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Systems Design Study of the Pioneer Venus Spacecraft

Final Study Report

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Volume III. Specifications

PIONEER VENUS SPACECRAFT PERFORMANCE REQUIREMENTS
SYI-60

PRELIMINARY PIONEER VENUS
SCIENTIFIC INSTRUMENT-PROBE AND SPACECRAFT
INTERFACE DOCUMENT

29 July 1973

Contract No. NAS2-7249

Prepared for

AMES RESEARCH CENTER
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

TRW SYSTEMS GROUP

MARTIN MARIETTA

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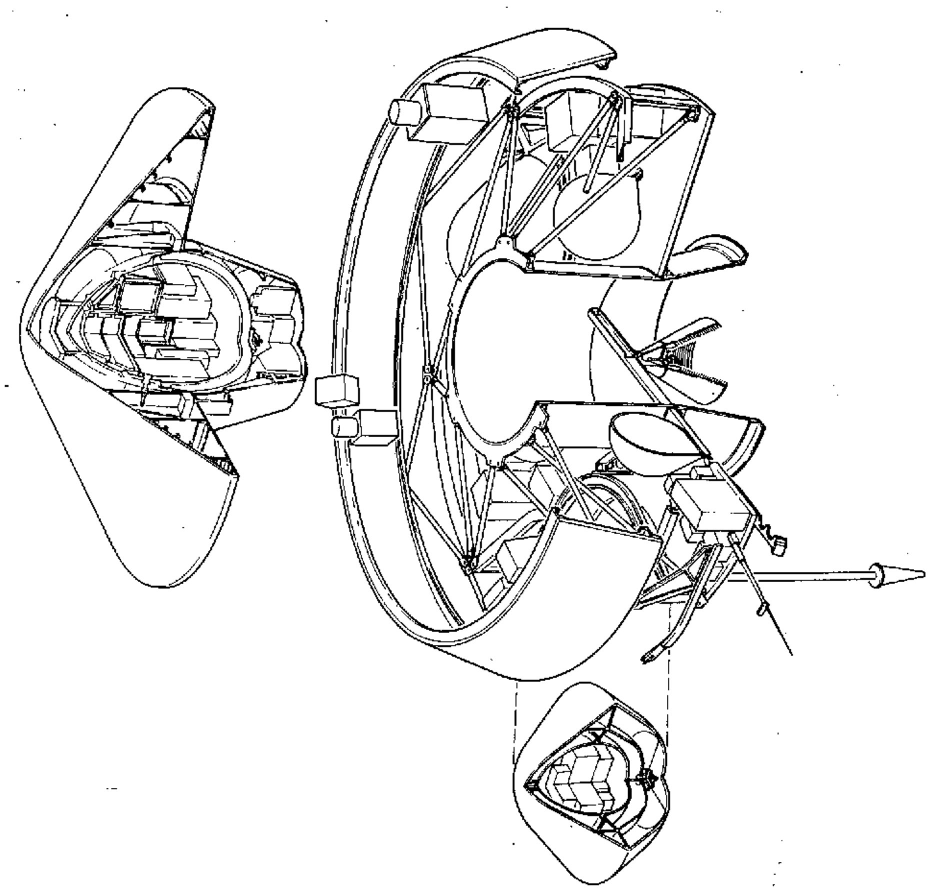

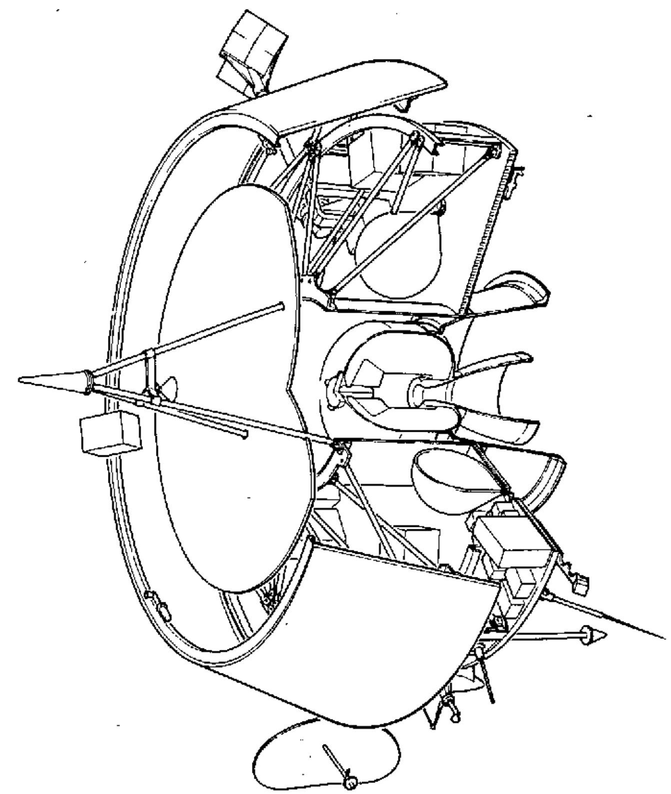

LIST OF VOLUMES

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ACRONYMS AND ABBREVIATIONS

A	ampere analog
abA	abampere
AC	alternating current
A/C	Atlas/Centaur
ADA	avalanche diode amplifier
ADCS	attitude determination and control subsystem
ADPE	automatic data processing equipment
AEHS	advanced entry heating simulator
AEO	aureole/extinction detector
AEDC	Arnold Engineering Development Corporation
AF	audio frequency
AGC	automatic gain control
AgCd	silver-cadmium
AgO	silver oxide
AgZn	silver zinc
ALU	authorized limited usage
AM	amplitude modulation
a.m.	ante meridian
AMP	amplifier
APM	assistant project manager
ARC	Ames Research Center
ARO	after receipt of order
ASK	amplitude shift key
at. wt	atomic weight
ATM	atmosphere
ATRS	attenuated total refractance spectrometer
AU	astronomical unit
AWG	American wire gauge
AWGN	additive white gaussian noise
B	bilevel
B	bus (probe bus)
BED	bus entry degradation

ACRONYMS AND ABBREVIATIONS (CONTINUED)

BER	bit error rate
BLIMP	boundary layer integral matrix procedure
BPIS	bus-probe interface simulator
BPL	bandpass limiter
BPN	boron potassium nitrate
bps	bits per second
BTU	British thermal unit
C	Canberra tracking station — NASA DSN
CADM	configuration administration and data management
C&CO	calibration and checkout
CCU	central control unit
CDU	command distribution unit
CEA	control electronics assembly
CFA	crossed field amplifier
cg	centigram
c.g.	center of gravity
CIA	counting/integration assembly
CKAFS	Cape Kennedy Air Force Station
cm	centimeter
c.m.	center of mass
C/M	current monitor
CMD	command
CMO	configuration management office
C-MOS	complementary metal oxide silicon
CMS	configuration management system
const	constant construction
COSMOS	complementary metal oxide silicon
c.p.	center of pressure
CPSA	cloud particle size analyzer
CPSS	cloud particle size spectrometer

ACRONYMS AND ABBREVIATIONS (CONTINUED)

CPU	central processing unit
CRT	cathode ray tube
CSU	Colorado State University
CTRF	central transformer rectifier filter
D	digital
DACS	data and command subsystem
DCE	despin control electronics
DDA	despin drive assembly
DDE	despin drive electronics
DDU	digital decoder unit
DDULBI	doubly differenced very long baseline interferometry
DEA	despin electronics assembly
DEHP	di-2-ethylhexyl phthalate
DFG	data format generator
DGB	disk gap band
DHC	data handling and command
DIO	direct input/output
DIOC	direct input/output channel
DIP	dual in-line package
DISS REG	dissipative regulator
DLA	declination of the launch azimuth
DLBI	doubly differenced very long baseline interferometry
DMA	despin mechanical assembly
DOF	degree of freedom
DR	design review
DSCS II	Defense System Communications Satellite II
DSIF	Deep Space Instrumentation Facility
DSL	duration and steering logic
DSN	NASA Deep Space Network
DSP	Defense Support Program
DSU	digital storage unit
DTC	design to cost
DTM	decelerator test model

DTP	descent timer/programmer
DTU	digital telemetry unit
DVU	design verification unit
E	encounter entry
EDA	electronically despun antenna
EGSE	electrical ground support equipment
EIRP	effective isotropic radiated power
EMC	electromagnetic compatibility
EMI	electromagnetic interference
EO	engineering order
EOF	end of frame
EOM	end of mission
EP	earth pointer
ESA	elastomeric silicone ablator
ESLE	equivalent station error level
ESRO	European Space Research Organization
ETM	electrical test model
ETR	Eastern Test Range
EXP	experiment
FFT	fast Fourier transform
FIPP	fabrication/inspection process procedure
FMEA	failure mode and effects analysis
FOV	field of view
FP	fixed price frame pulse (telemetry)
FS	federal stock
FSK	frequency shift keying
FTA	fixed time of arrival

G	Goldstone Tracking Station - NASA DSN gravitational acceleration
g	gravity
G&A	general and administrative
GCC	ground control console
GFE	government furnished equipment
GHE	ground handling equipment
GMT	Greenwich mean time
GSE	ground support equipment
GSFC	Goddard Space Flight Center
H	Haystack Tracking Station - NASA DSN
HFFB	Ames Hypersonic Free Flight Ballistic Range
HPBW	half-power beamwidth
htr	heater
HTT	heat transfer tunnel
I	current
IA	inverter assembly
IC	integrated circuit
ICD	interface control document
IEEE	Institute of Electrical and Electronics Engineering
IFC	interface control document
IFJ	in-flight jumper
IMP	interplanetary monitoring platform
I/O	input/output
IOP	input/output processor
IR	infrared
IRAD	independent research and development
IRIS	infrared interferometer spectrometer
IST	integrated system test
I&T	integration and test
I-V	current-voltage

ACRONYMS AND ABBREVIATIONS (CONTINUED)

JPL	Jet Propulsion Laboratory
KSC	Kennedy Space Center
L	launch
LD/AD	launch date/arrival date
LP	large probe
LPM	lines per minute
LPTTL MSI	low power transistor-transistor logic medium scale integration
LRC	Langley Research Center
M	Madrid tracking station - NASA DSN
MAG	magnetometer
max	maximum
MEOP	maximum expected operating pressure
MFSK	M'ary frequency shift keying
MGSE	mechanical ground support equipment
MH	mechanical handling
MIC	microwave integrated circuit
min	minimum
MJS	Mariner Jupiter-Saturn
MMBPS	multimission bipropellant propulsion subsystem
MMC	Martin Marietta Corporation
MN	Mach number
mod	modulation
MOI	moment of inertia
MOS LSI	metal over silicone large scale integration
MP	maximum power
MSFC	Marshall Space Flight Center
MPSK	M'ary phase shift keying
MSI	medium scale integration
MUX	multiplexer
MVM	Mariner Venus-Mars

ACRONYMS AND ABBREVIATIONS (CONTINUED)

NAD	Naval Ammunition Depot, Crane, Indiana
N/A	not available
NiCd	nickel cadmium
NM/IM	neutral mass spectrometer and ion mass spectrometer
NRZ	non-return to zero
NVOP	normal to Venus orbital plane
OEM	other equipment manufacturers
OGO	Orbiting Geophysical Observatory
OIM	orbit insertion motor
P	power
PAM	pulse amplitude modulation
PC	printed circuit
PCM	pulse code modulation
PCM-PSK-PM	pulse code modulation-phase shift keying-phase modulation
PCU	power control unit
PDA	platform drive assembly
PDM	pulse duration modulation
PI	principal investigator proposed instrument
PJU	Pioneer Jupiter-Uranus
PLL	phase-locked loop
PM	phase modulation
p.m.	post meridian
P-MOS	positive channel metal oxide silicon
PMP	parts, materials, processes
PMS	probe mission spacecraft
PMT	photomultiplier tube
PPM	parts per million pulse position modulation
PR	process requirements
PROM	programmable read-only memory
PSE	program storage and execution assembly

