initiated by the timer in preparation for the entry into the upper atmosphere and high-g deceleration. The timer error at this time could have built up to a maximum of 19.9 sec over the 20 day cruise period. At E + 5 sec, a signal from the accelerometer causes data rate/format selection. Events happen in rapid succession, as noted in Table 3-23 over the next 22 sec, culminating with reinitialization of the timer to remove errors and chute deployment at E +25 sec and beginning of the pressure vessel stabilized descent at E + 29 sec and approximately 66 km altitude. All experiments are turned on at this time and are gathering data.

At an altitude of 40 km and 36.1 min after entry, the chute is jettisoned and the final free-fall to the planet surface begins. Jettison of four window covers occurs at this time. At E + 73.3 min, impact occurs and the probe is powered down as shown (if still operational). The various formats used during the descent phase are shown in Figure 3-20.

30163-41(U)

FORMAT I

2W	30W
Sync + ID	Accelerometer Data

FORMAT II

2W	17W	9W	4W
Sync + ID	Shock Layer Rad	Accelerometer	Engr

FORMAT III

3Words	2W	22W	41W	17W	5W	3W	2W	33W
Sync + ID	Temp S	Mass Spect	Cloud Part	AU Ext	Nep	Hygro	Play Back	Sub-Com

FORMAT III SUBCOMMUTATOR

1W	1W	4W	4W	3W	20W	20W	2W	19W	2W	2W	2W	6W	33W	10W	3W
Sync ID	Temp	ACC	Press S	Tur	Solar Flux	Planet Flux	Trans	Engr	Nep	Mass Spect	Play Back	Sci House	Cloud Part	Aur Ext	Nep

POSSIBLE ALTERNATE SUBCOMMUTATOR (WOULD PROVIDE SAME MASS SPECTROMETER DATA RATE AS ABOVE SUBCOMMUTATOR IF DATA RATE WERE REDUCED FROM 276 TO 184 bps)

	44 W
SAME	Mass Spect

1 WORD = 10 BITS

FIGURE 3-20. LARGE PROBE DATA FORMATS

TABLE 3-23. LARGE PROBE DESCENT SEQUENCE

Subsequence	Time	Commands	Initiated By	Format	Data Rate BPS
Timer initiation	E-20 days	Initiate timer	Bus command	- -	- -
Post separation	E-20 days	Engineering electronics On RF On	Separation	- -	- -
Cruise	+10 min	RF Off Engineering electronics Off	Timer	- -	- -
Pre-entry I	E-2 days	Engineering electronics On Planet flux heater On Engineering electronics Off	Timer	- -	- -
Pre-entry II	E-15 min	Engineering electronics On RF On Sensors On	Timer	- -	- -
Pre-entry III	E-5 min	Level II set Deceleration module On Format I select Data rate I select	Timer	I	184
Blackout I	E+5 sec	Format II select Data rate II select Re-initiate timer	0.5 g	II	361.6 (recorded)
Blackout II	E+6 sec	Arm mortar Re-initiate timer (backup)	3 g	II	361.6
Post blackout	E+15 sec	Format I select Data rate III select Deceleration module Off	Timer	I	40 (recorded)
Timer initiation	E+21 sec	Re-initiate timer	3 g	I	40
Chute deployment (67.3 km)	E+25 sec	PCU On Fire chute deploy squibs	Timer	I	40
Interface disconnect	E+25.5 sec	Fire IFD squibs	Timer	I	40
Aeroshell jettison	E+28.3 sec	Fire jettison squibs	Timer	I	40
Descent (60 km)	E+29 sec	Format III select Data rate IV select Level II set Science On Window heater On Fire breakoff hat PCU Off	Timer	III	184
Mass spectrometer events	10 events	PCU On Fire inlet valve squib PCU Off 3 cmds per event	Mass spectrometer	III	184
Chute jettison	E+36.1 min	PCU On Fire chute jettison squibs PCU Off Step-up mass spectrometer power	0.5 atm	IV	184
Impact	E+73.3 min	Format I select Window heater Off Planet flux heater Off Science Off	Timer	I	184

Small Probe

The summary of the small probe descent profile is shown on Table 3-24. The slightly shorter descent time compared to Thor/Delta spacecraft reduces power requirements for the Atlas/Centaur spacecraft; however, greater power usage by subsystem equipment still requires larger batteries than the Thor/Delta design. Since no parachute is employed on the small probe, the descent velocity is determined by mass and aerodynamic considerations resulting in a slightly faster velocity for the Atlas/Centaur probe. This higher velocity coupled with the same power and data rate results in a slightly lower but adequate data rate margin for the Atlas/Centaur design. Figure 3-21 is a representation of the descent profile with major point noted for reference.

Table 3-25 is a tabulation of the descent sequence for the small probe. It is similar to the large probe with simplification resulting from fewer experiments and operational needs. Figure 3-22 presents the data formats for the small probe.

TABLE 3-24. SMALL PROBE DESCENT PROFILE

Profile Summary	Thor/Delta	Atlas/Centaur
Ballistic coefficient (kg/m^2)	142.2	153.7
Descent time, min	74.9	69.0
Time on chute, min	-	-
Jettison altitude, km	-	-
Design velocity, m/sec	141.8	154.0
Transmitter rf power, w	6.5	6.5
Data rate, bps	16.0	16.0
Margin at design velocity, bps	8.0	7.0

TABLE 3-25. SMALL PROBE DESCENT SEQUENCE

Subsequence	Time	Commands	Initiated By	Format	Data Rate bps
Timer Initiation	E-23 days	Initiate timer	Bus Command	—	—
Magnetic and RF Calibration	E-20 days	Engineering electronics on RF on Science on Format I select	Separation	I	16
Cruise	+10 min	Science off RF off Engineering electronics off	Timer	—	—
Pre-Entry I	E-45 min	Engineering electronics on	Timer	—	—
Pre-Entry II	E-15 min	RF on Science on PCV on Fire despin thrusters PCV off	Timer	—	—
Pre-Entry III	E-2 min 20 sec	Format II select RF off	Timer	II	16 (recorded)
Descent	E+19.5 to E+30.5 sec	RF on Format I select Window heater on PCV on Fire temp probe/neph. cover squib PCV off Reinitiate timer	2.0 g 65.3 km, $\gamma_e = 78.5$ 69.5 km, $\gamma_e = 28.5$	I	16
Impact	E+74.2 to E+74.9 min	Window heater off Format II select	Timer	II	16

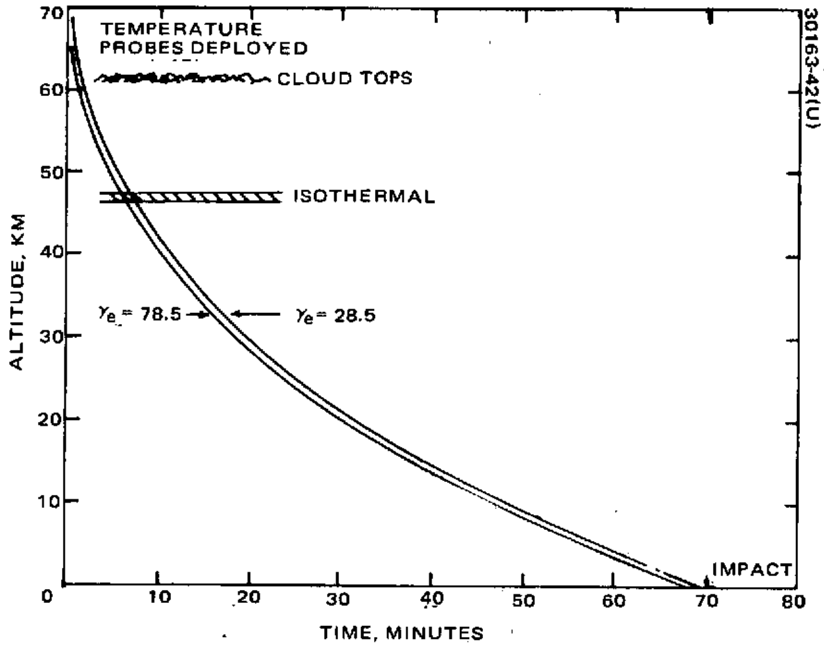


FIGURE 3-21. ATLAS/CENTAUR SMALL PROBE DESCENT PROFILE

30163-43(U)

FORMAT II

2W	30W
Sync + ID	Accelerometer Data

FORMAT I

2W	9W	5W	4W	3W	3W	2W	2W	6W	9W
Sync + ID	Nep	Accel	Magn	Temp	Press	Osc	Play Back	Engr	Spares

1 WORD = 10 BITS

FIGURE 3-22. SMALL PROBE DATA FORMATS

REFERENCES

3-1. "Task No. LV-2, Atlas/Centaur Mission Analysis," HS-507A-002-10, 19 March 1973.

3-2. "Task No. LV-3, Nominal Mission Profiles," HS-507A-0022-11, 6 April 1973.

3-3 "Pioneer Venus," report of a study by the Science Steering Group, Ames Research Center, June 1972.

3-4 "Task No. MS-3, Mission Launch Dates," HS-507-0022-103, 2 February 1973.

3-5 "Task No. MS-5, Nominal Probe Target Locations," HS-507-0022-97, 5 February 1973.

3-6 Pioneer Venus Final Report, Volume 3, "System Analysis and Requirements."

3-7 "Task No. MS-25, 1978 Orbiter Transit Trajectory Selection," HS-507-0022-131, 12 March 1973.

4. SPACECRAFT DESCRIPTIONS

An overall description of the probe bus, probes and orbiter spacecraft is contained in this subsection of the report. These descriptions are primarily oriented toward the Atlas/Centaur configurations; however, comparisons are made to the Thor/Delta versions in sufficient depth to enable an understanding of the major differences between the spacecraft designs for the two launch vehicles.

A summary of the probe bus spacecraft is contained in Subsection 4.1, giving the major characteristics and performance of the bus. This is followed by a summary description of the configuration augmented as needed by layout drawings and exploded views to show internal details. A shelf layout is shown to illustrate some of the hardware mounting provisions. A final subsection gives a functional block diagram of the system with a description of the major functions of each portion. This is concluded with a summary of electrical power requirements and spacecraft reliability. Further details of the reliability analysis may be found in Reference 4-1.

Similar presentations for the large and small probes, and orbiter spacecraft are contained in Subsections 4.1 and 4.2, respectively.

The final subsection, 4.3, gives a summary of the primary considerations for selecting the baseline configurations for the probe bus and orbiter spacecraft.

4.1 PROBE SPACECRAFT

The probe spacecraft consists of the probe bus, one large probe, three small probes, and a group of science experiments. Descriptions of the bus, the large probe, and the small probes are given in the paragraphs that follow in this subsection. The accommodation of the experiments by the probe bus and probes will be covered in Subsection 5.7.

Probe Bus

The probe bus is a spin stabilized vehicle using sun and star sensors for attitude reference. It is a basic bus providing a platform for the science experiments and the probes. It is designed to accommodate the experiment payload of 12.6 kg (27.7 lb.) and one large and three small probes weighing a

TABLE 4-1. PROBE BUS CHARACTERISTICS AND PERFORMANCE

	Thor Delta Spacecraft	Atlas Centaur Spacecraft	Key Cost Savings Differences
General			
Overall length, cm (in.)	241.3 (95)	294.6 (116)	153 versus 22.6 kg contingency
Diameter, cm (in.)	213 (84)	254 (100)	Al versus Be structure
Usable spacecraft mass - dry, kg (lb)	363 (801)	730 (1610)	4.27 versus 2.88 m^2 shelf
Science, kg (lb)	11.6 (25.6)	12.6 (27.7)	
Probe mass, kg (lb)	227.1 (477.3)	367.8 (811)	10 versus 6 louvers
Equipment shelf area, m^2 (ft^2)	2.88 (31)	4.27 (46)	
Power			
Solar panel area, m^2 (ft^2)	3.07 (33)	3.48 (37.5)	Battery common to orbiter
Peak power, W	145	170	
Science, W	15.9	24.5	No battery boost
Ampere-hours	(Ag-Zn) 13.6	(Ni-cad) 7 (2)	
Radio			
Transmitter/receiver	S band	→	21 versus 28 percent FPA efficiency
RF power, W	1, 5, 10	→	
3 dB bicone	→	→	
Two omnis	→	→	
18 dB horn	→	→	
Command and data handling			No new LSI versus 10 new
Modulation - command	PSK/PM	→	
Bit rate, bps - command	1, 2, 4	→	
Data rates, bps - telemetry	8-2048	→	
Encoding	Convolutional	→	
Modulation	PCM/PSK/PM	→	
Attitude control/mechanisms	→	→	
Spin axis ⊥, ecliptic			Less magnetic
Spin rates, rpm	15 - 100	15 - 57	Cleanliness
Star sensor and sun sensors	Solid state	→	
Spin axis determination accuracy, deg.	0.9	→	Less deployment mechanism
Magnetometer boom length, m (ft)	1.07 (3.5)	4.4 (14.5)	
Propulsion			
Blowdown hydrazine	→	→	None (more expensive)
Six 22.2 N (5 lb) thrusters	→	→	
Tanks, cm^3 (in^3)	18,026 (1100)(2)	37,690 (2300)(2)	

total of 368 kg (811 lb). Its design employs maximum use of already developed hardware and commonality with the orbiter design to reduce total program costs.

Characteristics and Performance

Table 4-1 provides a description of the major characteristics and performance of the Atlas/Centaur probe bus design compared to the Thor/Delta. The table highlights those items providing cost savings for the Atlas/Centaur design. The usable spacecraft weight is about double that of the Thor/Delta version allowing the use of liberal allowances in weight and volume for the spacecraft hardware and still maintaining a 153 kg (337 lb) contingency. The Atlas/Centaur spacecraft employs aluminum throughout its design, whereas beryllium is used by the Thor/Delta design in selected areas. The larger diameter of the Atlas/Centaur allows usage of a larger equipment shelf. This in turn promotes easier hardware integration and use of more thermal louvers as needed for thermal control margin. The active thermal control by use of the louvers is simplified by maintaining the spin axis normal to the sun during the cruise phase except for relatively short periods during maneuvers. Large thermal loads are placed near louvers and heaters used as necessary for temperature control.

The power subsystem has 3.48 m^2 (37.5 ft^2) of solar panel which produces 170 W near Venus and 90 W near earth. Thus, additional power is provided over the Thor/Delta design to enable use of less efficient spacecraft hardware (hence lower cost). The additional weight capability of the Atlas/Centaur allows use of nickel cadmium batteries based on an existing design. These batteries provide a design that is common to the orbiter and their higher output voltage eliminates the need for battery boost on discharge. Electrical interconnections between the bus/probes allow charging of the probe batteries, telemetry monitoring, and other necessary functions.

The rf subsystem operates at S-band frequencies with two-way doppler capability for navigation purposes. RF power outputs are 1, 5, or 10 W, depending on the link requirements. A lower efficiency power amplifier is used on the Atlas/Centaur design to save cost in the development phase. The 10 W mode will be used on bus entry along with the 18 dB horn for maximum data capability. Omnidirectional antennas provide 4π steradian command coverage for the entire mission. The 3 dB gain bicone is used to transmit telemetry during cruise.

Real time ground command capability is available using PSK/PM modulation at a 1, 2, or 4 bps rate. This is supplemented during the launch phase by onboard command storage. Nine telemetry formats will be used to provide capability for the varying experiment complements at various phases of the mission. Convolution encoding is employed with PCM/PSK/PM modulation. Data rates vary from 8 to 2048 bps as necessary to support mission needs. Less program cost risk is incurred by the more conservative design approach used. The Thor/Delta spacecraft design uses 10 new LSI elements, whereas no new ones are required for the Atlas/Centaur design.

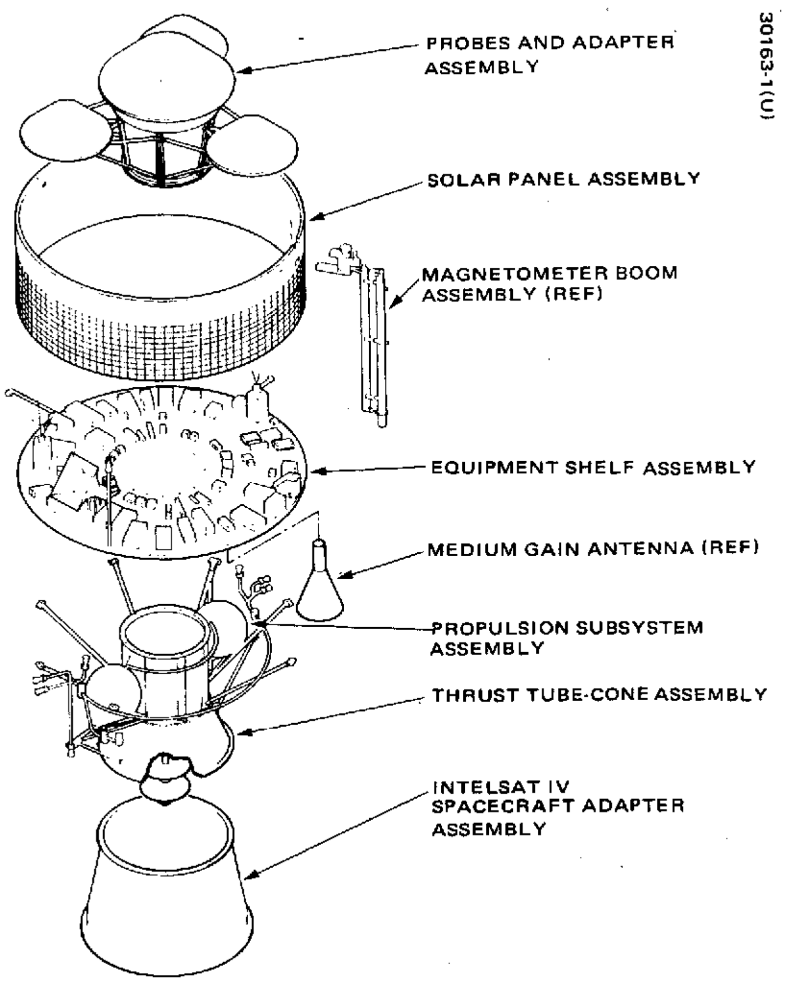


FIGURE 4-1. PROBE BUS SPACECRAFT

The spacecraft is spin stabilized with the spin axis aligned normal to the ecliptic except for short maneuver periods. Inertial attitude will be determined by solid-state sun and star sensors. Spin axis attitude will be determined to an accuracy of 0.9 deg. Spin rates will be variable between 15 and 57 rpm, depending on the mission event. The Atlas/Centaur design employs a magnetometer boom that positions the sensor 4.4 m (14.5 ft) from the solar panel surface as compared to 1.07 m (3.5 ft) for the Thor/Delta. This additional distance greatly reduces spacecraft magnetic cleanliness requirements and reduces program costs significantly. In addition, the bicone antenna on the Thor/Delta design requires a deployment mechanism, whereas none is required for the Atlas/Centaur.

Six hydrazine thrusters (developed for Intelsat IV) are used for midcourse maneuvers, spin rate changes, and precession control. A total of 16 kg (35 lb) of hydrazine is provided for velocity changes totaling 36 m/sec., spin rate changes of 114 rpm, and precession control totaling 396 deg. From a cost standpoint, the larger sizes of the tanks and tubing makes the Atlas/ Centaur slightly greater.

Configuration Description

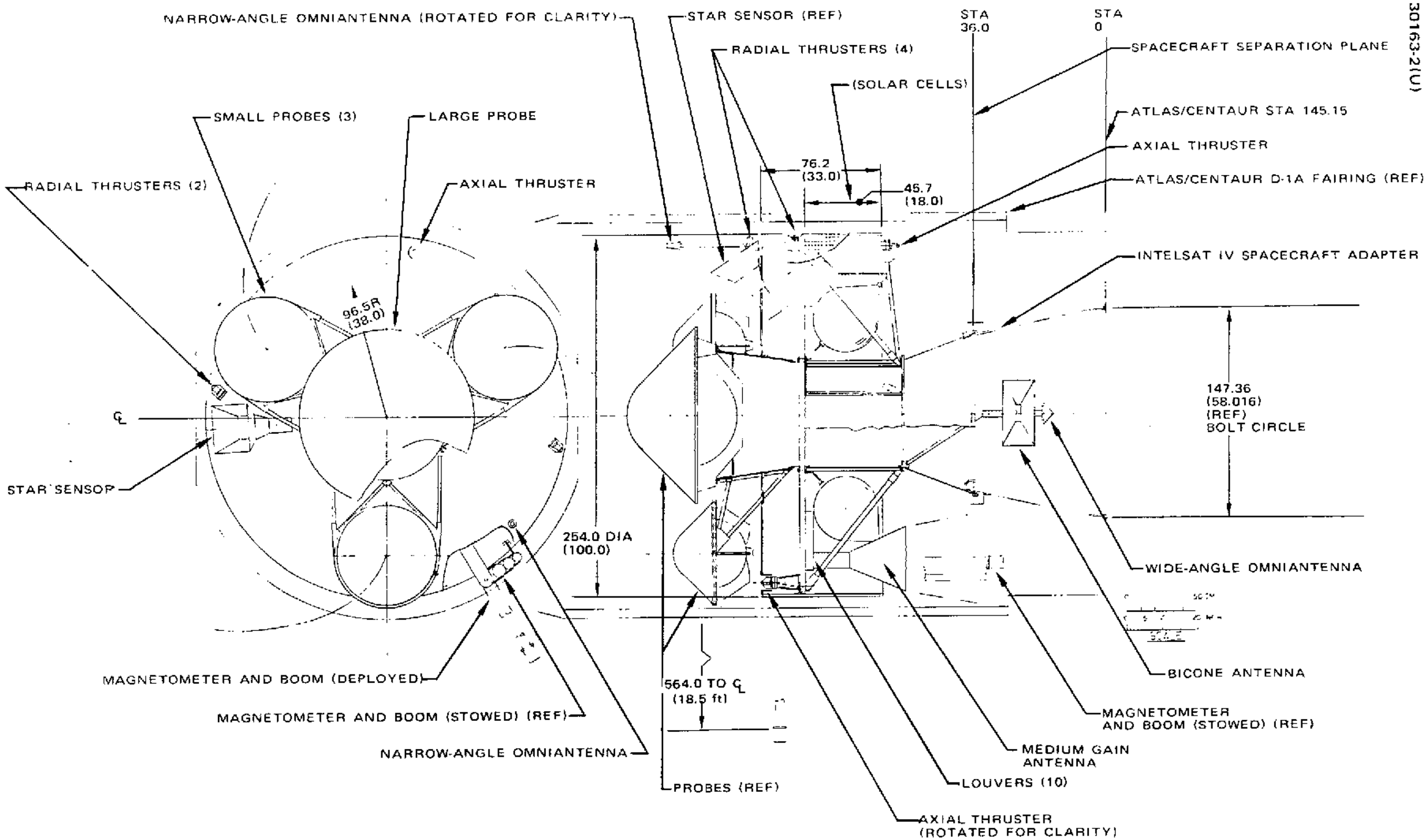
The probe bus spacecraft consists of five major assemblies, each of which constitutes a module which can be fabricated, assembled, and tested independently prior to the final spacecraft integration. Starting from the forward in Figure 4-1, the assemblies are:

1) Probes and adapter assembly
2) Solar panel assembly
3) Equipment shelf assembly
4) Thrust tube - cone assembly
5) Intelsat IV spacecraft assembly

This spacecraft configuration is derived from HS-339 basic bus the development of which has been sponsored and completed by Hughes Aircraft Company. The following modifications are incorporated:

1) The aft diameter of the thrust cone is increased from 95.89 cm (37.75 in) to 112.40 cm (44.25 in) so that existing Intelsat IV spacecraft adapter and its separation mechanism hardware can be used.

2) The diameters of solar panel and equipment shelf are increased from 215.9 cm (85 in) and 209.6 cm (82.5 in) to 254 cm (100 in) and 247.7 cm (97.5 in), respectively.



DIMENSIONS IN CENTIMETERS
AND ( ) INCHES

FIGURE 4-2. HS-507A PROBE BUS CONFIGURATION

Table 4-2 is a summary of the structural hardware derivation of both the Thor/Delta and Atlas/Centaur designs. Both designs are based on Telesat type structures with differences mostly in size. The Atlas/Centaur uses additional strength in the designs to provide greater margins of safety and reduce analysis and structural testing. In addition, the Atlas/Centaur uses aluminum in its design whereas beryllium is used in some areas of the Thor/Delta design. Figure 4-2 is a layout of the probe bus configuration. It can be used in conjunction with Figure 4-1 in understanding the following discussions.

Probes and Adapter Assembly. This assembly is unique for the probe bus spacecraft and consists of a large probe attached to the forward ring of a central cylindrical adapter and three small probes equally spaced around and supported from the cylindrical adapter by tripods.

All probes interface with the adapter through release mechanisms for separation near Venus.

The aft ring of the adapter rests on the equipment shelf and is bolted to the forward ring of the thrust tube-cone assembly.

Solar Panel Assembly. The solar panel assembly consists of a cylindrical substrate of 254 cm (100 in.) diameter by 83.8 cm (33 in.) long with solar cells bonded to the lower part of the exterior surface of the substrate with the rest of the panel [approximately a 33 to 38 cm (13 to 15 in.) wide band along the forward edge] being void of solar cells. All cutouts for science experiments, radial thrusters, magnetometer boom, and spacecraft preflight access are located in this area.

The solar panel assembly is supported by the equipment shelf at eight places along the interior surface approximately 25.4 cm (10 in.) from the forward edge of the drum. It forms a compartment for the equipment with the shelf and the forward (spun) thermal barrier.

The substrate is common for the probe bus and orbiter spacecrafts but the solar arrays are unique for each because of unequal number of solar cells and different cutout locations for science experiments and radial thrusters.

Equipment Shelf. The equipment shelf shown in Figure 4-3 is supported by the forward ring of the thrust tube-cone assembly. It is 3.8 cm (1.5 in.) thick honeycomb of 247.7 cm (97.5 in.) diameter. Both the core and face sheets are aluminum. Eight aluminum tubular struts equally spaced on the aft periphery of the shelf terminate at the thrust tube-cone assembly along the tube-cone transition circle. The forward side of each strut pickup point on the shelf is attached to the solar panel assembly.

The forward side of the shelf also supports subsystem components as shown and science experiments which are fastened to the threaded inserts post-molded in the shelf core. The magnetometer boom is supported by an aluminum bracket on the forward side of the shelf through a cutout in the solar panel.

TABLE 4-2. PROBE BUS STRUCTURAL HARDWARE DERIVATION

Item	Thor/Delta	Atlas/Centaur
Equipment shelf	Telesat type 213.4 cm (84 in.) diameter, Al	Telesat type 254.0 cm (100 in.) diameter, Al
Shelf support struts	New (six, Be)	Telesat type (eight, Al)
Thrust tube	New 60.96 cm (24 in.) diameter, Be	Domestic Satellite 76.2 cm (30 in.) diameter, Be
Cone (thrust tube-to-adapter)		Domestic Satellite Modification (Al)
Solar panel cylinder	Telesat type 213.4 cm (84 in.) diameter	Telesat type 254.0 cm (100 in.) diameter
Large/small probe attach structure	New (Be)	New (Al)

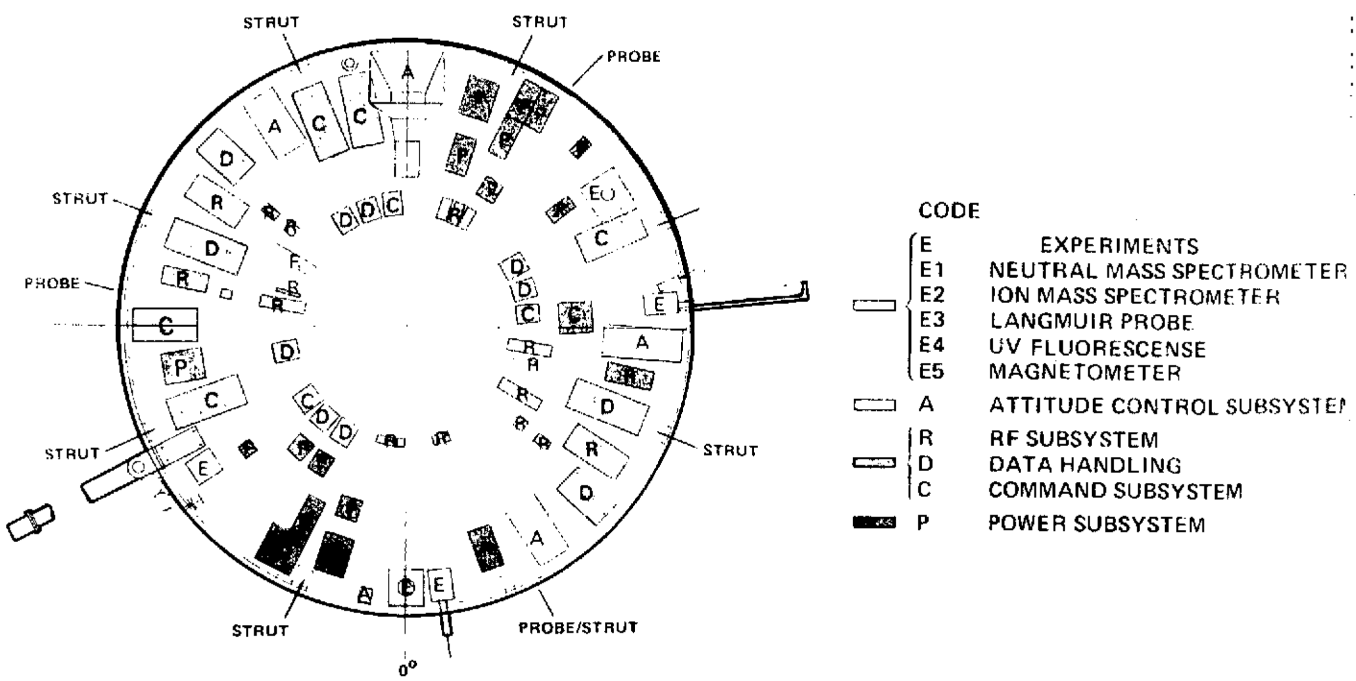


FIGURE 4-3. PROBE SPACECRAFT EQUIPMENT SHELF ARRANGEMENT - ATLAS/CENTAUR

Aluminum doublers bonded to the forward face sheet are used as necessary in the component footprint areas to effectuate heat conduction and distribution for thermal balance.

The arrangement of subsystem components and science experiments is such that a single layout will accommodate both probe bus and orbiter spacecrafts. Units peculiar to either bus are simply added to the common units.

The aft surface of the shelf is used to install the required number of thermal louvers, propulsion fuel lines, medium gain antenna and thermal blankets.

Thrust Tube-Cone Assembly. This primary structure which is common to both the probe bus and orbiter spacecrafts is a 73.7 cm (29.0 in.) diameter by 68.6 cm (27 in.) long aluminum cylinder whose aft end flares to a cone of 112.39 cm (44.25 in.) diameter x 50.8 cm (20.0 in.) height.

The forward end of the assembly supports the equipment shelf and the aft end forms the spacecraft separation plane and interfaces to Intelsat IV spacecraft adapter. The separation mechanism group at this plane includes separation springs, switches, and a clamp which are identical to flight proven Intelsat IV parts.

Two propellant tanks, an aft directed axial thruster, and a fill and drain valve are supported by machined fittings and tubular struts of aluminum which are attached to the thrust tube and cone assembly. The remaining units, i.e., a pressure transducer, two latch valves, four radial thrusters, and a forward directed thruster are supported from the equipment shelf.

Intelsat IV Spacecraft Adapter Assembly. The forward ring of this assembly supports the probe bus separation plane and its aft ring is bolted to the Atlas/Centaur mission peculiar (Intelsat IV) adapter. This spacecraft adapter assembly facilitates the use of flight proven separation mechanism hardware without any modification.

Table 4-3 is a summary of the spacecraft subsystem and experiment masses. A comparison is shown between the Atlas/Centaur and Thor/Delta designs.

Functional Description

Figure 4-4 is a functional block diagram for the probe bus showing all major power, signal, and configurational interfaces with and between the science instruments and subsystem hardware. The diagram divides the spacecraft into established subsystems.

TABLE 4-3. MASS COMPARISON, THOR/DELTA VERSUS ATLAS/CENTAUR

Item/Subsystem	Thor/Delta		Atlas/Centaur		Difference	
	Kg	lb	Kg	lb	Kg	lb
RF	8.9	19.5	10.0	22.1	+1.1	+2.6
Antenna	3.1	6.8	3.4	7.4	+0.3	+0.6
Data handling	5.8	12.8	7.7	17.0	+1.9	+4.2
Command	7.0	15.5	12.3	27.2	+5.3	+11.7
Attitude control, mechanisms	11.2	24.6	14.7	32.3	+3.5	+7.7
Structure	36.5	80.4	96.4	212.5	59.9	+132.1
Power	16.0	35.3	27.2	60.0	+11.2	+24.7
Cabling	4.7	10.3	9.1	20.0	+4.4	+9.7
Thermal control	10.3	22.6	19.0	41.9	+8.7	+19.3
Propulsion (dry)	9.1	20.0	10.4	22.9	+1.3	+2.9
Orbit insertion motor case						
Bus total	112.6	247.8	210.2	463.3	+97.6	+215.5
Large probe	114.6	252.6	198.7	438.0	+84.1	+185.4
Small probe (three)	101.9	224.7	169.1	372.9	+67.2	+148.7
Spacecraft subtotal	329.1	725.1	577.0	1274.2	+248.9	+549.1
Contingency	22.6	50.3	153.0	337.6	+130.0	+287.3
Experiments (bus only)	11.6	25.6	12.6	27.7	+1.0	+2.1
Spacecraft total (dry)	363.3	801.0	743.6	1639.5	+380.3	+838.5
Propellant	20.7	45.7	15.9	35.1	-4.8	-10.6
Pressurant	0.1	0.1	0.1	0.1	0.0	0.0
Orbit insertion expendables						
Spacecraft total (wet)	384.1	846.8	759.6	1674.7	+375.5	+827.9
Spacecraft adapter	13.2	29.0	31.3	69.0	+18.1	+40.0
Telemetry and C-band	8.4	18.5			-8.4	-18.5
Launch vehicle payload	405.7	894.3	790.9	1743.7	+385.2	+849.4

FOLDOUT FRAME 1

FOLDOUT FRAME 2

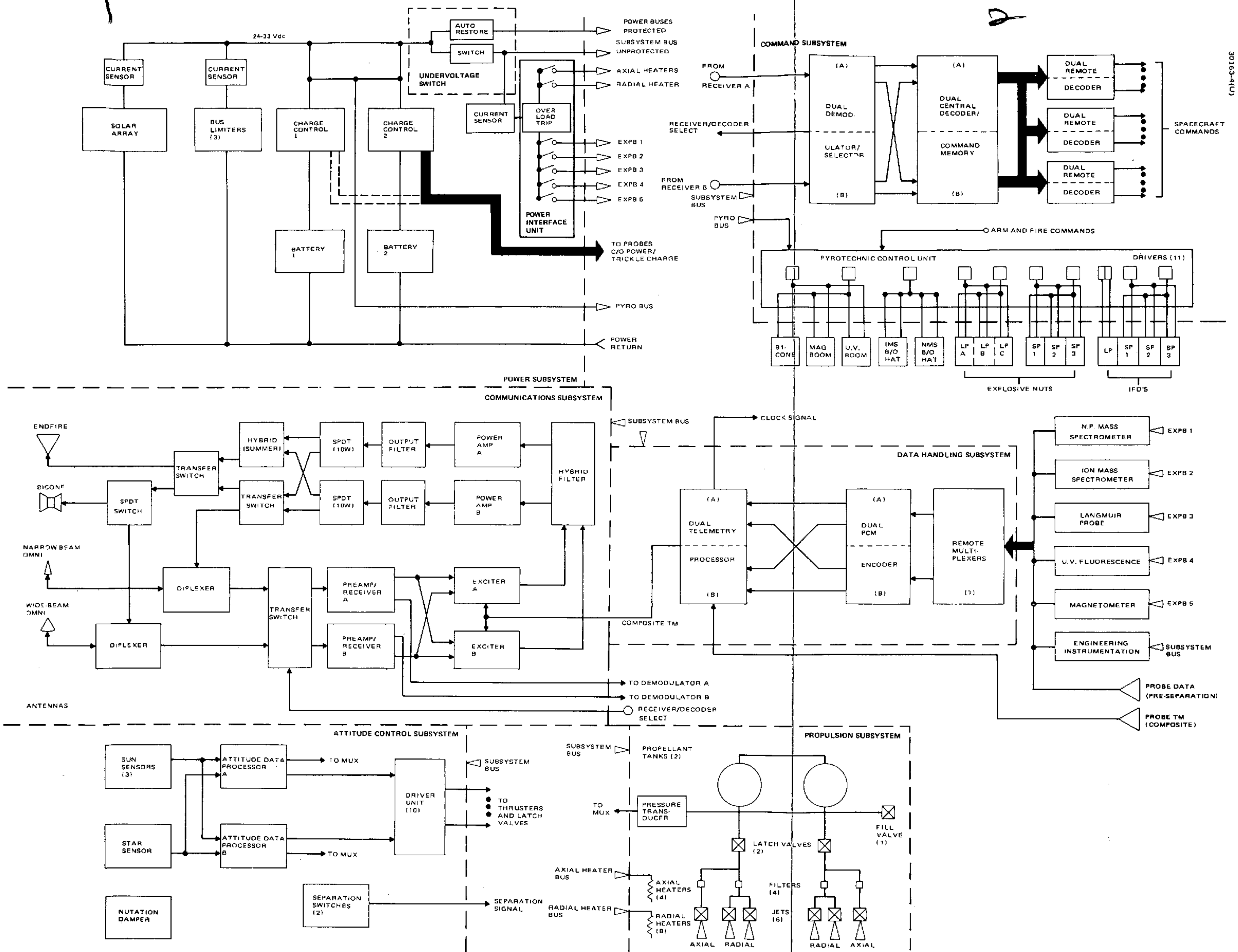


FIGURE 4-4. PROBE BUS FUNCTIONAL BLOCK DIAGRAM

The probe bus uses an unregulated 24 to 33 Vdc power subsystem. The upper voltage limit is controlled by bus limiters while the battery discharges when the solar array output drops below 24 Vdc. The probe bus utilizes a battery charger while on solar power to maintain charge in the spacecraft battery. The battery charger unit also supplies all four probes with checkout power and trickle charge current during the pre-probe separation transit phase of the mission. Three bus limiters are required. Power to the pyrotechnic control unit is drawn directly from the battery. All heaters and each science experiment has individual power switching.