ACRONYMS AND ABBREVIATIONS (CONTINUED)

PSIA	pounds per square inch absolute
PSK	phase shift key
PSU	Pioneer Saturn-Uranus
PTE	probe test equipment
QOI	quality operation instructions
QTM	qualification test model
RCS	reaction control subsystem
REF	reference
RF	radio frequency
RHCP	right hand circularly polarized
RHS	reflecting heat shield
RMP-B	Reentry Measurements Program, Phase B
RMS	root mean square
RMU	remote multiplexer unit
ROM	read only memory rough order of magnitude
RSS	root sum square
RT	retargeting
RTU	remote terminal unit
S	separation
SBASI	single bridgewire Apollo standard initiator
SCP	stored command programmer
SCR	silicon controlled rectifier
SCT	spin control thrusters
SEA	shunt electronics assembly
SFOF	Space Flight Operations Facility
SGLS	space ground link subsystem
SHIV	shock induced vorticity
SLR	shock layer radiometer
SLRC	shock layer radiometer calibration

ACRONYMS AND ABBREVIATIONS (CONTINUED)

SMAA	semimajor axis
SMIA	semiminor axis
SNR	signal to noise ratio
SP	small probe
SPC	sensor and power control
SPSG	spin sector generator
SR	shunt radiator
SRM	solid rocket motor
SSG	Science Steering Group
SSI	small scale integration
STM	structural test model
STM/TTM	structural test model/thermal test model
STS	system test set
sync	synchronous
TBD	to be determined
TCC	test conductor's console
T/D	Thor/Delta
TDC	telemetry data console
TEMP	temperature
TS	test set
TTL MSI	transistor-transistor logic medium scale integration
TLM	telemetry
TOF	time of flight
TRF	tuned radio frequency
TTM	thermal test model
T/V	thermo vacuum
TWT	travelling wave tube
TWTA	travelling wave tube amplifier
UHF	ultrahigh frequency
UV	ultraviolet

ACRONYMS AND ABBREVIATIONS (CONTINUED)

VAC	volts alternating current
VCM	vacuum condensable matter
VCO	voltage controlled oscillator
VDC	volts direct current
VLBI	very long baseline interferometry
VOI	Venus orbit insertion
VOP	Venus orbital plane
VSI	Viking standard initiator
VTA	variable time of arrival
XDS	Xerox Data Systems

PRELIMINARY DRAFT

PIONEER VENUS SPACECRAFT PERFORMANCE REQUIREMENTS

SY1-60

21 March 1973
Revised 4 April 1973
Revised 18 June 1973

Notes

1) Preliminary drafts of this specification were provided informally to ARC during the course of the study. This revision includes a better definition of the preferred designs as of 18 June 1973. Only some of the changes reflecting the new requirements of PV-1000.00 and related RFP documents have been incorporated because of study schedule constraints.

2) Some numerical data has been underlined (e.g., 20.0 W) to indicate preliminary values subject to change. These preliminary values represent nothing more than educated guesses in some cases, but have been included in preference to TBD's (To Be Determined) to increase the usefulness of the document.

PIONEER VENUS SPACECRAFT PERFORMANCE REQUIREMENTS

OUTLINE

OUTLINE (Continued)

1. SCOPE

This specification establishes the Pioneer Venus Spacecraft requirements for the following areas:

a) Design criteria and performance requirements for the Pioneer Venus spacecraft systems and subsystems for a 1978 Multiprobe Mission and a 1978 Orbiter Mission

b) Spacecraft System Interfaces

c) Scientific Instrument Integration

2. APPLICABLE DOCUMENTS

The following documents of the issue, as specified, form a part of this specification (SY 1-62) to the extent specified herein.

a) Supplementary Requirements Document, PV-1006 which consists of the following:

PV-1006.01	General Mission Requirements
PV-1006.02	Scientific Instrument Requirements
PV-1006.03	DSN Interface Requirements
PV-1006.04	Launch Vehicle Requirements
PV-1006.05	Launch Operations Requirements
PV-1006.06	Flight Operations Requirements
PV-1006.07	Ground Support Equipment Requirements

b) AFETRM 127-1, "The Range Safety Manual of the AFETR", September 1, 1972

c) NASA SP-8011, "Models of the Venus Atmosphere", revised September 1972

d) DSN Document No. 810-5, "Deep Space Network/Flight Project Interface Design Handbook", Revision C, April 15, 1972

e) NASA TM X-64627, "Space and Planetary Environment Criteria Guidelines for Use in Space Vehicle Development", November 15, 1971

f) KMI 1710.1B/SF, "General Safety Plan of KSC", October 9, 1970

g) NHB 5300.4(1A), "Reliability Program Provisions for Aeronautical and Space System Contractors", April 1970

h) NHB 5300.4(1B), "Quality Program Provisions for Aeronautical and Space System Contractors", April 1969

i) NHB 5300.4(3A), "Requirements for Soldered Electrical Connections" and ARC Supplement 1, September 15, 1972

j) AHB 5328.1, "Preferred Parts and Materials List", September 1972

k) AHB 5328-3, "Screening Inspection for Electronic Parts", November 1972

l) Fed. Std. No. 209A, "Clean Room and Work Station Requirements, Controlled Environment", August 10, 1966

m) MIL-P-116F, "Methods of Preservation", 1 February 1973

n) MIL-STD-129F, "Marking for Shipment and Storage", 30 March 1973

o) MIL-STD-1186, "Cushioning, Anchoring, Bracing, Blocking and Waterproofing, Appropriate Test Methods", 28 October 1972

3. REQUIREMENTS

3.1 SYSTEM DESCRIPTION

The Pioneer Venus spacecraft system includes vehicles for a 1978 multi-probe mission and a 1978 orbiter mission. A spacecraft bus with common subsystem designs is used to support both missions. An Atlas/ Centaur launch vehicle is used for each mission.

For the multi-probe mission the bus carries three small probes and a single large probe, as shown in Figure 3.1-1. Each probe has a heat shield and is designed to protect its experiment complement during entry and descent to the surface of Venus. The bus carries some experiments and also enters the Venusian atmosphere to return high altitude data before it is destroyed.

For the orbiter mission, the bus carries a larger complement of experiments, is provided with increased data handling capability, and is inserted into orbit around Venus by a solid propellant rocket motor. The spacecraft remains in orbit for over one Venusian year (243 earth days) in order primarily to return low altitude scientific data from a wide range of latitudes and longitudes. Figure 3.1-2 shows the overall orbiter mission configuration.

3.1.1 Probe Bus Description

The probe bus in combination with its complement of scientific instruments and with the probes comprises a spinning spacecraft. At launch the probe bus carries a large probe and three small probes, as described in Sections 3.1.2 and 3.1.3. The probe bus itself consists of:

a) A structures subsystem to support and enclose in a suitable environment the bus scientific instruments, the probes and other spacecraft components. The structures subsystem also includes the launch vehicle adapter and separation mechanisms. The structures subsystem includes support fittings and separation mechanisms to support and release the probes.

b) An attitude control subsystem to provide the required spacecraft attitudes throughout the mission and to provide small velocity increments during interplanetary cruise and prior to probe release and bus entry.

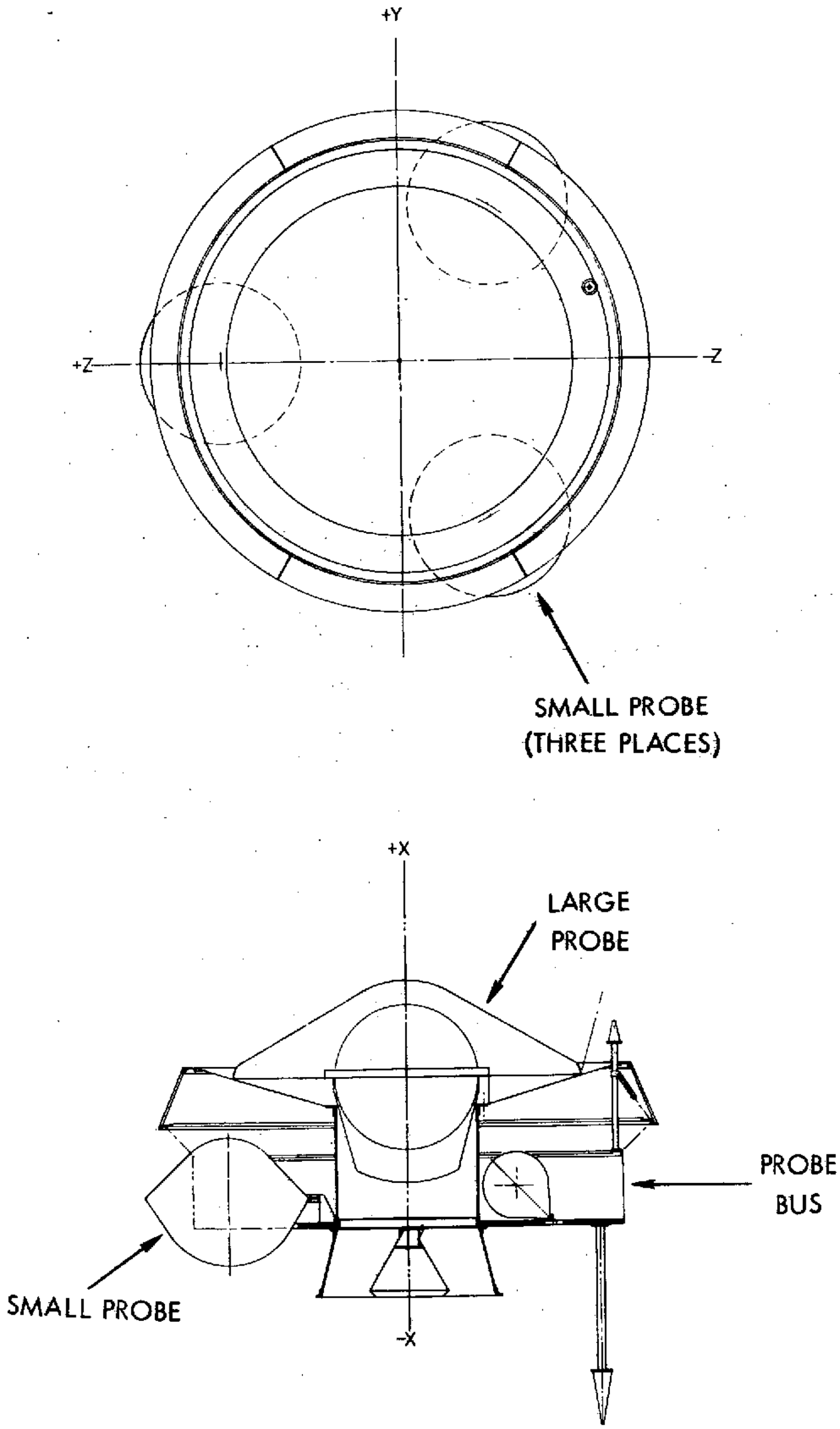

Figure 3.1-1. Multi-Probe Mission Spacecraft

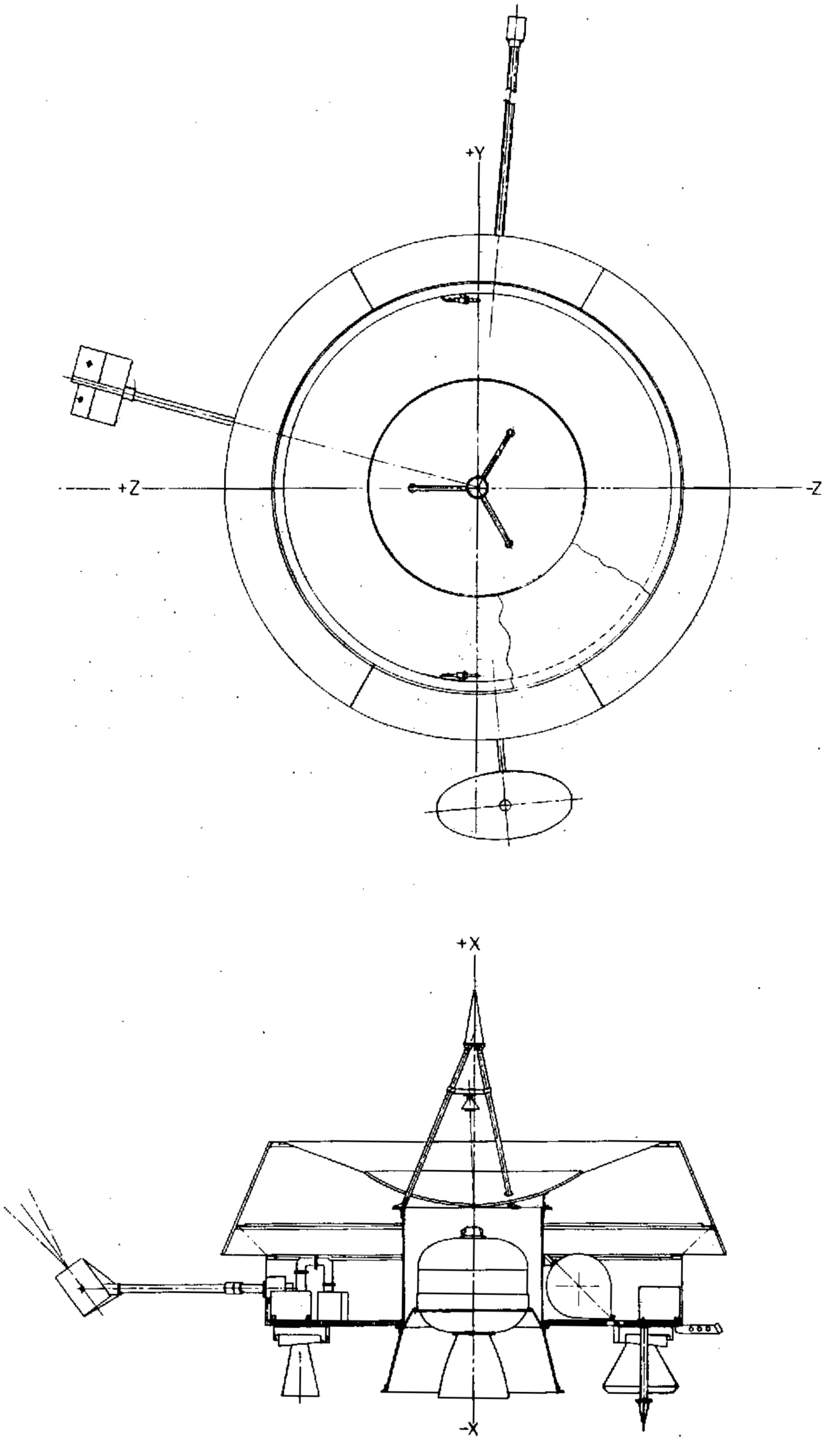

Figure 3-12. Orbiter Mission Spacecraft

c) An electrical power subsystem which includes a solar array, a battery, and equipment to convert, regulate, and distribute electrical power to other bus subsystems, to the probes, and to the bus scientific instruments. This subsystem also includes the electrical connectors between the bus and the probes.

d) A communications subsystem which includes antennas, transmitters, and receivers to provide two-way communication between the bus and DSN stations.

e) A command subsystem which decodes commands received from the communications subsystem and distributes them to other bus subsystems, to the bus experiments, and to the probes. Capability to store time tagged commands for later execution is also included.

f) A data handling subsystem to process data from bus scientific instruments, probe checkout data, and data from bus engineering sensors and to telemeter the results to earth via the communications subsystem.

g) A thermal control subsystem which provides the surface coatings, insulation, heaters, and louvers to provide a controlled thermal environment for the bus equipment and scientific instruments and to meet the probe thermal interface requirements during launch and interplanetary cruise prior to probe release.

3.1.2 Large Probe Description

The large probe is released from the bus about 25 days prior to entry in an attitude appropriate for entry at essentially zero angle of attack. This attitude is stabilized by the spin momentum retained by the large probe at separation. The major elements of the large probe are an aeroshell/heat shield to provide aerodynamic braking and thermal protection during entry, and a heat and pressure resistant descent capsule which is separated from the aeroshell by a parachute which further decelerates the capsule. Final descent is made without the parachute because of the dense atmosphere. In more detail, the large probe consists of:

a) An aeroshell subsystem to support and enclose the descent capsule, and other probe equipment during launch, cruise and atmospheric entry and to support the heat shield subsystem. The aeroshell also defines the aerodynamic shape during entry.

b) A decelerator subsystem which consists of a deployment mortar and parachute used to separate the aeroshell from the descent capsule and to decelerate the descent capsule.

c) A descent capsule subsystem which consists of a pressure vessel with internal structure to support scientific instruments and probe electronic equipment and a combination aerodynamic cover/radome which encloses the pressure vessel, communications antenna and the thermal insulation which surrounds the pressure vessel. Viewports and other penetration are also part of the descent capsule subsystem.

d) A mechanisms subsystem which consists of attachment, release, and deployment mechanisms which control the mechanical operations and staging of the large probe during entry and descent.

e) A heat shield subsystem which consists of forebody and afterbody ablative materials which are bonded to the aeroshell to provide thermal protection during entry. Exposed surfaces of the heat shield are also coated to provide appropriate thermal properties during cruise, both before and after separation from the bus.

f) A thermal control subsystem consisting of coatings, insulation, and heaters to provide a controlled thermal environment for the probe equipment and scientific instruments mounted inside the descent capsule and within the aeroshell.

g) An electrical power subsystem consisting of a battery and equipment to control and distribute electrical power to separation devices, probe electronics, and to the scientific instruments.

h) A data handling and command subsystem which consists of acceleration sensors, timers and other equipment to control timing and sequencing of all probe subsystems and to collect and encode data from the science instruments and probe engineering sensors. The data handling and command subsystem stores data as necessary which cannot be transmitted in real time and relays both stored and real time data to the probe communications subsystem.

i) A communications subsystem including a transmitter, a receiver, and an antenna to provide a two-way coherent communications link with DSN stations after separation from the bus.

3.1.3 Small Probe Description

The three small probes are identical (except for downlink frequencies) and are similar in many ways to the large probe. One important exception is that the aeroshell and heat shield are retained throughout entry and descent. Ports are opened after peak heating to expose the scientific instruments. The small probes are spin stabilized after release from the bus, carry identical science payloads, and consist of the following:

a) An aeroshell subsystem to support and enclose the pressure vessel and other probe equipment throughout the mission from launch until impact on the surface of Venus.

b) A pressure vessel subsystem consisting of a pressure vessel and internal structure to enclose and support the science instruments and probe electronic equipment.

c) A mechanisms subsystem which includes equipment to deploy and expose the science instruments during descent.

d) A heat shield subsystem consisting of forebody and afterbody ablative materials which are bonded to and supported by the aeroshell and which provide protection from aerodynamic heating during entry.

e) A thermal control subsystem consisting of coatings, insulation, phase change material, and heaters to provide a controlled thermal environment for the probe equipment and scientific instruments.

f) An electrical power subsystem consisting of a battery and equipment to control and distribute electrical power to the probe equipment and the scientific instruments.

g) A data handling and command subsystem which consists of acceleration sensors, timers, and other equipment to control timing and sequencing of all probe subsystems and to collect the encode data from the science instruments and probe engineering sensors. The data handling and command subsystem stores data as necessary which cannot be transmitted in real time and relays both stored and real time data to the probe communications.

h) A communications subsystem which includes a transmitter and an antenna to transmit data from the scientific instruments and probe engineering sensors to DSN stations.

3.1.4 Orbiter

The orbiter consists of a modified probe bus which together with its complement of scientific instruments comprises a spinning spacecraft. During normal interplanetary cruise and orbital operation the orbiter flies with its spin axis pointing toward earth. A gimballed platform is provided for the scientific instruments which must be aligned with the ram velocity near periapsis.

a) Structures - Changes in the structures subsystem to accommodate different scientific instruments, the addition of a high gain antenna reflector, and a solid rocket motor for orbit insertion.

b) Attitude Control - Add a drive and the associated electronics for the gimballed ram experiment platform. Add a drive and the associated electronics for the radar altimeter antenna.

c) Electric Power - The battery and solar array sizes are increased to accommodate the increased science instrument and communication requirements.

d) Communication - Increase transmitter power and add a high gain (directional) antenna and related equipment in order to provide increased data rate capability. A conscan processor is added in order to measure main antenna pointing relative to the earthline.

e) Command - No change to the command subsystem.

f) Data Handling - Data storage capability is added to the data handling subsystem to store data during earth occultation and to temporarily store data acquired at rates too high for real time transmission.

g) Thermal Control - Detailed changes in thermal control subsystem to accommodate the new science, spacecraft equipment, and the Venus environment.

h) Solid Rocket Motor - A solid rocket motor is added for orbit insertion.

3.2 COMPATIBILITY

3.2.1 Spacecraft/Launch Vehicle

Both spacecraft shall be compatible with the Atlas/Centaur launch vehicle, as specified in PV 1006.04.

3.2.2 Spacecraft/DSN

The probes, probe bus, and orbiter spacecraft shall be compatible with the Deep Space Network, as specified in PV 1006.03.

For the probe mission, one discrete nominal transmit and receive frequency shall be allocated to the probe bus and another to the large probe. A total of four additional nominal transmit frequencies shall be allocated for use by the small probes. Only three will actually be used during the mission with the fourth small probe frequency reserved for the flight spare.

3.2.3 Probe Bus/Scientific Instrument Interfaces

The probe bus shall be compatible with the scientific instruments, as specified in PV 1006.02.

3.2.4 Probe/Scientific Instrument Interfaces

The large and small probes shall be compatible with the scientific instruments, as specified in PV 1006.02.

3.2.5 Orbiter/Scientific Instrument Interfaces

The orbiter spacecraft shall be compatible with the scientific instruments, as specified in PV 1006-02.

3.3 PROBE AND PROBE BUS INTERFACES

3.3.1 Weight and Mass Properties

The probes shall not exceed the weights allocated to them in Section 3.4.9.2.

Entry and descent dynamic considerations impose constraints on probe center of mass locations and mass properties. In that regard, the large probe shall be designed to meet the requirements of Section 3.6.4.2 and the small probes to meet the requirements of Section 3.6.5.2. In addition, the pertinent probe mass properties data including limits on allowable variations shall be documented and controlled in the Probe/ Probe Bus Interface Control Document (ICD) (IF1-51) in order to meet dynamic requirements during interplanetary cruise and probe separation. Tolerances and allowable variations of probe mass properties shall be controlled during design and manufacture to assure that the requirements of the Probe/Probe Bus ICD are met.