The communications subsystem uses four antennas; endfire, bicone, and wide and narrow beam omnis. The dual preamplifier receiver and exciter, two power amplifier arrangement is common to both the probe bus and orbiter spacecraft.

The probe bus and orbiter spacecraft have similar attitude control and propulsion subsystems. Three sun sensors and one star sensor with twin attitude data processor/jet control electronics units and a single driver unit form the basic attitude control subsystem. The probe bus requires four radial, one forward, and one aft axial control jets. A two propellant tank, two latch valve, one pressure transducer, and one fill valve configuration is common to both spacecraft propulsion subsystems.

A dual telemetry processor, dual PCM encoder, and seven remote multiplexers form the basic data handling subsystem for both the probe bus and orbiter. A dual demodulator/selector, a dual central decoder/command memory, and three dual remote decoders form the basic command subsystem. It contains pyrotechnic control units which with power from the battery and arm and fire commands from the remote decoders, provide drivers to fire the spacecraft pyrotechnic functions. The probe bus requires eleven drivers for squib firing.

The output data for all science experiments are fed to the remote multiplexers of the data handling subsystem. Formatting circuits provide data timing and clock signals to several of the science experiments. Power is provided to science by individual switching.

A breakdown of the electrical power required by the various units is shown in Table 4-4. The table summarizes the critical points in the mission from launch through small probe separation. The solar panel power needed near earth is shown to be about 70 W and that near Venus to be about 165 W. The battery sizing is determined by the power needed at large and small probe separation. At this time, with the sun angles as shown, 143.5 W-hr hours are required or about 41 percent of the installed 352 W-hr capability.

A reliability summary of the probe bus subsystems is shown in Table 4-5.

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TABLE 4-4. PROBE BUS POWER SUMMARY

Subsystems	Mission Events and Times						
	Launch to Spacecraft Separation	Spacecraft Separation to Sun Acquisition	Cruise No. 1A	5 Days TCM No. 1	Cruise No. 5	Large Probe Separation	108 Days, TCM No. 5 and Small Probe Separation
	40 min.	60 min.	2 days	50 min.	1 day	3.2 hr.	4.8 hr.
Radio, W	20.3	20.3	20.3	45.2	78.8	78.8	84.8
Data handling command, W	15.6	15.6	14.0	15.6	15.6	19.0	15.6
Attitude control, W	Off	6.3	3.8	6.3	3.8	6.3	6.3
Thermal control, W	3.5	3.5	3.5	3.5	3.5	3.5	3.5
Experiments	Off	Off	Off	Off	4.0	Off	Off
Spacecraft total, W	39.4	45.7	41.6	70.6	105.7	107.6	110.2
Power subsystem, W	12.0	13.0	22.0 (1)	16.0	45.0 (2)	6.0	7.0
Contingency, W	5.1	5.9	6.4	8.7	15.1	11.4	11.7
Total, W	56.5	64.6	70.0	95.3	165.8	125.0 (3)	129.0 (4)
W-hr	37.3	64.6		79.4		13.8	127.9
Transients, W-hr		0.6		6.3		1.1	0.5
Total, W-hr	102.3			85.7		143.5	

(1) Charge one battery at a time at C/12 rate
(2) Both batteries charged at C/12
(3) Sun angle 45 deg; solar panel power = 120 W
(4) Sun angle 37 deg; solar panel power = 102 W

TABLE 4-5. PROBE BUS RELIABILITY SUMMARY

Subsystems	Reliability
Radio frequency	0.9981
Data handling	0.999
Command	0.99975
Attitude control	0.9994
Power	0.9942
Propulsion	0.9973
Mechanical	0.9997
Thermal	0.9995
Cabling	0.9992
Orbit insertion	-
Total	0.9863

TABLE 4-6. LARGE PROBE DECELERATION MODULE AND PRESSURE VESSEL REQUIREMENTS

Deceleration module
11.2 km/sec entry speed
-35 deg ≥ γ_E* ≥ -60 deg
Subsonic parachute deployment (M = 0.7)
Aerodynamically stable
Aeroshell/pressure vessel separation
Parachute separation at 40 km altitude
Pressure vessel module
Encapsulate and protect experiments, subsystems
Aerodynamically stable within ±10° envelope
Spin at 15 rpm maximum
95 atm maximum pressure
773 °K maximum temperature
*γ_E = entry angle

Large Probe Configuration

In the design of the probes for the Atlas/Centaur configuration, it was attempted to take advantage of the large mass and volume capability of the booster to ease design problems on the probes themselves. Some of the more important considerations are shown below:

1) The pressure vessel structures for both the large and small probes were strengthened and enlarged to eliminate precision tolerances in fabrication and to increase margins of safety with the goal of minimizing testing.

2) Sufficient volume was provided to easily accommodate mounting in addition to cabling and connectors. This facilitated assembly and integration activities and simplified operations.

3) The increased volume allowed easier positioning of the science and housekeeping hardware to meet mission and functional requirements.

Requirements and Design Evolution

The intent of this section is to summarize those design requirements that influence overall probe configuration and to summarize the evolution iterations. The overall science requirements for the probe mission were discussed in Subsection 3.1 of this report. The detail science instrument requirements and the integration of these into the probe design is discussed in Subsection 5.7.

Table 4-6 summarizes the range of entry angles and entry speed of the probe into the Venus atmosphere. The major requirements of the deceleration module are to provide an aerodynamically stable system to decelerate the pressure vessel upon entry into the Venus atmosphere and to provide for aeroshell/pressure vessel separation at a predesignated altitude.

The pressure vessel is required to encapsulate and protect science instruments and subsystem hardware during the descent to the planet's surface. It must withstand 95 atm maximum pressure and 773°K maximum temperature. Additional stability and spin rate requirements are shown in the Table 4-6.

Table 4-7 is a summary of the evolution of the large probe design. The first column is the major parameters as they existed at the first Atlas/Centaur review on 19 December 1972. At this time the ground rule was to make the probes as large as reasonably possible and to maintain a minimum box separation of 2.5 cm (1.0 in.). This resulted in a pressure vessel inside diameter of 63.5 cm (25.0 in.) to package a volume of 38,394 cm^3 (2343 in^3). This provided a packing factor (total volume ÷ packaged volume) of 3.48 and a total mass of 250 kg (552 lb). Probe internal volume was more than adequate and all hardware fit within the volume with relative ease. As the spacecraft design developed, however, it became apparent that the probes were too large and imposed severe mass, volume, and stability constraints. It became necessary to consider reducing the size of the probes. A more realistic mass bogey was imposed on the probes and a design study initiated to determine if a feasible

TABLE 4-7. LARGE PROBE SIZE, MASSES AND VOLUMES

Major Parameters	Dates of Development					
	19 December 1972		4 January 1973		26 February 1973	
Base diameter, cm (in.)	140.7	(55.4)	132.0	(52.0)	123.2	(48.5)
Pressure vessel diameter, cm (in.)						
Inside	63.5	(25.0)	58.4	(23.0)	61.0	(24.0)
Outside	76.7	(30.2)	69.9	(27.5)	69.9	(27.5)
Volume available, cm^3 (in^3)	133,996	(8177.0)	104,336	(6367.0)	118,544	(7234.0)
Volume packaged, cm^3 (in^3)	38,394	(2343.0)	38,329	(2339.0)	44,867	(2738.0)
Science	28,416	(1612.0)	24,499	(1492.0)	24,547	(1498.0)
Housekeeping	11,978	(731.0)	13,880	(847.0)	20,319	(1240.0)
Equipment packing factor		3.48		2.72		2.65
Entry mass total, Kg (lb)	250.4	(552.0)	220.8	(487)	196.8	(434.0)
Pressure vessel module	145.6	(321.0)	128.8	(284)	111.1	(245.0)
Deceleration module	104.3	(230.0)	92.0	(203)	85.7	(189.0)
Minimum box separation, cm (in.)	2.54	(1.0)	1.3	(0.5)	1.3	(0.5)
Knockdown factor		0.15		0.15		0.4

design could be developed that stayed within the mass constraints. Volume required by the science instruments had decreased due to further definition, however, housekeeping needs had increased resulting in overall packaging requirements remaining the same. The results of this study are tabulated as the 4 January entry. The inside diameter of the pressure vessel was decreased to 58.4 cm (23.0 in.) resulting in a packing factor of 2.72 and a minimum box separation of 1.3 cm (0.5 in.). Total mass of the large probe reduced to 221 kg (487 lb). This design appeared to satisfy spacecraft constraints and at the same time allowed sufficient latitude in probe internal layout. It appeared to be a satisfactory design.

By the time of the midterm presentation on 26 February, the housekeeping had grown to 20,319 cm (1240 in.^{3}) making a growth of the pressure vessel necessary. The internal diameter was increased to 61 cm (24.0 in.) while keeping the volumetric efficiency and minimum box spacing the same. Even with the volume increase it was possible to achieve a mass decrease. This was due to design changes such as aeroshell nose cone angle, insulation thickness, and knockdown factor (ratio of expected pressure vessel strength to theoretical). This mass reduction further enhanced spacecraft margins. Significant probe growth could be tolerated without exceeding spacecraft capability.

Design Summary

Table 4-8 is a summary of the deceleration module characteristics. The 45 deg cone is employed for commonality with the small probe resulting in 123.2 cm (48.5 in.) diameter base diameter. The structure employs standard aluminum monocoque construction for design simplicity and cost savings at some mass increase. The 4.6 m (15 ft) main parachute insures positive separation and should result in decreased development risk. Figure 4-5 is a configuration layout of the large probe showing the key dimensions.

Table 4-9 is a summary of the pressure vessel module characteristics. The sphere with perforated ring results in a 0.55 drag coefficient and a stable configuration. The shell is made of titanium with five ports and six windows requiring four electrical penetrations. The 61 cm (24.0 in.) inside diameter results in a packing factor of 2.65 and a minimum spacing between units of 1.3 cm (0.5 in.). This additional space alleviates layout and integration problems experienced on the Thor/Delta design. Figure 4-6 is a layout of the large probe showing the internal box spacing and line of sight of some of the science units.

Figure 4-7 is a functional block diagram for the large probe showing all major power, signal, and configurational interfaces within and between the science instruments and subsystem hardware. The power subsystem provides a 28 V ±2 percent regulated bus to the subsystems and science experiments. Power to stepper motors, idle mode timer circuits, heaters, and to the pyrotechnic control unit (PCU) is unregulated from the battery. Regulation is accomplished by the boost regulator and all power switching is done through the power interface unit. A separate switch to the PCU acts as an arming device for all probe pyrotechnic events.

TABLE 4-8. DECELERATION MODULE CHARACTERISTICS

Aero configuration	45 degree cone 0.5 nose radius/base radius Cylindrical/conical afterbody 158.5 N/m^2 (33.1 lb/ft^2) W/C_DA
Base dimensions	123.2 cm (48.5 in) diameter 82.3 cm (32.4 in) length
Heat shield	Phenolic nylon
Structure	Aluminum monocoque
Main parachute	4.57 m (15 ft) diameter Dacron Disk gap band configuration
Pilot parachute	Mortar deployed 0.84 m (2.75 ft) diameter Conical ribbon configuration

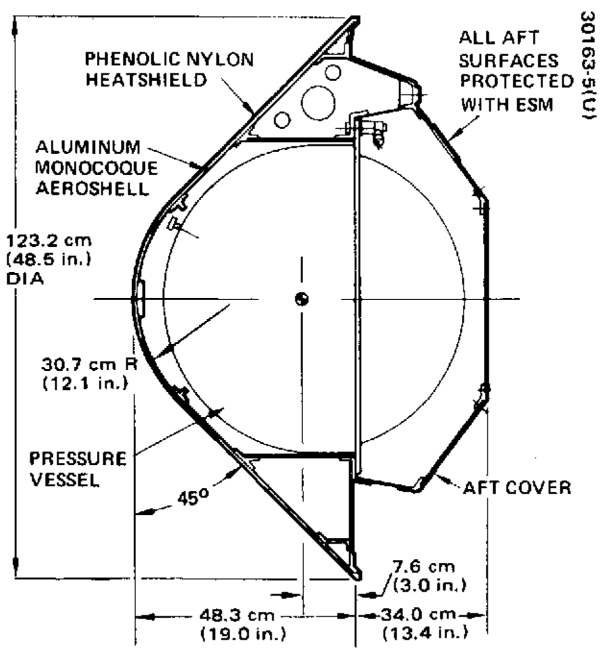

FIGURE 4-5. LARGE PROBE - ATLAS/CENTAUR

TABLE 4-9. PRESSURE VESSEL MODULE CHARACTERISTICS

Aero configuration	Sphere, perforated ring 0.55 drag coefficient
Structure	Titanium monocoque shell Min K cold wall insulation Five ports and six windows Four electrical penetrations
Diameter, cm (in.)	
Inside	61.0 (24.0)
Outside	69.9 (27.5)
Volume available, cm^3 ($in.^3$)	118, 544 (7234)
Volume packaged, cm^3 ($in.^3$)	
Science	24, 547 (1498)
Housekeeping	20, 319 (1240)
Equipment packing factor*	2.65
Entry mass, kg (lb)	196.8 (434)
Pressure vessel module	111.1 (245)
Deceleration module	85.7 (189)

*Not including shelves

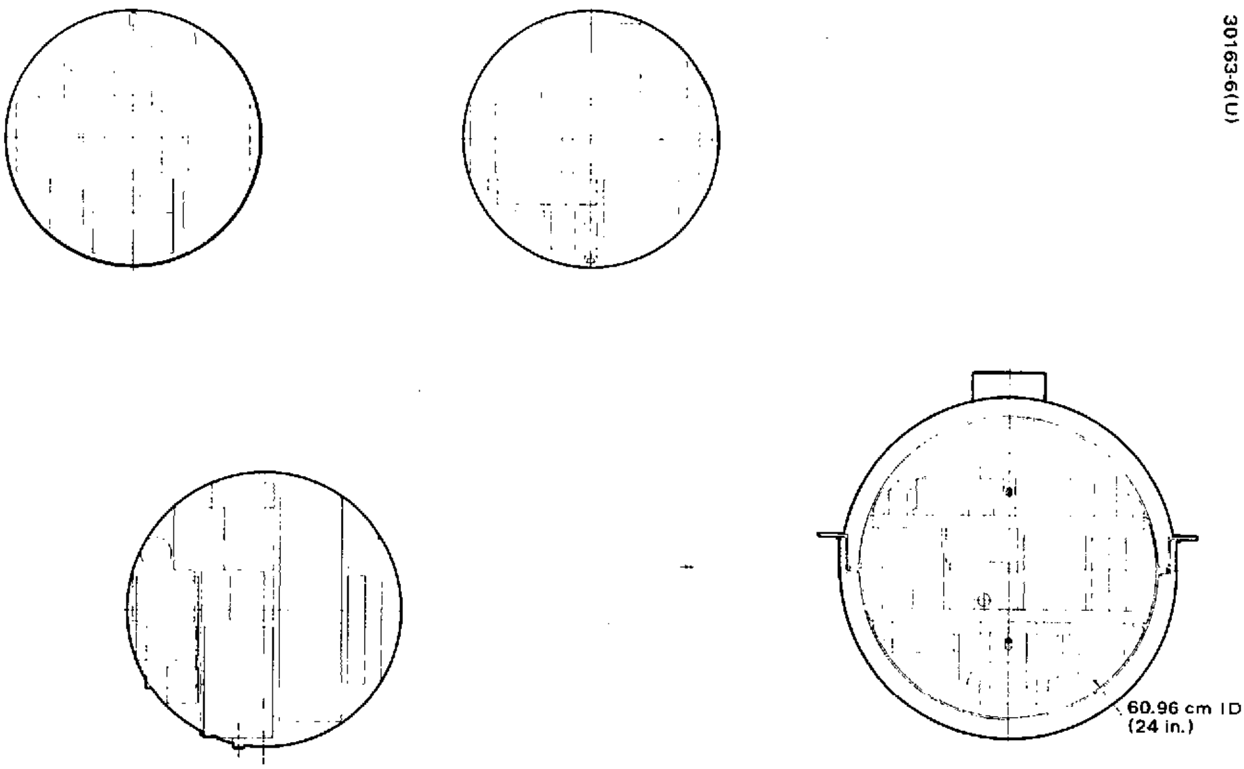

FIGURE 4-6. ATLAS/CENTAUR PRESSURE VESSEL LAYOUT - LARGE PROBE

FOLDOUT FRAME 1

FOLDOUT FRAME 2

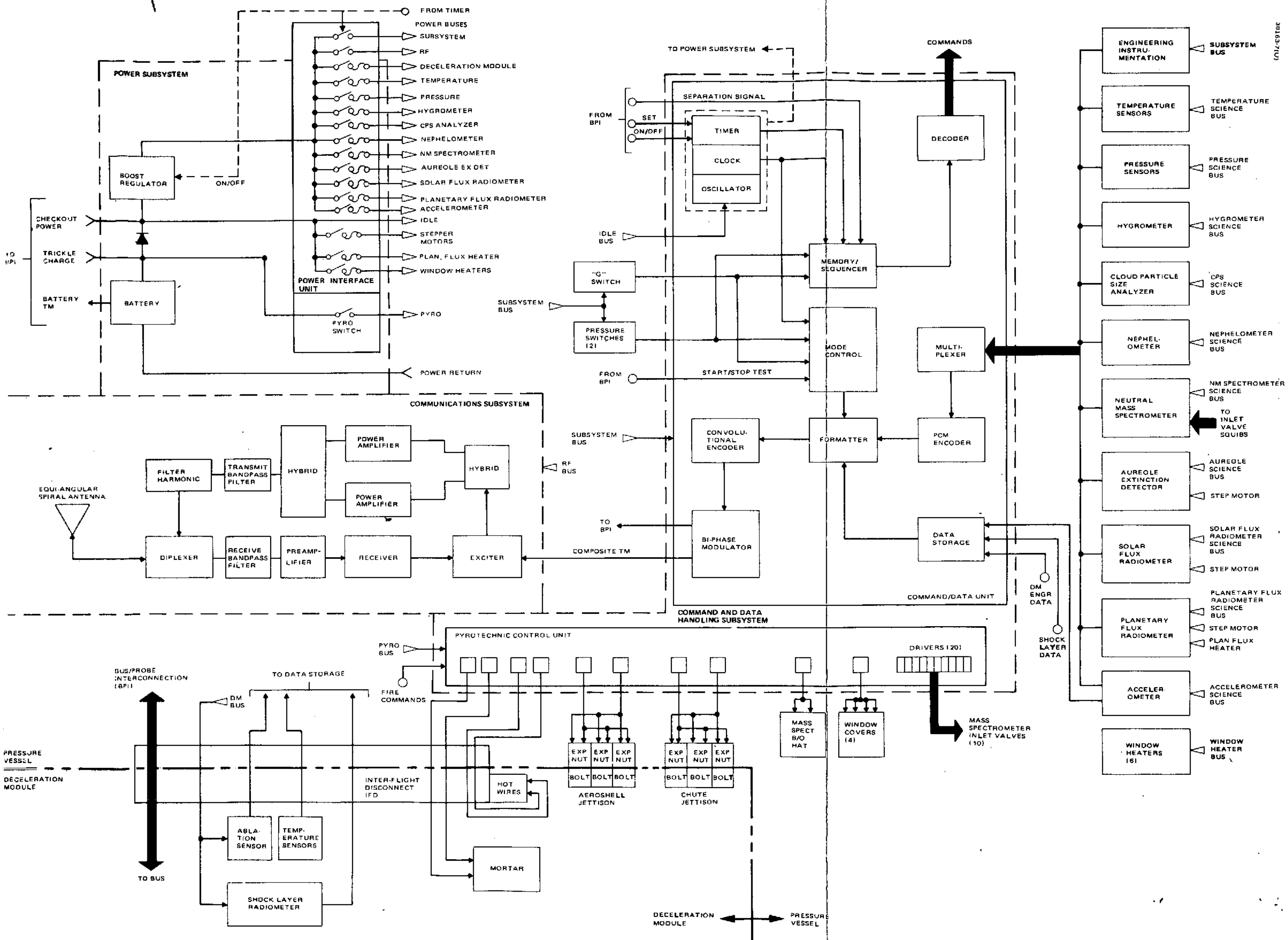

FIGURE 4-7. LARGE PROBE FUNCTIONAL BLOCK DIAGRAM

The communications subsystem comprises an equiangular-spiral antenna, diplexer, preamplifier, receiver, exciter, filters, hybrids and two power amplifiers. The command and data handling subsystem comprises a command/data unit and a PCU. The command/data unit provides multiplexing, data storage, encoding, formatting and modulating functions for data handling as well as timing, clock, memory, sequencing and decoding functions for the command requirements of the probe missions. The probe PCU houses 20 drivers for the various science and engineering pyrotechnic functions shown in Figure 4-7. Bus/probe interconnections are shown on the figure. These include all signal and power lines required between the probe bus and the probe during the pre-separation cruise phase of the mission.

The output data for all science experiments are fed to the multiplexing circuits of the data handling subsystem. Accelerometer data and shock layer radiometer data are inputs to data storage. Formatting circuits provide data timing and clock signals to several of the science experiments. Power is provided to the experiments by individual switching to provide operational flexibility.

A breakdown of the electrical power required by the various units is shown in Table 4-10. This table summarizes the mission needs from post-separation RF tracking to post impact on the planet's surface. The total watt-hours needed is 397, with the battery being sized to supply 437 W-hr at a depth of discharge of 80 percent. A breakdown of the unit masses is shown in Table 4-11 adding up to a total of 199kg (438 lb) on the bus. A reliability summary of the probe subsystems is shown in Table 4-12.

Key Cost Saving Differences from the Thor/Delta Design

Table 4-13 is a comparision of the key characteristics and performance of the Atlas/Centaur and Thor/Delta designs. The major cost saving differences between the two designs are noted. The aeroshell cone angle is 45 deg on the Atlas/Centaur design. This reflects a common aerodynamic design with the small probe and reduces total aero testing. In addition, the material for the Atlas/Centaur aeroshell is aluminum with that of the Thor/Delta being beryllium, a considerable cost saving. Thicker phenolic nylon is used on the Atlas/Centaur heatshield to minimize development testing.

The 61.0 cm (24.0 in.) internal diameter of the pressure vessel promotes easier unit integration and greater accessibility. The pressure vessels of both spacecraft are made from titanium monocoque. A more conservative knock-down factor of 0.4 is used on the Atlas/Centaur spacecraft to minimize development testing. For the same reason, thicker insulation is used for the pressure vessel.

In the power subsystem, more volume is available for packaging the batteries, thus easing the battery/packaging problems. If the rf area, use of a lower efficiency power amplifier reduces design and testing required and results in a cost saving. In the command and data handling area the primary cost saving results from the use of more volume for packaging and less packaging constraints. Use of more standard packaging techniques is possible.

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TABLE 4-10. LARGE PROBE POWER BUDGET

	Mission Events and Times									
	Post-Separation RF Tracking	Cruise (Idle)	E-2 Days Planetary Flux Rad Htr On	Pre-entry 1	Pre-entry 2	Entry	Descent 1 70→40 km	Descent 2 40→20 km	Descent 3 20→0 km	Post Impact
Subsystems	10 min.	18 days	2 days	10 min.	5 min.	0.5 min.	36.8 min.	12.4 min.	24.0 min.	60.0 min.
Radio, W	74.7	-	-	74.7	74.7	74.7	74.7	74.7	74.7	74.7
Deceleration module and shock layer rad., W	-	-	-	-	1.5	1.5	-	-	-	-
Equipment, W	1.5	-	-	1.5	1.5	1.5	1.5	1.5	1.5	1.5
Thermal, W	-	-	-	-	-	-	55.0	55.0	51.0	-
Command/data, W	5.0	0.1	0.1	5.0	5.0	5.0	5.0	5.0	5.0	5.0
Power, W	12.6	-	-	13.4	13.6	14.2	21.4	22.3	22.3	13.4
Experiments, W	-	-	0.1	5.1	5.1	9.1	58.3	64.3	64.3	5.0
Contingency, W	9.4	0.01	0.02	10.0	10.1	10.6	21.6	22.3	21.9	-
Total, W	103.2	0.11	0.22	109.7	111.5	116.6	237.5	245.1	240.7	99.6
W-hr	17.2	47.5	10.8	18.3	9.3	1.0	145.8	50.7	96.4	100.0
Cumulative total, W-hr	17.2	65.0	75.8	94.1	103.4	104.4	250.2	300.9	397.3	N.A.

TABLE 4-11. LARGE PROBE MASSES

	Current Mass	
	kg	lb.
Deceleration module	85.7	189.0
Heat protection	22.9	50.6
Structure	39.7	87.6
Aft cover	9.2	20.2
Parachute	5.4	12.0
Equipment	7.3	16.1
Experiments	1.1	2.5
Pressure vessel module	111.1	245.0
Structure	37.5	82.7
Insulation	16.6	36.6
Radio	5.0	11.1
Antenna	0.8	1.7
Data command	4.1	9.1
Power	18.1	39.8
Cabling	2.3	5.0
Experiments	24.5	54.0
Penetrations	2.3	5.0
Entry mass	196.8	434.0
Separation	1.8	4.0
Mass on bus	198.6	438.0

TABLE 4-12 LARGE PROBE RELIABILITY BY MISSION PHASE

Phase	Environment	Duration	Definition of Successful Performance	Large Probe Reliability
Preseparation	Probe bus transit phase	105 days	Survival of equipment required in following phases	0.9763
Separation from probe bus	Pyrotechnic loads	20 ms	Successful operation of separation system	0.9986
Coast	Low powered mode	20 days	Survival of equipment required during descent	0.9963
Entry	Descent equipment turned on	~15 min	Survival of equipment required during descent	~1.0
Descent	High power mode	1.75 h	Experiment data transmitted	0.99997
Parachute deployment	Pyrotechnic loads	-	Parachute deployed	0.9975
Aeroshell separation	Pyrotechnic loads	-	Aeroshell separated from deceleration modules	0.9987
Parachute separation	Pyrotechnic loads	-	Separation of parachute from pressure vessel	0.9987
Heat shield	-	-	Protect thermally the large probe deceleration module	0.995
Large probe mission				0.9616

TABLE 4-13. LARGE PROBE CHARACTERISTICS AND PERFORMANCE

	Thor Delta Spacecraft	Atlas Centaur Spacecraft	Key Cost Saving Differences
General			
Aeroshell cone, deg.	55	45	Common aerodynamics
Heatshield	Phenolic nylon	Same, 50 thicker safety margin	Minimize development testing
Deceleration module base diameter, cm (in.)	116.8 (46)	123.2 (48.5)	Lower cost
Deceleration module structure	Be	Al	
Main parachute diameter, m (ft)	3.5 (11.5)	4.6 (15.0)	
Pressure vessel id, cm (in.)	54.6 (21.5)	60.9 (24)	Greater accessibility
Pressure structure vessel	Titanium, monocoque, K=0.7	Same, K=0.4	
Insulation	Min K, 2.3 cm thick	Same, 5.1 cm thick	Minimize development
Total mass, kg (lb)	114.6 (252.6)	198.6 (438)	
Pressure vessel mass, kg (lb)	76.0 (167.5)	111.1 (245)	
Science payload mass, kg (lb)	22.5 (49.6)	25.6 (56.5)	
Volume available, cm^3 (in^3)	85,179 (5198)	118,544 (7234)	
Science payload volume, cm^3 (in^3)	22,860 (1395)	24,547 (1498)	
Power			
Peak power, W	212	245	More volume for packaging
Science payload, W	49.6	55.3	
A-hr (Ag-Zn)	19.2	30	
Radio			
Transmitter	S-band, 10W	→	21 versus 28 percent efficiency
Two-way doppler tracking	→	→	
4.8 dB equiangular spiral	→	→	
Command and data handling			
No ground commands	→	→	More volume for packaging
96 command register	→	→	
20 sec accuracy clock	→	→	
Encoding	Convolutional	→	
Modulation	PCM/PSK/PM	→	
Data rates, bps	276/184	184	
Semiconductor memory, bits	4096	4096	

TABLE 4-14. SMALL PROBE DECELERATION AND PRESSURE VESSEL REQUIREMENTS

Deceleration module
11.2 km/sec entry speed
-20.3 deg $\geq \gamma_E^* \geq$ -90 deg
Aerodynamically stable, supersonic and subsonic, within ±10 deg envelope subsonically
Pressure vessel module
Encapsulate and protect experiments, subsystems
69.0 min total descent time
95 atm maximum pressure
773°K maximum temperature

* γ_E = entry angle

Small Probe Configuration

In the design of the probes for the Atlas/Centaur configuration, it was attempted to take advantage of the large weight and volume capability of the booster to ease design problems on the probes themselves. Some of the more important considerations are shown below:

1) The pressure vessel structures for both the large and small probes were strengthened and enlarged to eliminate precision tolerances in fabrication and to increase margins of safety with the goal of minimizing testing.

2) Sufficient volume was provided to easily accommodate mounting in addition to cabling and connectors. This facilitated assembly and integration activities and simplified operations.

3) The increased volume allowed easier positioning of the science and housekeeping hardware to meet mission and functional requirements.

Requirements and Design Evolution

The intent of this section is to summarize those design requirements that influence overall probe configuration and to summarize the evolution iterations. The overall science requirements for the probe mission was discussed in Subsection 3.1 of this report. The detail science instrument requirements and the integration of these into the probe design is discussed in Subsection 5.7.

Table 4-14 summarizes the range of entry angles and entry speed of the probe into the Venus atmosphere. The major requirement of the decelerate the pressure vessel upon entry into the Venus atmosphere and during the terminal descent to the surface of the planet.

The pressure vessel is required to encapsulate and protect science instruments and subsystem hardware during the descent to the planet's surface. It must withstand 95 atm maximum pressure and 773° k maximum temperature.

Table 4-15 is a summary of the evolution of the small probe design. The first column is the major parameters as they existed at the first Atlas/Centaur review on 19 December 1972. At this time the ground rule was to make the probes as large as reasonably possible and to maintain a minimum box separation of 2.5 cm (1.0 in.). This resulted in a pressure vessel inside diameter of 38.1 cm (15.0 in.) to package a volume of 28,939 cm^3 (1766 in^3). This provided a packing factor (total volume ÷ packaged volume) of 4.18 and a total mass of 65 kg (144 lb). Probe internal volume was more than adequate and all hardware fit within the volume with relative ease. As the spacecraft design developed, however, it became apparent that the probes were too large and imposed severe mass, volume, and stability constraints. It became necessary to consider reducing the size of the probes. A more realistic mass

TABLE 4-15. SMALL PROBE SIZE, MASS AND VOLUMES

Major Parameters	Dates of Development					
	19 December 1972		4 January 1973		12 February 1973	
Base diameter, cm (in.)	87.6	(34.5)	83.3	(32.8)	72.6	(28.6)
Pressure vessel diameter, cm (in)						
Inside,	38.1	(15.0)	35.6	(14.0)	38.1	(15.0)
Outside	48.3	(19.0)	45.7	(18.0)	46.5	(18.3)
Volume available	28,939	(1766.0)	23,531	(1436.0)	28,939	(1766.0)
Volume packaged, cm^3 (in^3)	6,915	(422.0)	7,866	(480.0)	10,536	(643.0)
Science	1,966	(120.0)	1,966	(120.0)	1,163	(71.0)
Housekeeping	4,949	(302.0)	5,899	(360.0)	9,373	(572.0)
Equipment packing factor		4.18		3.0		2.75
Entry mass (total each), kg (lb)	65.3	(144.0)	57.6	(125)	54.4	(120.0)
Pressure vessel module	37.2	(82.0)	31.3	(69)	32.2	(71.0)
Deceleration module	28.1	(62.0)	25.4	(56)	22.2	(49.0)
Minimum box separation, cm (in)	2.54	(1.0)	1.3	(0.5)	1.3	(0.5)
Knockdown factor		0.15		0.15		0.4

bogey was imposed on the probes and a design study initiated to determine if a feasible design could be developed that stayed within the mass constraints. Volume required by the science instruments had remained the same; however, housekeeping needs had increased, resulting in overall packaging requirements increasing as shown in Table 4-15 as the 4 January entry. The inside diameter of the pressure vessel was decreased to 35.6 cm (14.0 in.), resulting in a packing factor of 3.0 and a minimum box separation of 1.3 cm (0.5 in.). Total mass of the small probe reduced to 57.6 kg (125 lb). This design appeared to satisfy spacecraft constraints and at the same time allowed sufficient latitude in probe internal layout.

By the time of the midterm presentation on 26 February, the housekeeping had grown to 9373 cm^3 (572 $in.^3$), making a growth of the pressure vessel necessary. The internal diameter was increased to 38.1 cm (15.0 in.) while slightly decreasing the volumetric efficiency and keeping the minimum box spacing the same. Even with the volume increase it was possible to achieve a mass decrease. This was due to design changes such as aeroshell nose cone angle, insulation thickness, and knockdown factor (ratio of expected) pressure vessel strength to theoretical). This mass reduction further enhanced spacecraft margins. Significant probe growth could be tolerated without exceeding spacecraft capability.

Design Summary

Table 4-16 is a summary of the deceleration module characteristics. The 45 degree cone is employed for aerodynamic stability resulting in a 72.6 cm (28.6 in.) diameter base diameter. The structure employs titanium construction for design simplicity and cost savings at some mass increase. Figure 4-8 is a configuration layout of the small probe showing the key dimensions.

Table 4-17 is a summary of the pressure vessel module characteristics. The sphere with fins results in a stable configuration. The shell is made of titanium with one port and one window requiring three electrical penetrations. The 38.1 cm (15.0 in.) inside diameter results in a packing factor of 2.75 and a minimum spacing between units of 1.3 cm (0.5 in.). This additional space alleviates layout and integration problems experienced on the Thor/Delta design. Figure 4-9 is a layout of the small probe showing the internal box spacing.

Figure 4-10 is a functional block diagram for the small probe showing all major power, signal, and configurational interfaces within and between the science instruments and subsystem hardware. The power subsystem provides a 28 V ±2 percent regulated bus to the subsystems and science experiments. Power to heaters, and to the pyrotechnic control unit (PCU) is unregulated from the battery. Regulation is accomplished by the boost regulator and all power switching is done through the power interface unit. A separate switch to the PCU acts as an arming device for all probe pyrotechnic events.

The communications subsystem comprises a loop-vee antenna, stable oscillator, exciter, and one power amplifier. The command and data handling

TABLE 4-16. DECELERATION MODULE CHARACTERISTICS

Aero configuration	45 deg cone Stabilizing fins 0.5 nose radius/base radius Spherical afterbody 127.8 N/m^2 (26.7 lbs/ft^2)W/C_DA
Base dimensions	72.64 cm (28.6 in.) diameter 54.1 cm (21.3 in.) length
Heat shield	Phenolic nylon
Structure	Titanium nonocoque

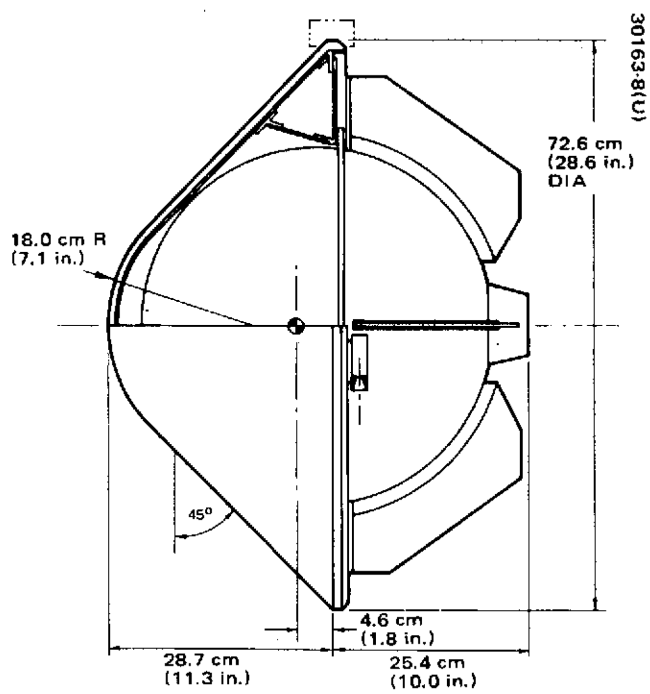

FIGURE 4-8. SMALL PROBE - ATLAS/CENTAUR

TABLE 4-17. PRESSURE VESSEL MODULE CHARACTERISTICS

Structure	Titanium shell Min K cold wall insulation One port and one window Three electrical penetrations	
Diameter, cm (in.)		
Inside	38.1	(15.0)
Outside	46.48	(18.3)
Volume Available, cm^3 ($in.^3$)	28,938	(1766)
Volume packaged, cm^3 ($in.^3$)		
Science	1163	(71)
Housekeeping	9373	(572)
Equipment packing factor*	2.75	
Entry mass, kg (lb)	54.4	(120)
Pressure vessel module	32.2	(71)
Deceleration module	22.2	(49)

*Not including shelves

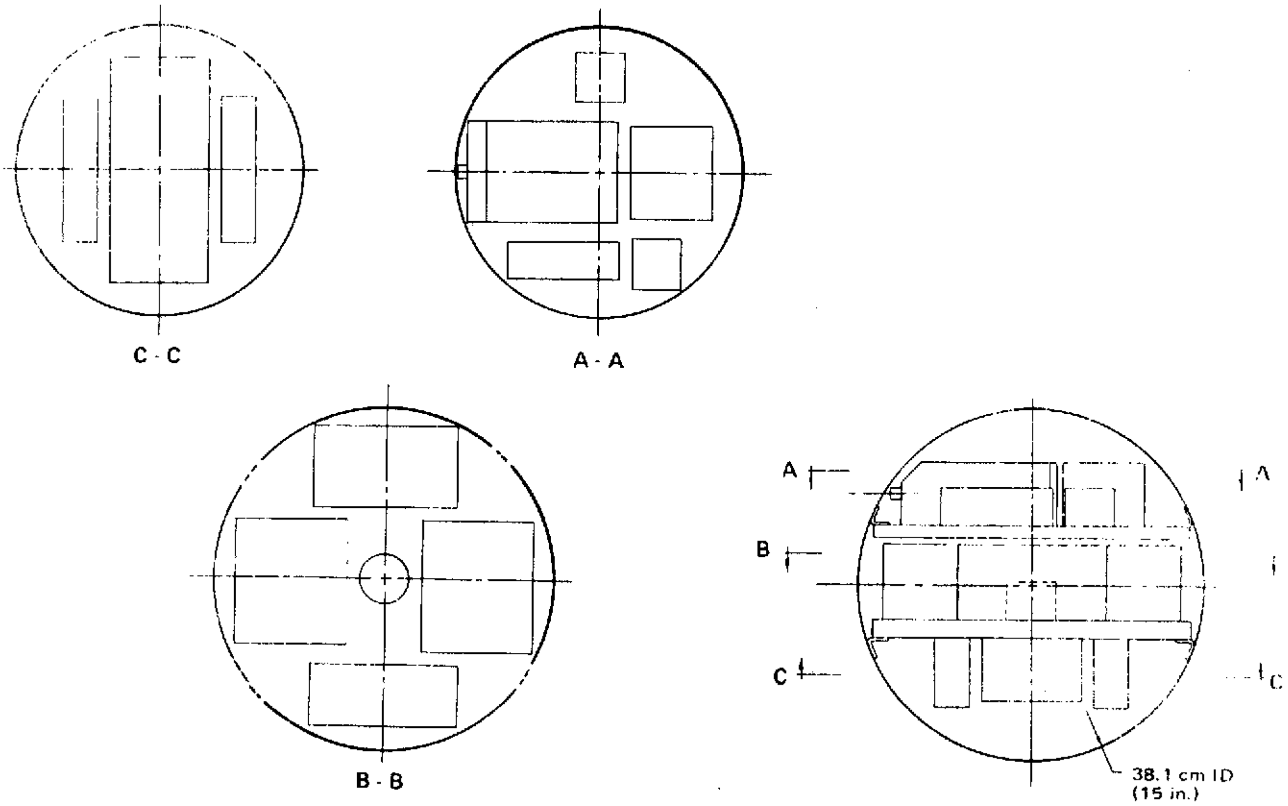

FIGURE 4-9. ATLAS/CENTAUR PRESSURE VESSEL LAYOUT - SMALL PROBE

FOLDOUT FRAME

FOLDOUT FRAME 2

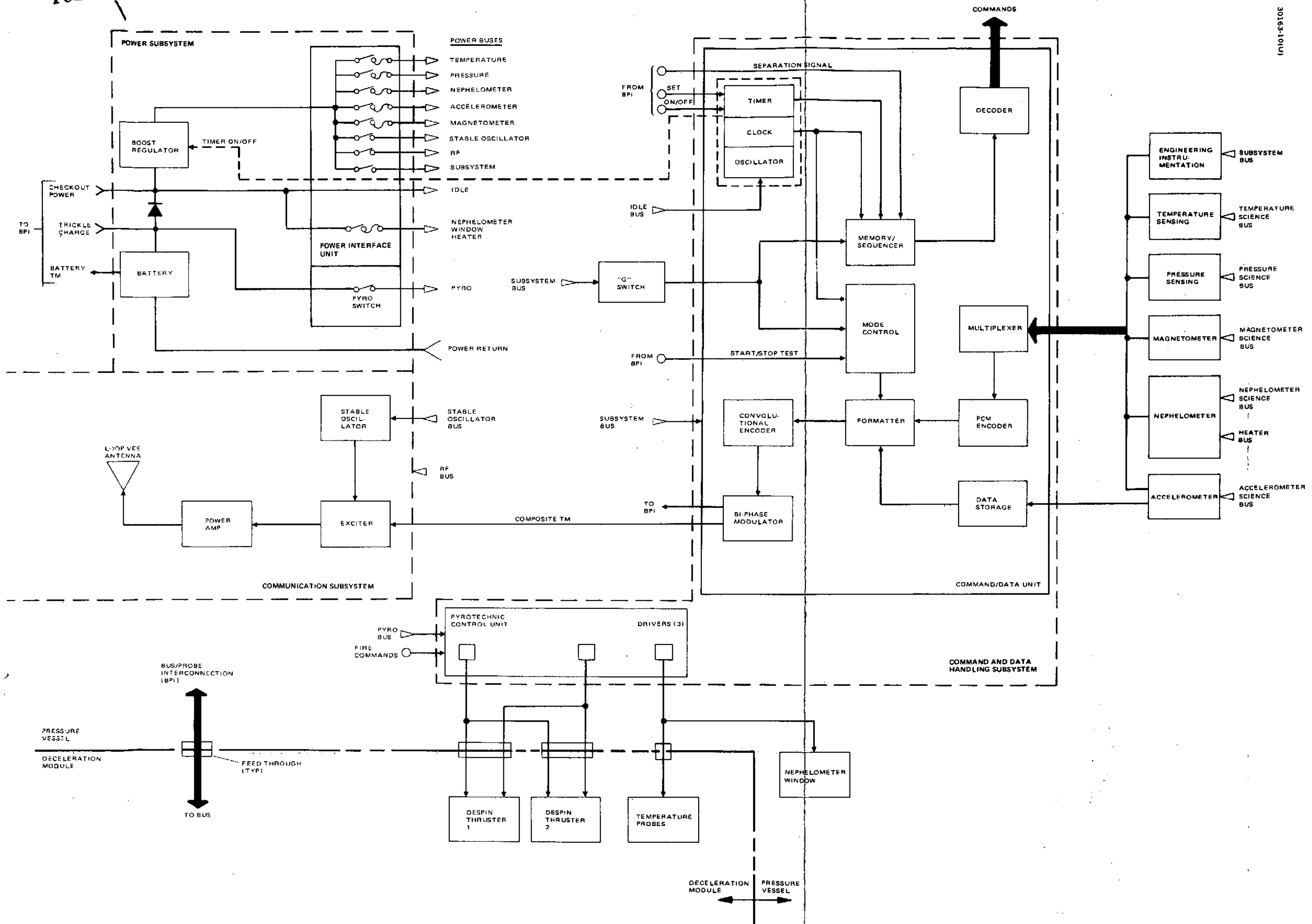

FIGURE 4-10. SMALL PROBE FUNCTIONAL BLOCK DIAGRAM

subsystem comprises a command/data unit and a PCU. The command/data unit provides multiplexing, data storage, encoding, formatting and modulating functions for data handling as well as timing, clock, memory, sequencing and decoding functions for the command requirements of the probe missions. The probe PCU houses three drivers for the various science and engineering pyrotechnic functions shown in Figure 4-10. Bus/probe interconnections are shown on the figure. These include all signal and power lines required between the probe bus and the probe during the pre-separation cruise phase of the mission.

The output data for all science experiments are fed to the multiplexing circuits of the data handling subsystem. The accelerometer data is an input to data storage. Formatting circuits provide data timing and clock signals to several of the science experiments. Power is provided to the experiments by individual switching to provide operational flexibility.

A breakdown of the electrical power required by the various units is shown in Table 4-18. This table summarizes the mission needs from post-separation magnetometer calibration through descent to the planet's surface. The total watt-hours needed is 169, with the battery being sized to supply 176 W-hr at a depth of discharge of 80 percent. A breakdown of the unit masses is shown in Table 4-19, adding up to a total of 56 kg (124 lb) on the bus. A reliability summary of the small probe subsystems is shown in Table 4-20.

Key Cost Saving Differences from the Thor/Delta Design

Table 4-21 is a comparison of the key characteristics and performance of the Atlas/Centaur and Thor/Delta designs. The major cost saving differences between the two designs are noted. The aeroshell cone angle is 45 deg on both vehicles. This is needed for subsonic stability and is a common aerodynamic design with the large probe, thus reducing total aero testing. In addition, the material for the Atlas/Centaur aeroshell is titanium, while that of the Thor/Delta is beryllium resulting in a considerable cost saving. Thicker phenolic nylon is used on the Atlas/Centaur heat shield to minimize development testing.

The 38.1 cm (15.0 in) internal diameter of the pressure vessel promotes easier unit integration and greater accessibility. The pressure vessels of both spacecraft are made from titanium. A more conservative knockdown factor of 0.4 is used on the Atlas/Centaur spacecraft to minimize development testing. For the same reason, thicker insulation is used for the pressure vessel.

In the power subsystem, more volume is available for packaging the batteries, thus easing the battery packaging problems. In the RF area, use of a lower efficiency power amplifier reduces design and testing required and results in a cost saving. In the command and data handling area the primary cost saving results from the use of more volume for packaging and less packaging constraints. Use of more standard packaging techniques is possible.

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TABLE 4-18. SMALL PROBE POWER BUDGET

Subsystems	Mission Events and Times					
	Post-Separation Mag. Calib. 10 Min	Cruise (Idle) 480 Min	Stable Osc. Warmup 30 Min	Pre Entry 15 Min	0.5 Min	Descent 69 Min
Radio, W	39.6	--	2.0	39.6	2.0	39.6
Thermal, W	--	--	--	--	12.0	12.0
Command/data, W	3.2	0.1	3.2	3.2	3.2	3.2
Power, W	7.7	0.0	1.2	7.7	2.7	7.7
Experiments, W	5.5	--	--	5.5	9.9	5.5
Contingency, W	5.6	0.01	0.6	5.6	3.0	6.8
Total, W	61.6	0.11	7.0	61.6	32.8	74.8
W-hr	10.3	52.8	3.5	15.4	0.3	86.0
Cumulative total, W-hr	10.3	63.1	66.6	82.0	82.0	169.3

TABLE 4-19. SMALL PROBE MASSES

	Current Mass	
	kg	lb
Deceleration Module	22.2	49.0
Heat Protection	9.3	20.4
Structure	12.3	27.3
Miscellaneous	0.6	1.3
Pressure Vessel Module	32.2	71.0
Structure	10.3	22.6
Insulation	5.7	12.5
Radio	1.9	4.2
Antenna	0.3	0.6
Data /Command	2.7	5.9
Power	8.8	19.5
Cabling	0.4	1.0
Experiments	1.9	4.2
Penetrations	0.2	0.5
Entry mass	54.4	120.0
Separation /Despin	2.0	4.3
Mass on Bus	56.4	124.3

TABLE 4-20. SMALL PROBE RELIABILITY BY MISSION PHASE

Phase	Environment	Duration	Definition of Successful Performance	Small Probe Reliability
Preseparation	Probe bus transit phase	105 days	Survival of equipment required in following phases	0.9478
Separation and despin	Pyrotechnic loads	--	Successful operation of separation system	0.9973
Coast	Low powered mode	20 days	Survival of equipment required during descent	0.9927
Entry	Descent equipment turned on	~15 min	Survival of equipment required during descent	~1.0
Descent	High powered mode	~1.25 h	Experiment data transmitted	0.9997
Heat shield	--	--	Thermally protect the probes deceleration module	0.9851
Small probe mission				0.9241

TABLE 4-21. SMALL PROBE CHARACTERISTICS AND PERFORMANCE

	Thor/Delta Spacecraft	Atlas/Centaur Spacecraft	Key Cost Saving Differences
General			
Aeroshell cone, deg	45		Stability, commonality
Heat shield	Phenolic nylon	Same, 50% safety margin	Minimize development testing
Deceleration module base diameter, cm (in.)	60.96 (24)	72.64 (28.6)	Allow c. g. flexibility
Deceleration module structure	Be	Ti	Lower cost
Pressure vessel id, cm, (in.)	32.5 (12.8)	38.1 (15.0)	Greater accessibility
Pressure vessel structure	Titanium monocoque, K=0.7	Same, K=0.4	Minimize development testing
Insulation	Min K, 3.6 cm (1.4 in.) thick	Same, 5.1 cm (2.0 in.) thick	Minimize development
Total mass, kg (lb)	33.9 (74.9)	56.4 (124.3)	
Pressure vessel mass, kg (lb)	23.6 (52)	32.2 (71.0)	
Science payload mass, kg (lb)	2.2 (4.9)	2.9 (6.3)	
Volume available, cm^3 (in^3)	16747 (1022)	28939 (1766)	
Science payload volume, cm^3 (in^3)	967 (59)	1655 (101)	
Power			
Peak power, W	57	75	More volume for packaging
Science payload, W	4.3	7.5	
A-hr (Ag-Zn)	10	12	
Radio			
Transmitter	S-band 5 W	→	21 Versus 28 percent efficiency
One-way doppler tracking	→	→	
2.5 dB loop vee antenna	→	→	
Command and data handling			
No ground commands	→	→	More volume for packaging
64 Command register	→	→	
20 Sec accuracy clock	→	→	
Encoding	Convolutional	→	
Modulation	PCM/PSK/PM	→	
Data rates, bps	16	→	
Semiconductor memory, bits	512	→	

TABLE 4-22. ORBITER CHARACTERISTICS AND PERFORMANCE

	Thor/Delta Spacecraft	Atlas/Centaur Spacecraft	Key Cost Saving Differences
General			
Overall length, cm (in.)	277 (109)	330 (130)	91.9 versus 10.0 kg contingency
Diameter, cm (in.)	213 (84)	254 (100)	Al versus Be structure
Usable spacecraft mass - dry, kg (lb)	179.6 (396.3)	375.5 (827)	
Science, kg (lb)	31.1 (68.6)	35.0 (77.2)	4.27 versus 2.88 m^2 shelf
Equipment shelf area, m^2 (ft^2)	2.88 (31)	4.27 (46)	12 versus 8 louvers
Power			
Solar panel area, m^2 (ft^2)	3.76 (40.5)	3.994 (43)	No battery boost
Peak power, W	179	193	
Science, W	48.5	51.5	
A-hr	(Ni-cad) 10.0	(Ni-cad) 7 (2)	
Radio			
Transmitter/receiver	S-band	→	21 versus 28 percent FPA efficiency
RF power, W	1, 5, 10	→	
23.5 dB MDA	→	→	
Two omni	→	→	
Command and data handling			
Modulation - command	PSK/PM	PSK/PM	No new LSI versus 10 new
Bit rate, bps - command	1, 2, 4	1, 2, 4	
Data rates, bps	8-2048	8-2048	
Memory	Core	Core	
Encoding	Convolutional	→	
Modulation	PCM/PSK/PM	PCM/PSK/PM	
Attitude control/mechanisms			
Spin axis ⊥ ecliptic	→	→	
Spin rates, rpm	5-100	5-25	Less magnetic cleanliness
Star sensor and sun sensors	Solid state	→	
Spin axis determination accuracy, deg	0.9	0.9	Ti versus Be bearing assembly
Magnetometer boom length, m (ft)	1.07 (3.5)	4.4 (14.5)	
Propulsion			
Blowdown hydrazine	→	→	
22.2 Ne (5 lb) thrusters	→	→	
Tanks, cm^3 (in^3)	18025 (1100)(2)	37690 (2300)(2)	None (more expensive)
Insertion motor	TE-M-521	TE-M-616	

4.2 ORBITER

The orbiter basic design goals remain unchanged from the probe bus. These goals emphasize science payload accommodation, commonality between the two missions, maximum use of existing hardware, and use of the additional mass and power capability of the Atlas/Centaur approach to arrive at a low cost/reliable design.

Characteristics and Performance

Table 4-22 provides a description of the major characteristics and performance of the Atlas/Centaur orbiter design compared to the Thor/Delta. It is similar to Table 4-1 previously presented on the probe bus. Much of the discussion of the probe bus is relevant due to the commonality of the two spacecraft and will not be repeated here. Only those major differences will be noted.

The major changes from the probe bus design include substitution of a high gain, mechanically despun antenna (MDA) for the bicone and endfire antennas. A memory is required in the data handling subsystem for occultation and eclipse data storage. An additional two louvers are required for thermal control of the spacecraft because of greater electrical power. Larger solar panels are incorporated to generate this power. In the propulsion area an orbit insertion motor is required along with one additional hydrazine thruster for orbital maneuvers.

Spin rates for the orbiter mission will be variable between 5 and 25 rpm, depending on the mission event. The solid propellant motor is the TE-M-616 used on the Canadian Technology Satellite with an ISP of 289.9 sec and ΔV of 1706.9 m/sec.

A total of 28.5 kg (63 lb) of hydrazine has been provided for velocity changes of 130 m/sec, spin rate changes of 45 rpm, and precession control of 130 m/sec. An additional axial thruster is required over the probe bus design for satisfaction of these requirements.

The accommodation of the experiments by the orbiter is discussed in subsection 5.7.

Configuration Description

The orbiter consists of six major assemblies, four of which are essentially identical with those of the probe bus. Only those differences will be noted in this subsection. For further discussion refer to subsection 4.1 of this report. Starting from the forward in Figure 4-11, the assemblies are:

1) Despun section assembly
2) Solar panel assembly
3) Equipment shelf assembly

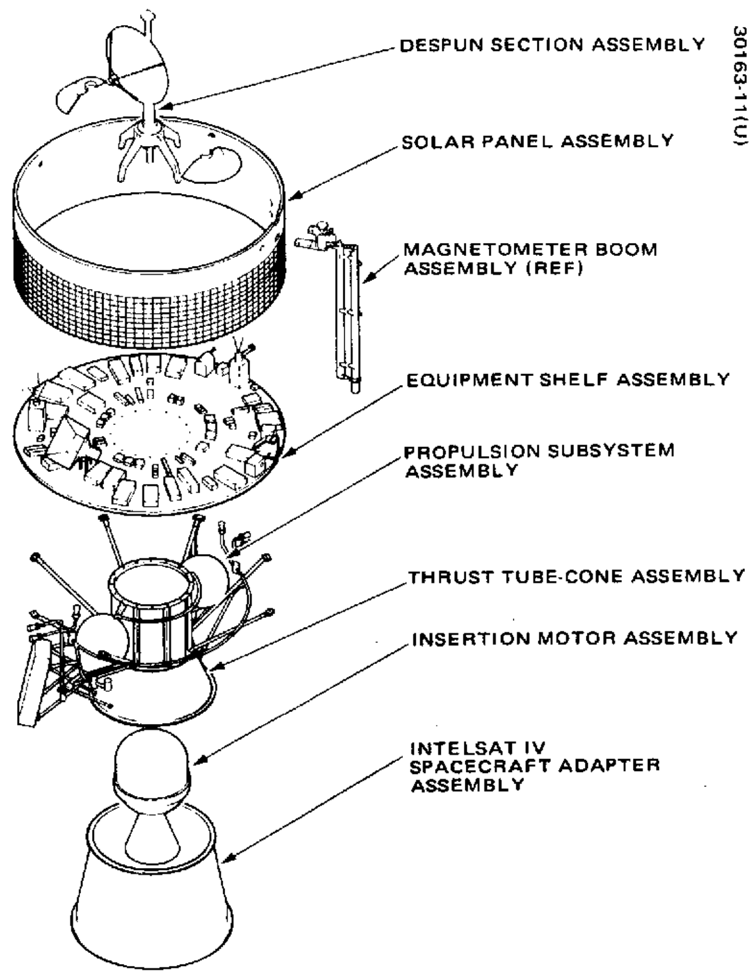

FIGURE 4-11. ORBITER SPACECRAFT

4) Thrust tube - cone assembly

5) Insertion motor assembly

6) Intelsat IV spacecraft adapter assembly

Table 4-23 is a summary of the structural hardware derivation of the Thor/Delta and Atlas/Centaur designs. Figure 4-12 is a layout of the orbiter configuration. These can be used in conjunction with Figure 4-11 in understanding the following discussions.

Despun Section Assembly

This assembly is unique for orbiter spacecraft and consists of antenna mast assembly, despin motor bearing assembly whose forward end (despun shaft) support the aft end of the antenna mast assembly, and bearing support assembly.

The four legs of the bearing support assembly are attached to the forward end of the thrust tube-cone assembly through four clearance openings in the shelf.

The antenna mast assembly consists of an aluminum tubing mast of 127 cm (50 in.) long, 81.3 cm (32 in.) diameter parabolic high gain antenna at the middle of the mast and a wide angle omni antenna at the forward end.

The despun section assembly is dynamically balanced and electrically calibrated as a single unit prior to final installation on the spacecraft in a similar manner as for the HS-333. The despin motor bearing is a modification of HS-333 bearing and the bearing support is identical to HS-333 bearing support.

Solar Panel Assembly

The substrate is common for the probe bus and orbiter and is as described in the probe bus subsection. The solar arrays are unique for each because of the unequal number of solar cells and different cutout locations for science experiments and radial thrusters.

Equipment Shelf Assembly

The arrangement of subsystem components and science experiments is such that a single layout will accommodate both the probe bus and orbiter spacecrafts. Units peculiar to either bus are added to the common units. The orbiter version is shown in Figure 4-13.

TABLE 4-23. ORBITER STRUCTURAL HARDWARE DERIVATION

Item	Thor/Delta	Atlas/Centaur
Equipment shelf	Telesat type 213 cm (84 in.) diameter, Al	Telesat type 254 cm. (100 in.) diameter, Al
Shelf support struts	New (six, Be)	Telesat type (eight, Al)
Thrust tube	New 60.9 cm (24 in.) diameter, Be	Domestic satellite 76.2 cm (30 in.) diameter, Al
Cone (thrust tube-to-adapter		Domestic satellite Modified, Al
Solar panel cylinder	Telesat type 213 cm (84 in.) diameter	Telesat type 254 cm (100 in.) diameter
Large small probe Attach structure	New (Be)	New (Al)
BAPTA support	New (Be)	Telesat
HGA support	New (Al)	New (Al)

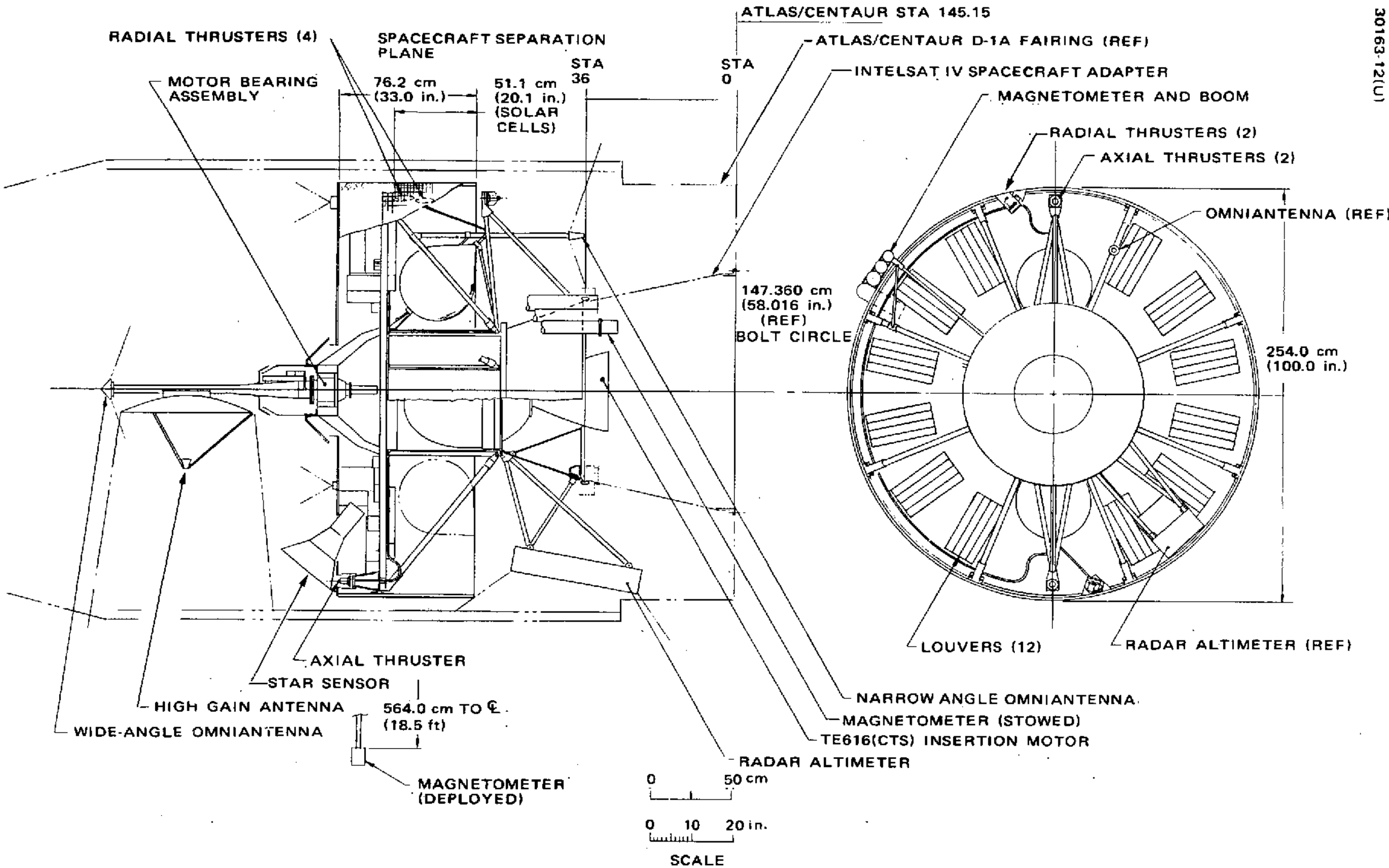

FIGURE 4-12. HS-507A ORBITER CONFIGURATION

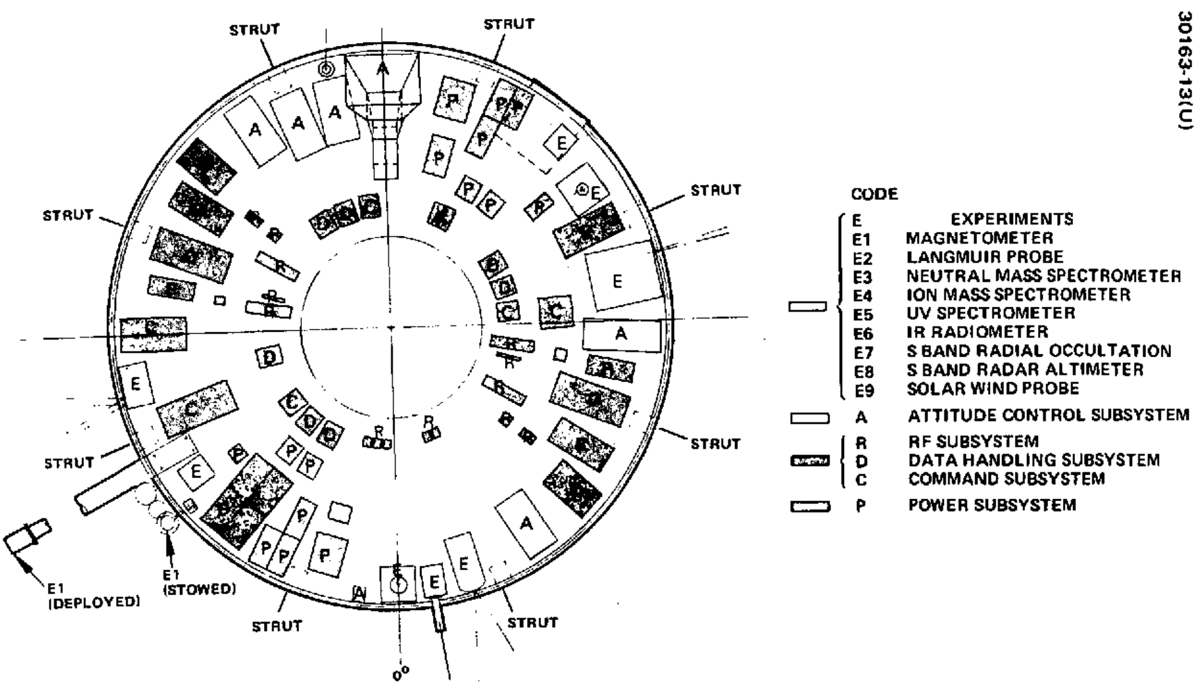

FIGURE 4-13. ORBITER SPACECRAFT EQUIPMENT SHELF ARRANGEMENT - ATLAS/CENTAUR

Thrust Tube-Cone Assembly

This primary structure is common to both probe bus and orbiter and is described in the probe bus subsection.

Insertion Motor Assembly (TE-M-616)

This assembly is used for the orbiter spacecraft only and is identical to the apogee kick motor and interface scheduled for a Hughes domestic satellite.

Intelsat IV Spacecraft Adapter Assembly

This is identical to the probe bus and is described in subsection 4.1. Table 4-24 is a summary of the spacecraft subsystem and experiment masses. A comparison is shown between the Atlas/Centaur and Thor/Delta designs.

Functional Description

Figure 4-14 is a functional block diagram for the orbiter showing all major power, signal, and configuration interfaces. It is similar to the probe bus diagram previously discussed in subsection 4.1 of this report. The major differences are as follows:

1) Electrical Power. Different switching arrangement to accommodate the different experiment complement.

2) Data Handling. Addition of the data storage function.

3) Communications. Different antenna configuration and one less transfer switch.

4) Attitude Control. Addition of BAPTA and despin electronics to accommodate the despun antenna.

5) Propulsion. Addition of one axial thruster and the orbit insertion motor.

A breakdown of the electrical power required by the various units is shown in Table 4-25, which summarizes the critical points in the mission from launch through operation in orbit around Venus. The solar panel power needed near earth is shown to be about 92 W and that in orbit around Venus to be a maximum of 189 W during apoapsis orbit corrections. The battery is sized by the power needed at a phase of the mission occurring late in the 225 day orbital life. This is an apoapsis eclipse of 4.0 h duration followed within a few hours by a periapsis subsolar pass. This is shown on Figure 4-15. The 4 h apoapsis eclipse is encountered as shown in the figure during which the spacecraft minimum loads require power such that the batteries are discharged to about 60 percent depth of discharge. This is followed by a couple of hours of battery charge which partially replenish the batteries. At the subsolar point solar panel heating causes a power dropoff such that

TABLE 4-24. MASS COMPARISON, THOR/DELTA VERSUS ATLAS-CENTAUR

	Thor/Delta		Atlas/Centaur		Difference	
	kg	lb	kg	lb	kg	lb
RF	8.5	18.8	9.7	21.4	+1.2	+2.6
Antenna	2.5	5.6	3.5	7.6	+1.0	+2.0
Data handling	9.9	21.8	13.2	29.0	+3.3	+7.2
Command	6.6	14.5	11.0	24.2	+4.4	+9.7
Attitude control, mechanisms	18.1	40.0	26.6	58.6	+8.5	+18.6
Structure	32.5	71.7	83.3	183.7	+50.8	+112.0
Power	22.4	49.4	29.1	64.2	+6.8	+14.8
Cabling	6.8	14.9	13.6	30.0	+6.9	+15.1
Thermal control	11.3	24.9	20.9	46.0	+9.6	+21.1
Propulsion (dry)	9.8	21.7	11.2	24.6	+1.4	+2.9
Orbit insertion motor case	10.1	22.3	26.7	58.8	+16.6	+36.5
Bus total	138.5	305.6	248.8	548.1	+110.3	+242.5
Large probe						
Small probe (three)						
Spacecraft subtotal	138.5	305.6	248.8	548.1	+110.3	+242.5
Contingency	10.1	22.1	91.7	202.5	+81.6	+180.2
Experiments (bus only)	31.1	68.6	35.0	77.2	+3.9	+8.6
Spacecraft total (dry)	179.7	396.3	375.5	827.8	+195.8	+431.5
Propellant	24.3	53.6	28.3	62.4	+4.0	+8.8
Pressurant		0.1	0.2	0.4	+0.1	+0.3
Orbit insertion expendables	88.7	195.6	326.9	720.7	+238.2	+525.1
Spacecraft total (wet)	292.8	645.6	730.9	1611.3	+438.1	+965.7
Spacecraft adapter	13.2	29.0	31.3	69.0	+18.1	+40.0
Telemetry and C-band	8.4	18.5			-8.4	-18.5
Launch vehicle payload	314.4	693.1	762.2	1680.3	+447.8	+987.2

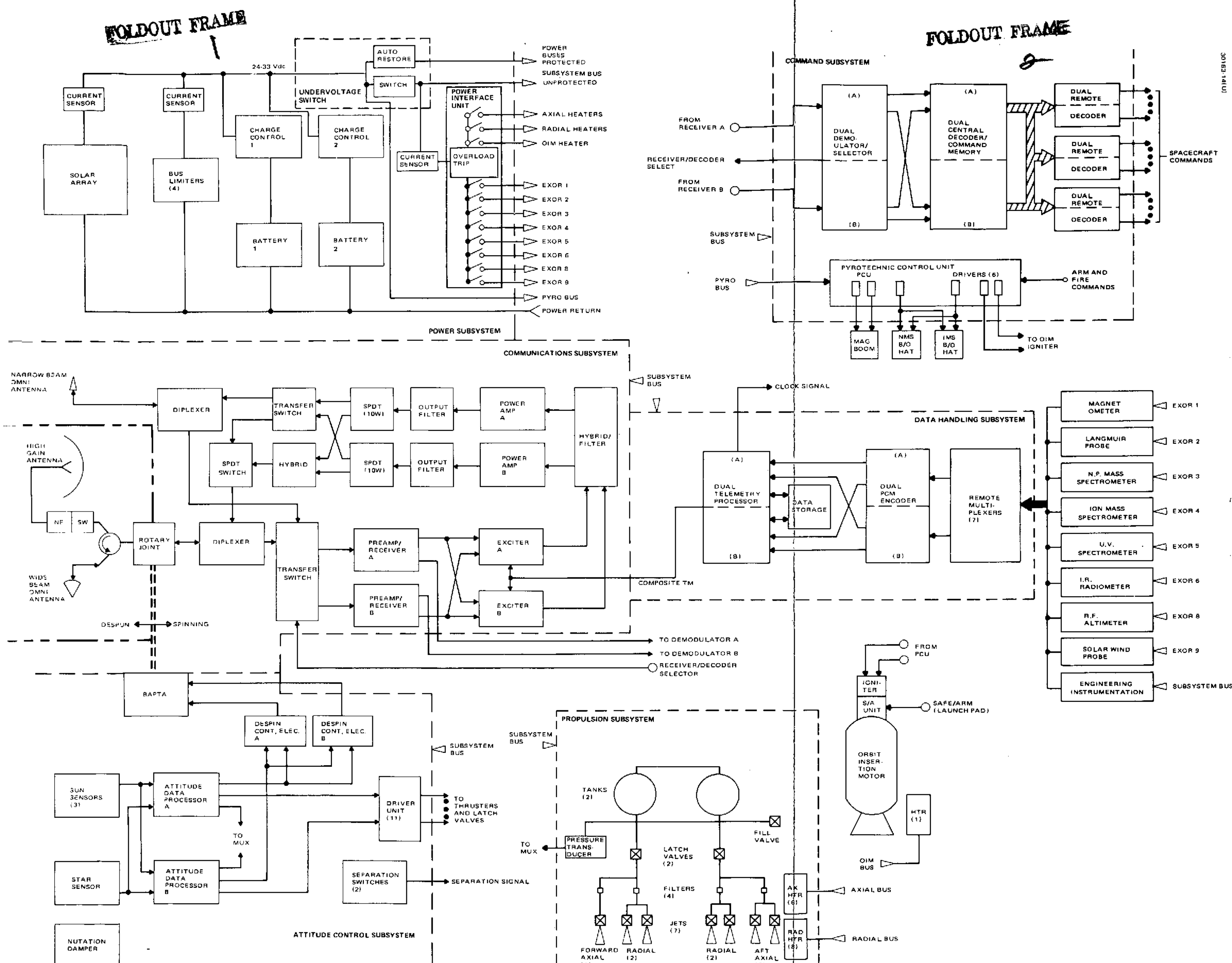

FIGURE 4-14. ORBITER FUNCTIONAL BLOCK DIAGRAM

TABLE 4-25. ORBITER SPACECRAFT POWER BUDGET

Subsystems	Mission Events And Times										
	Launch to Spacecraft Separation	Spacecraft Separation to Sun Acquisition	Cruise 1	5 Days, TCM 1	Periapsis Eclipse Under 1000 km	Periapsis Eclipse Under 5000 km	Apoapsis	Periapsis Subsolar Pass	Apoapsis Eclipse	Rest of Apoapsis	Apoapsis Orbit Correction
	40 min.	60 min.	4.5 day	47 min.	20 min.	10 min.	23 hour	30 min.	4.0 hour	19.5 hour	1.9 hour
Radio, W	20.2	20.2	20.2	45.1	45.1	45.1	45.1	78.7	7.0	45.1	78.7
Data handling, command, W	15.6	15.6	17.0	15.6	17.0	17.0	17.0	17.0	13.5	17.0	17.0
Attitude control, W	Off	10.3	10.3	13.8	10.3	10.3	10.3	10.3	0.8	10.3	10.3
Thermal control, W	14.5	14.5	14.5	14.5	4.5	4.5	4.5	4.5	5.5	4.5	4.5
Experiments, W	Off	Off	Off	Off	56.0	25.0	25.0	56.0	7.0	16.0	16.0
Spacecraft total, W	50.3	60.6	62.0	89.0	132.9	101.9	101.9	167.5	33.8	92.9	126.5
Power subsystem, W	12.0	12.0	22.0 (1)	20.0	36.0	28.0	45.0 (1)	37.0	11.5	45.0 (2)	45.0 (2)
Contingency, W	6.2	7.3	8.4	10.9	16.9	13.0	14.7	20.4	4.5	13.8	17.2
Total, W	68.5	79.9	92.4	119.9	185.8	142.9	161.9	223.9	49.8	141.7	188.7
W-hr	45.2	80.0	-	95.9	61.3	24.7	-	112.0	199.2	-	-
Transients, W-hr	-	-	-	3.6	-	-	-	-	-	-	-
Total - W-hr	125.2			99.5	85.6			112.0 (3)	99.2 (3)		

(1) Charge one battery at a time at C/12

(2) Charge both batteries at C/12

(3) DOD is approximately 60 percent both times

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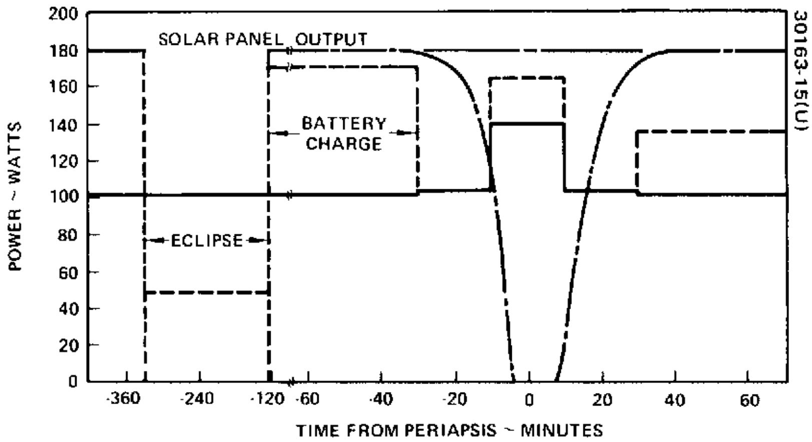


FIGURE 4-15. ORBITAL PHASE POWER HISTOGRAM

the batteries are discharged again to a level near 60 percent depth of discharge. Following the periapsis pass the solar panel power returns and the batteries are fully recharged prior to the next apoapsis eclipse some 20 h later. This pattern is repeated over roughly a 10 day period until the apoapsis eclipse season is over. A reliability summary of the orbiter subsystems is shown in Table 4-26.