3.3.2 Mechanical Attachment and Separation Mechanism

3.3.2.1 Large Probe

The large probe shall be compatible with three-point attachment and support from the probe bus. The probe bus shall provide the mechanisms for attachment and support of the large probe throughout all mission phases leading to large probe release. The probe bus attachment and support mechanisms shall provide for release of the large probe on command in a manner which assures that the requirements of Section 3.3.6 will be met.

The electrical umbilical required in Section 3.3.7 shall be designed to operate in a manner which assures that the requirements of Section 3.3.6 are met. This may be done by assuring mechanical separation of the electrical connections prior to probe release.

The large probe size shall not exceed the envelope given in Figure 3.3-1.

3.3.2.2 Small Probe

Each small probe shall be compatible with four pairs of support pads located as shown in Figure 3.3-2 around the largest diameter with one pad from each pair bearing on the probe forebody and the other on the

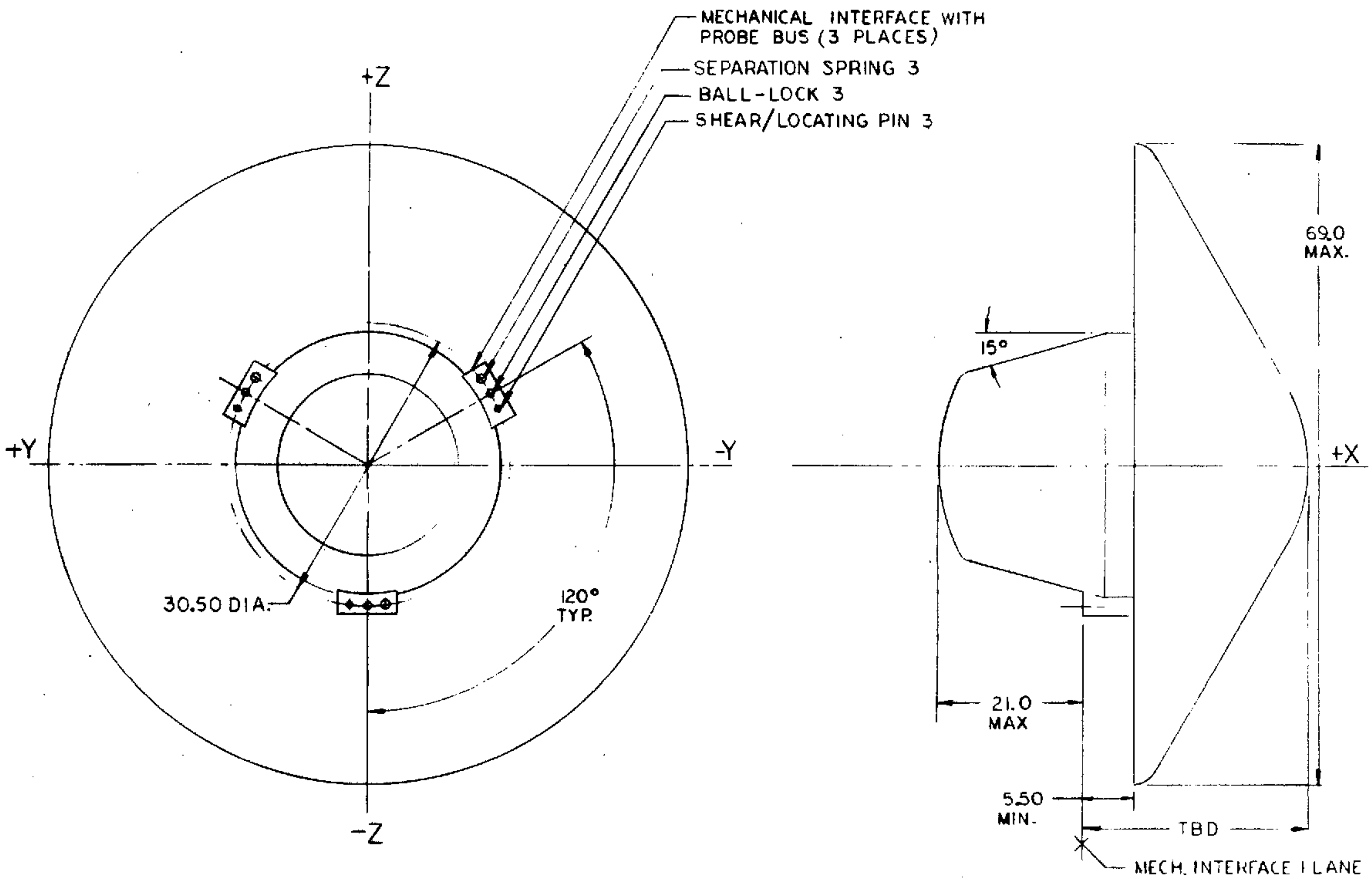

Figure 3.3-1. Large Probe Envelope Limitations

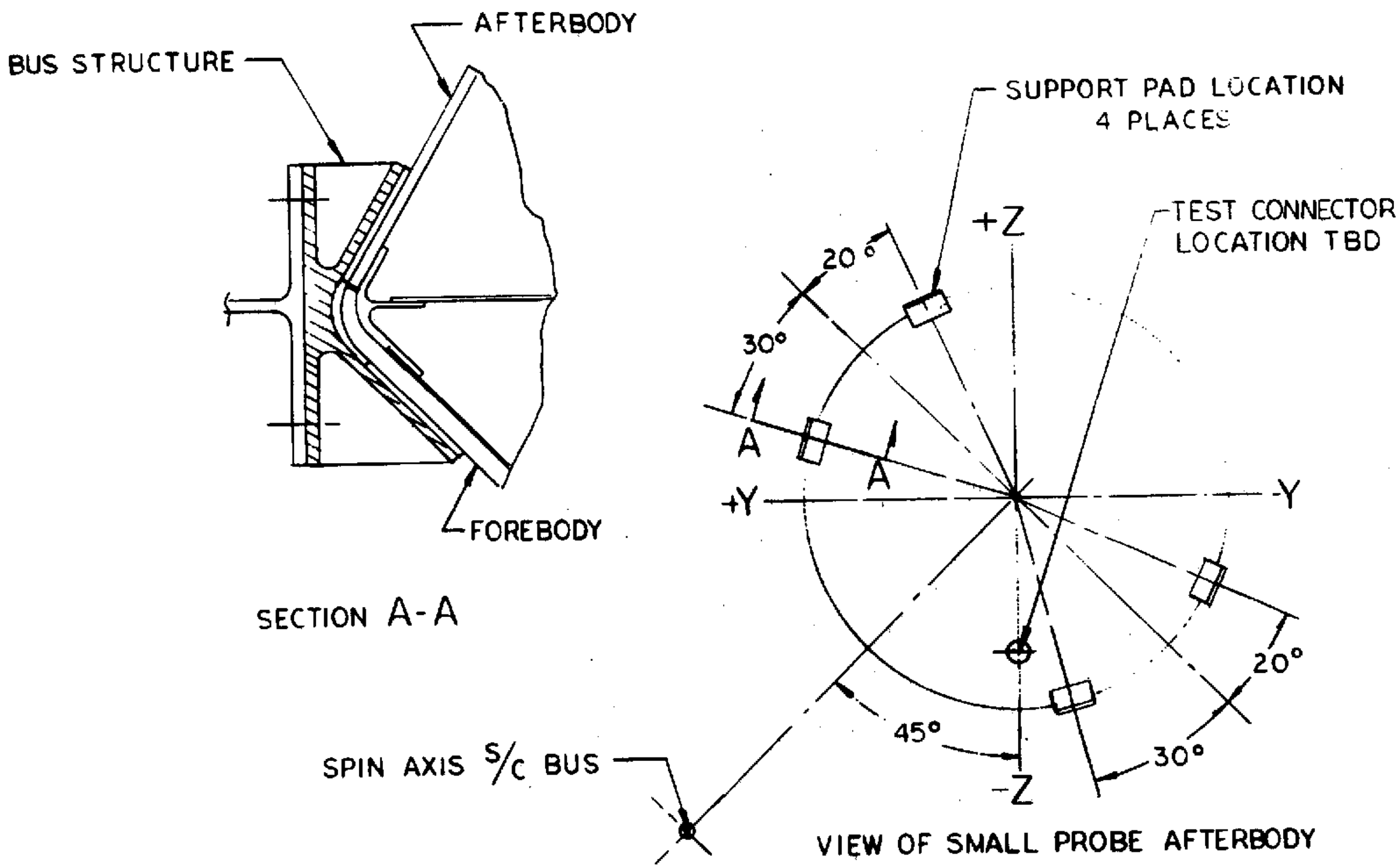

Figure 3.3-2. Placement of Small Probe Support Pads

probe afterbody. High density material with low susceptibility to creep shall be used for the heat shield in the areas contacted by the support pads.

The probe bus shall provide the support pads and associated structure and mechanisms in order to support each small probe throughout all mission phases up to the time of its release. The support pad material(s) and design shall be compatible with the interfacing small probe heat shield material. This compatibility includes meeting the requirement that no heat shield surface damage or deformation which might significantly affect entry performance shall be caused by the support pads. The probe bus shall provide for the release of each small probe on command in a manner which assures that the requirements of Section 3.3.5 will be met.

The small probe shall not exceed the envelope given in Figure 3.3-3.

The electrical umbilical required in Section 3.3.6 shall be designed to operate in a manner which assures that the requirements of Section 3.3.5 are met. This may be done by assuring mechanical separation of the electrical connections prior to probe release.

3.3.3 Power

The probe bus shall provide 28 VDC electrical power to each probe in order to meet the requirements in Section 3.3.6. Provision shall also be made in the probe and probe bus designs to permit checkout and charging of the probe batteries at any time up until shroud installation.

3.3.4 Thermal Interfaces

3.3.4.1 Large Probe

The forward portion of the large probe may be exposed to the space environment after launch vehicle shroud release and for the entire interplanetary cruise portion of the mission until the large probe release. The exposed surfaces of the large probe shall be compatible with this exposure and shall meet the requirements of Section 3.6.4.8. Thermal characteristics of these exposed surfaces shall be documented and controlled in the Probe/Probe Bus ICD (IF1-51).