4.3 CONFIGURATION TRADEOFFS

Configuration Selection History

The primary considerations for selecting the baseline configurations for the probe bus and orbiter are:

1) Low cost configurations which provide for all associated subsystems:
   a) Minimum cost for tests
   b) Use of off-the-shelf components
   c) Compatibility with launch vehicle
2) Commonality between orbiter and probe bus
3) Adaptability to available insertion motor for orbiter
4) High degree of accommodating for the subsystems involved; adequate shelf area for thermal louvers and high roll-to-pitch ratio for attitude control.

An additional consideration is also given to the ease of fabrication, accessibility and prelaunch serviceability.

In Table 4-27, three configurations, namely two versions of the modified HS-339 spacecraft, which development has been sponsored and completed by Hughes and a modified HS-312 (Intelsat IV) spacecraft configuration are qualitatively compared with regard to aforementioned aspects. The HS-312 version is very expensive compared to the others.

In Figures 4-16 and 4-17, a modified HS-339 probe bus and orbiter with a 216 cm (85 in.) diameter solar panel are shown, respectively. Figure 4-18 depicts the modified HS-312 configurations with the probe bus and orbiter superimposed.

In Table 4-28, the three configurations are compared to see whether the HS-312 (Intelsat IV) modified design has any advantages which may offset its higher cost over the two versions of HS-339 modification. The former design, however, exhibits less desirable characteristics both in roll-to-pitch ratios and payload mass margins as shown in the table.

TABLE 4-26. ORBITER RELIABILITY SUMMARY

Subsystems	Reliability
Radio frequency	0.9947
Data handling	0.8684
Command	0.9953
Attitude control	0.9915
Power	0.9875
Propulsion	0.9941
Mechanical	0.9997
Thermal	0.9982
Cabling	0.9979
Orbit insertion	0.9993
Total	0.8327

TABLE 4-27. CANDIDATE CONFIGURATIONS

Bus Configuration	Modified HS-339 254 cm (100 in.) Diameter Panel	Modified HS-339 216 cm (85 in.) Diameter Panel	Modified HS-312 (Intelsat IV)
Relative integration cost (including modification)	1.1	1.0	2.0
Commonality between probe bus and orbiter	Good	Good	Good; heavier probe adapter needed
Adaptability to insertion motor, TE 616 (CTS)	Good	Good	Fair; motor adapter needed
Launch vehicle interface	Fair (thrust cone modification necessary)	Fair (thrust cone modification necessary)	Good; no change
Thermal aspects:			
Maximum number of louvers can be integrated	18 (growth potential)	12 (adequate)	18 (growth potential)
Control mode for equipment shelf	Complex; RCS tanks not in compartment	Complex; RCS tanks not in compartment	Less complex; all in compartment
View factor for louvers	Eight struts and two tanks obstruct view	Eight struts and two tanks obstruct view	Eight shelf ribs limit view
Accessibility	Better	Good	Poor
Equipment shelf area, m^2 (ft^2)	4.27 (46)	2.88 (31)	3.25 (35)
Manufacturing:			
Fab and assembly	Easier (modular approach)	Easier (modular approach)	Difficult (unitized)
Harness	Simple	Simple	Complex
Mass Properties:			
Roll-Pitch Ratio: Orbiter at separation	1.32 (1.27)*	(1.08)*	(1.5)*
Roll-Pitch Ratio: Probe bus at separation	1.37 (1.35)*	(1.23)*	(0.82)*
Roll-Pitch Ratio: Without large probe and fuel	1.63	1.50	1.10
Roll-Pitch Ratio: Without all probes and fuel	1.80	1.75	1.60
Mass margin, kg (lb)			
Orbiter	91.8 (202.3)	99.8 (220)	67.1 (148)
Probe bus	153.1 (337.6)	162.4 (358)	124.7 (275)

*Magnetometer boom stowed.

TABLE 4-28. MASS PROPERTY COMPARISON
(Three Candidate Configurations)

	HS-339 Mod* 254 cm. Panel				HS-339 Mod 216 cm. Panel				HS-312 (Intelsat IV) Mod			
	Probe Bus		Orbiter		Probe Bus		Orbiter		Probe Bus		Orbiter	
	Kg	Lb	Kg	Lb	Kg	Lb	Kg	Lb	Kg	Lb	Kg	Lb
Subsystems other than structure	113.8	250.8	138.8	305.6	111.2	245.2	137.3	302.7	111.2	245.2	137.3	302.7
Structure	96.4	212.5	83.3	183.7	89.8	198.0	76.7	169.0	127.5	281.0	109.3	241.0
Insertion motor case	-	-	26.7	58.8	-	-	26.7	58.8	-	-	26.8	58.8
Bus total	210.2	463.3	248.8	548.1	201.0	443.2	240.7	530.7	238.7	526.2	273.4	602.7
Large probe	198.7	438.0	-	-	198.7	438.0	-	-	198.7	438.0	-	-
Small probe	169.1	372.9	-	-	169.1	372.9	-	-	169.1	372.9	-	-
Spacecraft subtotal	578.0	1274.2	248.8	548.1	568.8	1254.1	240.7	530.7	606.5	1337.1	273.4	602.7
Contingency	153.1	337.6	91.9	202.5	162.2	357.7	99.7	219.9	124.8	275.2	67.1	147.9
Experiments (bus only)	12.6	27.7	35.0	77.2	12.6	27.7	35.0	77.2	12.6	27.2	35.0	77.2
Spacecraft total (dry)	743.7	1639.5	375.5	827.8	743.6	1639.5	375.5	827.8	741.4	1639.5	375.5	827.8
Propellant	15.9	35.1	28.3	62.4	15.9	35.1	28.3	62.4	15.9	35.1	28.3	62.4
Pressurant	0.1	0.1	0.2	0.4	0.1	0.1	0.2	0.4	0.1	0.1	0.2	0.4
Orbit insertion expendable	-	-	326.9	720.7	-	-	326.9	720.7	-	-	326.9	720.7
Spacecraft total (wet)	759.6	1674.7	730.9	1611.3	759.6	1674.7	730.9	1611.3	759.6	1674.7	730.9	1611.3
Spacecraft adapter	31.3	69.0	31.3	69.0	31.3	69.0	31.3	69.0	31.3	69.0	31.3	69.0
Launch vehicle payload	791.0	1743.7	762.2	1680.3	790.9	1743.7	762.2	1680.3	790.9	1743.7	762.2	1680.3
Roll/pitch ratio at separation	1.35	1.35	1.27	1.27	1.23	1.23	1.08	1.08	0.82	0.82	1.5	1.5

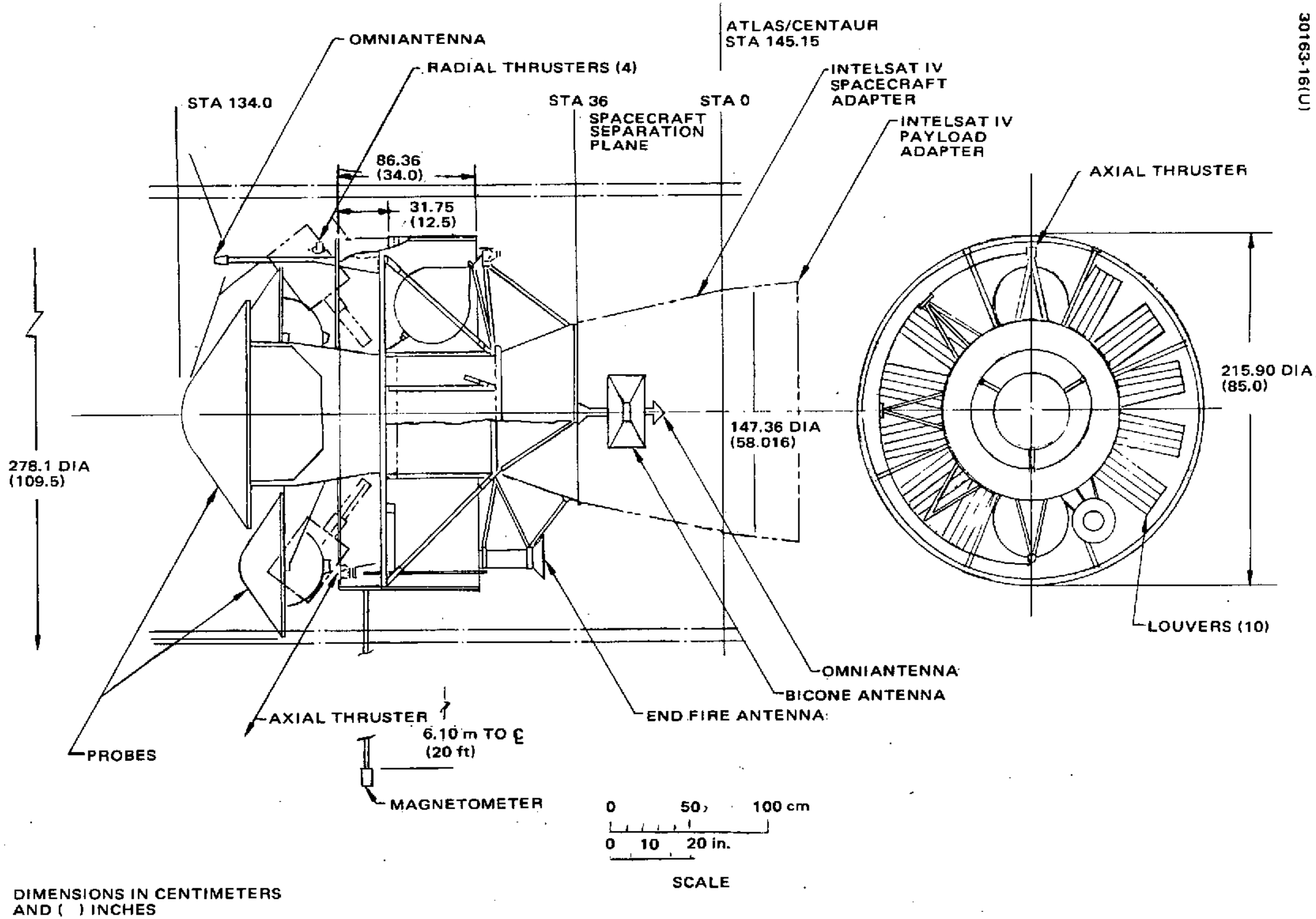
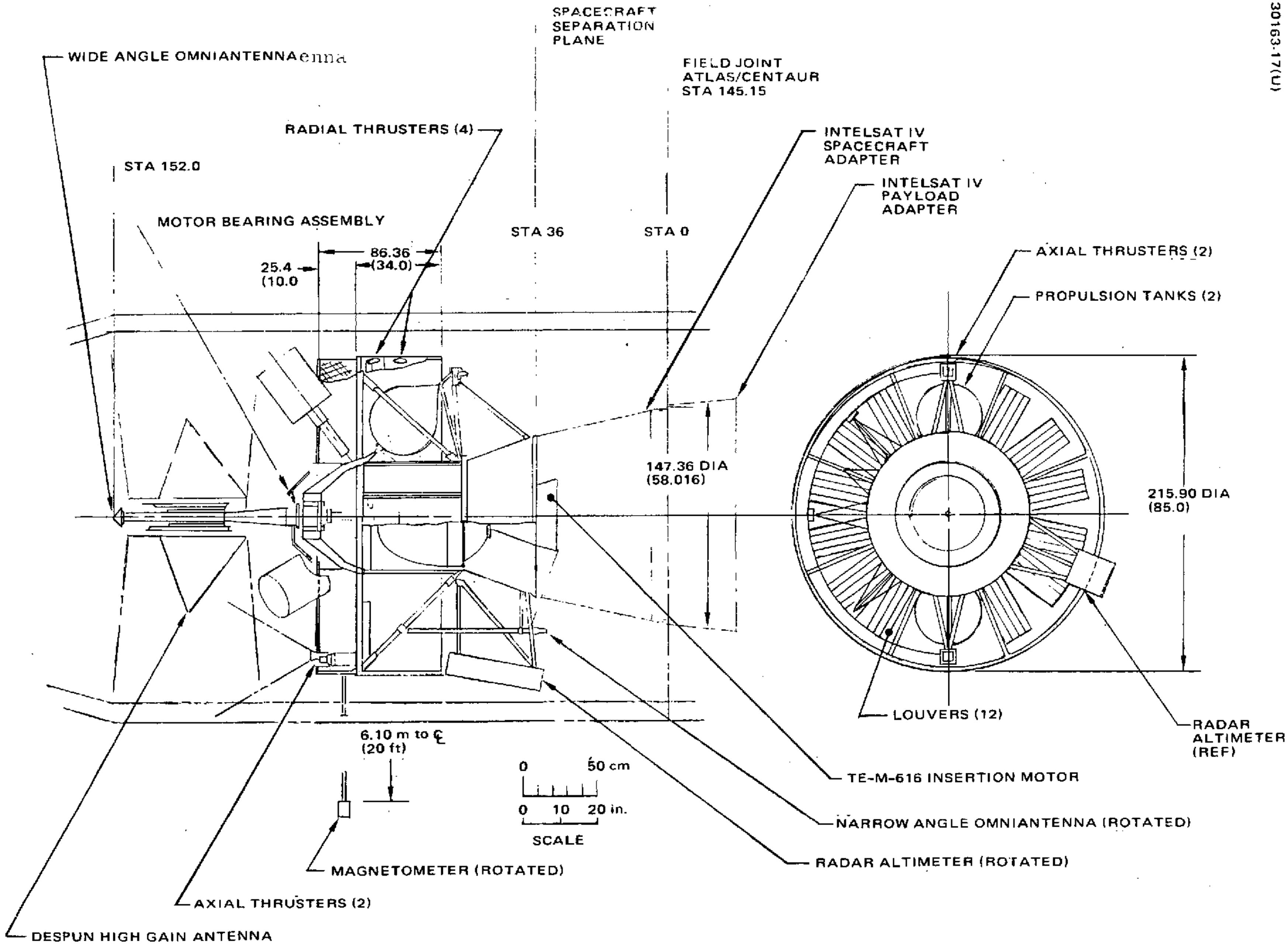


FIGURE 4-16. PROBE BUS-MODIFIED HS-333 WITH 216 CENTIMETER (85 INCH) SOLAR PANEL



DIMENSIONS IN CENTIMETERS
AND ( ) INCHES

FIGURE 4-17. ORBITER-MODIFIED HS 333 WITH 216 CENTIMETER (85 INCH) SOLAR PANEL

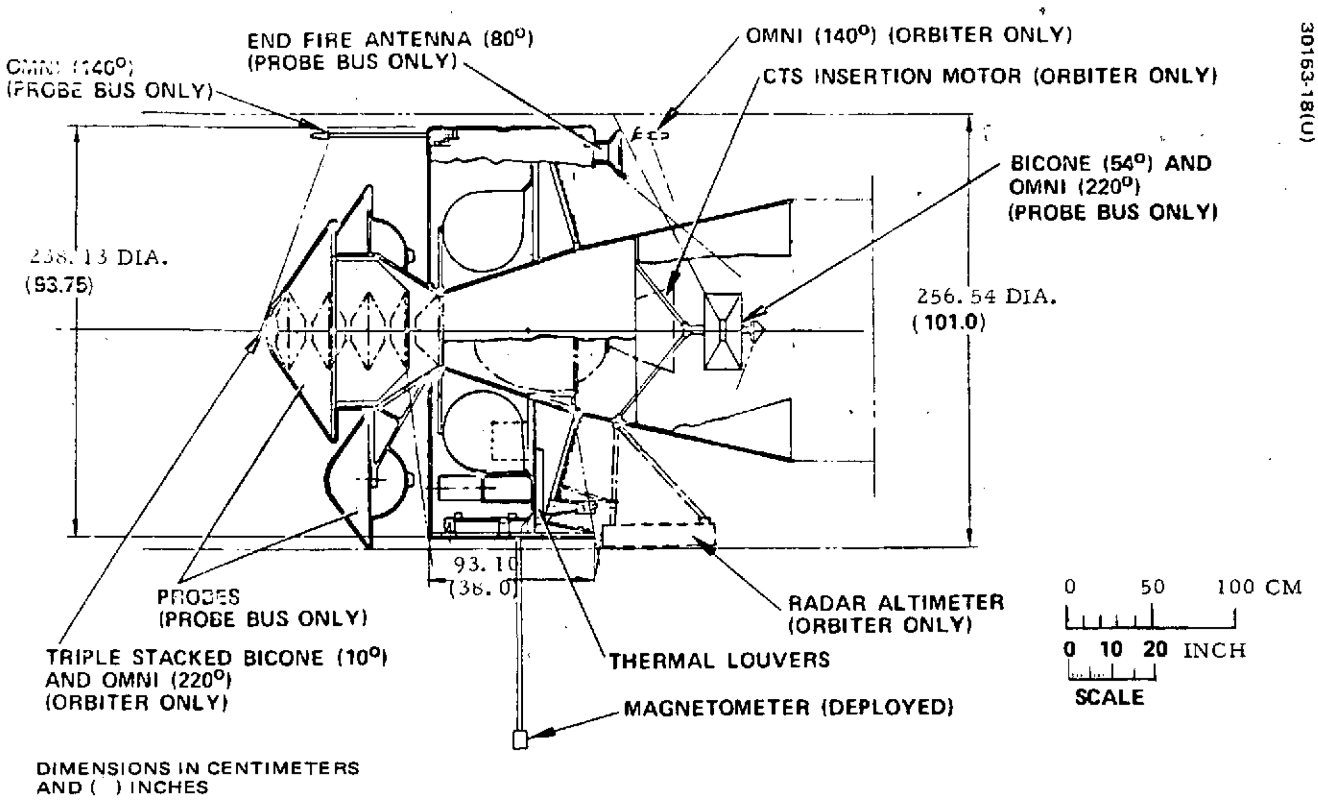


FIGURE 4-18. MODIFIED HS-312 (INTELSAT IV)

Finally, the modified HS-339 version with a 254 cm (100 in.) diameter solar panel is selected as the baseline configuration over the remaining HS-339 modified version with a 216 cm (85 in.) solar panel because the former exhibits better dynamic characteristics (higher roll-to-pitch ratio) and yields higher degree of confidence in thermal control capability while providing more usable shelf area.

Component Derivation

To support the low cost and high reliability (or minimum thermal and structural tests) approach represented in the baseline configuration, each subsystem is broken down to the components of major significance for which either the present qualification status or the extent of modifications required for program use are listed in Table 4-29. As it is seen in the table, the use of newly designed components are minimized to reduce the total cost.

REFERENCES

4-1. R. Prior, "Preliminary System Analysis - Task Number LV20.1," Pioneer Venus Mission Systems Design Study, Hughes Aircraft Company, 1 May 1973. (HS 507-0022-152.)

TABLE 4-29. HS-339 MODIFICATIONS

Subsystem	Item	Probe Bus Description of Modification	Orbiter Description of Modification
Spacecraft adapter and separation group		Use Intelsat IV without changes	
Structure	Thrust end cone	Base o. d. (separation plane) is 112. 39 cm (44. 25 in.) in plane of (37. 75 in.)	
	Separation ring	112. 39 cm (44. 25 in.) o. d. from 95. 89 cm (37. 75 in.)	
	Motor attach ring	Flange angle change to accommodate 112. 39 cm (44. 25 in.) o. d., hole pattern change	
	Thrust tube stiffener, etc.	Delete (8) stiffener; increase skin thickness	
	Shelf support strut and fittings	(8) strut 12. 7 cm (5 in.) longer (8) fitting angle change to accommodate larger shelf diameter	
	Shelf	247. 7 (97. 5 in.) o. d. in place of 209. 6 cm (82. 5 in.) o. d. honeycomb revised insert locations	
	Motor bearing assembly support	Not required	No change. Use as is.
	Solar panel substrate	254 cm (100 in.) o. d. 83. 8 cm (33 in.) long in place of 213 cm (85 in.) o. d. by 168. 9 cm (66. 5 in.)	
	Substrate support (8)	No change	
	RCS tank supports	Provided for two tanks only Modified configurations to support 255577-6 tanks, instead of 255577-1 tanks.	
	RCS thruster and valve supports	New designs to accommodate two axial and four radial thrusters	New designs to accommodate four axial and four radial thrusters
	Omni antenna support	New design	New design
	Mid-gain antenna support	New design	Not required
	Bicone and omni antenna support	New design	Not required
	High gain antenna and omni; antenna mast	Not required	New design
	Magnetometer (sensor unit support) boom	New design	New design
	Probe adapter	New design	Not required
	Radar altimeter supports	Not required	New design
	Despun thermal barrier support	Not required	Similar to HS-339
	Forward (spun) thermal barrier support	New design. Al honeycomb F.G. face sheet; angular disk; extend to probe adapter Provide openings for experiments, sensors, and thrusters.	New design. Al honeycomb F.G. face sheet; angular disk; extend beyond despun thermal barrier. Provide openings for experiments, sensors and thruster

*Common to probe bus and orbiter

Table 4-29. Continued

Subsystem	Item	Probe Bus	Orbiter
		Description of Modification	Description of Modification
Structure (continued)	Insertion motor cavity barrier support	Similar to HS-339; cover entire thrust cone opening; provide openings for bicone and omni antenna supports.	Similar to HS-339; cover opening between thrust cone and insertion motor.
	Aft (spun) barrier support	Not required	Not required
	Insertion motor nozzle (thermal) cover	Not required	New design
Thermal control	Despun barrier	Not required	Similar to HS-333 multilayer super insulation
	Forward spun barrier	New design; multilayer super insulation along inside surfaces of equipment space; between large probe and adapter shelf.	New design; multilayer super insulation along inside surfaces of equipment space.
	Aft (spun) barrier	New design; multilayer super insulation along all surfaces of cavity formed by solar panel substrate and thrust tube and cone assembly, except louver areas.	
	Insertion motor cavity barrier	New design; multilayer super insulation along exterior of motor cavity barrier supports	New design; multilayer super insulation along interior wall of thrust tube and cone assembly and motor surface
	All exposed items including RCS tanks, lines, structural elements, and antennas	New design; wrapped with multilayer super insulation	
	Louver	Same as (10) HS-318 type	Same as (12) HS-318 type
	Footprint doublers	New design; aluminum	
	All exposed surfaces which cannot be wrapped	Provide thermal control surfaces by polishing, painting or other means as necessary	

*Common to probe bus and orbiter

5. SUBSYSTEM DESIGN

This subsection provides a description of the major features of the hardware subsystems and some of the issues concerning accommodation of the scientific experiments. It is primarily directed toward the Atlas/Centaur designs; however, data showing hardware derivation for both the Atlas/Centaur and Thor/Delta versions are contained in each subsystem discussion.

The material in each subsystem description contains hardware requirements and subsystem functional block diagrams. The major hardware characteristics are shown along with a mass summary of each subsystem. Data for the probe bus and orbiter spacecraft are contained as well as the probes. A discussion of major cost saving differences between the two booster designs is contained herein.

Subsection 5.7 lists the mass, power, volume, and data rate requirements imposed by the various experiments. The major experiment integration problems are identified and those requiring particular attention from an integration standpoint are discussed.

5.1 ELECTRICAL POWER SUBSYSTEM

Probe Bus/Orbiter Spacecraft

Table 5-1 lists the primary requirements for the probe bus and orbiter spacecraft. In the case of the probe bus, the solar panel power required is 170 W at Venus and 90 W near earth. The battery energy required is 144 W-hr at probe separation. The orbiter requires 95 W of solar panel power near earth and 193 W in Venus orbit. Battery energy required is 196 W-hr to power the spacecraft loads during the 4-h apoapsis eclipse.

Figure 5-1 is the block diagram for the baseline power subsystem. It is primarily a design derived from the hardware developed for the OSO program. Primary power is supplied by the solar panel while in sunlight. Power is supplied to the loads in the voltage range of 24 to 33 V, the upper voltage being determined by the voltage limiters and the lower by the dual redundant nickel-cadmium (Ni-Cd) batteries. Each battery is serviced by a charge controller to provide battery charging and is diode connected to the bus during discharge. Loads are divided according to mission criticality and connected to one of two busses. The unprotected bus is removed during undervoltage

TABLE 5-1. PIONEER VENUS MISSION POWER SUBSYSTEM REQUIREMENTS

Orbiter mission

- Life - 125 days in-transit (type 1 trajectory)
- Power required
  - Solar array
    - 193 W at Venus
    - 95 W near Earth
  - Battery
    - 196 W-hr (apoapsis eclipse)
- Provide dc power at 24 to 33 Vdc (main bus power)

Multiprobe mission

- Life - 130 days in-transit (bus)
  - Probes
    - 110 days in-transit (deactivated)
    - 20 days free flight (activated)
    - 1.00 to 1.5 hr activated at planet encounter
- Power required, bus
  - Solar array
    - 170 W at Venus
    - 90 W near Earth
  - Battery
    - 144 W-hr (probe separation)
- Provide dc power at 24 to 33 Vdc

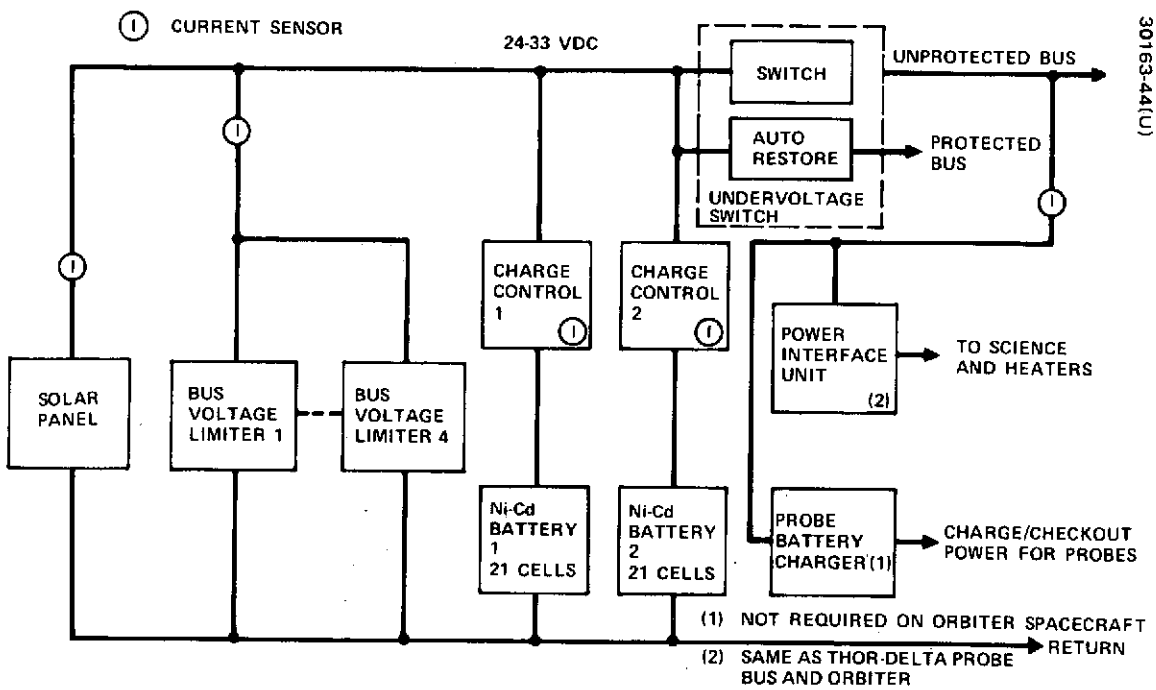


FIGURE 5-1. ATLAS/CENTAUR ORBITER/PROBE BUS POWER SUBSYSTEM

TABLE 5-2. SPACECRAFT POWER SUBSYSTEM CHARACTERISTICS

Unit	Orbiter Mission	Multiprobe Mission
Solar panel		
Power capability, W	193 (EOL)	170 (EOL)
Mass (less substrate), kg (lb)	5.3 (11.6)	4.5 (10.0)
Solar cell type	n-p 2 ohm-cm 2.03 mm (8 mil) thick 15.2 mm covers	Same
Battery type	Ni-cad (2)	Ni-cad (2)
Size, A-hr	7 (14 total)	7 (14 total)
Cells/battery	21	21
DOD, percent	57 (4 h eclipse)	41 (probe separation)
Mass, kg (lb)	14.06 (31)	14.06 (31)
Charge controller	Constant current	Constant current
Power interface unit	Switching/trip	Switching/fuses
Discharge control	Battery diode	Battery diode
Undervoltage switch	Protected bus	Protected bus
Bus limiter	Four shunt (264 W)	Three shunt (198 W)
Probe battery charger	-	Constant Voltage-current limited
Bus voltage range, Vdc	24 to 33	24 to 33
Total power electronics mass, kg (lb)	9.79 (21.6)	8.62 (19.0)
Total power subsystem mass, kg (lb)	29.12 (64.2)	27.21 (60.0)

conditions and requires manual resetting, whereas, the more critical loads are provided with an automatic restore capability. The science loads are serviced by a power interface unit which provides each experiment with a commandable switch for on/off control. The orbiter is provided with commandable circuit breakers for protection, whereas the probe bus protection is implemented by fusing. The probe bus requires a battery charger for charging the probe batteries during the cruise phase.

Table 5-2 lists the characteristics of the power subsystem. Table 5-3 lists the solar panel characteristics. Separate power estimates are shown for the solar panel for radiation degradation and no radiation degradation conditions. Table 5-4 gives a breakdown of the power subsystem masses.

Probes

The major requirements and design features of the large probe are shown in Table 5-5. It features a bus voltage of 28 Vdc ± 2 percent with centralized switching and fusing. The peak power requirement of 245 W is supplied by a silver-zinc (Ag-Zn) battery at an 80 percent depth of discharge. Total energy required is 437 W-hr.

Figure 5-2 shows a block diagram of the system. The battery power is distributed through an unregulated and regulated bus. Unregulated loads include window heaters, planetary flux heater, stepper motors, and the command subsystem timer which is connected to the idle bus. Following the 20 day coast period, signals from the timer enable the boost regulator and the subsystem bus and enable regulated power to be supplied as needed by the loads. The regulator boosts and regulates the voltage to 28 Vdc ±2 percent. Each experiment is provided with a separate switch for operational flexibility and separate fusing for protection.

Table 5-6 and Figure 5-3 shows the requirements, characteristics, and block diagram for the small probe power subsystem. The design is similar, being scaled down by the smaller power requirements and fewer experiment switches.

Major Cost Saving Differences Between the Atlas/Centaur and Thor/Delta Designs

Tables 5-7 and 5-8 show the hardware derivation for the buses and the probes for both the Thor/Delta and Atlas/Centaur designs. A cost saving results from elimination of the discharge regulator from the Atlas/Centaur designs. This is made possible by the use of higher voltage batteries and direct diode coupling on discharge. The existing Telesat battery design is used for the Atlas/Centaur for both the probe bus and orbiter.

In the case of the probes, the primary cost saving results from easing of size and mass constraints on the hardware design.

TABLE 5-3. SOLAR PANEL CHARACTERISTICS

Solar cell	
Cell type	2 x 2 cm, n-p, 2 ohm - cm, 2.03 mm (8 mil) thick
Coverglass	1.52 mm (6 mil) thick, 0211 microsheet
Cell output at 25°C at 1 sun	119.5 mA at 460 mV
Cell adhesive	GE RTV 511/577
Coverglass adhesive	Dow Corning RTV 63489
Interconnect	0.51 mm, chem-etched copper, solder plated

Solar Panel Item	Orbiter	Probe Bus
Total cells	9072	7812
Series X parallel	84 x 108	84 x 93
Power at 28 V		
Earth w/o radiation	107	100
Earth w/radiation	101	95
Venus w/o radiation	213	183
Venus w/radiation	193	170
Panel length, cm (in)	51.0 (20.1)	44.6 (17.6)
Panel weight, kg (lb) (less substrate)	5.75 (12.7)	5.16 (11.4)

TABLE 5-4. SPACECRAFT POWER SUBSYSTEM MASS SUMMARY

Unit	Orbiter kg	Orbiter (lb)	Orbiter Qty	Probe Bus kg	Probe Bus (lb)	Probe Bus Qty
Charge controller	3.4	(7.5)	3	3.4	(7.5)	2
Bus limiters	2.5	(5.5)	4	1.9	(4.2)	3
Current sensors	0.4	(0.9)	3	0.4	(0.9)	3
Power interface unit	2.7	(6.0)	2	0.9	(2.0)	
Batteries (Ni-Cd)	14.1	(31.0)	2	14.1	(31.0)	2
Undervoltage switch	0.7	(1.7)		0.7	(1.7)	
Probe battery charger	-	-		1.3	(2.7)	
Solar panel (less substrate)	5.3	(11.6)		4.5	(10.0)	
Total kg (lb)	29.1	(64.2)		27.2	(60.0)	
Large Probe						
Discharge regulator				3.6	(8.0)	
Power interface unit				2.7	(6.0)	
Pyro switch unit				0.7	(1.5)	
Battery				10.4	(23.0)	
Total kg (lb)				17.4	(38.5)	
Small Probe						
Discharge regulator				2.0	(4.5)	
Power interface unit				0.9	(2.0)	
Pyro switch unit				0.7	(1.5)	
Battery				5.0	(11.0)	
Total kg (lb)				8.0	(19.0)	

TABLE 5-5. LARGE PROBE POWER SUBSYSTEM

Requirements

- 28 Vdc ±2 percent bus to subsystems, experiments
- Centralized switching and fusing
- 245 W peak power
- 437 W-hr energy storage
- 17 power switches

Design

- Battery
  - 30.0 A-hr Ag-Zn (545 W-hr)
  - 80 percent depth of discharge
  - 13 cells (1.4 V per cell)
  - Open circuit stand trickle charge
- Charge controller
  - Constant voltage current limited
- Discharge regulator
  - Boost add-on discharge

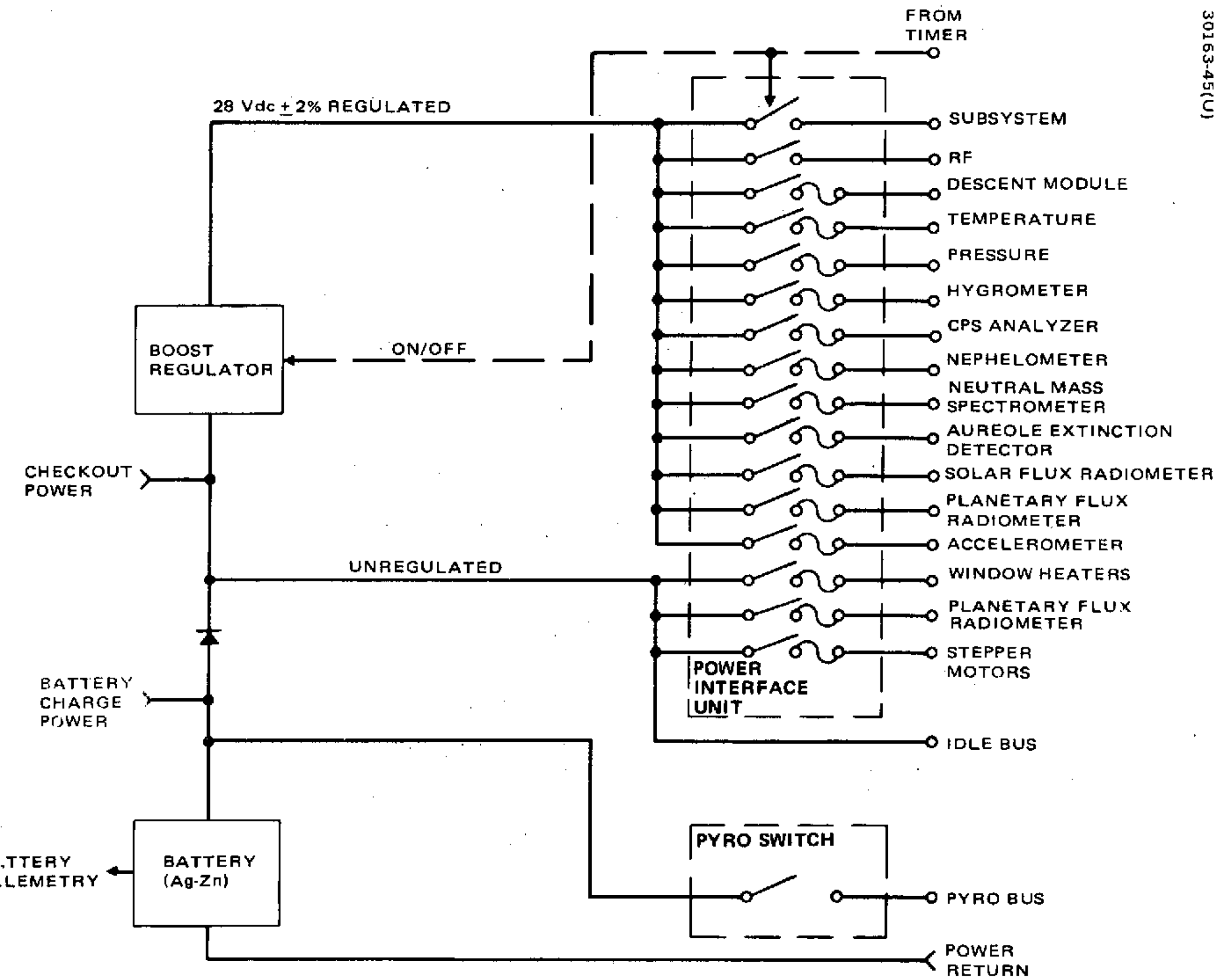


FIGURE 5-2. ATLAS/CENTAUR LARGE PROBE ELECTRICAL POWER SUBSYSTEM

TABLE 5-6. SMALL PROBE POWER SUBSYSTEM

Requirements

- 28 Vdc ±2 percent bus to subsystems, experiments
- Centralized switching and fusing
- 75 W peak power
- 176 W-hr energy storage
- 10 power switches

Design

- Battery
  - 12.0 A-hr Ag-Zn (220 W-hr)
  - 80 percent depth of discharge
  - 13 cells (1.4 V per cell)
  - Open circuit stand/trickle charge
- Charge controller
  - Constant voltage current limited
- Discharge regulator
  - Boost add-on discharge

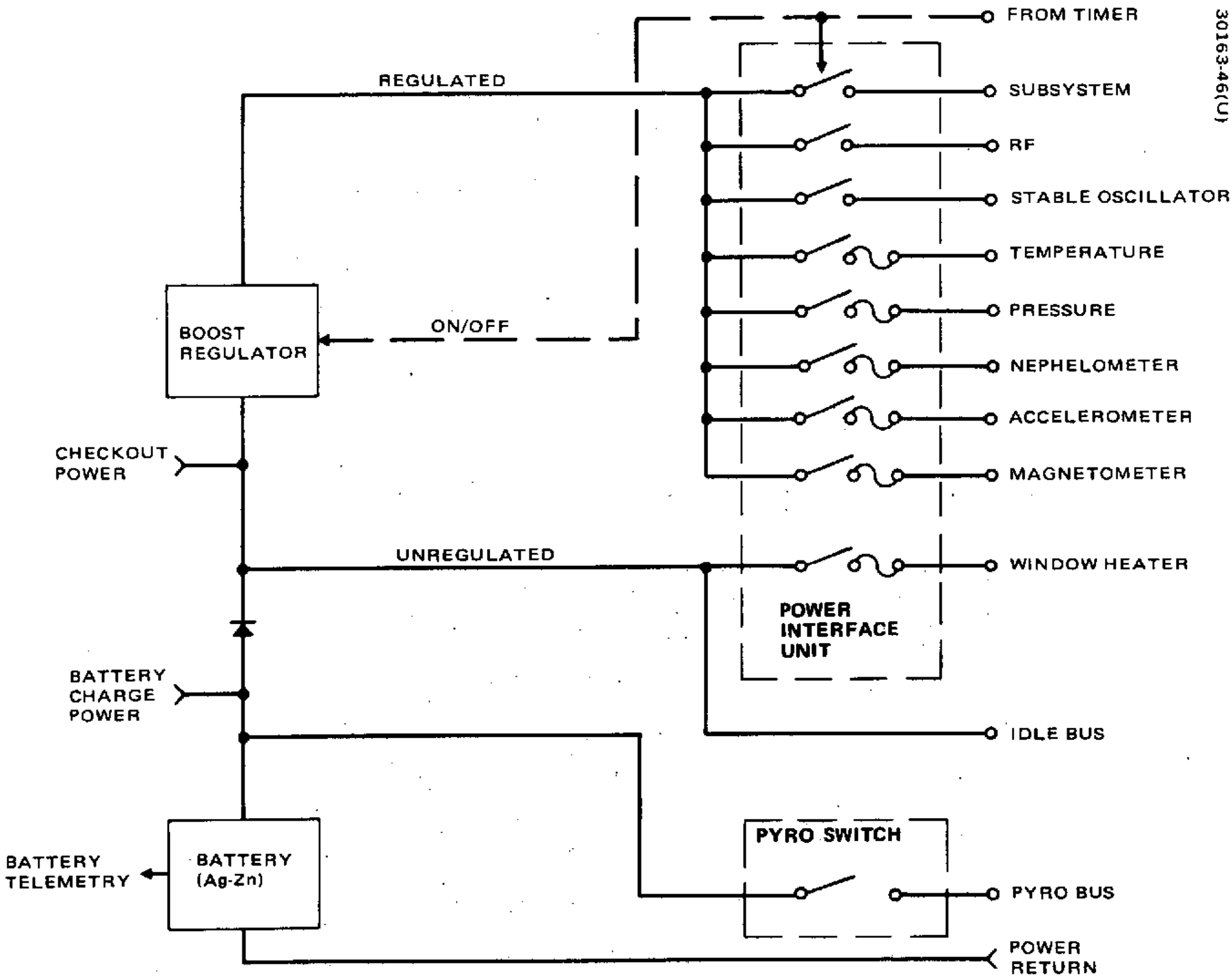


FIGURE 5-3. ATLAS/CENTAUR SMALL PROBE ELECTRICAL POWER SUBSYSTEM

TABLE 5-7. SPACECRAFT POWER HARDWARE DERIVATION

Unit	Thor/Delta		Atlas/Centaur	
	Probe Bus	Orbiter Bus	Probe Bus	Orbiter Bus
Discharge regulator	Classified (40 percent modification and repackage)	Same as probe bus	Same as Thor/ Delta	Same as Thor/ Delta
Charge/discharge controller	OSO (charge portion only) add current limiting for Ag-Zn, 10 percent modification	Same as probe bus (add current limiting for Ni-cad)	OSO-add current limiting for Ni-cad	Same as probe bus
Bus limiters	OSO (5 percent modification)	OSO (5 percent modification)	OSO (5 percent modification)	OSO (5 percent modification)
Undervoltage switch			OSO (5 percent modification change limits)	
Power interface unit	New design (similar to Telesat)	OSO overload control unit (3 percent modification)	New design (similar to Telesat)	OSO overload control unit (3 percent modification)
Current sensors	OSO (3 percent modification range change)	OSO (3 percent modification range change)	OSO (3 percent modification range change)	OSO (3 percent modification range change)
Probe battery charger	Main bus charger used	N/A	New (60 percent circuits from OSO)	N/A
Battery	Silver-Zinc new design (13.6 A-hr, 13 cells)	Ni-cad - like Telesat design (10 A-hr, 18 cells)	Ni-cad - existing Telesat design (7 A-hr) 2 batteries, 21 cells	Same as probe bus
Solar panel	2 x 2 cm cells, 2.03 mm(8 mil) thick, 1.52 mm (6 mil) covers - like Telesat Design	Same, larger than probe bus	Same, larger than Thor/Delta	Same, larger than probe bus

TABLE 5-8. PROBES POWER HARDWARE DERIVATION

Unit	Thor/Delta		Atlas/Centaur	
	Large Probe	Small Probe	Large Probe	Small Probe
Discharge regulator	Classified type - but new design	Modified (reduced) from large probe	Classified type - but new design	Modified (reduced) from large probe
Power interface unit	New design	Modified (reduced) from large probe	New design	Modified (reduced)
Current sensors	OSO (minimum change)	OSO (minimum change)	Oso (minimum change)	OSO (minimum change)
Pyro switch	New design	New design	New design	New design
Battery	Ag-Zn - New design (19.2 A-hr, 13 cells)	Ag-Zn - New design (10 A-hr, 13 cells)	Ag-Zn - new design (33 A-hr, 13 cells)	Ag-Zn - New design (14.4 A-hr, 13 cells)

TABLE 5-9. DESIGN PARAMETERS

Design Ground Rules

- Full mission spacecraft command capability in any attitude
- Near-earth telemetry coverage in any attitude (launch, near-earth, early TCM's)
- Full mission coverage in nominal cruise attitude
- Coverage for unique scheduled situations (probe release, probe spacecraft entry, orbit insertion)

Mission Requirements

- Compatible with deep space network (DSN) configuration specified for 1975-80 period
- Maximum use of 26 m net; 64 m net used only for mission critical events
- Utilize S-band for all telecommunications. Limit X-band to possible radio science applications
- Maximum commonality between telecommunications subsystems on each vehicle

Operability Considerations

- Separate transmit and receive functions as much as possible
- Provide circular polarization for all links for operational simplicity
- Size beamwidths for minimum operational impact

5.2 COMMUNICATIONS SUBSYSTEM

Design Parameters/Characteristics

The primary design and mission requirements are listed in Table 5-9. Continuous command capability is required for the full mission in any attitude. Telemetry coverage is required in any attitude near earth, however, later in the mission, it is required only for nominal mission attitudes.

Maximum use of the 26 m net is required with the 64 m net used for mission critical events. S-band is utilized for all telecommunications and compatibility is required with the deep space network (DSN) configuration. X-band is limited to possible radio science applications. Circular polarization is required for all links for operational simplicity.

Table 5-10 is a summary of the critical parameters. The probe bus, orbiter, and probes utilize the same power amplifier module for commonality. The probe bus and orbiter have two of these modules and can generate various rf powers as needed. The large probe has two of these modules, whereas, the small probe has one. The antenna types, gains and beamwidths are shown and are designed to meet mission needs. The probe bus and orbiter can generate various data rates in the range of 8 to 2048 bps, whereas the probes require generation of only one data rate to satisfy mission needs.

Probe Bus/Orbiter Spacecraft

Table 5-11 is a summary of probe bus antenna and rf power requirements at various points in the mission. During early phases of the mission when communication distances are relatively short, 1 W of rf power and the spacecraft omni antennas are sufficient to serve mission needs. As the mission proceeds and communication distances increase, the bicone antenna and 5 W are used. Finally, at probe entry, the full capability of the system is employed by using the medium gain horn, 10 W and the 64-m net. This provides sufficient data rate capability to serve all science and engineering needs.

Figure 5-4 is a block diagram of the probe bus rf subsystem. Uplink information is received via the narrowband or wideband omni antenna and processed through the diplexers and switches to the preamplifiers/receivers. From the receivers, the baseband command data is supplied to the command subsystem and the tracking data to the exciters. The exciters also receive composite telemetry data from the data handling subsystem. The output of one exciter is supplied to the hybrid and to the power amplifiers where levels of 1, 5, or 10 W of rf power are selected by ground command. After appropriate filtering and switching, the downlink data is provided to one of the four antennas as selected by mission needs for transmission back to earth.

Table 5-12 is a summary of orbiter antenna and rf power requirements and Figure 5-5 is a block diagram of the orbiter rf subsystem. Similar comments apply as for the probe bus. The major subsystem differences is the deletion of one transfer switch due to the different antenna complement. The

TABLE 5-10. CRITICAL PARAMETERS SUMMARY – RF SUBSYSTEM

	Probe Bus	Orbiter	Large Probe	Small Probe
Transmitter power, rf W	1.75/7.0/14.0	1.75/7.0/14.0	14.0	7.0
Preamplifier nf, dB	3.5	3.5	3.5	
System noise temperature, °K	800	600	600	
Antenna type(s)	Omnis/bicone/ medium gain horn	Omnis/MDA	Equiangular spiral	Loop-vee
Antenna gain, dBi	-6/3/18	-6/23.5	4.8	2.7
Antenna beamwidth, deg	-30/ elev/20	-/11	40	40
Navigation	Two-way doppler	Two-way doppler	Two-way doppler	One-way doppler
Downlink modulation	PCM/PSK/PM	PCM/PSK/PM	PCM/PSK/PM	PCM/PSK/PM
Coding/decoding	Convolutional/ sequential	Convolutional/ sequential	Convolutional/ sequential	Convolutional/ sequential
Uplink modulation	PCM/PSK/PM	PCM/PSK/PM	Carrier only	
Data rates, 2^n	8-2048	8-2048	184	16
ERP, dBm (10 W)	33.5/42.7/58.0	32.7/62.4	45.2	–
(5 W)	30.6/39.8/55.1	29.8/59.5	–	41.1
(1 W)	24.6/33.8/49.1	23.8/53.5	–	–
Power, dc W (10 W)	78.8	78.7	74.7	–
(5 W)	45.2	45.1	–	37.9*
(1 W)	20.3	20.2	–	–
Weight, kg (lb)	10.0 (22.05)	9.7 (21.35)	4.85 (10.7)	1.56 (3.45)
Volume, cm^3 (in^3)	8404 (513)	8129 (496)	4193 (256)	1015 (62)

* +2 W during 0.5 h warmup.

TABLE 5-11. PROBE BUS ANTENNA USAGE

Mission Phase	Antenna Usage	Nominal RF Power, W
Launch and acquisition	Omnis/26 m (DSS 42)	1 or 5
Near-Earth	Omnis/26 m	1 or 5
Early (large) midcourse maneuvers	Omnis/64 m	5
Cruise	Bicone/26 m	1, 5, or 10
Later (small) midcourse maneuvers	Bicone/26 or 64 m	5
Probe checkout (prior to probe release)	Bicone/64 m	10
Probe release	Widebeam omni/64 m	10
Probe bus entry	Medium gain horn/64 m	10

TABLE 5-12. ORBITER ANTENNA USAGE

Mission Phase	Antenna Usage	Nominal RF Power, W
Launch and acquisition	Omnis/26 m (DSS 51)	1 or 5
Near-Earth	Omnis/26 m	1 or 5
Early (large) midcourse maneuvers	Omnis/64 m	5
Cruise	MDA/26 m	1 or 5
Later (small) midcourse maneuvers	MDA/26 or 64 m	5
Orbit insertion	Widebeam (despun) omni/64 m	10
Orbital operations	MDA/26 m	5
RF occultations	MDA/26 or 64 m	5

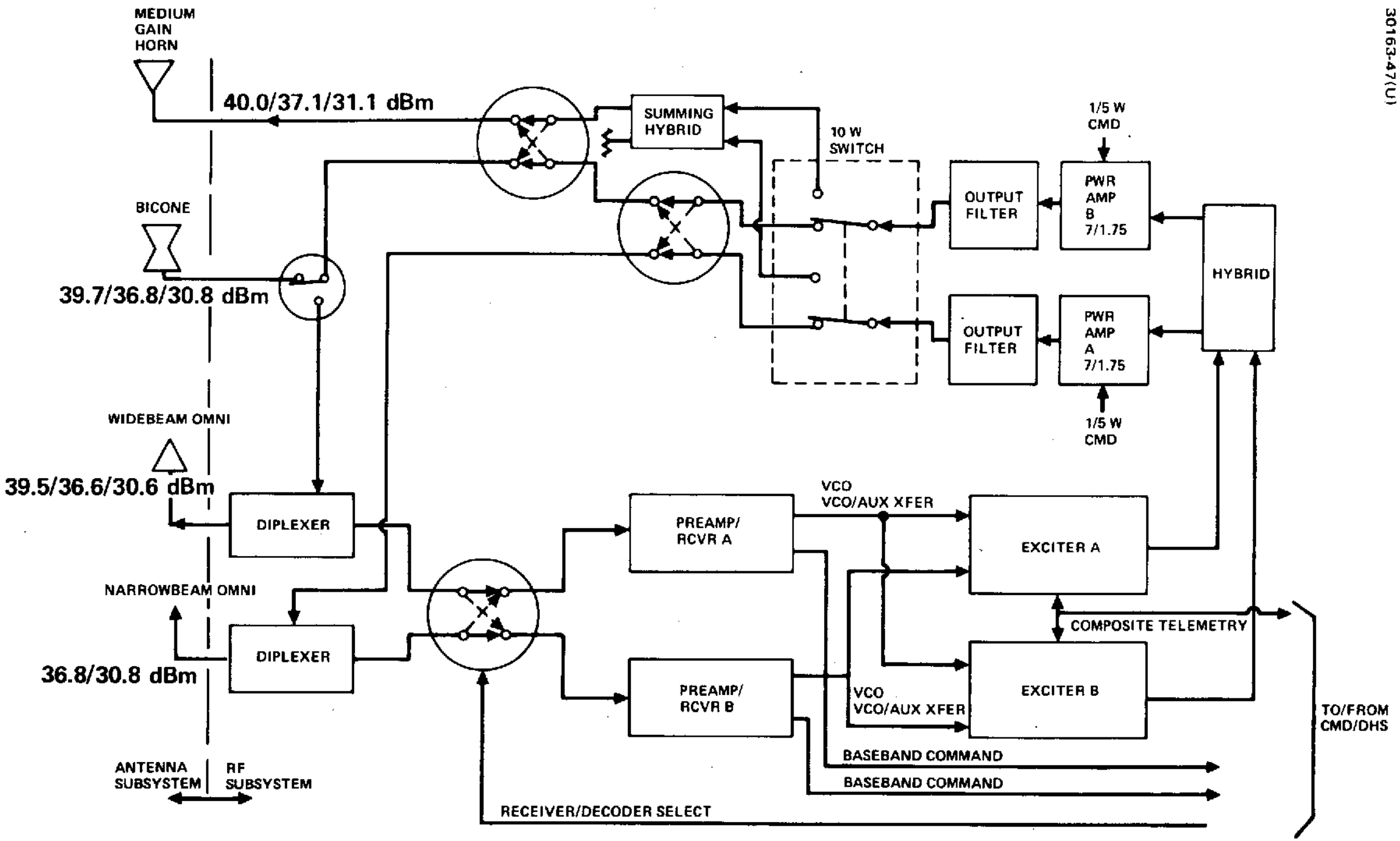


FIGURE 5-4. PROBE BUS COMMUNICATION SUBSYSTEM

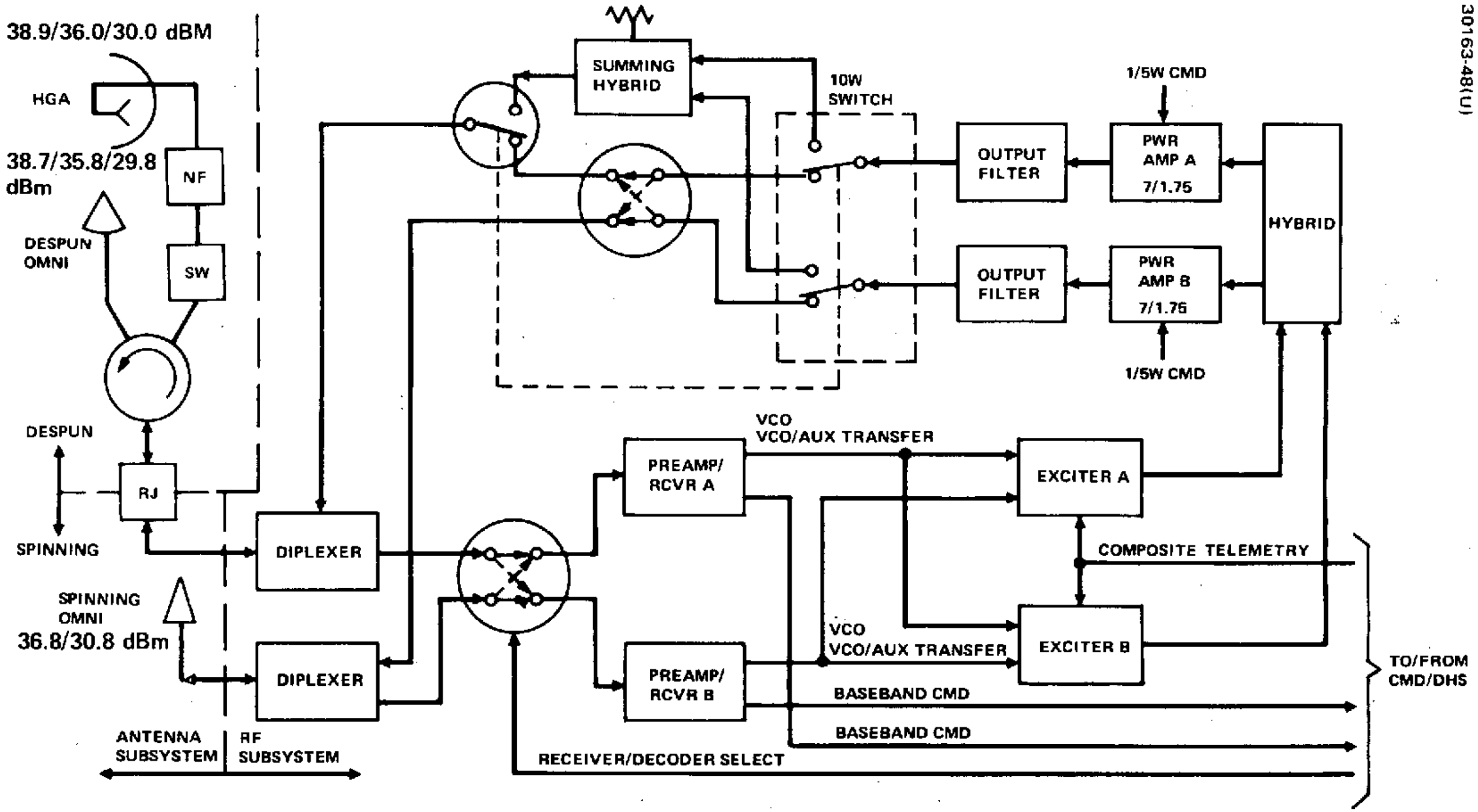


FIGURE 5-5. ORBITER COMMUNICATION SUBSYSTEM

TABLE 5-13. ATLAS/CENTAUR RF SUBSYSTEM

	Unit Description					Probe Bus Totals						Orbiter Totals					
		Mass		Size				Mass		Size				Mass		Size	
	Power, W	kg	lb	cm	in	Qty.	Power, W	kg	lb	cm^3	in^3	Qty.	Power, W	kg	lb	cm^3	in^3
Exciter	4	1.99	4.4	27.3x12.7x6.3	10.75x5x2.5	2	4	3.99	8.8	4424	270	2	4	3.99	8.8	4424	270
Receiver	3						6						6				
Hybrid	-	0.02	0.05	3.81x2.54x1.3	1.5x1.0x0.5	2	-	0.05	0.1	25	1.5	2	-	0.05	0.1	25	1.5
Filter, TxBP	-	0.45	1.0	20.3x5.1x3.7	8x2x1.45	2	-	0.9	2.0	753	46	2	-	0.9	2.0	753	46
Filter harmonic	-	0.05	0.1	10.1x1.3x1.3	4x0.5x0.5	2	-	0.09	0.2	35	2	2	-	0.09	0.2	35	2
Filter, RCBP	-	0.45	1.0	20.3x5.1x3.7	8x2x1.45	2	-	0.9	2.0	753	46	2	-	0.9	2.0	753	46
Circulator isolator	-	0.11	0.25	5.1x5.1x1.9	2x2x0.75	4	-	0.45	1.0	196	12	4	-	0.45	1.0	196	12
SPDT Switch	0.1	0.11	0.25	4.6x4.6x3.8	1.8x1.8x1.5	3	0.3	0.34	0.75	245	15	3	0.3	0.34	0.75	245	15
Transfer Switch	0.1	0.31	0.7	10.7x5.1x5.1	4.2x2x2	3	0.3	0.95	2.1	826	50.4	2	0.2	0.64	1.4	551	33.6
Preamplifier	0.5	0.11	0.25	7.9x3.8x2.5	3.1x1.6x1.0	2	1.0	0.22	0.5	164	10	2	1.0	0.22	0.5	164	10
Coax cables		-	-	0.95	3/8		-	0.73	1.6				-	0.73	1.6		
P.A. low power (1.75 W)	8.7																
P.A. high power (7 W)	33.6	0.68	1.5	20.3x7.6x3.2	8x3x1.25	2	8.7	1.36	3.0	983	60	2	8.7	1.36	3.0	983	60
					1 W mode		20.3 +24.9	10.0	22.05	8404	513		20.2 +24.9	9.69	21.3	8129	469
					5 W mode		45.2 +33.6						45.1 +33.6				
					10 W mode		78.8						78.8				

bicone and medium gain horn is replaced by a despun high gain antenna and rotary joint assembly. Switching is necessary between the despun omni and the HGA at various points in the mission.

The physical parameters of power, mass and dimensions for the probe bus and orbiter are shown in Tables 5-13 and 5-14. Table 5-14 includes data for the large and small probe antennas as well.

Probes

Figure 5-6 is the block diagram for the large probe. Uplink signals are received, amplified, and supplied to the hybrid along with data from the data handling subsystem. These signals provided to the power amplifiers, hybrid, filters, and to the antenna for transmission to earth. Power to the antenna is 40.4 dBm. Table 5-15 gives data on mass, power, volume for the large probe hardware.

Figure 5-7 is a block diagram for the small probe. Only downlink operation is required for the small probe. Table 5-16 gives data on mass, power, and volume.

Major Cost Saving Differences Between the Atlas/Centaur and Thor/Delta Designs

Table 5-17 summarizes the major design differences between the Atlas/Centaur and Thor/Delta designs. It was possible to take advantage of the greater capability of the Atlas/Centaur and select a lower efficiency, heavier, lower cost power amplifier design. In the case of the receiver/exciter, it was possible to use the existing Viking package on the Atlas/Centaur; whereas the Thor/Delta design required modification to reduce mass. Along the same line, the filters for the Atlas/Centaur version are heavier and larger and less costly.

The hardware derivation for the probe bus and orbiter is shown in Table 5-18 and for the probes in 5-19. A high degree of commonality and use of existing equipment results in low cost designs for both the Thor/Delta and Atlas/Centaur spacecraft.

5.3 DATA HANDLING AND COMMAND SUBSYSTEMS

Probe Bus and Orbiter Spacecraft

The requirements for the data handling subsystem are essentially identical for the probe bus and orbiter and are shown in Table 5-20. The major difference is that the orbiter requires data storage for orbital operation and requires additional data formats. The modulation type, data rates, encoding, and word size are identical for both.

Figure 5-8 is a block diagram for the data handling subsystem (DHS). It consists of seven remote multiplexers that sample inputs from 224 channels.

TABLE 5-14. ANTENNA SUBSYSTEM HARDWARE SUMMARY

	Mass		Size	
	kg	lb	cm	in
Bicone[1]	1.7	3.8	45.7 dia x 25.4 high	18 dia x 10 high
Medium gain horn[1]	0.9	2.0	45.7 dia x 66.0 long	18 dia x 26 long
Widebeam omni[1, 2]	0.3	0.6	10.2 dia x 5.1 high	4.0 dia x 2.0 high
Narrowbeam omni[1, 2]	0.2	0.4	5.1 dia x 9.1 high	2.0 dia x 3.6 high
High gain antenna[2]	1.4	3.1	82.6 dia x 45.7 deep	32.5 dia x 18 deep
Circulator[2]	0.1	0.3	5.1 x 5.1 x 1.9	2 x 2 x 0.75
Switch (SPDT)[2]	0.1	0.2	4.6 x 4.6 x 3.8	1.8 x 1.8 x 1.5
Notch filter[2]	0.5	1.0	25.4 x 6.4 x 1.9	10 x 2.5 x 0.75
Equiangular spiral[3]	0.7	1.5	15.2 dia x 5.1 high	6.0 dia x 2.0 high
Loop-vee[4]	0.2	0.5	8.1 dia x 4.3 high	3.2 dia x 1.7 high

1 Probe bus antennas

2 Orbiter antennas

3 Large probe antenna

4 Small probe antenna

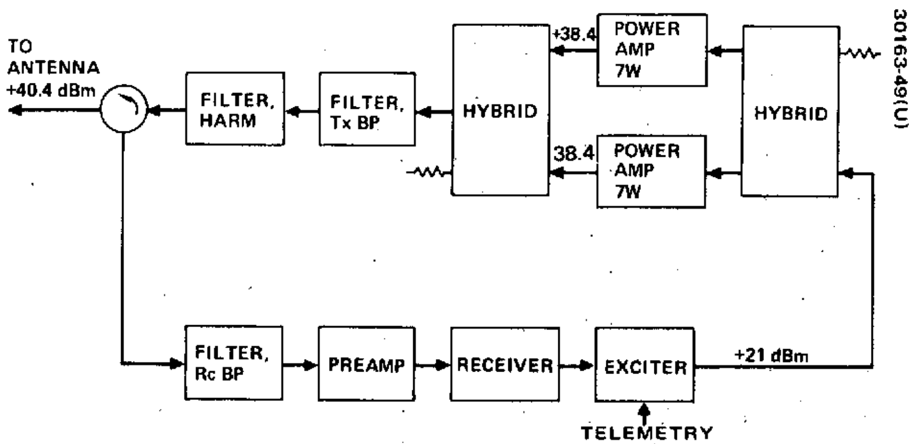


FIGURE 5-6. LARGE PROBE RF SUBSYSTEM

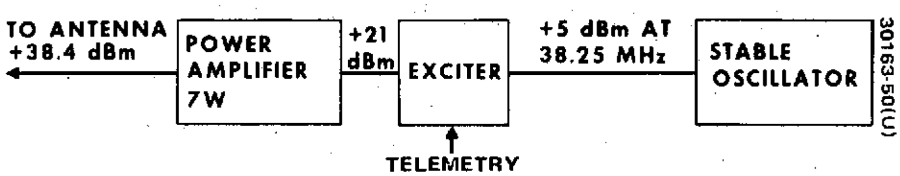


FIGURE 5-7. SMALL PROBE RF SUBSYSTEM

TABLE 5-15. RF SYSTEM – LARGE PROBE

	Unit Description					Totals					
	Power, W	Mass		Dimensions		Qty	Power, W	Mass		Size	
		kg	lb	cm	in			kg	lb	cm^3	in^3
Exciter Receiver	4 3	2.0	4.4	27.3x12.7x6.35	10.75x5x2.5	1	7.0	2.0	4.4	2212	135
Hybrid	--	0.1	0.2	3.8x2.5x1.3	1.5x1.0x0.5	2	-	0.2	0.4	98	6
Filter TxBP	-	0.5	1.0	17.3x5.1x3.2	6.8x2x1.25	1	-	0.5	1.0	376	23
Filter, harmonic	-	0.05	0.1	10.1x1.3x1.3	4.0x0.5x0.5	1	-	0.05	0.1	16	1
Filter RcBP	-	0.4	0.8	17.3x5.1x3.2	6.8x2x1.25	1	-	0.4	1.0	377	23
Circulator	-	0.1	0.25	5.1x5.1x1.9	2x2x0.75	1	-	0.1	0.25	49	3
Preamplifier	0.5	0.1	0.25	7.9x4.1x2.5	3.1x1.6x1.0	1	0.5	0.1	0.25	82	5
Coax cables								0.3	0.7		
Power amplifier	33.6	0.7	1.5	20.3x7.6x3.2	8x3x1.25	2	67.2	1.4	3.0	983	60
							74.7	5.15	11.1	4193	254

TABLE 5-16. RF SUBSYSTEM – SMALL PROBE

	Unit Description					Totals					
	Power, W	Mass		Dimensions		Qty	Power, W	Mass		Size	
		kg	lb	cm	in			kg	lb	cm^3	in^3
Exciter	4.0	0.5	1.0	11.9x8.6x3.8	4.7x3.4x1.5	1	4.0	0.5	1.0	393	24
Stable oscillator	0.25*	0.3	0.75	5.1x5.1x5.1	2x2x2	1	0.25*	0.3	0.75	131	8
Coax cables	-						-	0.1	0.2		
Power amplifier	33.6	0.7	1.5	20.3x7.6x3.2	8x3x1.25	1	33.6	0.7	1.5	491	30
							37.85*	1.6	3.45	1015	1015

*Plus 2 W during 0.5 h warmup

TABLE 5-17. RF SUBSYSTEM DESIGN DIFFERENCES BETWEEN THOR/DELTA AND ATLAS/CENTAUR

	Thor/Delta	Atlas/Centaur
Power amplifiers		
Efficiency, percent	28	21
Mass, kg (lb)	0.5 (1.0)	0.7 (1.5)
Receiver/exciter		
Viking transponder	Modify to orginal Philco mounting configuration concept	Use as is
Mass, kg (lb)	1.8 (4.0)	2.0 (4.4)
Bandpass filter		
Mass, kg (lb)	0.4 (0.8)	0.5 (1.0)
Size, cm^3 (in^3)	279 (17)	376 (23)

TABLE 5-18. SPACECRAFT COMMUNICATIONS HARDWARE DERIVATION

Unit	Thor/Delta		Atlas/Centaur	
	Probe Bus	Orbiter	Probe Bus	Orbiter
High gain antenna	NA	Intelsat IV Modify frequency, beamwidth, feed and mechanical interface	NA	Same as Thor/Delta
Bicone	Data system Modify frequency, beamwidth, and mechanical interface	NA	Same as Thor/Delta	NA
Wide angle omni (conical crossed slots)	Surveyor →	→	→	→
Narrowbeam omni (conical spiral)	HS-350 Modify frequency, and beamwidth →	→	→	→
Medium gain horn	Intelsat IV Modify frequency and beamwidth	NA	Same as Thor/Delta	NA
Rotary joint	NA	Telesat. Modify Frequency, and simplify to single channel	NA	Same as Thor/Delta
Receiver/exciter	Viking. Repackage to reduce weight →	→	Viking. Use as is. →	→
Switches	Pioneer, Helios →	→	→	→
Power amplifier	Classified, increase gain, efficiency, decrease power, weight →	→	Heavier, less efficient than Thor/Delta →	→
Preamplifier	Brazilian Trade intermodulation for noise figure →	→	→	→
Filters				
Bandpass	ATS-E Modify frequency, bandwidth →	→	Larger, heavier than Thor/Delta →	→
Low pass	Telesat Modify frequency	→	→	→
Notch	NA	Telesat Modify frequency	NA	Same as Thor/Delta
Circulator	OSO →	→	→	→

TABLE 5-19. PROBE COMMUNICATION HARDWARE DERIVATION

Unit	Thor/Delta		Atlas/Centaur	
	Large Probe	Small Probe	Large Probe	Small Probe
Equiangular spiral	New (in development)	NA	Same as Thor/Delta	NA
Loop vee	NA	New (in development)	NA	Same as Thor/Delta
Exciter	NA	Skynet	NA	Same as Thor/Delta
Stable oscillator	NA	New	NA	Same as Thor/Delta
Receiver/exciter	Same as Thor/Delta spacecraft	NA	Same as Atlas/Centaur spacecraft	NA
Filters	Same as Thor/Delta spacecraft	NA	Same as Atlas/Centaur spacecraft	NA
Circulator	Same as Thor/Delta spacecraft	NA	Same as Atlas/Centaur spacecraft	NA
Preamplifier	Same as Thor/Delta spacecraft	NA	Same as Atlas/Centaur spacecraft	NA
Power amplifier	Same as Thor/Delta spacecraft	Same as Thor/Delta spacecraft	Same as Atlas/Centaur spacecraft	Same as Atlas/Centaur spacecraft

TABLE 5-20. SPACECRAFT DATA HANDLING FUNCTIONAL REQUIREMENTS

Requirement	Baseline
Modulation type	PCM/PSK/PM
Data rates	8-2048 bps
Formats	
Probe bus	Nine 32-word frames
Orbiter	Twelve 32-word frames
Orbiter storage	362 (393) Kbps
Encoding	Convolutional
Word size	8 bits

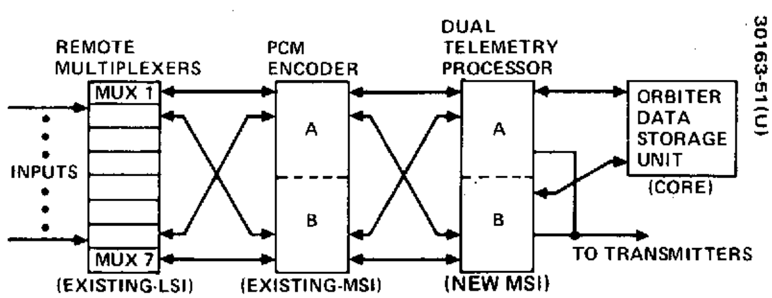


FIGURE 5-8. MODULAR DATA HANDLING SUBSYSTEM

TABLE 5-21. DATA HANDLING SUBSYSTEM HARDWARE

Item	Mass		Power, W	Volume	
	kg	lb		cm^3	in^3
Remote multiplexers	1.4	3.0	<0.35	967	59
Dual telemetry processor	3.3	7.2	5.7	4670	285
Data storage unit	(5.4)	(12.0)	(2.0)	(5309)	(324)
PCM encoders	3.1	6.8	2.6	3506	214
TOTAL	7.8(13.2)	17.0(29.0)	8.65(10.65)	9143(14452)	558(882)

( ) = Orbiter value if different from probe bus.