The probe bus shall maintain appropriate attitudes throughout cruise so that the average solar input, in combination with heat supplied through the attach fittings from the bus, will keep the bulk temperature within the

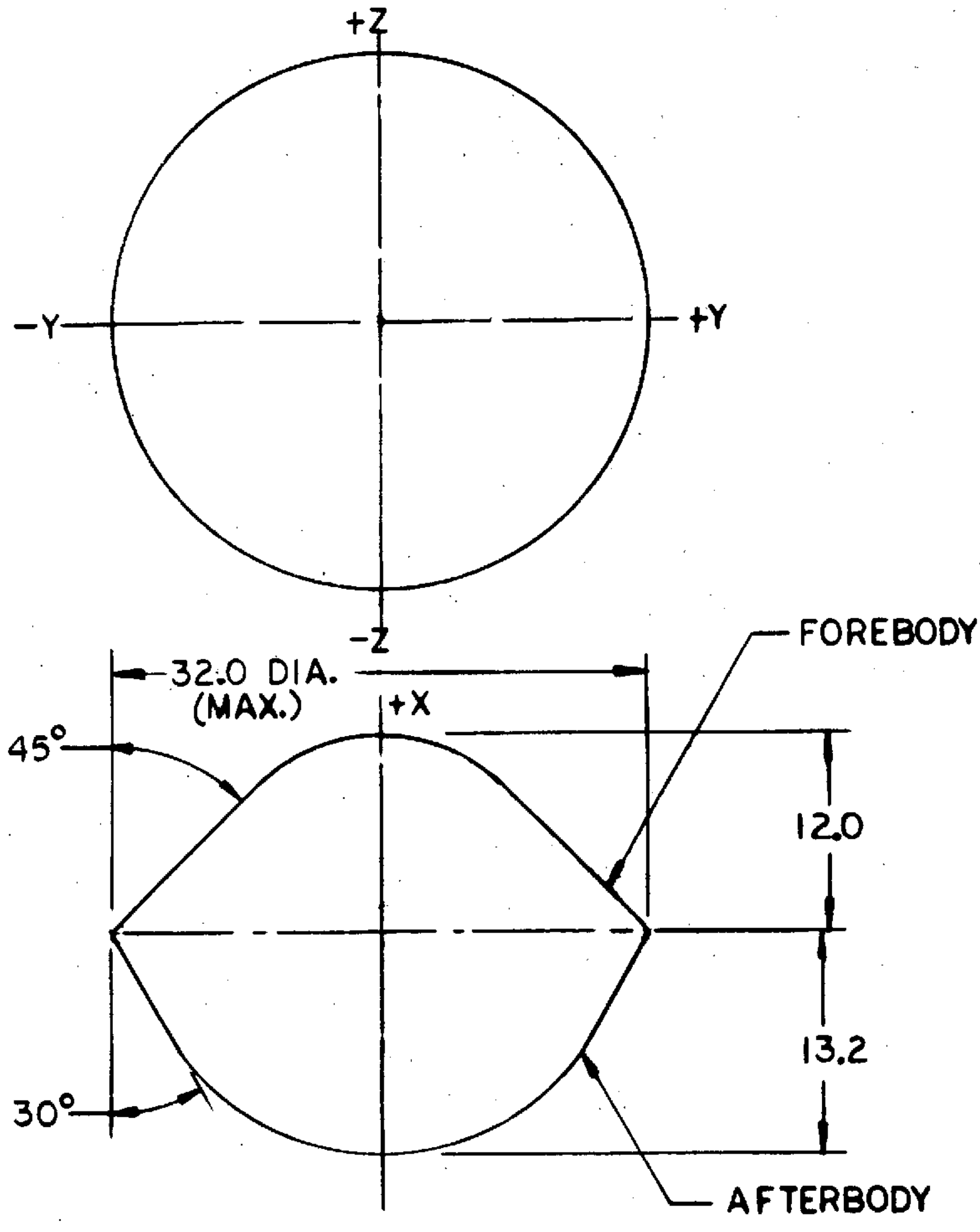

Figure 3.3-3. Small Probe Envelope Limitations

large probe aeroshell within the limits of -18 and 29°C (0 and 85°F) at all times prior to release from the bus.

3.3.4.2 Small Probe

The probe bus shall provide a thermal enclosure for each small probe through all mission phases up to 0.5 hour before release of the particular small probe. The thermal environment provided by the probe bus shall maintain the bulk temperature of each small probe between -18 and 29°C (0 and 85°F) at all times prior to release of the thermal cover on the small probe enclosure.

3.3.5 Separation Attitude and Dynamics

The nominal probe attitudes and velocities at probe release are functions of the requirements of Section 3.6.2 and of DSN tracking capabilities as specified in PV 1006.03, Spacecraft/DSN interface. The probes shall be compatible with those requirements and with the effects of attitude, spin rate, and velocity uncertainties within the bounds specified in Table 3.3-1.

The probe bus shall be capable of ensuring that uncertainties in probe attitude, spin rate, and velocity immediately after separation do not exceed the requirements of Table 3.3-1, due to errors or uncertainties in probe bus performance. Earth based calibrations or measurements of probe bus inflight performance may be used to aid in meeting this requirement but errors in Table 3.3-1 shall include effects of all residual uncertainties in probe bus attitude prior to probe release, in release timing, and in tip-off forces and moments during release.

Table 3.3-1

	Max. Angular Momentum Vector Direction Error (deg, 3σ)	Max. Spin Axis Nutation Angle About Momentum Vector (deg, 3σ)	Spin Rate (RPM, 3σ)	Maximum Velocity Error* (m/sec, 3σ)
Large probe	2	2	10.0 ± 0.1	0.02
Each small probe	3	3	10.0 ± 0.2	0.05

*Does not include bus position and velocity errors at release.

3.3.6 Test Provisions

An electrical umbilical connection shall be provided between the probe bus and each probe. Mechanical and electrical details of this umbilical shall meet the requirements of Probe/Probe Bus ICD (IF1-51), and the probe bus shall be responsible for mechanical separation of this umbilical connection prior to or during probe release.

As a minimum, this electrical umbilical shall permit prelaunch and cruise phase tests of probe battery and functional integrity of probe subsystems.

The probe bus shall provide appropriate command capability to stimulate and control each probe and shall provide data handling capability so that probe test data may be relayed to EGSE prior to launch or to earth via DSN stations during cruise. These capability requirements are summarized in Table 3.3-2.

Table 3.3-2. Summary of Probe Test Command and Telemetry Capabilities Required of Probe Bus

	Ground Test Mode	Cruise Mode
Maximum number of commands for each probe	64	64
Command format	8 bits serial	8 bits serial
Command bit rate	One command each 22 seconds	One command each 22 seconds
Telemetry block length	192 bits	192 bits
Telemetry bit rate	16 to 2048 bps at 50% duty cycle	16 to 2048 bps at 50% duty cycle

3.3.7 Dynamic Environments

The dynamic (structural) environments imposed on the probes during test, launch, cruise, and separation from the bus shall meet the requirements of Probe/Probe Bus ICD (IF1-51).

3.4 GENERAL DESIGN REQUIREMENTS AND CRITERIA

3.4.1 Failure Modes, Reliability, and Redundancy

3.4.1.1 Failure Modes

Modes of operation shall be provided for in the basic design or shall be available with the basic design such that a single failure of any electronic part of electromechanical control system assembly will not result in catastrophic failure of the probe bus or orbiter to complete their respective missions.

3.4.1.2 Reliability

Probe and orbiter spacecraft, exclusive of scientific instruments, shall as a goal exceed the minimum probability given in Table 3.3-3 of performing the nominal mission profile. The numerical probability of success shall account for degradation due to the environment and shall be calculated using failure rates provided by the spacecraft contractor, and subject to approval by the ARC/PPO. The probe heat shields and pressure vessels may be excluded from the numerical reliability model, but separation and deployment mechanisms shall be included. Assurance that these specific items meet overall mission reliability requirements will be assumed as a result of satisfying the product assurance requirements of Section 4.0.

Table 3.3-3. Reliability Goals for Probe and Orbiter Spacecraft

	Probability of Completing Nominal Mission
Probe bus	0.93
Large probe	0.90
Each small probe	0.90
Orbiter spacecraft	0.90

3.4.1.3 Redundancy

Redundancy shall be used judiciously so as to maximize the probability of achieving the mission objectives within the overall constraints of the mission profile. As a minimum, all pyrotechnically operated devices such as pin pullers, bolt cutters, etc., whose failure to operate would jeopardize continued mission success, shall be redundant or shall have redundant initiators and redundant firing circuits. Pyrotechnically operated devices which are part of a scientific instrument and whose operation affects only that instrument need not be redundant. The solid rocket motor used for orbit insertion shall have redundant arming and redundant firing circuits.

3.4.2 Interchangeability

Each small probe shall be interchangeable with any other small probe except that a total of 4 different transmitter frequencies shall be used, of which 3 frequencies shall be assigned for flight and the fourth frequency shall be reserved for the spare small probe.

Except where specifically required by differences in mission profile, the probe bus and orbiter spacecraft hardware shall be completely interchangeable with functionally identical systems, assemblies, or subassemblies. Within each spacecraft, interchangeability between functionally identical equipment is required.

3.4.3 Factor of Safety and Parts Derating

3.4.3.1 Electronic Parts

The contractor shall maintain a Pioneer Venus Approved Parts List. All parts and derating factors shall be subject to ARC/PPO approval. The Pioneer Venus Approved Parts List shall indicate the part type, manufacturer, and derating criteria. The parts selected for the list shall be of the JAN-TX and ERMIL type with precap visual inspection on integrated circuits.

3.4.3.2 Probe Pressure Vessels

The critical load condition for the probe pressure vessels occurs at the termination of the descent phase when the probes reach the surface of Venus. For this design condition the nominal pressure of 1400 psi

(9.6×10^6 Newtons/meter^2) may be treated as the ultimate design load for the pressure vessels. The pressure vessels shall survive a single exposure to ultimate load at the design temperature limit without catastrophic failure. The thermal control provisions for the pressure vessels shall have adequate margins to assure that the design temperature limits on the pressure vessels are not exceeded.

3.4.3.3 Pressure Vessels Involving Human Safety

Propellant tanks and other pressure vessels which may be pressurized prior to launch shall meet the requirements of AFETRM 127-1.

3.4.3.4 General Structural Factors of Safety

Reasonable factors of safety between rated performance and operating performance shall be applied to all structural elements not already specifically referenced in Sections 3.4.3.2 or 3.4.3.3. These factors of safety shall be subject to approval of the ARC/PPO.

3.4.4 Access and Testing

The spacecraft shall be designed with due consideration of the requirements and schedule for the integration and test program to provide:

a) Ease of access for installation and removal of subsystems and assemblies including scientific instruments and probes.

b) Ease of handling and transporting between and during the individual tests.

c) A capability of verifying overall system performance and integrity by testing major subsystems independently of the complete system so as to reduce the time required for and difficulty and complexity of full-system tests.

3.4.5 Operation in Flight

The requirements of this section are predicated on operation of the spacecraft in normal modes without having suffered any on board equipment failures.

3.4.5.1 General Requirements

The probe and orbiter spacecraft shall be designed to operate in all non-malfunction modes after separation from the launch vehicle so that all mission critical maneuvers or events can be safely accomplished without requiring tracking by or transmission of commands from any ground station to the spacecraft.

a) At any particular time

b) At any fixed period

c) For a time exceeding four hours

d) More frequently than twice per week.

In addition, for all subsystems, in the normal design mode of operation:

e) Required performance monitoring from telemetry data shall be accomplished within the above tracking constraints.

f) Transmission of a command from the ground at a time set by the monitored performance shall not be required.

g) A series of commands at precise time intervals shall not be required.

3.4.5.2 Special Mission Operations

Operations associated with midcourse velocity maneuvers, probe retargeting and release, probe and probe bus entry, and orbit insertion are excluded from the constraints of Section 3.4.5.1.

3.4.5.3 Readout of Orbiter Scientific Data

For the orbiter spacecraft daily readout of scientific instrument data may be assumed. The readout may occur at any time of day when the spacecraft is not occulted from earth. The readout shall not interfere with ongoing scientific measurements or normal operation of the spacecraft. No degradation to the system other than loss of scientific data shall occur if readout of the scientific data is not made.

As a goal, it shall be possible to read out the scientific data from an entire orbit from a single DSIF station during a single pass over that station. Use of a 64 m station may be assumed in order to meet this requirement. As an alternate, at any time in the orbiter mission it shall be possible to read out all the scientific data from each orbit assuming normal spacecraft operation and daily coverage by two 26 m DSIF stations.

3.4.5.4 Orbiter Scientific Instrument Commands

Certain of the scientific instruments which take data only near periapsis may be operated at reduced power during other portions of the orbit and their data need not be transmitted or stored for delayed transmission during portions of the orbit removed from periapsis as reflected in the requirements of PV 1006.02.

The orbiter spacecraft shall be designed so that if the spacecraft is operated normally, scientific instrument power control and data handling commands are not required from the ground at any particular time in order to assure continued mission success, except for possible loss of scientific data during periods when daily readout by DSN is not done.

The orbiter spacecraft shall provide operational capability and scientific instrument command storage capability so that during normal operation without spacecraft or scientific instrument failures, command and control of the scientific instruments may be accomplished within the constraints of Section 3.4.5.1 without any loss of valid scientific data.

3.4.5.5 Orbit Adjustment

The orbiter spacecraft shall be capable of performing velocity changes in order to adjust orbit period and periapsis altitude according to the requirements of Section 3.6.6.3 without interrupting communications with the earth.

3.4.6 Parts and Materials

3.4.6.1 General

The system shall be fabricated entirely with parts and materials and by processes that are spaceflight qualified for the specific application intended. The use of magnetic materials and materials subject to outgassing problems shall be selected consistent with the scientific objectives of the mission. All parts and materials shall be subject to approval by the ARC/PPO.

3.4.6.2 Commercial Parts

In general, use of commercial grade parts and materials shall be confined to fasteners, standard fittings, and minor hardware items. Standard material stock may be used with appropriate certification.

3.4.6.3 Prohibitions

The following items shall not be used:

a) Parts made of or coated with zinc, tin, or cadmium.

b) Magnesium parts without suitable surface treatment, if exposed to the environment.

c) Locktite or other thread binding materials.

3.4.7 Magnetic Properties

The magnitude of the magnetic field at the position of the magnetometer sensor on the orbiter, exclusive of the fields induced by the scientific instruments, shall not exceed $\underline{5}\gamma$ during flight.

The magnetic fields shall not exceed the above limits for all commandable and normal operating modes in flight which would not otherwise present acquisition of valid magnetometer data. Magnetometer data need not be obtained during midcourse maneuvers, probe release, or retargeting maneuvers; orbit insertion, or periapsis maintenance maneuvers.

3.4.8 Electromagnetic Compatibility (EMC)

3.4.8.1 General Requirements

Each spacecraft subsystem shall be electromagnetically compatible with all other spacecraft subsystems, and scientific instruments as specified in PV 1006.02. Electromagnetic compatibility requires that:

a) The normal operation of each spacecraft subsystem shall not be adversely affected by signals or voltage variations generated by other subsystems or scientific instruments as part of their normal operation

b) No spacecraft subsystem shall disturb normal operation of other subsystems or scientific instruments by the emission of signals or voltage variations other than those produced to perform its intended function.

The electromagnetic compatibility requirements shall be met for all commandable and normal operating modes in orbital flight.

3.4.8.2 Grounding and Specific EMC Requirements

In order to meet the requirements of Sections 3.4.7 and 3.4.8.1, the spacecraft electrical power subsystem and all using units shall, as a goal, use a single point ground system. Deviations from this goal are permissible providing that the requirements of Sections 3.4.7 and 3.4.8.1 are met. Signal grounds (involving currents of less than 500 μa) may be distributed, i.e., individual spacecraft units or scientific instruments may use the spacecraft structure for signal grounds.

In order to meet the requirements of Section 3.4.8, each spacecraft subsystem and the units which comprise it shall meet the requirements of TBD and TBD (EMC design criteria and requirements modified from Power 10/11).

3.4.9 Dynamics and Mass Properties

3.4.9.1 Expendables

The expendables shall include sufficient propellant to provide velocity and attitude control capabilities as specified in Sections 3.6.3.3 and 3.6.3.4 for the probe mission, and Sections 3.6.5.3 and 3.6.5.4 for the orbiter mission.

3.4.9.2 Weight and Weight Allocations

The spacecraft system weight for the probe and orbiter missions shall be in accordance with the launch vehicle performance capabilities for the selected launch opportunities, as defined in Section 3.6.1.

3.4.9.2.1 Probe Mission Spacecraft Weight and Weight Allocation

The spacecraft system weight, which includes the probe bus system, the large and small probes, and the launch vehicle/spacecraft adapter, shall not exceed 818.3 kilograms (1804 pounds), as specified in Section 3.6.1.1.

The following weight allocations within the spacecraft system shall not be exceeded.

	Kilograms	Pounds
Probe bus (Section 3.6.3.9)	226.8	500
Large probe (Section 3.6.4.9)	310.3	684
Small probes (3) (Section 3.6.5.9)	234.0 (78 each)	516 (172 each)
Launch vehicle/spacecraft adapter	47.2	104
	818.3	1804

3.4.9.2.2 Orbiter Mission Spacecraft Weight and Weight Allocation

The spacecraft system weight, which includes the orbiter spacecraft and the launch vehicle/spacecraft adapter, shall not exceed 552.2 kilograms (1224 pounds)

The following weight allocations shall be observed:

	Kilograms	Pounds
Orbiter spacecraft (Section 3.6.6.8)	508.0	1120
Launch vehicle/spacecraft adapter	47.2	104
	555.2	1224

3.4.9.3 Static Balance Constraints Imposed by Launch Vehicle

The spacecraft static balance shall comply with the requirements as imposed by the launch vehicle design constraints. The spacecraft center of gravity lateral displacement from the center-line of the spacecraft attach fitting that mates with the launch vehicle/spacecraft interstage structure may be as large as 254 millimeters (10 inches), provided the structural limitations of the Centaur are not exceeded and the separation tipoff rates as specified in PV 1006.04 are met.

3.4.9.4 Rigidity

The rigidity of the spacecraft shall be such that all resonant frequencies of the structure and assemblies are above 4 Hz in the launch configuration. As a design goal, the primary structural frequencies should be above 25 Hz in the axial direction and above 4 Hz in the lateral direction for the spacecraft rigidly attached at the separation plane.

3.4.10 Environments

3.4.10.1 Shipping and Handling

The spacecraft shall withstand a 4.5g load factor at the separation plane or the hard points, applied in any direction.

3.4.10.2 Storage

The spacecraft shall be capable of being stored in a suitable, spacecraft contractor supplied facility for a period of three years with minimal maintenance (periodic exercise of electrical equipment). Assemblies particularly susceptible to ambient environment may be removed from the spacecraft and stored separately under appropriate conditions (such as solar panels which possibly could not tolerate the specified humidity environment for protracted periods). Batteries are excluded from this storage requirement.

3.4.10.3 Test Environments

The spacecraft shall withstand the test environments specified in Section 4.0.

3.4.10.4 Launch Environments

The spacecraft shall withstand the checkout, on-stand, and powered flight environments as specified in PV 1006.04.

3.4.10.5 Interplanetary Environment

The spacecraft shall withstand the interplanetary environments as defined in NASA TM X-64627 "Space and Planetary Criteria Guidelines for Use in Space Vehicle Development 1971 Revision," November 15, 1971, with the exception of Section 1.3.2.1. The following shall be substituted for Section 1.3.2.1:

Solar High Energy Particle Radiation

Composition: Predominantly Protons (H^+)

Predicted Integrated Yearly Flux at 1.0 AU:

Annual Totals (protons/cm^2)

Year	Energy > 10 Mev	Energy > 30 Mev	Energy > 60 Mev
1978	2.1×10^9	2.9×10^8	9.2×10^7
1979	2.5×10^9	5.0×10^8	2.0×10^8

Note: For radiation environments at other than 1.0 AU, scaling according to the inverse square of the distance from the Sun may be employed.

Note: In the event of any conflicts between TM X-64627 and SP-8011, SP-8011 shall prevail.

3.4.10.6 Venus Atmosphere

The most probable molecular mass Venus atmosphere (Model I) given in NASA SP 8011 (revised September 1972) shall be used for a space craft design.

3.4.11 Planetary Quarantine

Because of the nature of the Venus environment, no special provisions beyond normal spacecraft manufacturing practice shall be required to avoid contamination of Venus by earth-based organisms.

3.4.12 Ground Support Equipment (GSE) Requirements

3.4.12.1 Ground Support Equipment for Launch Operations

As a minimum, ground support equipment shall be provided at CKAFS to permit functional tests of the subsystems with the spacecraft in the launch configuration and to permit removal and replacement of any probe with the spacecraft on the launch vehicle with the shroud removed.

3.4.12.2 Other GSE Requirements

TBD

3.4.13 Venting

All internal volumes within the aeroshell of each probe but outside of the pressure vessels shall be vented to avoid pressure differentials during test, launch, or entry which could cause sufficient structural deformation to jeopardize successful operation of the probe.

All internal volumes within the probe bus and the orbiter spacecraft shall be vented to avoid pressure differentials during test or launch which could cause sufficient structural deformation to jeopardize successful operation of either spacecraft.

3.5 ACCOMMODATIONS FOR SCIENTIFIC INSTRUMENTS

3.5.1 Probe Bus Scientific Instruments

3.5.1.1 General and Specific

The requirements set forth herein for individual scientific instruments are intended to be of a general nature for use in preliminary or gross design of the spacecraft. The specific requirements for each scientific instrument will be specified in PV 1006.02, will fit within the boundaries of these general requirements, and will be the governing requirements for the final design of the spacecraft system.

3.5.1.2 Capability

The bus shall have the capability to accommodate and meet the requirements of the individual scientific instruments as defined in the Supplementary Requirements Document, PV-1006.02, Scientific Instrument Requirements. The bus shall be capable of telemetering to earth the scientific data obtained throughout the critical entry period.

3.5.1.3 Mechanical Requirements

a) The bus shall be capable of mounting and supporting up to six scientific instruments, weighing a total of 40 lbs (18.1 kg) and occupying a total volume of 1281 in^3 (21,000 cm^3).

b) Each instrument will include suitable mounting tabs, or through-holes, for fastening the instrument to the bus. Required mounting screws or bolts shall be furnished as part of the bus system. Location, orientation and mounting accuracy for each scientific instrument shall be as specified in the individual instrument descriptions. (PV-1006.02)

c) Fields of view, viewing windows, and sampling inlet ports for the scientific instruments shall be provided to meet the requirement as indicated in the individual instrument descriptions. (PV-1006.02)

d) The bus system shall include provisions for stowing, retaining, and protecting scientific instrument assemblies and appendages for launch, and for unlatching and deploying sensors into suitable operating positions as specified in the individual instrument descriptions. This requirement includes storage for launch and subsequent unlatching of the spring-loaded sensor whip for the electron temperature probe and appropriate signals provided to trigger mechanisms integral to the instruments to remove protective covers.

3.5.1.4 Thermal Control Requirements

The bus system shall include thermal control to maintain temperatures of all internally mounted instruments, measured at the mounting surface, to between -40 and +50°C nonoperating, and between -20 and +50°C operating.

3.5.1.5 Electrical Requirements

a) The bus shall provide to the scientific instruments at least 30 watts of electric power at +28 VDC, ± 10 percent, as measured at the instrument input connectors when in the cruise attitude and from approximately 60 minutes before Venus entry until the end of the bus mission.

b) The bus shall provide the electric power to each scientific instrument as specified in the individual instrument descriptions. (PV-1006.02)

c) The bus shall provide individual power circuit protection provisions for each of the scientific instruments, so that a malfunction within any one instrument will not jeopardize performance of any other instrument or bus subsystem.

d) The bus shall include suitable wire harness to distribute power, timing signals, and data signals to and from each scientific instrument.