Timing signals are generated by the PCM encoders and are supplied to the users as needed. The DHS multiplexes analog and digital data inputs at data rates of 8 to 2048 bps. Both serial and bilevel digital data are handled. All analog inputs are converted into 8-bit words.

The DHS encodes, modulates, and controls the amplitude of PCM data and formats the data into a serial bit stream. This bit stream is supplied to the communications subsystem for transmission. Nine 32-word frame formats are generated by the probe bus DHS and 12 by the orbiter. The orbiter DHS provides a data storage memory with a 393 kilobit capability.

The mass, power, and volumes for the DHS units are shown in Table 5-21. Any differences of the orbiter units are noted by the parenthetical numbers.

The requirements for the command subsystem are shown in Table 5-22. Both real time and stored commands are required with initiation by ground control or by timer. The requirement exists for both pulse and magnitude commands. A storage size of 64 words is required for the command memory with an execution timing accuracy of ±1.0 sec.

Figure 5-9 is a block diagram for the command subsystem. The major functions include the provision of circuitry for the selection of the antenna and receiver combination in the absence of an uplink. This combination is interchanged to maximize the probability of uplink acquisition. Once the units are locked on to a valid signal, the subcarrier is demodulated into clock and command data at a 1, 2, or 4 bps data rate. Capability for the storage of command sequences is provided as well as the processing and checking of commands prior to distribution. The distribution of pulse commands and magnitude data to users is accomplished by six remote decoders. The command subsystem also provides power switching to fire pyrotechnic devices. The mass, power, and volumes for the command subsystem units are shown in Table 5-23. Any difference in the orbiter units are noted by the parenthetical numbers.

Probes

The probe data handling requirements are shown in Table 5-24. Both probes employ PCM/PSK/PM modulation and require only one data rate for all mission modes. The formats required are three for the large and two for the small. Storage of data during critical events is handled by a memory whose capacity is 4096 bits for the large probe and 512 for the small.

The command sequencer requirements for the probes are shown in Table 5-25. All events are required to be initiated by stored pulse commands which are time or event actuated. Timer resolution is 1.6 sec for 20 days or 0.25 sec for 30 min with an accuracy of 17.3 sec for 20 days.

Figure 5-10 is a block diagram for the probe data handling and command subsystem. This subsystem multiplexes and encodes at 10-bit accuracy analog and digital data inputs. Data storage is provided for recording data

TABLE 5-22. SPACECRAFT COMMAND FUNCTIONAL REQUIREMENTS

Requirement	Baseline
Command Mode	Real time/stored
Command initiation	Ground/timer/event
Word size	36 bits
Bit rate	1, 2, 4 bps
Modulation	PSK
Probability of false command	1×10^{-9} (Pioneer)
Number of pulse commands	
Probe	159
Orbiter	174
Number of magnitude commands	
Probe	10
Orbiter	9
Magnitude size	10 bits/20 bits
Storage size	64 words
Execution timing	
Real time	Not determined
Stored	$\pm$ 1.0 sec

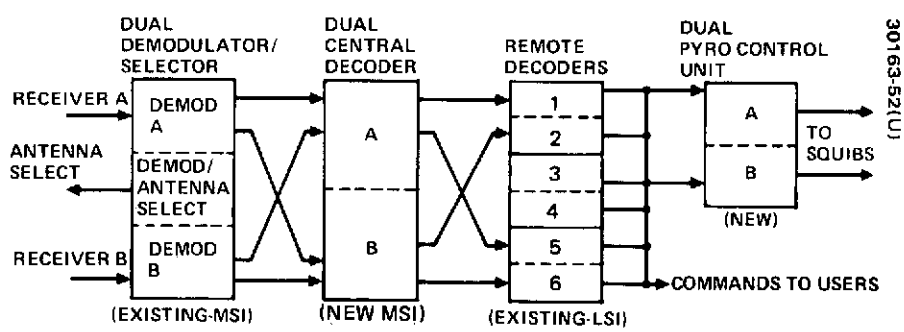


FIGURE 5-9. MODULAR COMMAND SUBSYSTEM

TABLE 5-23. COMMAND SUBSYSTEM HARDWARE

Item	Mass		Power, W	Volume	
	kg	lb		cm^3	in^3
Dual modulator/selector	2.1	4.6	3.0	3081	188
Dual central decoder	4.3	9.4	3.6	5997	366
Remote decoders	1.9	4.2	<0.3	1721	105
Dual pyro control unit	4.1(2.7)	9.0(6.0)	0	6391(4261)	390(260)
TOTAL	12.4(11.0)	27.2(24.2)	6.9	17190(15060)	1049(919)

( ) = Orbiter value if different from probe bus

TABLE 5-24. PROBE DATA HANDLING FUNCTIONAL REQUIREMENTS

Requirement	Baseline
Modulation type	PCM/PSK/PM
Data rates	184 (16) bps
Encoding	Convolutional
Formats	3 (2)
Memory	4096 (512) bits
Word size	10 bits
Subcarrier frequency	4416 (256) Hz

( ) = Small probe value if different from large

TABLE 5-25. PROBE COMMAND FUNCTIONAL REQUIREMENTS

Requirement	Baseline
Command Mode	Stored
Type	Pulse
Initiation	Time or Event
Number of Commands	44* large, 15 small
Number of Events	27* large, 7 small
Total Command Executions	83* large, 25 small
Timer Resolution	1.6 sec/20 days or 0.25 sec/30 min
Timer Accuracy	17.3 sec/20 days

*Includes Mass Spectrometer Events

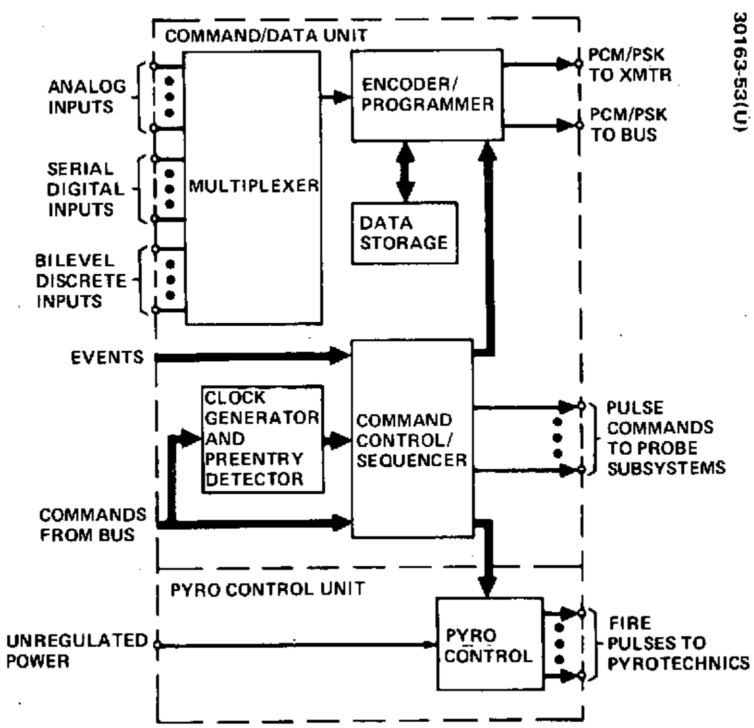


FIGURE 5-10. LARGE/SMALL PROBE COMMAND AND DATA HANDLING

TABLE 5-26. LARGE/SMALL PROBE COMMAND AND DATA HANDLING SUBSYSTEMS MASS, VOLUME, AND POWER SUMMARY

	Mass		Volume		Power, Cruise/Descent, W
	kg	lb	cm^3	in^3	
Large probe					
Data handling unit	2.9	6.4	2720	166	0.06/5.0
Pyro control unit	1.0	2.2	1196	73	-
Total	3.9	8.6	3916	239	0.06/5.0
Small probe					
Data handling unit	2.3	5.0	1852	113	0.06/3.2
Pyro control unit	0.2	0.5	163	10	-
Total	2.5	5.5	2015	123	0.06/3.2

during the blackout phase. Three stored formats are provided for downlink data for the large probe and two for the small. A timer is provided for the preentry sequence and it provides timing to the command control/sequencer during descent. This unit initiates pulse commands from real-time events and stored sequences. The pyro control unit provides power switching to fire pyrotechnic devices.

Table 5-26 shows the mass, volume, and power data for the large and small probe units.

Major Cost Saving Differences Between the Atlas/Centaur and Thor/Delta Designs

The spacecraft data handling hardware derivation is shown in Table 5-27. Most of the units are based on OSO hardware with some modifications. The major difference between the Thor/Delta and Atlas/Centaur designs is in the use of LSI in the telemetry processor. It is felt that not using LSI in the Atlas/Centaur design leads to a lower risk and potentially lower cost approach. Larger volume of Atlas/Centaur units also provides lower costs for productization of the units.

Table 5-28 shows the hardware derivation for the spacecraft command hardware. As in the case of the data handling, it is based largely on OSO hardware with the dual demodulator being from the VO '75 program. The major difference between Thor/Delta and Atlas/Centaur is in not using LSI in the central decoder for the Atlas/Centaur design.

In the case of the probe hardware shown in Table 5-29, major cost savings were obtained by not using LSI and in the simplifying of the packaging of the probe hardware.

5.4 THERMAL CONTROL SUBSYSTEM

Thermal Requirements/Design Conditions

Table 5-30 is a summary of thermal design conditions and solar orientation for the probe bus and orbiter at various points in the mission. During the cruise phase, later trajectory correction maneuvers (TCMs), and orbital operation (orbiter only), the sun angle will be constrained to 90 ± 3 deg and represent less severe design constraints than other phases. During the launch phase and the first TCMs, the sun angle may vary between wide limits as shown and impose more severe design constraints. During probe separation and subsequent bus maneuvers, the sun angle is such that some design constraints are imposed on the spacecraft. During the firing of the orbit insertion motor (orbiter only), the sun shines in the aft end of the spacecraft for up to a 2-h period. The additional sun loading causes a rapid rise in temperature and approaches the design limit temperature at the end of the 2-h period. Table 5-31 is a summary of the temperature limits of the various spacecraft hardware. Since the science instruments will be mounted to the exterior of the spacecraft for the most part to obtain the necessary fields of

TABLE 5-27. SPACECRAFT DATA HANDLING HARDWARE DERIVATION

Unit	Thor/Delta		Atlas/Centaur	
	Probe Bus	Orbiter	Probe Bus	Orbiter
Remote multiplexer	OSO (7 single units)	OSO (7 single units)	OSO (7 single units)	OSO (7 single units)
Dual PCM encoder	OSO (dual unit)	Same as probe bus	OSO (two single units)	Same as probe bus
Dual telemetry processor	New design (uses 70 percent OSO circuits)	Same as probe bus	New design (uses 70 percent OSO circuits; no LSI)	Same as probe bus
Data storage	Not required	Electronic memories-POV (modified design)	Not required	Electronic memories-POV (modified design)

TABLE 5-28. SPACECRAFT COMMAND HARDWARE DERIVATION

Unit	Thor/Delta		Atlas/Centaur	
	Probe bus	Orbiter	Probe Bus	Orbiter
Dual demodulator/ antenna selector (DSS compatible)	Motorola - POV (use 98 percent VO '75 circuits)	Motorola - POV (use 98 percent VO '75 circuits)	Motorola - POV (use 98 percent VO '75 circuits)	Motorola - POV (use 98 percent VO '75 circuits)
Dual central decoder	New packaging (use 50 percent OSO circuits)	Same as probe bus	New packaging (use 50 percent OSO circuits; no LSI)	Same as probe bus
Dual remote decoder	OSO (three dual units)	Same as probe bus	OSO (six single units)	Same as probe bus
Pyro control	New design (three slice unit)	(two slice unit)	New design (use 70 percent OSO circuits; three slice unit)	(two slice unit)

TABLE 5-29. PROBES COMMAND/DATA HANDLING HARDWARE DERIVATION

Unit	Large Probe		Small Probe	
	Thor/Delta	Atlas/Centaur	Thor/Delta	Atlas/Centaur
Command/Data	New Design (PCB and LSI)	New Design (MICAM and no LSI)	New Design (uses 50 percent large probe circuits) PCB and LSI)	New Design (uses 50 percent large probe circuits) (MICAM and no LSI)
Pyro Control	New Design (uses 50 percent bus circuits)	New Design (uses 50 percent bus circuits)	New Design (uses 80 percent large probe circuits)	New Design (uses 80 percent large probe circuits)

TABLE 5-30. THERMAL DESIGN CONDITIONS SOLAR ORIENTATION

Event	Mission	Sun Angle, deg*	Duration	Time Of Occurrence (After Launch)
Launch/ascent	Probe Orbiter	20 to 35 80	1.5 h	--
Cruise	Probe Orbiter	90 $\pm$ 3	Continuous	0 to 20 days before entry 0 to orbit insertion
First TCM	Both	10 to 170	60 min	5 days
Second TCM	Both	90 $\pm$ 3	40 min	20 days
Subsequent TCMs	Both	90 $\pm$ 3	--	--
Orbit insertion motor firing	Orbiter	103.5	2 h	Encounter
Orbit	Orbiter	90 $\pm$ 3	Continuous	--
Large probe separation	Probe	45	3 h	20 days before entry
Small probe separation	Probe	37	4.5 h	20 days before entry
Bus targeting	Probe	38	1 hr	18 days before entry
Preentry cruise 1	Probe	45 to 58	Continuous	18 to 10 days before entry
Preentry cruise 2	Probe	50 to 66	Continuous Variation	10 days to entry

* Measured from spin axis, probe/HGA end

TABLE 5-31 TEMPERATURE LIMITS

Equipment	Temperature			
	Operating		Nonoperating	
	°C	°F	°C	°F
Science				
IR radiometer	- 29 to + 31	- 20 to +110	-40 to +60	-40 to +160
Radar altimeter	TBD	TBD	TBD	TBD
Magnetometer				
Sensor	- 20 to + 20	- 4 to + 68	-40 to +60	-40 to +140
Electronics	0 to + 30	32 to 86	- 4 to +60	- 4 to +140
Solar wind probe	- 29 to + 31	- 20 to +110	-40 to +74	-40 to +165
Ion mass spectrometer	0 to + 50	32 to 122	-10 to +60	14 to 140
Neutral particle				
Mass spectrometer	- 4 to + 52	25 to 125	-37 to +71	-35 to +160
UV spectrometer	TBD	TBD	TBD	TBD
UV fluorescence	TBD	TBD	TBD	TBD
Shelf				
Electronics	+ 5 to + 38	40 to 100		
Batteries	+ 5 to + 32	40 to 90		
Propulsion				
Tanks	+ 5 to + 38	40 to 100		
Thruster valves	+ 5 to + 60	40 to 140		
Catalyst beds	- 7 to + 60	20 to 140		
Lines	+ 5 to + 60	40 to 140		
BAPTA	+ 5 to + 38	40 to 100		
Solar panel	-158 to +135	-250 to +275		
Antennas				
High gain	-158 to +175	-250 to +350		
Bicone	-158 to +110	-250 to +230		

view, etc., they will be designed to a wider range of temperatures than some of the electronic units positioned on the equipment shelf and in the interior of the spacecraft cavity. The solar panels and antennas, however, will be designed to a wide range of temperatures as shown in the table.

Thermal Control Summary

A comparison of the thermal designs for the Atlas/Centaur orbiter and Thor/Delta orbiter is shown in Figure 5-11. The basic design is similar, with the Atlas/Centaur design being more conservative to obtain the greater margins. These greater margins make it more reasonable to delete some of the thermal testing normally required on spacecraft programs resulting in a lower cost design. In both designs, the spin axis is maintained normal to the sunline during cruise and Venus orbit resulting in the simplest design approach. The selected design considers all power/solar variations for the full mission and has adequate margin for all sun angles and eclipse transients.

Low cost overall is achieved by use of flight proven hardware such as the louver modules, heaters, and thermal blankets. Identical or similar hardware has been used on classified programs, Intelsat IV and Telesat.

Passive designs are used on the probes and they are controlled passively while attached to the spacecraft. After separation and prior to entry, they are controlled passively by their own design features. Figure 5-12 is a history of the representative temperatures expected for the large probe during the preentry phase. Starting at a temperature near the battery survival limit, it gradually warms up during the cruise phase. It experiences the coldest temperature shortly after launch during a TCM and experiences the hottest temperature at probe release. After probe release, the probes cool down to about 5°C (40°F) at entry, the desired temperature. This profile exhibits that adequate margin exists at all phases of the mission.

Solar interreflections in the spacecraft cavity limit the lower module capability for off-normal sun angles. The Atlas/Centaur design has capability for adding up to six more louvers on the orbiter and eight on the probe bus should the existing complement appear inadequate. A photometric technique is used in conjunction with a scaled down thermal model to determine spacecraft cavity solar loads.

Table 5-32 is a summary of mass and power requirements for the probe bus and orbiter.

5.5 PROPULSION SUBSYSTEM

Propulsion Requirements

The propulsion subsystem is required to provide impulse for control of spin axis precession, spin rate, and provide velocity increment for trajectory control. Table 5-33 is a summary of requirements for both the probe bus and orbiter spacecrafts.

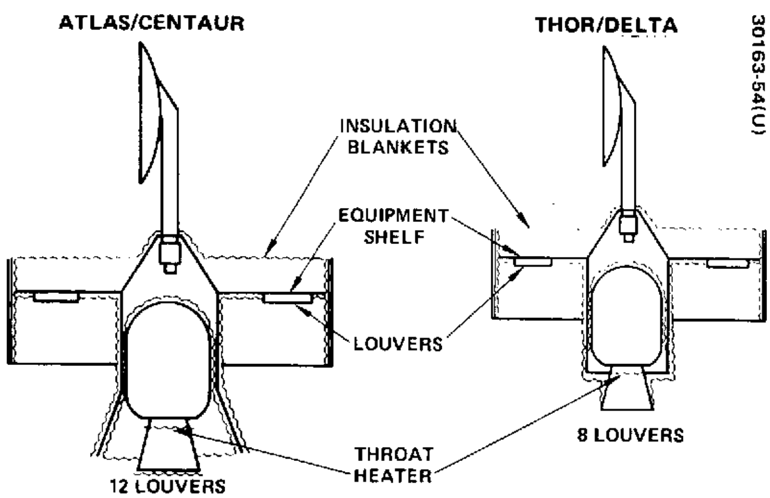


FIGURE 5-11. ORBITER THERMAL DESIGNS

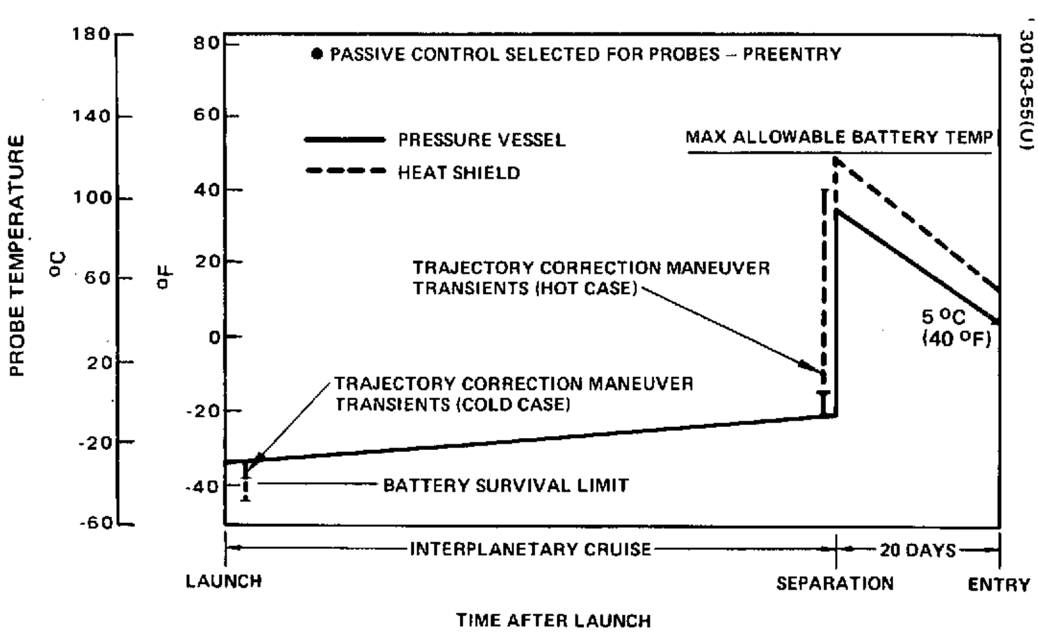


FIGURE 5-12. LARGE PROBE PREENTRY TEMPERATURE HISTORY

TABLE 5-32. SPACECRAFT THERMAL CONTROL PARAMETERS

Element	Orbiter			Probe Bus		
	Mass		Power, Watts	Mass		Power, Watts
	kg	lb		kg	lb	
Louvers[1,2]	3.5	7.8	-	2.9	6.5	-
Shelf doublers	4.5	10.0	-	3.5	7.7	-
Blankets	11.0	24.2	-	11.0	24.2	-
Coatings	0.9	2.0	-	0.9	2.0	-
Heaters	In propulsion		-	In propulsion		-
Thrusters	-	-	5.8	-	-	4.5
Nozzle throat	-	-	10.0	-	-	-
Total	19.9	44.0	15.8	18.3	40.4	4.5

1. 12 louvers for Atlas/Centaur and 8 for Thor/Delta for orbiter
2. 10 louvers for Atlas/Centaur and 6 for Thor/Delta for probe bus

TABLE 5-33. SPACECRAFT PROPULSION REQUIREMENTS

Multiprobe bus
Spin axis precession, 496.5 deg
Spin rate control, 114 rpm
Velocity increment, 36.0 m/sec
Orbiter
Spin axis precession, 855 deg
Spin rate control, 45 rpm
Velocity increment, 130.4 m/sec
Orbit insertion motor
1706.9 m/sec ΔV
724 kg (1596.5 lb) spacecraft

Propulsion Design

The design of the propulsion subsystem is based on existing, flight proven hardware. The thrusters, valves, and filters are from the Intelsat IV program and the tanks are from a military satellite. Figure 5-13 depicts the block diagram for the propulsion subsystem. It is a monopropellant hydrazine subsystem using a blowdown centrifugal feed. It is composed of two propellant tanks of 37,096 cm^3 (2300 in^3) capacity interconnected, via tubing, to latch valves, propellant plenums, filters, and axial and radial thrusters. The probe bus employs two axial and four radial thrusters whereas the orbiter employs an additional axial thruster to satisfy on-orbit attitude control requirements. The nominal thrust level is 22.2 N (5 lbf). The propellant plenums are employed for inital spacecraft spinup. A small amount of propellant is stored here to impart a low spin rate upon separation from the Atlas/Centaur booster. Once a low spin rate is obtained, the main propellant reserve is made available via centrifugal force to complete the spinup maneuvers. This technique was used successfully on four Intelsat IV flights.

A summary of propulsion components and mass summary is shown in Table 5-34.

Orbit Insertion Motor

A summary of orbit insertion motor mission requirements and motor design parameters is shown in Table 5-35. The orbit insertion motor selected is a Thiokol/TE-M-616. The capability of this motor is such that it can, without modification, provide the necessary velocity increment to maximum spacecraft mass (limited by the launch vehicle capability). It is offloaded in propellant by about 2 percent of the maximum load capability. This motor selection satisfies the original program goal to use an existing design to minimize cost, and will result in a flight proven motor. A sketch of the selected motor is shown in Figure 5-14.

5.6 ATTITUDE CONTROL/MECHANISMS

Requirements

Table 5-36 gives a summary of the attitude control subsystem (ACS) requirements. The ACS is required to provide data to enable spin axis determination. Sun angle measurements to an accuracy of ± 0.5 deg is required over the range of sun angles of $15 < \phi \leq 165$ deg from the spin axis and star angle measurements to the same accuracy over a range of $40.5 < \phi \leq 65.5$ deg from the spin axis. An overall accuracy of $\leq \pm 0.9$ deg is required when the spin axis is within 15 deg of ecliptic normal. These requirements provide sufficient accuracies to enable initial spacecraft acquisition, midcourse maneuvers, orbit insertion, probe targeting, and science pointing including a pointing reference for the spinning radar altimeter to ± 1.0 deg.

The ACS must provide attitude and ΔV control for spacecraft maneuvers. It must control the average spin axis attitude to ± 2.5 deg and control

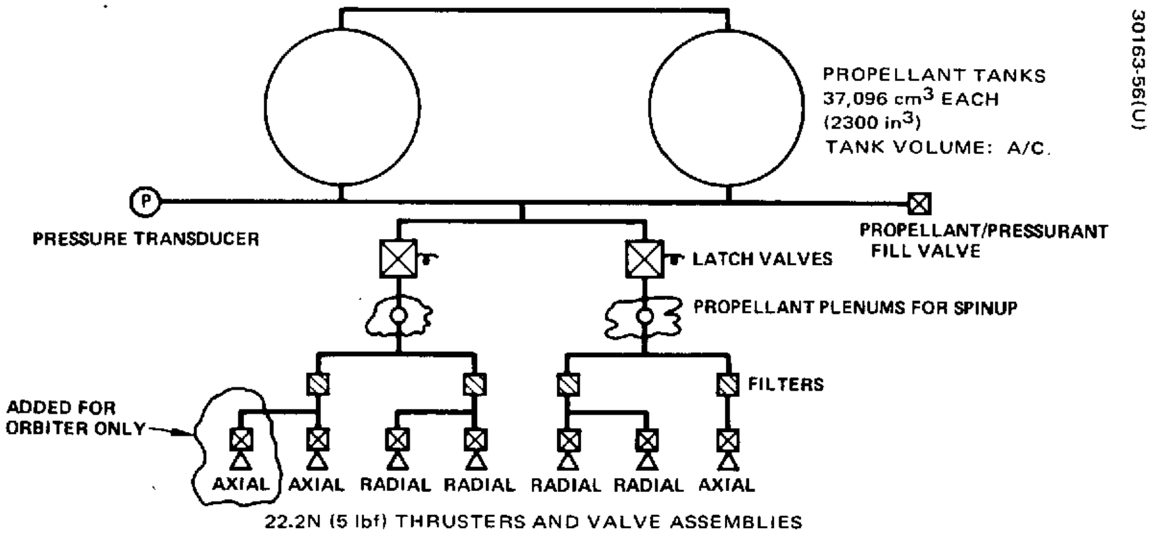


FIGURE 5-13. PROPULSION SUBSYSTEM

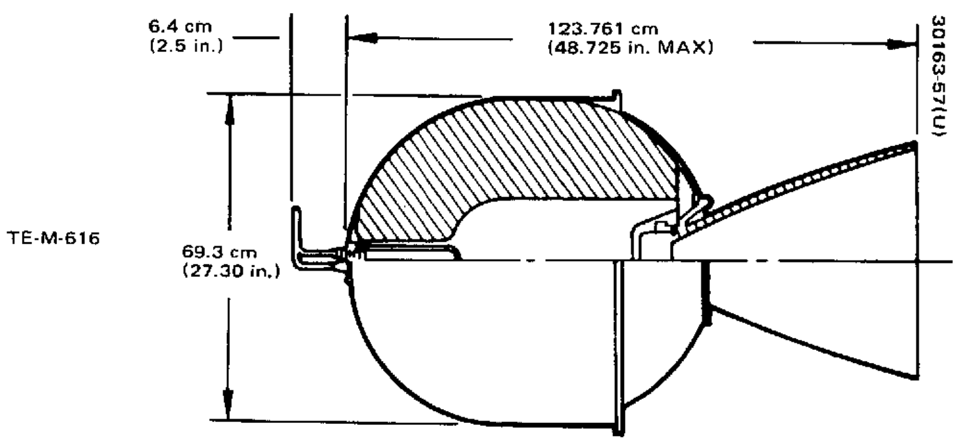


FIGURE 5-14. ORBIT INSERTION MOTOR CONFIGURATIONS

TABLE 5-34. PROPULSION SUBSYSTEM MASS SUMMARY

Components	Probe Bus			Orbiter		
	Units	Mass		Units	Mass	
		kg	lb		kg	lb
Propellant tanks	2	4.3	(9.5)	2	4.3	(9.5)
Thrusters	6	1.6	(3.6)	7	1.9	(4.2)
Propellant valves	6	1.4	(3.0)	7	1.6	(3.5)
Latch valves	2	0.5	(1.2)	2	0.5	(1.2)
Pressure transducer	1	0.2	(0.5)	1	0.2	(0.5)
Fill valve	1	0.1	(0.3)	1	0.1	(0.3)
Filters	4	0.5	(1.2)	4	0.5	(1.2)
Tubing	-	0.3	(0.7)	-	0.5	(1.0)
Fittings	-	0.5	(1.0)	-	0.5	(1.0)
Valve and Catalyst bed heaters	6	0.2	(0.4)	7	0.2	(0.5)
Temperature sensors	-	0.1	(0.2)	-	0.1	(0.2)
Thruster insulation	6	0.5	(1.0)	7	0.5	(1.2)
Propellant plenums	2	0.1	(0.3)	2	0.1	(0.3)
Total subsystem		10.3	(22.9)		11.0	(24.6)

TABLE 5-35 ORBIT INSERTION MOTOR CHARACTERISTICS

Parameter	Performance
Mission requirements	
Thrust, N (lbf)	<39,031 (<8775 (10g))
Propellant and expended inerts, kg (lb)	326.9 (720.7)
Total impulse, N/sec (lb-sec)	9,293,624 (208,930)
Temperature range, ^{o}C (^{o}F)	5 to 30 (40 to 80)
Baseline motor design parameters	
Vendor model	Thiokol/TE-M-616
Maximum thrust, N (lbf)	32,368 (7277)
I_{sp} (effective), sec	289.9
Maximum propellant load, kg (lb)	332.9 (733.9)
Actual propellant load, kg (lb)	326.9 (720.7)
Inerts expended, kg (lb)	3.2 (7.1)
Burnout weight, kg (lb)	26.7 (58.8)
Nozzle expansion ratio	41.9
Total impulse, N/sec (lb-sec)	9,563,630 (215,000)
Operating environmental temperature, ^{o}C (^{o}F)	-7 to 38 (20 to 100)

TABLE 5-36. ATTITUDE CONTROL SUBSYSTEM REQUIREMENTS SUMMARY

Function	Requirement	Derivation
Spin axis attitude determination	Measure sun angle over range $15^\circ < \varphi < 165^\circ$ from spin axis to $\pm 0.5^\circ$ Measure star angle over range $40.5^\circ < \varphi < 65.5^\circ$ from spin axis to $\pm 0.5^\circ$ * Overall < $\pm 0.9^\circ$ including geometry within 15° of ecliptic normal	Initial spacecraft acquisition Midcourse maneuvers Antennae pointing Orbit insertion Probe targeting and separation Science pointing
Attitude/ΔV control	Control average spin axis attitude to $\pm 2.5^\circ$ Control jet thrust vector to ± 4 percent magnitude, $\pm 4^\circ$ direction in inertial space	Same as above Orbit period trim Periapsis altitude adjust
Spin speed control	Vary spin speed over range of 5 to 57 rpm to < ± 0.6 rpm	Cruise attitude stability (25 rpm) Large probe separation (15 rpm) Small probe separation (57 rpm) Experiments (5 rpm)
Nutation damping	< 20 minute time constant at 57 rpm < 45 minute time constant at 5 rpm * Residual nutation 0.1° in 3 hours after any maneuver	Transients induced by maneuvers Science pointing stability
Despun antenna control	Control ℄ of beam to $\pm 2^\circ$ of earthline in spin plane	Mechanically despun antenna on orbiter
Spin angle reference	* Measure and control inertial spin angle reference marks for experiments to $\pm 0.4^\circ$	Probe separation Strobe signal for experiments

* Provides pointing reference for spinning radar altimeter to ±1.0 deg.

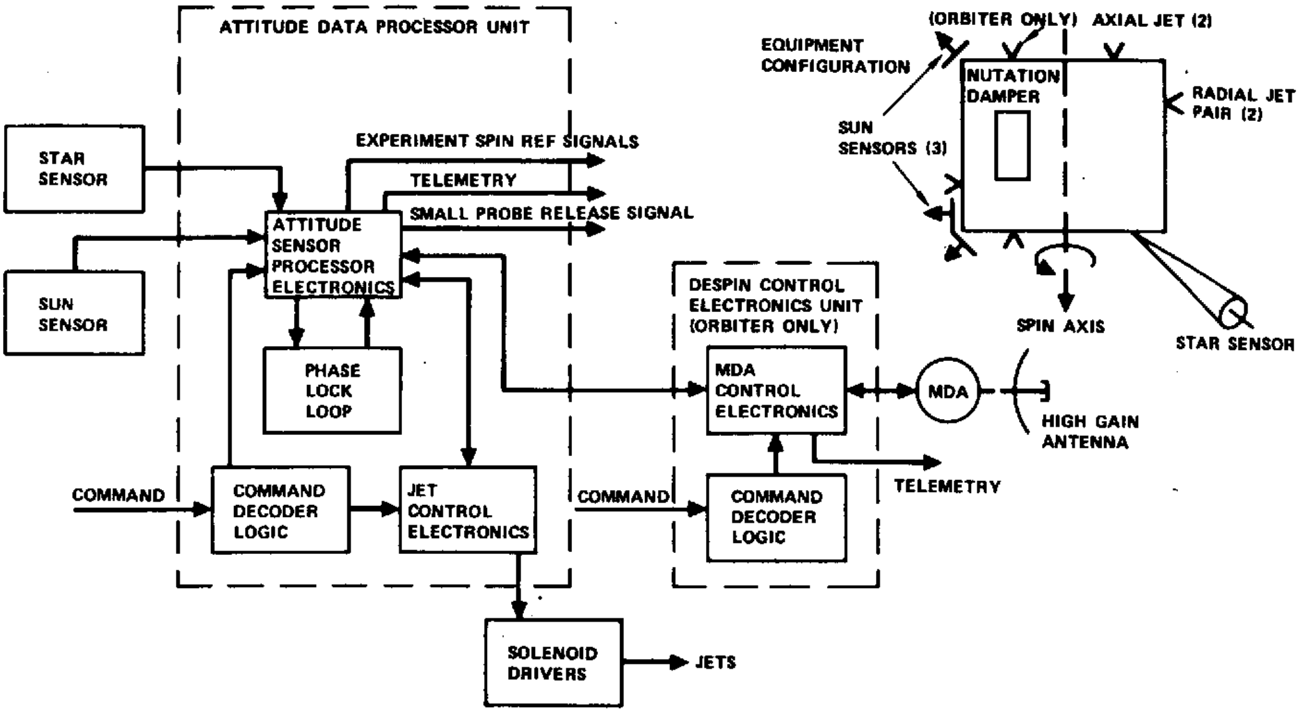


FIGURE 5-15. ATTITUDE CONTROL SYSTEM

jet thrust vector to ± 4 percent in magnitude and ± 4 percent in direction in inertial space. In addition, to providing for the maneuvers noted above, it also provides for orbit period trim and periapsis altitude adjust for the orbiter spacecraft.

Spin speed control is a requirement imposed on the ACS and the spin speed must be varied over a range of 5 to 57 rpm to an accuracy of $<\pm 0.6$ rpm. The range of spin speed provides for cruise attitude stability, large and small probe separation, and science needs.

Nutation damping is required by the ACS to reduce the effect of transients induced by maneuvers and to provide for science pointing stability. A time constant of <20 min is required at 57 rpm and one of <45 min at 5 rpm. In addition, the residual nutation present after any maneuver must be reduced to ≤0.1 deg in 3 h.

Control of the mechanically despun antenna on the orbiter must be provided to control the centerline of the beam to ± 2 deg of earthline in the spin plane. In addition, spin angle reference for such events as probe separation and experiment strobe signals must be provided to an inertial angle accuracy of ± 4 deg.

The magnetic cleanliness requirements of the spacecraft demands that the ACS provide a mechanism for the deployment of the magnetometer. This mechanism is required to provide a 4.4 m (14.5 ft) radial distance between the magnetometer and the outer surface of the solar panel. Provisions for routing wires to the magnetometer are required.

Functional Description of the ACS Hardware

Figure 5-15 is a block diagram of the attitude control subsystem. The ACS uses the sun and stars as attitude references. The sensors employ solid-state detectors for light weight and high reliability. This approach provides the most mission flexibility for least cost and makes maximum use of flight-proven designs and technology. Use of the OSO design photomultiplier tube (PMT) star sensor was considered but was rejected form an overall cost standpoint. The outputs from the sensors go into the attitude sensor processor electronics. Spacecraft commands are provided as needed to the processor. The processor generates spin reference signals for the experiments as well as small probe release signal and telemetry status information. Control signals are provided to the jet control electronics portion which in turn provides signals to control firing time of the propulsion jets (described in subsection 5.5 of this report).

The orbiter employs a mechanically despun antenna (MDA) (described in subsection 5.2 of this report) which requires a despin control electronics unit. The MDA control electronics supplies control signals to the MDA and telemetry status information as derived from processor inputs and commands. The motor employed is a modification of the Telesat motor and is a brushless dc design. This motor design takes advantage of considerable flight experience on the Telesat and Intelsat IV programs. The despun antenna control

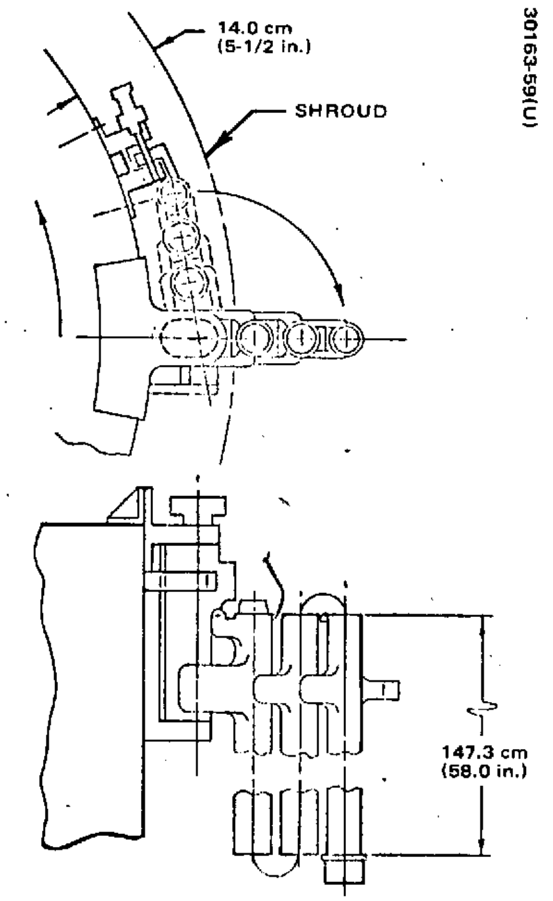


FIGURE 5-16. MAGNETOMETER BOOM LAUNCH POSITION

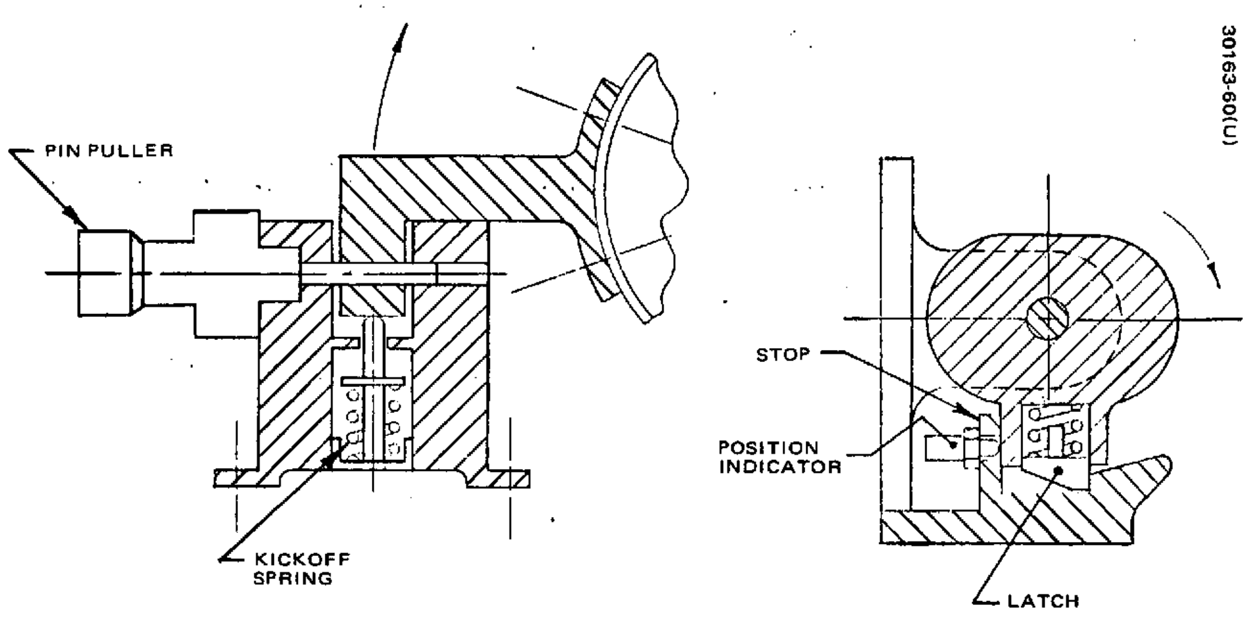


FIGURE 5-17. MAGNETOMETER BOOM RELEASE AND LOCKING MECHANISM

employs a magnetic shaft position encoder with a passive pickup. This latter device has also been used previously on a number of Hughes programs.

Magnetometer deployment is made possible via a three-segment boom assembly, folded and tucked in the 14 cm (5.5 in) clearance between the vehicle shroud and bus solar panel as shown in Figure 5-16. Each of the three segments is 147.3 cm (58 in) in length for a total of 4.4 m (14.5 ft) in the unfolded position. The assembly is released for deployment by actuation of a pyrotechnic pin puller as shown in Figure 5-17. A kickoff spring in conjunction with the centrifugal force rotates and locks the assembly into the deployment position after shroud separation. Centrifugal force deploys the assembly. Positive locking mechanisms are utilized for maintaining boom position after deployment and withstanding subsequent operational environments. The main magnetometer boom hinge and lock mechanism is shown in Figure 5-18 and the other hinge alignment and lock mechanisms are shown in Figure 5-19. The alignment and locking sleeve details are shown. These hinges and locking mechanisms are sized for a peak g-force at end of deployment of 2 g and axial thrust during orbit insertion motor firing of 10 g.

Table 5-37 gives a summary of the physical characteristics of mass size, and power required for the ACS units. All items are needed for the orbiter spacecraft as shown, whereas, the last two items are not used on the probe bus.

Hardware Derivation

A summary of the hardware derivations for the Thor/Delta and Atlas/Centaur configurations are shown in Table 5-38. Both configurations make extensive use of space proven technology and hardware. One major difference leading to a cost saving is the use of the modified Telesat BAPTA for the Atlas/Centaur while using a scaled down Telesat design on the Thor/Delta. The Atlas/Centaur has enough mass margin to allow use of titanium in the BAPTA, whereas, the Thor/Delta mass constraints require the use of beryllium. In addition, the Atlas/Centaur employs a nondeployable bicone antenna on the probe bus, whereas the Thor/Delta uses a deployable antenna because of launch configuration volume restrictions.

5.7 EXPERIMENT ACCOMMODATION

The varied requirements for mass, volume, power and fields of view of the bus and probe science payloads require careful consideration in the configuration layouts of these vehicles. These requirements are satisfied by the Atlas/Centaur design with adequate margins for expansion or growth. Additional science payload over and above the Thor/Delta configuration nominal instruments could be accommodated with relative ease.

The general approach to experiment integration is to identify and resolve all payload accommodation problems by an understanding of the experiment and science requirements and relating them to integration requirements of the multiprobe and orbiter spacecraft. This section of the report summarizes the experiment requirements and discusses those requiring particular consideration from an integration standpoint.

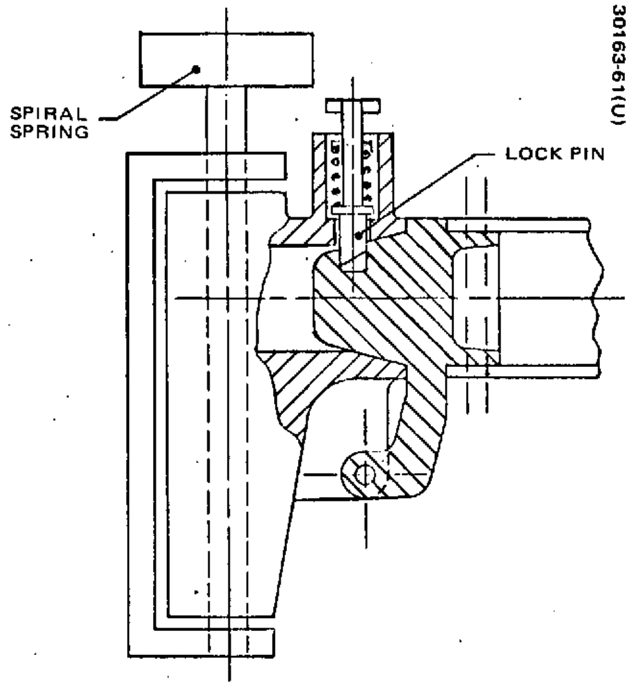


FIGURE 5-18. MAGNETOMETER BOOM HINGE AND LOCK MECHANISM

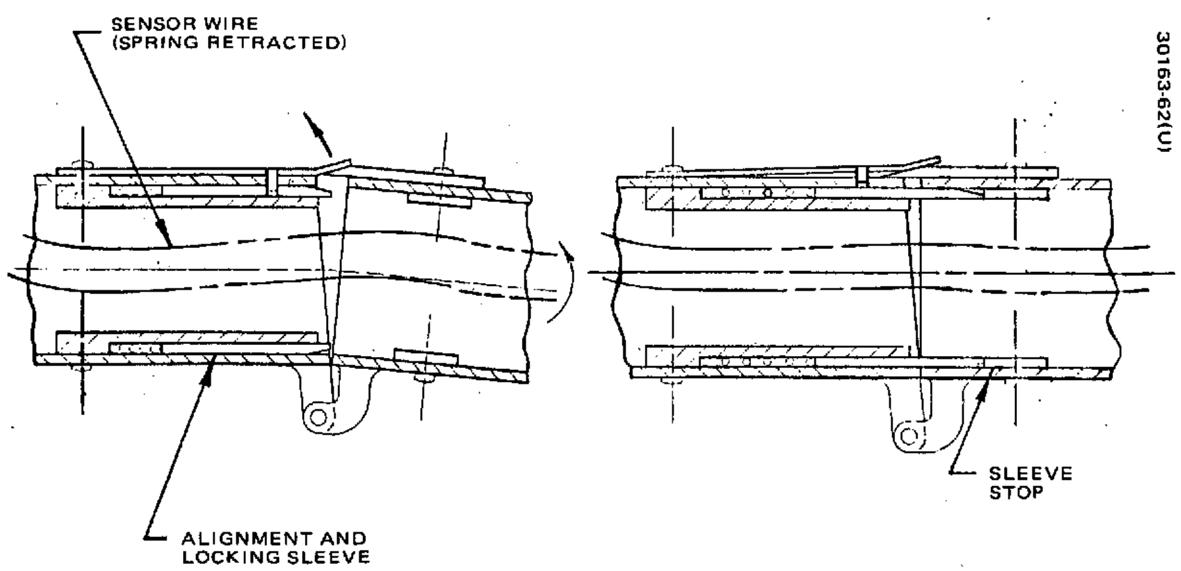


FIGURE 5-19. MAGNETOMETER BOOM ALIGNMENT AND LOCK MECHANISM

3

TABLE 5-37. ACS PHYSICAL CHARACTERISTICS

Component	Number per Spacecraft	Mass		Size		Power
		kg	lb	cm	in	Watts
Sun sensors	3	0.1/unit	0.2/unit	4.1 x 7.1 x 4.8	1.6 x 2.8 x 1.9	-
Star sensors	1			44.5 x 30.5 x 30.5	17.5 x 12.0 x 12.0	1.0
Attitude data processor	2	1.6/unit	3.6/unit	22.9 x 15.2 x 7.6	9.0 x 6.0 x 3.0	4.0
Solenoid drivers (jet and latch valves)	1	1.0	2.3	36.8 x 15.2 x 3.8	14.5 x 6.0 x 1.5	2.0 (1 percent duty cycle)
Nutation damper	1	1.8	4.0	7.6 x 2.6 dia.	30.0 x 1.0 dia.	-
BAPTA (orbiter only)	1	8.8	19.5	20.3 dia. x 30.5 long	8.0 dia. x 12 long	2.0
Despin control electronics (orbiter only)	2	1.9/unit	4.2/unit	25.4 x 15.2 x 7.6	10.0 x 6.0 x 3.0	7.0

TABLE 5-38. CONTROLS HARDWARE DERIVATION

Item	Thor/Delta		Atlas/Centaur		Comments
	Probe Bus	Orbiter	Probe Bus	Orbiter	
Nutation damper	Modified Telesat	Modified Telesat	Modified Telesat	Modified Telesat	Different sizing and fraction fill
BAPTA	N/A	Scaled down Telesat	N/A	Telesat plus modifications	Atlas/Centaur modifications - add slip rings, rotary joints, shaft encoder
Sun sensors	Telesat modified	Telesat modified	Telesat modified	Telesat modified	Modify sensor brackets
Star sensor	New (solid state)	New (solid state)	New (solid state)	New (solid state)	
Solenoid drivers (6 probe bus) (7 orbiter)	Modified Intelsat IV	Modified Intelsat IV	Modified Intelsat IV	Modified Intelsat IV	Some logic changes; add solenoid driver for orbiter
Attitude data processor	Modified MILSAT	Modified MILSAT	Modified MILSAT	Modified MILSAT	
Despin control electronics	N/A	Modified TACSAT	N/A	Modified TACSAT	

Probe Bus

Table 5-39 lists the mass, power, volume and data rate requirements imposed by the probe bus experiments. The numbers in parentheses show the Atlas/Centaur requirements if different from the Thor/Delta. Table 5-40 lists the two instruments imposing the most difficult integration problems. The uv flourescence requires a spin rate near 60 rpm and is satisfied by the 57 rpm spin rate of the baseline design. It desires an orientation perpendicular to the velocity vector and further desires that the spacecraft spin axis should be displaced 10 deg from the spacecraft velocity vector. The 10 deg displacement would place a severe requirement on the communication antenna beamwidth and would result in an aerodynamic problem as well during entry. Relative orientation of the spacecraft to the velocity vector must be adjustable to an accuracy of 1 percent and known to an accuracy of 0.5 deg. The requirement to know rotation rate about the spin axis to 2 percent accuracy is satisfied by the baseline design. The instrument desires entry into the Venus atmosphere on the night side for proper operation. This requirement conflicts with other science requirements for day side and maximum time.

The magnetometer requires tight magnetic control of the spacecraft subsystem hardware on the Thor/Delta design; however, the Atlas/Centaur deploys the magnetometer 5.65 m (18.5 ft) from the spin axis. Deployment of the magnetometer at this distance greatly reduces magnetic control problems and saves a large sum of money on spacecraft magnetic control. Figure 5-20 gives a comparison of Atlas/Centaur and Thor/Delta magnetic control and boom lengths with associated costs showing a $1.4M saving using Atlas/Centaur.

Figure 5-21 is a sketch showing the accommodation of the science instruments on the probe bus. The two experiments discussed above are shown as well as the other three and the communication antenna locations. All fields of view are satisfied without mutal interference between the various devices. The neutral mass spectrometer requires deployment of one of the small probes before its field of view is clear of interference.

Probes

Table 5-41 is a listing of the mass, power, volume, and data requirements of the large probe science instruments. Both Thor/Delta and Atlas/Centaur data are shown. A tabulation of those imposing unique integration considerations is shown as Table 5-42. The accelerometer is presently located at the probe c.g. as desired. In this location, it uses a segment of prime shelf space and requires arranging of other units around it. The accurate alignment required of 0.017 deg is very difficult to satisfy and to acquire with normal alignment techniques. In addition, there exists the possibility of interference with the neutral mass spectrometer. This device is presently located on the lower shelf and requires a 7.6 cm (3 in) penetration. The large volume needed for this device impacts probe size and layout. In addition, the large number of squib drivers required for its inlet operation is a significant impact on probe mass and volume.

TABLE 5-39. PROBE BUS SCIENCE PAYLOAD

Instrument	Mass		Average Power, W	Volume		Data, bps
	kg	lb		cm^3	in^3	
Neutral mass spectrometer	4.9 (5.4)	11.0 (12.0)	5.9 (12.0)	5,735 (8,193)	350.0 (500)	840
Ion mass spectrometer	1.4 (1.5)	3.0 (3.2)	2.0	3,933	240.0	670
Langmuir probe	1.6	3.5	2.0 (2.5)	1,819	111.0	320
Ultraviolet fluorescence experiment	1.4 (1.6)	3.0 (3.5)	2.5 (4.0)	1,966	120.0	240
Magnetometer	2.3 (2.5)	5.1 (5.5)	3.5 (4.0)	4,310	263.0	14
Total science payload	11.6 (12.6)	25.6 (27.7)	15.9 (24.5)	17,763 (20,221)	1084.0 (1,234)	2,084

( ) = Atlas/Centaur value if different from Thor/Delta

TABLE 5-40. PROBE BUS EXPERIMENT ACCOMMODATION CONSIDERATIONS

Experiment	Baseline Design	Comments
Ultraviolet fluorescence	Boom deployed perpendicular to spin axis and 57 rpm	Spin rate near 60 rpm. Velocity vector orientation. Knowledge of spacecraft parameters. Dark side entry.
Magnetometer	Sensor deployed on boom 5.65 m (18.5 ft) from spin axis.	Reduces magnetic control problem.

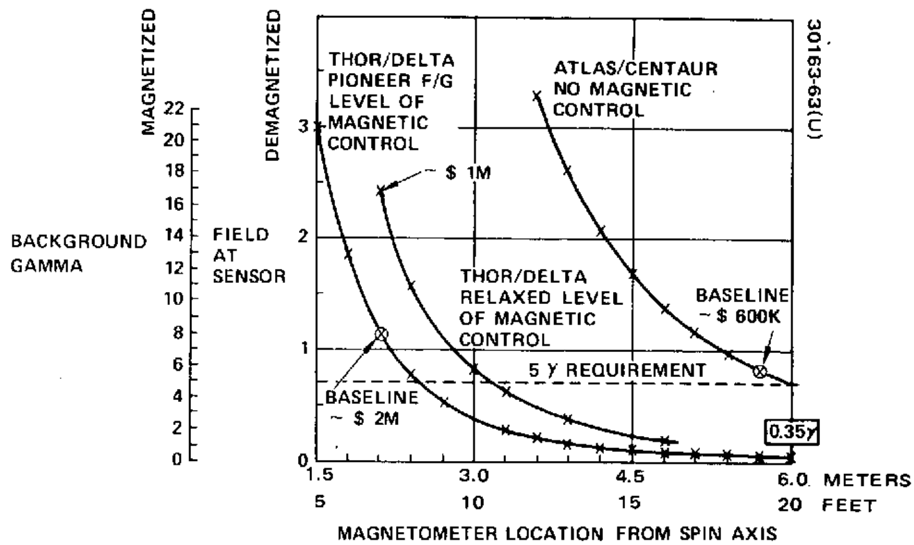


FIGURE 5-20. PROBE BUS AND ORBITER BACKGROUND VERSUS BOOM LENGTH

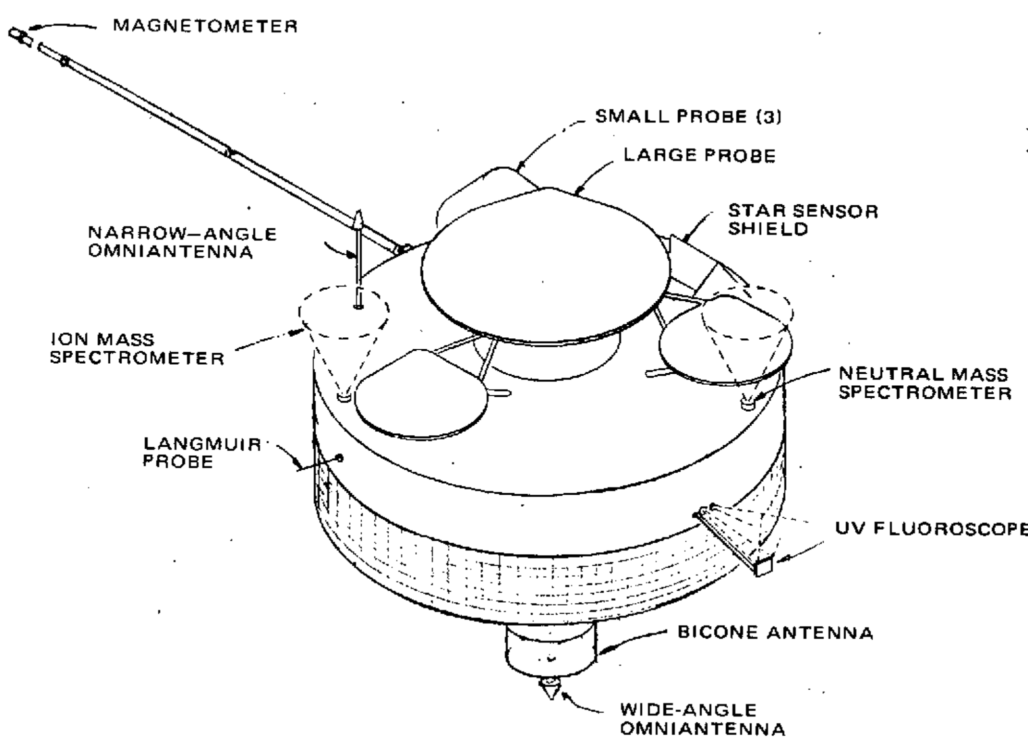


FIGURE 5-21. PROBE BUS SCIENCE ACCOMMODATION

TABLE 5-41. LARGE PROBE SCIENCE PAYLOAD

Instrument	Mass kg	Mass lb	Average Power, W	Volume cm^3	Volume in^3	Data, bps
Temperature sensing system	0.6 (0.7)	1.3 (1.5)	1.0	197	12.0	2.1
Pressure sensing system	0.8 (0.9)	1.8 (2.0)	1.0	229 (262)	14.0 (68)	1.1
Accelerometer	1.1	2.5	2.3 (3.0)	655	40.0	2.6 (1000 storage)
Neutral mass spectrometer	7.7 (9.1)	17.0 (20.0)	12.0 (14.0)	10,650	650.0	42.0
Cloud particle size spectrometer	3.6 (4.5)	8.0 (10.0)	20.0	3,277 (3,932)	200.0 (240)	67.0
Solar flux radiometer	1.8 (2.3)	4.0 (5.0)	4.0 (4.5)	1,966 (2,945)	120.0 (180)	4.0
Planetary flux radiometer	2.3	5.0	4.0	1,966 (2,945)	120.0 (180)	4.0
Aureole extinction detector	1.8 (2.0)	4.0 (4.5)	2.0 (2.5)	1,966 (2,458)	120.0 (150)	25.6
Transponder						-
Nephelometer	1.1 (1.6)	2.5 (3.5)	2.0 (4.0)	1,311 (1,966)	80.0 (120)	10.8
Shock layer radiometer	1.1	2.5	1.0	442 (983)	27.0 (60)	(2100 storage)
Hygrometer	0.5	1.0	0.25	197 (328)	12.0 (20)	2.0
Total science payload	22.4 (26.1)	49.6 (56.5)	49.55 (55.25)	22,856 (27,329)	1395.0 (1,668)	161.2

( ) = Atlas/Centaur value if different from Thor/Delta

TABLE 5-42. LARGE PROBE EXPERIMENT ACCOMMODATION CONSIDERATIONS

Experiment	Baseline Design	Comments
Accelerometers	Located at probe c. g.	Accurate alignment required ($\sim$ 0.017 deg). Possible magnetic interference with mass spectrometer.
Neutral mass spectrometer	Located on lower shelf 7.6 cm (3 in.) penetration	Large volume impacts probe size and layout. Large number of squib drivers impacts probe weight and volume. Location of inlet possible magnetic interference with accelerometer.
Cloud particle size	Located on upper shelf	15.2 cm (6 in.) protrusion impacts probe dynamics. Mirror alignment - stability problem aerodynamic tests.
Experiment windows	Heaters and deployable window	See Table 5-43.

The cloud particle size analyzer, presently located on the upper shelf, requires a 15.2 cm (6 in.) protrusion which impacts probe dynamics. The mirror alignment and stability is anticipated to be a problem that must be verified during aerodynamic tests. The experiments requiring heaters and viewing windows impose constraints on probe design. These are enumerated in Table 5-43. This table lists the window sizes, present material design, and power required for heating for both the large and small probes. Six windows are required for the large and one for the small. The general configuration of the experiments of the large probe are shown in Figure 5-22. This sketch shows the protrusions required and viewing angles for the various science devices.

Table 5-44 lists the data for the small probe science instruments and Table 5-45 summarizes the main integration considerations. As in the large probe, the accelerometer location and alignment requirements impose constraints on probe layout and assembly. The magnetometer, located in the upper hemisphere, imposes severe magnetic control problems on the probe hardware design. The nephelometer requires a heated window and ejectable window cover as noted previously in Table 5-43. The location and some internal mounting details of the small probe science instruments are shown in Figure 5-23.

Orbiter

The orbiter science payload data is listed on Table 5-46. Those instruments imposing unique requirements are in Table 5-47. As in the case of the probe bus, the magnetometer is deployed radially at a distance of 5.65 m (18.5 ft) from the spin axis. In this position, the magnetometer and boom has an impact on bus dynamics. In addition, it must undergo a 10 g acceleration and bending moment at the time of orbit insertion motor firing. The use of this magnetometer boom greatly relieves the magnetic control problems of the spacecraft hardware and reduces program cost significantly.

The S-band radar altimeter employs an electronically scanned array antenna. The large antenna volume requires mounting on the aft end of the solar panel to avoid interference with the main despun communication antenna. Mounting on the forward end (as in the Thor/Delta design) requires extending the mast of the despun antenna to provide pattern clearance above the altimeter antenna. Figure 5-24 is a sketch showing the viewing requirements and general layout of the orbiter science instruments. Shown also for reference are the communication antennas and the star sensor sun shield.

TABLE 5-43. EXPERIMENT WINDOWS

Experiment	Number of Windows	Size, cm (in.)	Material	Remarks
		Large Probe		
Cloud particle size	1	3.6 (1.4)	Sapphire	9 W coil/jettisoned outer
Planetary flux	1	3.6 (1.4)	(CVD) Zn Se	18 W coil/jettisoned outer
Solar flux	1	6.4 x 2.5 (2.5 x 1.0)	Sapphire	22 W coil/jettisoned outer
Aureole	1	5.1 (2.0)	Sapphire	3 W film/jettisoned outer
Nephelometer	2			
Source		0.9 (0.35)	Sapphire	2 W coil/jettisoned outer
Detector		3.5 (1.4)	Sapphire	9 W coil/jettisoned outer
		Small Probe		
Nephelometer	1	3.5 (1.4)	Sapphire	9 W coil/removable cover

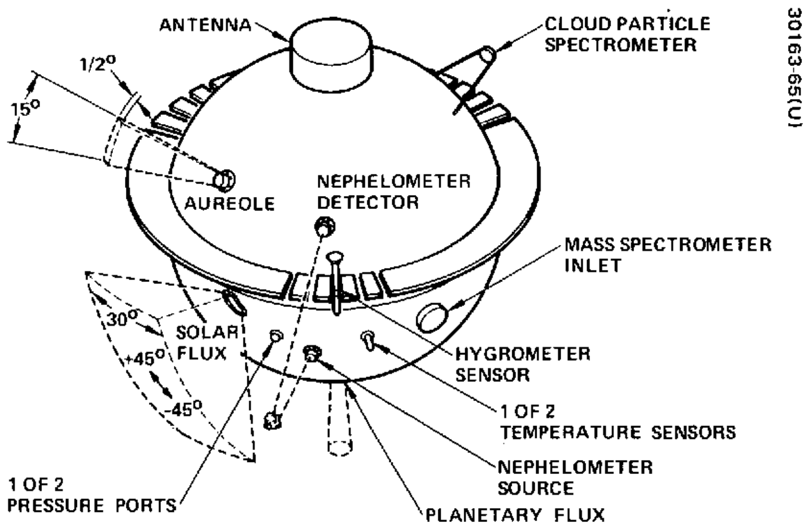


FIGURE 5-22. LARGE PROBE EXPERIMENT CONFIGURATION

TABLE 5-44. SMALL PROBE SCIENCE PAYLOAD

Instrument	Mass		Average Power, W	Volume		Data, bps
	kg	lb		cm^3	in^3	
Temperature sensing system	0.3 (0.4)	0.75 (0.9)	0.5	98 (164)	6.0 (10)	0.6
Pressure sensing system	0.4	0.9	0.5 (0.75)	115 (164)	7.0 (10)	0.6
Nephelometer	0.5 (0.6)	1.0 (1.4)	1.0 (2.0)	377 (575)	23.0 (35)	1.8
Accelerometer	0.2	0.4	1.0	33 (49)	2.0 (3)	1.0 (100)
Magnetometer	0.5 (0.5)	1.1 (1.2)	1.0 (1.2)	213	13.0	0.9
Stable oscillator	0.3 (0.7)	0.75 (1.5)	0.25 (2.0)	131 (492)	8.0 (30)	0.1
Total science payload	2.2 (2.8)	4.9 (6.3)	4.25 (7.45)	968 (1,650)	59.0 (101)	5.0 (100)

( ) = Atlas/Centaur value if different from Thor/Delta

TABLE 5-45. SMALL PROBE EXPERIMENT CONSIDERATIONS

Experiment	Baseline Design	Comments
Accelerometer	Located at probe c. g.	Accurate alignment required (~0.017 deg). Installation location.
Magnetometer	Located in upper hemisphere	Magnetic control problems
Nephelometer	Window heater cover ejected	See Table 5-43.

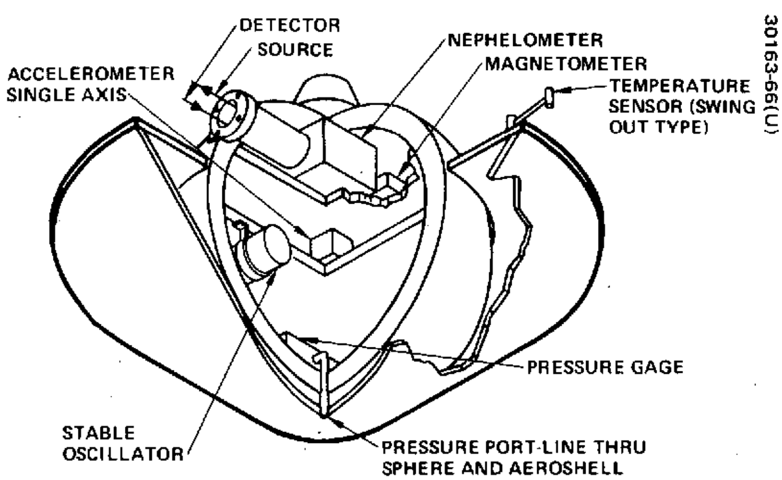


FIGURE 5-23. SMALL PROBE EXPERIMENT ACCOMMODATIONS

TABLE 5-46. ORBITER SCIENCE PAYLOAD

Instrument	Mass, kg	Mass, lb	Average Power, W	Volume, cm^3	Volume, in^3	Data, bps
Magnetometer	2.3 (2.5)	5.1 (5.5)	3.5 (4.0)	4,309	263.0	3
Langmuir probe	1.6	3.5	2.0 (2.5)	1,819	111.0	32
Neutral mass spectrometer	4.5 (5.4)	10.0 (12.0)	12.0	8,194	500.0	17
Ion mass spectrometer	1.4 (1.5)	3.0 (3.2)	1.0 (2.0)	3,277	200.0	14
Ultraviolet spectrometer	5.4	12.0	8.0	9,832	600.0	14
Infrared radiometer	4.1 (4.5)	9.0 (10.0)	6.0	6,554	400.0	7
S-band radio occultation	-	-	-	-	-	-
S-band radar altimeter	9.1	20.0	12.0	28,218	1722.0	300 to 400 km(1) 5 to 1000 km
Solar wind probe	2.7 (5.0)	6.0 (11.0)	4.0 (5.0)	3,932 (5,506)	240.0 (336)	-
Total science payload	31.1 (35.0)	68.6 (77.2)	48.5 (51.5)	66,135 (67,709)	4036.0 (4132)	387(1) 92

( ) = Atlas/Centaur value if different from Thor/Delta

(1) Higher data rate required for the 300 to 400 km region

TABLE 5-47. ORBITER EXPERIMENT ACCOMMODATION CONSIDERATIONS

Experiment	Baseline Design	Comments
Magnetometer	Sensor deployed on boom	Boom impacts bus dynamics. Relieves magnetic control problems.
S-band radar altimeter	Electronic antenna (mechanically scanned dish antenna)	Large antenna volume requires mounting on aft end to avoid interference with main communication antenna

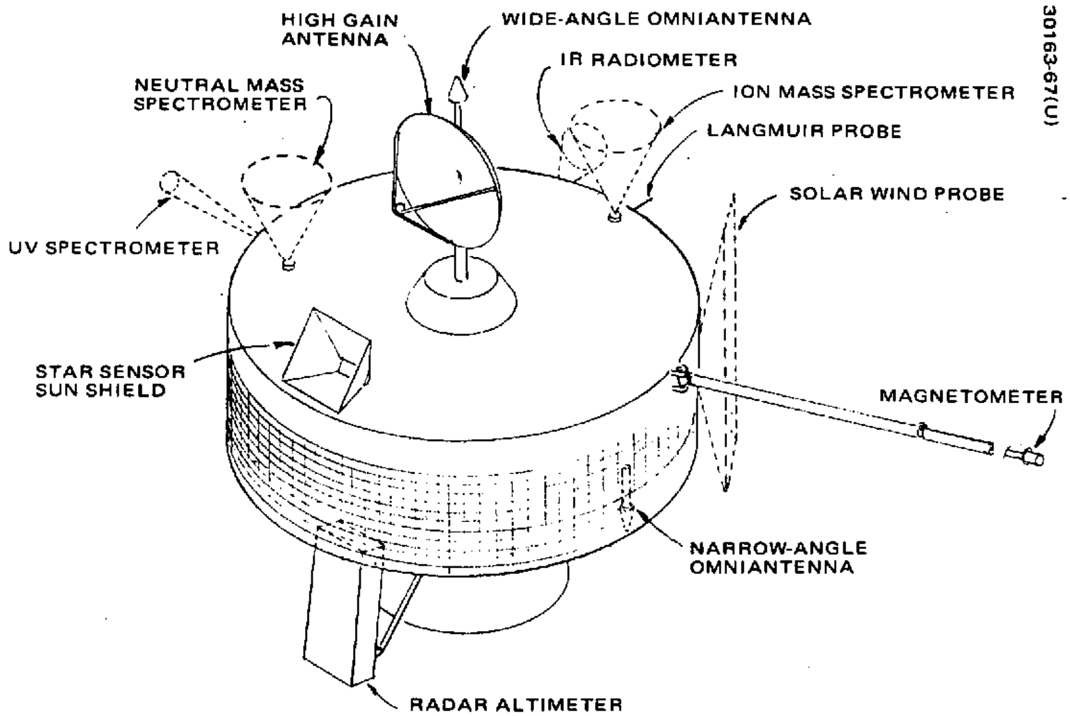


FIGURE 5-24. ORBITER SCIENCE VIEWING REQUIREMENTS

6. COST SAVINGS ASSESSMENT

The objective of the cost savings assessment in the launch vehicle utilization study was to determine the magnitude of the cost savings in the design and manufacture and launch of the Pioneer Venus probe and orbiter spacecraft for launch from the Atlas/Centaur rather than the Thor/Delta launch vehicle. This objective was attained as the result of an ROM (rough order of magnitude) costing exercise conducted in January 1973 for spacecraft configured for the two launch vehicles. To objectively evaluate the most cost effective program, several interviews and conferences were held with major subsystem and program managers and their staff to help establish cost saving factors. Numerous NASA cost savings type documents were also reviewed. The cost savings factors evolved were incorporated as ground rules in the costing exercise. The results are provided herein.

6.1 COST SAVINGS FACTORS

The initial approach to the concept of the Atlas/Centaur spacecraft design was to try to assess low cost influences by conducting a series of interviews with subsystem laboratory managers and their staffs and program managers of previous and on-going programs. The purpose of the interviews was for initial "brainstorming" and then detailed cataloging of "low-cost" (weight-not-a-factor) rationale and approaches to spacecraft design, manufacturing, and testing. The following paragraphs summarize the suggestions, ideas, and comments as a result of the interviews. Because many of the managers had similar ideas and suggestions, they have not been summarized by subsystem or program, nor in order of importance, but rather by hardware, procedural and programmatic groupings, deleting redundant comments. Interviews and conferences were held with the managers and their staffs of the Systems Engineering Laboratory, the Guidance and Controls Systems Laboratory, the Power and Propulsion Laboratory, the Systems Test Department, Space Environmental Simulation Laboratory, Digital Electronics Laboratory, Communications Laboratory, Structures and Thermo-Physics Laboratory, Product Effectiveness Organization, and several program managers within the Hughes Space and Communications Group.