3.5.1.6 Timing and Reference Signal Requirements

The bus system shall provide timing signals to the scientific instruments, including but not limited to:

a) Roll reference pulse

b) Master clock

c) Bit shift pulse train

d) Word gates

e) Frame rate pulse.

3.5.1.7 Data Requirements

a) The bus shall be capable of accepting data from the scientific instruments in digital, analog, or bilevel state form and converting, and arranging these data in appropriate telemetry formats for time multiplexed transmission to earth.

b) Baseline individual instrument data acquisition requirements are defined in PV-1006.02 for the individual instruments. The bus shall be capable of data transmission rates which are compatible with complete recovery of all scientific data collected during the altitude interval 250 to 130 km for entry angles as steep as -25 degrees.

c) The bus shall be capable of accepting analog data signals from scientific instruments and converting them to digital signals with an accuracy of 8 bits or better.

d) A minimum of 14 bilevel status bits shall be provided for scientific instruments.

3.5.1.8 Command Requirements

The bus system shall provide a minimum of 20 commands for bus-mounted scientific instruments, including an on-off command for each instrument.

3.5.1.9 Instrument Pointing and Orientation Requirements

The bus shall be capable of maintaining spin axis orientation within ±10 degrees of the bus inertial velocity vector during the altitude interval from 250 to 130 km. The instantaneous bus orientation shall be determinable to within ± 2 degrees during this period.

3.5.2 Large Probe Scientific Instruments

3.5.2.1 General and Specific

The requirements set forth herein for individual scientific instruments are intended to be of a general nature for use in preliminary or gross design of the large probe. The specific requirements for each scientific instrument will be specified in PV 1006.02, will fit within the boundaries of these general requirements, and will be the governing requirements for the final design of the spacecraft system.

3.5.2.2 Capability

The probe shall have the capability to accommodate and meet the requirements of the individual scientific instruments as defined in the Supplementary Requirements Document, PV-1006.02 "Scientific Instrument Requirements."

3.5.2.3 Mechanical Requirements

a) The probe shall be capable of mounting and supporting up to 12 scientific instruments weighing a total of 77.2 lbs (35.0 kg) and having a total volume of 2440 in^3 (40,000 cm^3). Each instrument will include suitable mounting tabs or through-holes for fastening to the probe. Required mounting devices shall be provided by the probe.

b) Fields of view, spectral transmission, and optical properties of windows as required by the individual instruments (specified in PV-1006.02) shall be provided by the probe. A principal goal in instrument window performance shall be to provide predictable transmission properties throughout the measurement period of each instrument. The design selected shall minimize the accumulation of solid and liquid matter on the surfaces. The design shall provide assurance that at the beginning of scientific observations in subsonic descent, each window assembly shall have a spectral transmission equal to at least 95 percent of the nominal expected for a clean window. Further, similar performance shall be provided at approximately 20 minutes later during the descent.

c) Inlets and electrical penetrations passing through the pressure vessel shall meet the requirements of the individual instruments.

d) The probe shall provide and include means for deploying booms and protective covers for instrument sensors required by the individual scientific instruments as specified in PV-1006.02.

3.5.2.4 Thermal Control

The thermal control requirements for the probe are as specified for the bus in Section 3.5.1.4.

3.5.2.5 Electrical Requirements

a) The probe shall provide to the scientific instruments at least 120 watts of electrical power at 28 VDC ±10 percent as measured at the instrument connectors from approximately 15 minutes before entry until impact on the surfaces of Venus. In addition, at least 120 watt-hours of energy, of which no more than 30 watt-hours will be used at any one time, shall be provided for in-flight checkout of the instruments.

b) The probe shall include suitable wire harness to distribute power to each scientific instrument.

c) The probe shall provide individual power circuit protection provision for each of the scientific instruments so that a malfunction within any instrument will not jeopardize any other instrument or probe subsystem. This protection subsystem shall provide an automatic one-time reset during the post-entry descent.

3.5.2.6 Timing and Reference Signal Requirements

The probe shall provide timing and reference signals as required by each scientific instrument as specified in PV 1006.02.

3.5.2.7 Data Requirements

a) The probe shall be capable of accepting data from the scientific instruments in digital, analog, or bilevel state form and converting and arranging these data in appropriate telemetry formats for time multiplexed transmission to earth.

b) The scientific data acquisition rate is a function of trajectory and of the ballistic coefficient in terminal descent. Baseline individual instrument data acquisition requirements are defined in the PV-1006.02 for the individual instruments. The probe shall be capable of data transmission rates which are compatible with complete recovery of all scientific data collected during the mission.

3.5.2.8 Descent Rate Modulation Requirements

The probe approach, entry, and descent profile shall be designed to accommodate scientific requirements within the capacity of the data handling and communication systems. Minimum residence times and scientific data returned expressed as a function of altitude are specified in PV-1006.02.

3.5.2.9 Operating Atmosphere Within the Pressure Vessel

The large probe shall be designed to maintain the pressure internal to the pressure vessel above at least 4 psi throughout the mission prior to entry and for all normal operating temperatures. The nominal internal atmosphere shall be dry nitrogen of one atmosphere when the pressure vessel is at room temperature.

During entry and descent to the surface the total pressure vessel leakage rate shall be less than $\underline{5 \times 10}^{-3}$ standard cc/sec excluding any leakage through instruments which are designed to admit atmospheric gases. The total leakage to the interior of the pressure vessel through those instruments will be less than $\underline{1 \times 10^{-3}}$ standard cc/sec.

As the pressure vessel and internal equipment temperatures increase during entry and descent the internal pressure will rise but will not exceed 30 psi. The large probe shall not be required to be designed to limit this natural pressure increase due to temperature except it shall limit ambient temperatures in the pressure vessel to less than $\underline{150^{o}}$F.

3.5.2.10 Stored and Sequenced Commands

a. Coast Timer

The large probe shall be capable of generating a sequence of discrete commands for the scientific instruments based on the output of timing circuitry which will be initialized or started prior to separation from the probe bus. The large probe shall be designed to permit the exact number and relative timing of the commands to be modified prior to final assembly of the large probe. If hardware changes are necessary to meet this requirement they shall be of a minor nature which does not require requalification or extensive retest. The following capabilities shall be provided:

a) Number of discrete commands - at least four

b) Maximum time from initialization to last stored discrete - 32 days

c) Minimum time between sequential commands - 1 ± .1 min

d) Maximum command time uncertainty relative to initialization - ± 5 min.

b. Entry Timer

The large probe shall be capable of generating a second sequence of discrete commands for the scientific instruments based on time elapsed from 50g increasing. The large probe shall include a sensor or sensors which shall produce a discrete signal when probe axial acceleration reaches 0.5 ± 0.50 g and this signal shall be used to start or initialize the command sequence. The large probe shall be designed to permit the exact number and relative timing of the commands to be modified prior to final assembly of the probe. If hardware changes are necessary to meet this requirement they shall be of a minor nature which does not require requalification or extensive retest. The following capabilities shall be provided:

e) Number of discrete commands — at least 40

f) Maximum time from initialization to last stored command — 120 minutes

g) Minimum time between sequential commands — 1 ± 0.1 seconds

h) Maximum command time uncertainty relative to initialization — 1 second.

3.5.2.11 Large Probe Apertures for Viewing and Sampling from Within Pressure Vessel

3.5.2.11.1 Number and Size

The large probe shall be designed to accommodate and be compatible with at least 8 penetrations for science instrument optical or sample apertures. One shall be 3.00 ± 0.005 inches in diameter for use with the mass spectrometer. Seven shall be of a standardized design and shall have at least a 1.0 inch diameter opening for use with optical windows or the mounting of protruding sensors.

3.5.2.11.2 Optical Windows

The large probe shall provide four optical windows to fit the standardized one inch pressure vessel penetrations and to provide the viewing capabilities required in 3.2.5.3. Each window shall consist of an inner and outer lens element. Each element shall be mounted so as to provide a pressure seal and each element shall be capable of withstanding the ambient pressure throughout the descent to the surface of Venus, including the thermal effects on the inner element assuming that it is not protected by the outer element.

The large probe shall be designed to heat the outer element of each optical window so that its temperature is at least $10^{o}C$ above ambient throughout the descent from an altitude of at least 75 km to the surface.

The enclosed volume between window elements shall be evacuated to reduce heat transfer. The enclosed volume shall be purged with dry nitrogen prior to forming the final pressure seal, and the pressure betwee the window elements shall not exceed 10^{-4} atmospheres at $20^{o}C$.

3.5.2.11.3 Viewing Requirements

The optical windows and associated pressure vessel penetrations shall be arranged to provide fields of view which have the following characteristics:

a) A window which provides a 2.5 deg (half angle) viewing cone unobstructed by probe surfaces and looking forward (i.e., downward) parallel to the spin axis. A mirror may be used between the inner and outer window elements to fold the optical path so that the axis of the pressure vessel penetration need not be parallel to the spin axis.

b) A window with an integral mirror assembly to provide two fields of view for the same instrument, each having a 15 deg (half angle) clear cone around the central lines of sight in a plane containing the probe spin axis, one line of sight 45 deg from the forward spin axis and the other 135 degrees from the forward spin axis.

c) Two windows adjacent to each other each providing a 0.5 deg (half angle) conical field of view arranged to look out radially between 85 and 95 deg relative to the spin axis. The central lines of sight of the windows shall cross TBD inches from the outer window elements.

3.5.2.12 Viewing Requirements for Instruments External to the Pressure Vessel

TBD

3.5.2.13 Descent Capsule Spin Rate

The large probe descent capsule shall be designed to maintain a clockwise (viewed from behind and above) spin rate of at least 5 rpm.

3.5.3 Small Probe Scientific Instruments

3.5.3.1 General and Specific

The requirements set forth herein for individual scientific instruments are intended to be of a general nature for use in preliminary or gross design of the small probes. The specific requirements for each scientific instrument will be specified in PV 1006.02, will fit within the boundaries of these general requirements, and will be the governing requirements for the final design of the spacecraft system.

3.5.3.2 Small Probe Scientific Accommodations

The scientific accommodations on the Small Probe shall be as specified for the Large Probe, Section 3.5.2, except as modified below:

a) Section 3.5.2.3 a) modified as follows: Each probe shall be capable of mounting and supporting up to 5 scientific instruments weighing a total of 5.2 pounds (2.4 kg) and having a total volume of 85 in^3 (1390 cm^3). Each instrument will include suitable mounting tabs or through-holes for fastening to the probe. Required mounting devices shall be provided by the probe.

b) Section 3.5.2.5 a) modified as follows: Each probe shall provide to the scientific instruments at least 10 watts of electrical power at 28 volts dc ±10 percent as measured at the instrument connectors from approximately 15 minutes before entry until impact on the surfaces of Venus. In addition, at least 10 watt-hours of energy, of which no more than 3 watt-hours will be used at any one time, shall be provided for in-flight checkout of the instruments.

c) Section 3.5.2.10 b) modified as follows: The number of discrete commands in item e) shall be reduced from at least 40 to at least 15.

d) Section 3.5.2.13 shall be deleted.

e) Section 3.5.2.11 shall be replaced by Section 3.5.3.3.

3.5.3.3 Small Probe Apertures for Viewing and Sampling from Within Pressure Vessel

3.5.3.3.1 Number and Size

The small probes shall be designed to accommodate and be compatible with at least 3 penetrations for science instrument optical or sample apertures which shall be of a standardized design and shall have at least

1.0 inch diameter opening for use with optical windows or the mounting of protruding sensors.