Hardware

1) Use off-the-shelf hardware.

   a) Use same part number or subsystem to achieve cost savings. A high degree of confidence would be achieved if the part were unchanged.

   b) Use previously used part with minimal changes; i.e., substitution of minor parts within a "black box" requiring little or no retesting.

   c) Use of same part number of subsystem with major changes or many part substitutions.

Items a), b) or c) above represent decreasing degrees of cost savings. The count number of new drawings or required drawing changes would be affected accordingly as would procurement procedures and/or assembly techniques.

2) Eliminate non-flyable prototype spacecraft (or conversely, fly the prototype). This would reduce the cost of almost an entire spacecraft and the supporting activities. This could be accomplished with reasonable confidence if permissible weight allowances provide for built-in design safety margins.

3) Eliminate use of exotic materials. Use of beryllium and/or titanium requires expensive and complicated design, higher cost of materials, and comprehensive expensive testing.

4) Design for easier accessibility (larger volume) for assembly, integration, and testing. This would reduce time and hence labor costs. It would provide for ease of making and breaking connectors; it would not require a lot of units to be removed to get to another unit; it would eliminate or reduce the requirements for the harness to turn sharp corners reducing the strength requirements of the harness as well as potential failures.

5) Utilization of standard bracketry, joints and fittings. This would reduce machining costs, permit riveted brackets instead of precise machining and permit "volume" purchasing with accompanying cost reductions.

6) Provide for fixed installations instead of deployable mechanisms wherever possible. This would reduce or eliminate costly mechanisms, devices, motors, pyros, etc., and eliminate expensive associated testing.

7) Eliminate high cost fasteners. I.e., it costs $60 per bolt for titanium and they are long lead time items. If the bolt is removed, it must be replaced with a new one. In vibration testing, it is necessary to supply new bolts each time the spacecraft is reoriented in direction for testing. Shipment to the launch site requires disassembly of certain parts or subsystems which require new fastening hardware. It would be desirous to eliminate any disassembly and select low cost and perhaps heavier weight hardware

8) Fabricate shelves or solar panels without honeycomb. This would lower manufacturing cost and reduce material costs. Intelsat IV spacecraft thermal protection required 2500 2.5 by 2.5 cm (1 x 1 in.) quartz mirrors mounted on the honeycomb thermal control sunshield. The process of mirror installation was tedious and time consuming. Inserts for fasteners in honeycomb is also a time consuming and hence costly process. Other techniques should be considered.

9) Delete structural/dynamic test model and thermal control test model to reduce costs. Higher weight factors would permit higher safety margins to be designed into the spacecraft permitting more analyses to eliminate the need for the expensive test models.

10) Use heat posts for the thermal test vehicle. This would save on fixture and test implementation costs.

11) Data (acquisition) channels during any testing should be minimal. I.e., use 25 instead of 50 channels of data if this is adequate. Exercise control of number of channels. This reduces instrumentation and data acquisition hardware requirements. Influence people to be frugal in their data requirements. Question how much of data is really needed and used.

12) Early evaluation of components and parts is important. Qualify parts early and not in system test. This procedure eliminates expensive redesign and component or subsystem requalification.

13) Reduce power constraints. This might permit usage of non-Hughes off-the-shelf hardware.

14) Utilize reduction of weight constraints to increase volume allowances. Larger volume would permit easier and hence less costly repair, replacement and maintenance.

15) Specify minimal requirements. Consider the OSO PCM and remote multiplexer for use "as is." A flexible data format would give better "off-the-shelf" possibilities. Program Office/System Engineering should specify the minimal requirements.

16) System Engineering, with assistance from Subsystems, should determine the specifications as early as possible and minimize changes, thus preventing redesign and permit early hardware testing.

17) Utilize minimal mass constraints to use weight to protect equipment from hostile environment, thus reducing use of exotic designs and materials.

18) Maximize use of subassembly testing to enhance yield and minimize changes resulting from system testing.

19) Lifting of weight and power constraint should minimize microminiaturization permitting more "off-the-shelf", lower cost, equipment.

20) If possible, select Hughes TWT and power supply for transmitter. This might reduce by almost 50 percent the costs of these items by reducing between 30 to 100 percent the nonrecurring costs if outside vendors were utilized. This might not be possible because of high-g environments experienced during entry at Venus.

21) Select the Philco/Ford transponder that has been developed for Viking. Nonrecurring costs could be eliminated.

22) RF noise performance specification should be written with sufficient dB margin, such that it would not be necessary to fine tune the system at the end of the project to meet requirements. This would lower costs because in the final design, we would not be "pushing" the state of the art. "The last dB costs the most." If the specification for performance of the RF transmitter design is limited to 30 percent efficiency or less, it would probably save about 10 percent of the cost of the transmitter.

23) "Telesat" antenna costs were markedly increased due to the stringent weight constrained design requirement." The size and mass capability of the Atlas/Centaur launch vehicle might permit less stringent antenna structural design requirements and thus reduce costs.

24) Provide large and small probe development models only and not full-up spacecraft. This would provide for the evaluation of the sealing, window design and instrument and connector penetration. It could be used for the prototype.

25) Handling and transportation equipment should be re-evaluated. The whole sequence of shipping and handling should be analyzed to determine if the whole spacecraft could be shipped to the launch site without disassembly and reassembly. It might be possible to design a one-way inexpensive throw-away container.

26) The contract should designate the specific launch vehicle and performance capabilities. Firm requirements would permit earlier design completion with fewer changes and lower costs.

27) In order to maximize use of existing hardware, do not require a strict specification. Survey the available hardware and adapt the spacecraft to accept the proven hardware. This might result in modifying mission capabilities or requirements, but could substantially lower many subsystem costs.

28) Use high reliability parts as early as possible. This would reduce testing, failures, replacements, etc. The initial high costs have a built-in long term savings factor.

29) An attractive cost saving factor is to specify an off-the-shelf orbit insertion motor (OIM) with ±10 percent on or off loading without requalification. This could save up to one million dollars in qualification testing.

30) Use lower cost stainless steel tanks for fuel (rather than titanium). (This, too, did not prove cost effective.)

31) If possible, purchase all parts for both spacecraft at the same time in order to take advantage of price breaks.

32) Attempt to optimize solar cell selection for cost versus efficiency. (The probe bus will have 7,800 cells and the orbiter will have 9,100 cells.)

33) Specify a standard experiment power regulator. Instead of a preset regulator for each experiment, have a standard regulator and the Program Office will determine where the interface is to be. (It was decided not to use experiment regulators.)

34) Attempt to consolidate system test and mission operation hardware and software requirements. If telemetry and functions were identical, some of the hardware and software for the following could be combined: fundamental handling, alarm limit telemetry, transit telemetry, format command, group command and configure experiment, etc.

35) Overdesign mechanical devices to minimize failures in vibration testing and, hence, redesign efforts.

36) Precision tolerances should be minimized (widened). Too many engineers specify tolerances that are more precise than are really required.

Procedures

1) Provide for customer representatives to be in residence. NASA Ames could benefit (as well as Hughes), by constant cognizance and ability to identify problems ASAP.

2) Schedule the spacecraft for assembly after last unit delivery is scheduled. This avoids delays in system assembly or testing awaiting unit delivery and reduces costs of people engaged in the operations.

3) Motivate personnel with cost effectiveness and job security. Personnel, when informed and kept informed, generally rise to the occasion.

4) Test the structural test model (if incorporated in the program) and the qualification spacecraft to 1.25 of flight acceptance level. (The launch vehicle components are seldom designed for any more than 1.25 factor.)

5) Consider 1 month burn-in for high reliability parts. This does not comprise a high percentage of program costs and money is saved by eliminating delays in system testing.

6) Motor bearing assembly low weight design could cost up to $500K and it increases installation time. The Atlas/Centaur should permit the use of the heavier (and existing) MBA.

7) Product effectiveness is expensive, but could be reduced if unit/part/system redundancy could be utilized by lifting weight restrictions.

8) Material selection and approval can be overburdened. On the OSO program, NASA/Goddard has to approve every callout on material and piece parts. This delays approvals of parts and material, and hence design and assembly. These delays are expensive. If NASA/Ames can accept the previous Hughes controls (Government) approved parts list, and only pass on non-approved parts, delays could be avoided and costs kept down.

9) On OSO, NASA/Goddard requires final approval of each vendor and does not permit Hughes the option of selecting lowest cost parts and materials.

10) Brief approval cycles for drawings, plans and specs should be specified. If NASA/Ames approved only specifications and with a 15-day limit, many delays could be eliminated and manpower requirements reduced as compared to OSO. Otherwise, Manufacturing obtains pre-release quotes which may get changed if drawings, parts and materials are not approved, etc

11) Pyrotechnic tests should be go/no-go. Reduce or eliminate large number of data channels because this type of data is not meaningful.

12) Acoustic testing without vibration testing should save money by taking higher risks. Not much energy is obtained below 100 cps. This might not be consistent with the deletion of the structural test model.

13) If the test setup is available after sine-vibration testing, it is less expensive to continue to do random vibration rather than acoustic testing.

14) Test levels should be negotiated with Ames and should be realistic in consonance with program goals. Lower levels of testing would result in less failures on units in test and would save money.

15) EMI testing should be for compatibility only and not by Mil Spec. The Mil Spec goes from dc to 20 GHz for test requirements and an inordinate amount of data is generated. On the OSO program, it required 14 days in two shifts to analyze the data. It is recommended to just "turn everything" on and assure that subsystems are compatible. Test in a sensible manner and do not require 2,000 pages of data which will not be adequately assessed and utilized.

16) It is recommended that the customer not get involved in unit/subassembly buildup. It could reduce the efficiency of the Hughes personnel and bog down the effort in red tape (long approval cycles, etc.).

17) It is recommended that the customer not get involved in change control below the unit level. The customer should only be involved with changes to the top specification, system level test plan and development plan.

18) Except in extraordinary situations, performance of potential vendors should be carefully screened and evaluated prior to selection. (But this is routine for Hughes). Vendors acceptable to Hughes on a fixed price multi-year program should be acceptable to most customers or any program.

19) Source inspection of purchase parts procedures should be carefully agreed to early in the program to avoid delays and inefficient usage of components and materials evaluation engineers.

20) Qualification testing of the OSO TØC unit is being deleted. This cost reduction approach by OSO may provide good historical data.

21) At contract negotiations or shortly thereafter, require that GFE be rigidly "specified" so that GFE meet the design and acceptance specification for interfaces, etc. NASA would have the responsibility for resolving GFE problems. Rigid adherence to this policy will result in no changes to the spacecraft interfaces and thus would permit adherence to the launch date, also.

22) Specify criteria for acceptance early in the program. This will provide for goals that can be set early and that can be accomplished.

23) On the Intelsat IV program, the customer (Comsat) has about 20 people in residence who were helpful on the program and helped expedite approvals. These people also learned very quickly what the cost of changes could be. NASA/Ames could also have residents who would be helpful to Ames' control and could be of assistance to Hughes in expediting the program.

24) Engineers in residence while manufacturing equipment and during assembly and test help to minimize problems. Their presence help keep costs down and minimize delays.

25) Intelsat IV goes through system acceptance tests which are approved by the customer but Hughes determines units and subsystem tests. There were no hot firings on propulsion, but a lot of leak tests were conducted. This approach reduces a large number of tests and the manpower associated in performing these tests.

Programmatic

1) The usage of a fixed price type contract was recommended for applicability to the Pioneer Venus program. The Intelsat IV contract had a heavy delivery time delay penalty - $7M for a 3-month slip which motivated Hughes considerably. The customer had the option to change the contract and had about 20 resident engineers but it was commented that perhaps 15 engineers might have been sufficient. The customer had the right to review adequacy of design, but did not have drawing or lower level specification approval rights. This procedure kept changes to a minimum and, of course, overall program costs were kept down.

2) Streamlining of engineering change requests and approvals procedures is important. In the Intelsat IV program, ECR's were processed in real time on drawings or specifications. During the first part of the program, the customer had control on changes. After the critical design review, the customer could not affect the design. However, the REA would have to substantiate why he wanted the change. Comsat would have 1 day to respond to the change. If Comsat wanted changes, Hughes negotiated costs. Comsat rejected 90 percent of suggested changes that would increase costs.

3) Determine optimum program time. Time equals money, but unrealistic scheduling could cost in excessive overtime versus realistic scheduling. Proper utilization of personnel could result in a longer but less expensive program. (This is not to be confused with program stretch-out which automatically increases the cost of any program due to built-in levels of effort.)

4) PERT is not suitable to this size program. It is expensive for the use obtained. It eventually was dropped on OSO and was not efficiently utilized on Intelsat IV. With proper incentive, laboratory and department maintain internal Gant and PERT type schedules which can be integrated into the overall program plan (and these are used).

5) Selected overtime is needed and used to achieve certain milestones, with an effective decrease in program costs.

6) Establish test program requirements early and do not deviate except under unusual change in requirements.

7) Precise statements of work and test parameters should be adhered to. Changes waste time and cost money.

8) Systems engineers should write top specifications with assistance of subsystem engineers. Early approval should be obtained from the customer and avoid changes.

9) The Program Office and System Engineering should know what the equipment performance capabilities are beforehand. Do not agree to new requirements which result in new inventions with uncertain results.

10) "ATS type of customer interface was much less costly than the OSO type of customer interface. Both contracts should be reviewed accordingly."

11) Internally generated company schedules and status reports should be used instead of additional customer required documents. This will help reduce costs of duplication.

12) The configuration should be frozen after the critical design review. Configuration management and accounting should be started at that time; 90 percent of the drawings have been completed by that time. (On OSO, NASA/Goddard started control on sketches and many extra people were required to maintain configuration management from that point on.)

6.2 SUBSYSTEM APPROACHES TO COST SAVINGS

As a result of the cost fact-finding sessions held with the project team members and pertinent laboratory managers responsible for the development of the project hardware, cost estimates were obtained for the Atlas/Centaur and Thor/Delta spacecraft configurations. In-depth cost details are provided in Volume 12, Cost Analysis of the final report. The cost savings accrued by utilizing an Atlas/Centaur launch vehicle are provided herein. Table 6-1 summarizes the cost savings (and pertinent increases) attained through evaluation of the cost estimates and many detailed iterations with the subsystem areas on their designs. In the following paragraphs, each subsystem is discussed to relate how the use of Atlas/Centaur has affected their costs.

Structures

The ROM cost estimate established a Thor/Delta versus Atlas/Centaur cost difference of $900K. These figures were based upon a 213 cm (84 in.) diameter Thor/Delta spacecraft and a 254 cm (100 in.) diameter Atlas/Centaur configuration. Since the 254 cm (100 in.) diameter configuration appears to be the more desirable of the Atlas/Centaur alternatives, the following discussion will concentrate on the cost differences between the Thor/Delta and the larger diameter Atlas/Centaur spacecraft configurations.

Five fewer probe hemispheres are required for qualification tests for both the large and small probes on the Atlas/Centaur design because of the larger structural margins. The reduced complexity in the design of the Atlas/Centaur probes will reduce the design effort by approximately 30 man-months. The reduced complexity is based upon the relatively larger size of the Atlas/Centaur probes. This portion of the structures task was thus reduced by $220K.

Based upon various complexity factors applied to known Telesat HS-333 program costs, an approximate $185K dollar savings was realized by the Atlas/Centaur configuration for the required structural design engineering effort of the probe bus and orbiter. The structural design engineering effort for the Thor/Delta was considered equal to the effort put forth on the HS-333 program. However, based upon the similarity of the thrust tube, longerons and shelf attachments on the Atlas/Centaur 254 cm (100) in. diameter configuration to the respective HS-339 program components, the Atlas/Centaur structural design engineering effort would be approximately 30 man-months less than the HS-333 program effort. An additional $200K savings was realized by the deletion of the design and material requirements for the use of beryllium as would be required for a lighter configuration on the Thor/Delta design.

An additional cost reduction of $240K is due to decreases in labor cost and computer ODC in the areas of stress analysis. The stress analysis effort is reduced for the probe bus, pressure vessels and orbiter spacecraft due to the larger structural mass allocation which allows "overdesign" in the the structure to higher margins of safety; i.e., the refinement of the stress analysis which is normally performed in the interest of minimizing mass need not be performed and conservative, simplifying assumptions can be made in establishing margins of safety.

TABLE 6-1. 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	2390	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	1780	9.6	Integration and test Materials
7. Test models	1500	100.0	Structural Thermal
8. Magnetic cleanliness	1430	68.0	Parts and materials Magnetic controls management
9. Mechanisms	300	5.2	Telesat BAPTA Bicone deployment
10. Miscellaneous (deductions)	1560	--	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

Although the structural test model will be deleted (see Subsection 6.2) a simplified static test is required for the purpose of calibrating the strain gages to be used later for notching during qualification vibration tests of the spacecraft. Development tests can be eliminated due to the high confidence in structural integrity resulting from high margins of safety. However, the cost of spacecraft qualification vibration tests remains unchanged.

A very small reduction is shown in the dynamic analysis task since a high degree of confidence must be maintained in the accuracy of design load prediction in order to justify a reduction in the stress analysis effort.

An estimated savings of $55K is attributed to the fact that the design integration effort on the probe bus, probes and orbiter for the Atlas/Centaur configuration is somewhat less than that on the Thor/Delta version. This savings is attributed to the increased size of the Atlas/Centaur which should make the location of housekeeping and experimental equipment less of an integration problem.

Deceleration Module

A total savings of $2390K was estimated for the deceleration modules because of the larger volume and higher mass allowed, common cone angles of the large and small probes and deletion of the use of beryllium in the Atlas/Centaur configurations.

With the larger volume inside the Atlas/Centaur pressure vessel, the units can be spread out with a spacing of 1.3 cm (0.5 in.) between units as a minimum. Harness routing as well as unit integration will hence be somewhat eased over the Thor/Delta configuration of the large and small probes.

With the use of common cone angle of 45 deg. for both probes, $240K is anticipated to be saved in aerodynamic testing. Static and dynamic parameters would only have to be ascertained for one set of cone angle conditions. Spin tunnel testing of the final configurations would also be minimized.

With increased margins of safety from 1.25 to 1.50 for both the heatshield ablative material and the structural integrity of the aeroshell, reduced development testing would result in $820K savings in the Atlas/Centaur configurations. No static tests are anticipated for the Atlas/Centaur configured aeroshell structure. Reduced thermal testing of the heatshield ablation material with increased margin of safety is required.

With the deletion of beryllium from the structure of deceleration modules for the Atlas/Centaur design, a cost savings of $1310K is anticipated.

Table 6-2 provides an additional breakdown of costs by the listing of cost savings approaches and hence cost saving factors that have influenced the Atlas/Centaur spacecraft design.

TABLE 6-2. PIONEER VENUS ESTIMATED COST SAVINGS BY USE OF ATLAS/CENTAUR VERSUS THOR/DELTA

	Cost Savings Approach	$K	Saved, $K
I.	Safety margin increase (30 percent)		1320
	Less dynamic and stress analysis and static tests	300	
	Reduction of heat shield and deceleration module development and testing (GE)	820	
	Reduce pressure vessel (probes) test specimens	200	
II.	Magnetic cleanliness reduction		1430
III.	Additional commonality		380
	Telesat ni-cad battery (bus and orbiter)	120	
	45 degree cone angle on large as well as small probes	260	
IV.	Existing hardware/proven design		1080
	Structural similarity (Telesat)	190	
	BAPTA (Telesat mod)	170	
	Reduced effort in Product Effectiveness and materials selection	580	
	Use Viking transponder - no new product design	140	
	Telesat batteries (included in III)		
V.	Increase mass and volume		2970
	Delete discharge regulator/boost voltage circuitry	500	
	Delete probe bus deployment mechanism of bicone antenna	130	
	Digital electronics ease of manufacture, accessibility, and test	820	
	Heavier batteries (included above)		
	24 versus 28 gauge wire (included in I)		
	Less exotic materials in spacecraft	210	
	Less exotic materials in probes	1310	
VI.	Increase in power/lower efficiency		90
	RF power amplifier (28 versus 21 percent), increase power, larger filters	90	
	Larger capacity battery (included above)		
VII.	System test reduction		1750
	Ease of system testing	250	
	Delete structural test model (and testing)	850	
	Delete thermal test model (and testing)	650	
VIII.	Ease of interfaces/integration		1200
	Experiments integration	330	
	Systems engineering/integration	720	
	Configuration/data management	150	
IX.	Additional risk pool		1310
	Beryllium		
	LSI - 14 new		
	Probe packaging		
	6.5 percent margin		
		Subtotal	11,530
X.	Sum of miscellaneous additional costs		-880
	Decreases plus some increases - propulsion, structure (larger diameter), thermal controls, etc.		
	Propulsion	-160	
	Orbit insertion motor	-137	
	Thermal controls	- 50	
	254 cm (100 inch) structure diameter	-150	
	Miscellaneous	-353	
		Total savings	$10,650

*Totals include probe bus with one large and three small probes (four subsystems) and one orbiter spacecraft

Communications Subsystems

In the communications subsystem, the largest cost difference between the Thor/Delta and Atlas/Centaur designs is in the transponder. For the Atlas/Centaur, the Viking transponder is used. Because of footprint requirements peculiar to the Viking program, the Philco-Ford transponder was heavier for Viking than need have been. Moreover, by packaging the receiver and exciter into one relatively large unit, the integration of this unit into the overall system, particularly in the large probe, is limited to configurations which accommodate its peculiar size and mounting configuration. This in turn tends to force the packing density in the large probe to decrease with a concomitant further mass penalty.

For Thor/Delta, it has been assumed that the Viking transponder will be repackaged to minimize the mass and to provide greater flexibility in integrating the transponder with the rest of the system. The estimated cost of this redevelopment is $125K for which the associated mass reduction in the transponder itself is estimated at 0.2 kg (0.4 lb). In the large probe, further mass reduction should be realizable due to the increased packaging flexibility and a resulting reduction in the overall sphere diameter.

The next most significant cost difference is for the power amplifiers. In the Atlas/Centaur, the overall efficiency of the power amplifier is 21 percent and the mass is 0.8 kg (1.5 lb). This compares with, respectively 28 percent and 0.5 kg (1.0 lb) in the Thor/Delta application. The Atlas/Centaur power amplifiers have been estimated to cost $70K less, in total, than the Thor/Delta versions. Aside from added design time to achieve the lower weight and higher efficiency, the Thor/Delta design will require considerably more test time and transistor selection in order to achieve the high efficiency performance.

The final cost difference of $25K is due to filters. In Thor/Delta, the packaging constraints in the large probe force the filters to be as small as possible. In order to do this without sacrificing filter performance, the filters must be designed with very little margin and great care must be taken to achieve the required high circuit Q. By relaxing the size requirements, as in the Atlas/Centaur, an easier design task has been assured with a higher manufacturing yield. These differences would account for the Atlas/Centaur cost savings.

During the course of the Atlas/Centaur study, a tradeoff on the type of RF and antenna subsystem arrangement for the orbiter was conducted to derive the most cost effective arrangement. Table 6-3 provides the cost comparison between a mechanically despun antenna with a 5-W transmitter compared to the triple stacked bicone antenna with a 100-W transmitter. The latter brute-force method was considered to evaluate whether a simple approach was feasible. To attain the required ERP, a high power transmitter was required for the fixed antenna. This design required considerably more thermal power dissipation and hence the thermal test model was considered a definite requirement. However, for the MDA approach, added complexity was required due to rotary joints, a BAPTA and despin control electronics. the lowest cost approach is the MDA.

TABLE 6-3. COSTS ON MDA VERSUS BICONE

	MDA 5-Watt Transmitter ($K)	Triple Stacked Bicone 100 Watt Transmitter ($K)
Transmitter	579.0	2,240.0
Rotary joint	117.0	N.A.
Omni transmit command switch	24.0	N.A.
Despin control electronics	796.3	N.A.
BAPTA	574.0	N.A.
Antenna	332.0	294.0
Thermal control - louvers	48.0	76.0
Thermal test model	None	370.0
Power subsystem:		
Solar panel power electronics	185.0	360.0
Battery	200.0	250.0
Structure		125.0
Total	2,855.3	3,663.0

Command and Data Handling Subsystems

An analysis of the cost differences in the Thor/Delta and Atlas/Centaur spacecraft for the command and data handling subsystems indicates a savings of $820K using Atlas/Centaur.

The reduction is in non-recurring labor only which makes up the cost difference except for the one instance where the trend is reversed in the orbiter data handling subsystem. That one exception is attributed to the subcontract cost and integration of different data handling memories. The lower engineering risk OSO memory has been selected for the orbiter Atlas/Centaur spacecraft with the inherent penalties of additional mass and cost.

In general, the non-recurring labor cost reduction attained for the Atlas/Centaur spacecraft is a reflection of minimized packaging constraints in terms of mass and volume. This reduces the iterative process required between circuit and product design to optimize the design.

The higher percentage cost difference in the probe designs is a reflection of the maximized packaging constraints plus unique environmental requirements imposed upon the probe hardware.

Power Subsystem

The Atlas/Centaur probe and orbiter power subsystem baseline has made maximum utilization of existing hardware developed on the OSO program resulting in a savings of $620K. The Thor/Delta probe and orbiter buses have been designed with low mass and low cost as primary factors. New developments were required to accommodate the low system power levels required. The power subsystem for the large and small probes for both Atlas/Centaur and Thor/Delta is a new design. A discussion of differences and design/development as of the mid-term presentation are as follows:

Large and Small Probes

New designs are required on both Atlas/Centaur and Thor/Delta for large and small probe discharge regulators, power interface units and high-g batteries. Atlas/Centaur silver-zinc batteries are slightly higher in cost due mainly to electrical size. Labor is slightly higher on the Thor/Delta versions due mainly to the smaller size of the vehicles, fabrication and maintainability considerations.

Orbiter and Probe Bus

Thor/Delta utilizes a boost discharge controller (not required on Atlas/Centaur) for a single battery, that requires major modification. Atlas/Centaur uses a (cheaper) diode coupled two battery system. In the Thor/Delta design, a single modified OSO battery charger is required whereas Atlas/Centaur requires two of the same. Thor/Delta orbiter uses a nickel-cadmium battery (lower cost), while the probe bus uses a silver-zinc battery (higher cost). Both Atlas/Centaur buses use the two-battery nickel-cadmium

(HS-333) design (lower total cost). Therefore, Atlas/Centaur buses can use a single battery charger design while Thor/Delta needs two (higher cost). Atlas/Centaur uses an undervoltage switch not used on Thor/Delta. Both recurring and non-recurring labor, subcontract and ODC cost are higher on the Thor/Delta version based on slightly more design, development and parts.

Atlas/Centaur requires more power on both orbiter and probe buses; therefore, higher solar array costs are incurred. In both cases, the orbiter is higher in cost.

Program Management

System Engineering and Experiment Integration

There are no basic differences in tasks associated with Thor/Delta and Atlas/Centaur for system engineering and experiment integration, as a matter of fact the same work statement could be used for both. The instrument types and numbers for both science payloads are the same. NASA/ARC assumed that with the additional mass and volume capability on Atlas/Centaur that development costs might be reduced if mass and volume were relaxed; as a result, a revised payload mass and volume was provided for certain instruments on Atlas/Centaur.

It was assumed that this would also reduce engineering manpower in certain tasks simply because more space and mass would allow easier access, installation, testing and resolution of interface engineering problems.

Accordingly, some system engineering and experiment integration engineering labor tasks were reduced by about 10 percent for Atlas/Centaur, with a cost savings of $1050K.

All other tasks in Atlas/Centaur which involved management and science mission tasks remained the same.

The value of 10 percent was arrived at by considering past experience on other programs where mass and volume were clearly critical problems, and correlating this with the Pioneer Venus application.

Configuration and Data Management

A cost savings of about $150K was estimated for configuration and data management by using the Atlas/Centaur launch vehicle. Discussions with the subsystems engineers indicated extensive use of existing equipment with a subsequent decrease in configuration management requirements. This would also lead to fewer requirements for data generation and submittals. This was estimated to be a reduction of about 1.5 man-months throughout the life of the program.

Product Effectiveness Materials

Due to the increased use of existing equipment for the Atlas/Centaur spacecraft, it was estimated that a somewhat lower level of effort would be required for reliability predictions, quality control and qualification of new materials. The basic Thor/Delta cost estimate was determined by the use of Components and Materials Laboratory standards tempered by experience gained on previous programs. The unavailability of definite design data regarding parts and materials selection caused the use of the OSO spacecraft design as a base for the estimate. Components and Materials Laboratory arrived at its reduced cost estimate (a savings of approximately 20 percent was realized) for the Atlas/Centaur booster by assuming the greater mass, power and volume margins afforded by this booster thereby increasing the possibility of using existing designs and equipment. Specifically, the reduction was based on:

1) Fewer number of new component parts and their associated documentation

2) Fewer required component part qualifications

3) Fewer studies of new materials and processes, particularly in view of the removal of the need for the use of beryllium

4) Fewer number of new materials and processes requiring documentation

5) Somewhat reduced application engineering support in component parts and materials

Inputs concerned with items such as correlation sample analysis and failure analysis were not changed. The estimated savings for Product Effectiveness and the Components and Materials task was $580K.

Elimination of Structural and Thermal Test Models and Testing

Due to the large structural mass allocation which allowed "over-design" in the structure to higher margins of safety and to similar allocations to the thermal control of the spacecraft, it was considered feasible to delete the structural and thermal test models of the probe (bus) spacecraft and the orbiter spacecraft. The deletion of the structural test models and the accompanying test planning and program effected a savings of approximately $810K. The deletion of the thermal test models, test planning, and testing, effected a savings of approximately $690K.

Magnetic Cleanliness

A rather extensive magnetic cleanliness control program had been developed, set up and estimated for the Thor/Delta spacecraft program. This included 1.07 m (3.5 ft) deployable boom plus expensive subsystem controls being established. Elements of the magnetic control program included a special component evaluation program, receiving inspection, assembly design support, vendor support, magnetic calculations, breadboard testing or testing of engineering models, unit testing, probe testing, spacecraft test procedures and spacecraft magnetic testing, micam process change and costs for special test equipment.

Use of a 4.4 m (14.5 ft) boom (5.6 m (18.5 ft) from spacecraft center) on the Atlas/Centaur spacecraft eliminates the requirement for most of the foregoing detail controls. It is estimated that by limiting magnetic controls to the small probes and the total spacecraft (by use of the long boom) approximately $1190K would be saved in the subsystems and about $240K would be saved in the Systems Engineering Magnetic Controls Management.

Mechanisms

The cost estimates for the attitude control subsystem, mechanisms, and sensors, assumes the same functional requirements apply to the spacecraft whether it is launched by a Thor/Delta or an Atlas/Centaur launch vehicle. However, the spacecraft physical constraints can be relaxed considerably if an Atlas/Centaur launch vehicle is used; therefore, the relaxed power and mass requirements resulted in a cost savings of about $300K.

Since many of the components, units and items which meet the baseline design requirements can be acquired "off-the-shelf," no cost savings would be realized on these items by relaxing the mass and power requirements. The following items are included in this category:

Probe Bus

- Sun sensors
- Nutation damper
- Solenoid drivers
- "G" switches
- Separation switches
- Squibs - pyro devices
- Pin pullers
- "G" switches (large probe)
- "G" switches (small probe)
- Separation switch (aeroshell)
- Squibs (nephelometer cover)
- Pin puller (nephelometer cover)
- Pressure gauge
- Pressure switches

Orbiter Bus

- Sun sensors
- Nutation damper
- Solenoid drivers
- Squibs - pyro devices
- Separation switches
- Pin pullers
- "G" switches

In addition, certain other costs are independent of spacecraft mass and power requirements. This category includes subsystem integration and subsystem test equipment.

Those items which must be designed to specifically meet the Pioneer Venus mission requirements are potential candidates for cost savings due to relaxed mass and power requirements. The following summarizes the rationale for the potential cost differences for using a Thor/Delta versus an Atlas/Centaur launch vehicle.

Probe Bus

Attitude Data Processor (ADP). The new unit design for Thor/Delta will have higher design and development costs to make use of LSI components. This results in a mass savings of approximately 1.09 kg (2.4 lb) per spacecraft. This will not be necessary for the Atlas/Centaur.

Bicone Antenna Deployment Mechanism. The development of a bicone antenna deployment mechanism will not be necessary for the Atlas/Centaur launched multiprobe spacecraft.

Magnetometer Deployment Mechanism. The boom length of 4.4 m (14.5 ft) on Atlas/Centaur requires more development and testing than the 1.07 m (3.5 ft) boom on Thor/Delta. However, this will result in relaxation of the magnetic cleanliness requirements on the Atlas/Centaur.

Thermal Louvers. Increased thermal dissipation on the Atlas/Centaur will require more louvers and resulting increase in cost. (See subparagraph Miscellaneous (Additions) that follows.)

Star Sensor. No cost difference is expected since design costs are primarily a function of performance requirements and reliability not mass and power. An off-the-shelf OSO PMT type of star sensor was considered for Atlas/Centaur, however, dual units would have been required for reliability of this unit. Therefore, cost-wise, a solid-state device was considered more cost effective.

Orbiter Bus

Bearing and Power Transfer Assembly (BAPTA). A new light mass design will be required for Thor/Delta to save approximately 4.1 kg (9 lb) which results in increased development costs. The Atlas/Centaur configuration will utilize an existing design (Telesat) with slight modifications.

Despin Control Electronics (DCE). The new unit design for Thor/Delta will have higher design and development costs to make use of LSI components. This results in a mass savings of approximately 0.5 kg (1.0 lb) per spacecraft. This will not be necessary for the Atlas/Centaur configuration.

Magnetometer Deployment Mechanism. Same comments as for the probe bus above.

Thermal Louvers. (See above under probe bus)

Miscellaneous (Deductions)

System Test

Careful analysis of the testing requirements of the Thor/Delta versus the Atlas/Centaur spacecraft configuration, resulted in a cost savings estimate of $210K. This is based on the larger probes and ease of handling them and the familiarity of personnel with many of the subsystems that would be utilized on the Atlas/Centaur configuration. An analysis of the test procedures indicates that one shift rather than two-shift testing would be feasible.

Risk Pool

Due to the unknown factors and time risk involved in developing new LSI circuitry as well as the uncertainty involved with the development and application of beryllium structures that would be involved in attempting to increase the mass margin of the Thor/Delta configuration, a "risk pool" factor of $1310K was included in the estimate of the Thor/Delta program. The easing of the mass and power constraints by using an Atlas/Centaur launch vehicle removes the necessity for developing new LSI circuitry and of using beryllium structures. Therefore, another $1310K savings is anticipated (refer to Table 6-2).

Miscellaneous (Additions)

The foregoing total savings to be incurred by utilization of an Atlas/Centaur configuration are somewhat offset by relatively minor cost increases due to the larger structure.

Thermal Control

Increased thermal dissipation on the Atlas/Centaur spacecraft (bus and orbiter) will require more louvers and resulting increase in cost. Six louvers are required on the Thor/Delta probe bus versus ten louvers on the Atlas/Centaur design. For the orbiter, eight versus twelve louvers are required respectively.

Propulsion

For both configurations, the propulsion subsystem consists of a monopropellant hydrazine, pressure blowdown system. The only difference consists of the tanks selected for storage of propellant and pressurant. Tanks for the Thor/Delta launched spacecraft are identical to those used on the HS-333 (Telesat) satellite; while tanks for the Atlas/Centaur launched spacecraft are presently being procured and qualified for the HS-356 (Marsat) satellite. The baseline design for the Thor/Delta design utilized three tanks for propellant storage. Since that time, changes in propellant requirements have permitted exercise of the option to use only two tanks. The Atlas/ Centaur design also utilizes two tanks for propellant storage, thus, an earlier cost advantage (due to fewer units and lower manufacturing costs) has now been reduced.

The baseline orbit insertion motor subsystems for both launch vehicles consist of solid propellant rocket motors. The present trajectory selections result in the choice of the Thiokol Model TE-M-616 motor for the Atlas/Centaur launched spacecraft; and exercise of the option to use the Thiokol Model TE-M-521 motor for the Thor/Delta launched spacecraft. Use of the TE-M-521 motor requires modification to the "off-the-shelf" design to accommodate the propellant off-load required for this mission. This modification consists of shortening the cylindrical section of the case. A case burst test, and vibration test have been included in the price to qualify the modification. The propellant off-load can also be accommodated by a redesigned igniter assembly, however, the case modification was selected due to vendor recommendation as being the least severe change of the two candidates. The TE-M-521 motor is flight-proven on the NATO I, Skynet I, and IMP HØJ satellites.

The TE-M-616 motor is presently in development and will be qualified for the Canadian Technology satellite program. No development or qualification costs, have been included in this estimate, however, due to the increased size, this motor is more costly than the TE-M-521 motor used on the Thor/Delta spacecraft.

Structures

The basic structure of the Atlas/Centaur spacecraft will cost somewhat more than that of the Thor/Delta because of the increased size and additional materials required. Also, in order to eliminate the structural and thermal test models, some additional stress and thermal analyses will be required

The foregoing three items result in an increase of about $880K versus a total decrease of $11,530K itemized in the preceding paragraphs.

Conclusion

As can be seen in Tables 6-1 and 6-2 a total cost savings for a two mission set utilizing an Atlas/Centaur launch vehicle is $10,650,000.