3.5.3.3.2 Optical Windows

Each small probe shall provide two optical windows to fit the standardized one inch pressure vessel penetrations and to provide the viewing capabilities required in 3.3.5.3. Each window shall consist of an inner and outer lens element. Each element shall be mounted so as to provide a pressure seal and each element shall be capable of withstanding the ambient pressure throughout the descent to the surface of Venus, including the thermal effects on the inner element assuming that it is not protected by the outer element.

The small probes shall be designed to heat the outer element of each optical window so that its temperature is at least 10°C above ambient throughout the descent from an altitude of at least 75 km to the surface.

The enclosed volume between window elements may be evacuated to reduce heat transfer. In any event the enclosed volume shall be purged with dry nitrogen prior to forming the final pressure seal, and the pressure between the window elements shall not exceed 1.1 atmospheres at 20°C.

3.5.3.3.3 Viewing Requirements

The optical windows and associated pressure vessel penetrations shall be arranged to provide fields of view which have the following characteristics:

a) Two windows adjacent to each other each providing a 0.5 deg (half angle) conical field of view arranged to look out radially between 85 and 95 deg relative to the spin axis. The central lines of sight of the windows shall cross TBD inches from the outer window elements.

3.5.3.4 Pressure Gauge Sensing Port

Each small probe shall provide a pressure gauge sensing port which shall be located at the forward stagnation point. As a minimum the pressure port shall be opened for pressure measurement prior to descent to TBD km and shall remain open for the rest of the nominal mission. The small probes shall not be required to seal the sensing port at any point in the mission except as necessary to provide for small probe integrity during entry. Each small probe shall provide a pressure tight line leading from the sensing port to the pressure gauge instrument inside the pressure vessel.

3.5.3.5 Deployment of Temperature Probe and Nephelometer

Each small probe shall deploy the temperature probe and nephelometer after peak heating during entry and before descending to an altitude of TBD km. When deployed the temperature probe shall protrude at least TBD inches from the surface of the forward portion of the aeroshell along a line normal to the local aeroshell surface so that it is exposed to the free stream.

3.5.4 Orbiter Spacecraft Scientific Instruments

3.5.4.1 General and Specific

The requirements set forth herein for individual scientific instruments are intended to be of a general nature for use in preliminary or gross design of the large probe. The specific requirements for each scientific instrument will be specified in PV 1006.02, will fit within the boundaries of these general requirements, and will be the governing requirements for the final design of the spacecraft system.

3.5.4.2 Scientific Accommodation

a) The orbiter shall have the capability to accommodate and meet the requirements of the individual scientific instruments as defined in PV 1006.02, "Scientific Instrument Requirements."

b) The orbiter shall be capable of performance which permits the establishment of the above specified orbit around Venus and which shall remain fully operational over a minimum of 243 earth days.

3.5.4.3 Mechanical Requirements

a) The orbiter spacecraft shall be capable of mounting and supporting up to 10 scientific instruments, weighing a total of approximately 100 pounds (45 kg) and occupying a total volume of approximately 3540 in^3 (58,000 cm^3).

b) The orbiter shall provide all mechanically positionable platforms or gimbals required for instrument sensor pointing and viewing.

c) The orbiter shall provide a mechanical boom upon which the magnetometer sensor of 1.7 pounds (0.77 kg) can be mounted. The boom shall be of sufficient length to place the magnetometer sensor at a distance from the spacecraft such that the requirements specified in Section 3.3.1.3 are met. The cable within the boom shall be considered part of the orbiter system.

d) Additional mechanical requirements for scientific instrument accommodation are as specified in Sections 3.5.1.3 b) through 3.5.1.3 d).

3.5.4.4 Thermal Control Requirements

The orbiter spacecraft shall provide thermal control to maintain temperatures of all internally mounted instruments, measured at the mounting surface, to between -40 and +50°C nonoperating and between -20 and +40°C operating.

3.5.4.5 Electrical Requirements

The orbiter shall be capable of supplying to the scientific instruments at their input connectors at least 40 watts of electrical power during the interplanetary phase of the mission and at least 130 watts during the in-orbit phase at a voltage of +28 VDC ±10%. Other electrical requirements for scientific instrument accommodation are as specified in Sections 3.2.2.2.3 b) through 3.2.2.2.3 d).

3.5.4.6 Timing and Reference Signal Requirements

The timing and reference signal requirements for the orbiter are as specified for the bus in Section 3.2.2.2.5.

3.5.4.7 Data Requirements

a) The bus shall be capable of accepting data from the scientific instruments in digital, analog, or bilevel state form and converting and arranging these data in appropriate telemetry formats for time multiplexed transmission to earth.

b) The scientific data acquisition rate is a function of the spacecraft location in orbit. Baseline individual instrument data acquisition requirements are defined in PV 1006.02, "Scientific Instrument Requirements." The orbiter shall be capable of data storage capacities and transmission rates which are compatible with the complete recovery of all scientific data collected during each orbit.

c) The orbiter shall be capable of accepting analog data signals from scientific instruments and converting them to digital signals with an accuracy of 8 bits or better.

d) A minimum of 30 bilevel status bits shall be provided for scientific instruments.

3.5.4.8 Command Requirements

The orbiter shall provide up to 75 commands for scientific instruments, including an on-off command for each instrument and a single command to turn off all instruments simultaneously.

3.5.4.9 Instrument Pointing

The orbiter shall meet the instrument pointing requirements as specified in PV 1006.02, "Scientific Instrument Requirements."

3.5.4.10 Radio Frequency Occultation

The spacecraft shall have the capability to perform a dual-frequency radio occultation experiment during the initial earth occultation season after arrival at Venus. The frequencies shall be S-band (the spacecraft telemetry downlink) and X-band (an RF system to be added specifically and only for the occultation experiment). The flight hardware required for both links shall be provided by the contractor. Requirements for this experiment are:

a) The S- and X-band links shall be capable of simultaneous operation.

b) The two links shall be compatible with DSN capabilities.

c) Each link shall be receivable for atmospheric refractions of up to 10 degrees.

d) The period of occultation measurements will normally be limited to the first 40 days in orbit.

e) The frequency relationship of the S- and X-band downlinks shall be the precise ratio of 3/11 for both coherent and noncoherent S-band communication modes.

f) The signal level at the DSN receiver shall be -178 dBm or greater for both links.

g) The 64-meter DSS antennas will be used for all occultation measurements.

3.6 PERFORMANCE REQUIREMENTS

3.6.1 Launch, Interplanetary Trajectories, and Orbit Parameters

3.6.1.1 Probe Mission

3.6.1.1.1 Trajectory

A Type I trajectory will be used for the probe mission.

3.6.1.1.2 Launch Opportunity

The 1978 opportunity shall be used as a basis for design of the probe mission spacecraft. The nominal launch and arrival dates are:

Launch dates: August 20 to 29, 1978
Arrival date: December 17, 1978
Departure vis viva energy, $C_3 \leq 9.6\ km^2/sec^2$
Arrival vis viva energy, $C_3 \leq 25.4\ km^2/sec^2$

3.6.1.1.3 Launch Vehicle and Interface

The design restraints and interfaces of the Atlas/Centaur shall be used as a basis for the multi-probe spacecraft interface design. The launch vehicle interface shall be at station 163.90 of the Centaur equipment module.

3.6.1.1.4 Spacecraft Mass

The mass of the multiprobe mission payload at launch, including spacecraft, adapter and separation hardware, expendables, and contingencies shall not exceed the allocations in Section 3.4.9.2.1.

3.6.1.1.5 Expendables

The expendables shall include sufficient propellant to perform maneuvers required in Section 3.6.3.3.3.2.

3.6.1.2 Orbiter Mission

3.6.1.2.1 Trajectory

A Type II trajectory (heliocentric transfer angle > 180 deg) shall be used as a basis for design of the orbiter mission spacecraft.

3.6.1.2.2 Launch Opportunity

The 1978 opportunity shall be used as a basis for design of the orbiter mission spacecraft. The nominal launch and arrival dates which define the 1978 Type II opportunity are:

Launch dates: May 26 to June 5, 1978
Arrival dates: December 11 to 14, 1978
Departure excess energy: $C_3 \leq 19.795\ km^2/sec^2$
Arrival excess energy: $C_3 \leq 10.966\ km^2/sec^2$
ΔV for orbit insertion at Venus, $\Delta V = 954 \pm$ TBD m/sec

3.6.1.2.3 Launch Vehicle and Interface

The design restraints and interface of the Atlas/Centaur shall be used as a basis for the spacecraft design. The launch vehicle interface shall be at station 163.90 of the Centaur equipment module.

3.6.1.2.4 Spacecraft Mass

The mass of the orbiter mission spacecraft at launch, including attach fitting, separation hardware, spacecraft, experiments, expendables, and contingencies shall not exceed the allocation given in Section 3.4.9.2.2 .

3.6.1.2.5 Expendables

The expendables shall include sufficient propellant to perform the maneuvers required in Section TBD.

3.6.1.3 Orbit Characteristics and Operational Requirements

3.6.1.3.1 Orbiter Mission Duration

The orbiter spacecraft shall be designed for a nominal mission life of 243 days in orbit about Venus after launch and interplanetary transit as specified in Section 3.6.1.2.

3.6.1.3.2 Planetary Approach

The orbiter spacecraft will be targeted so that the initial orbit inclination will be 120 $\pm$1 degree relative to the heliocentric Venus orbit plane and so that the nominal periapsis latitude after orbit insertion will be 45 degrees south. The initial attitude at periapsis will be 400 $\pm$ 100 km.

3.6.1.3.3 Orbit Period

The nominal orbit period shall be 24 hours. The orbiter spacecraft shall be designed to be compatible with orbit periods ranging from 20 to 28 hours.

3.6.1.3.4 Orbit Insertion

The orbiter spacecraft shall include a solid rocket motor to provide the velocity increment for orbit insertion. The spacecraft shall be designed so that it can be placed in the proper attitude for orbit insertion at least 20 hours prior to the actual rocket firing and remain in that attitude until 20 hours after orbit insertion without permanent degradation of any spacecraft equipment or scientific instruments.

Capability shall be provided to load and store redundant arming and redundant firing commands for the solid rocket motor at any time between 0.1 and at least 20 hours prior to the commanded firing time. Arming commands shall not be issued prior to the acceptable firing window. Firing commands shall be issued within ±20 seconds of the commanded time as measured from receipt of the original time increment command via RF link.

The solid rocket motor shall be sized to provide an initial orbit period of 24 ±1 hour if the orbiter spacecraft arrives with zero position and velocity errors after having used all of the propellant allocated for midcourse velocity adjustments prior to arrival at Venus.

3.6.1.3.5 Initial Orbit Adjustment

The orbiter spacecraft shall be designed with operating modes selectable by ground command which permit adjustments of initial orbit period and periapsis altitude. Sufficient propellant capacity shall be provided to adjust spacecraft velocity by at least 6 meters/second for initial orbit adjustment. The operating modes provided shall permit even larger velocity adjustments by using propellant allocated for other functions (such as excess midcourse propellant).

3.6.1.3.6 Periapsis Altitude Control

3.6.1.3.6.1 Basic Requirements

The orbiter spacecraft shall be designed with operating modes selectable by ground command which permit periodic velocity adjustments to be made in order to maintain periapsis altitudes within the range of 200 to 400 km throughout the nominal mission. Velocity adjustment capability equivalent to at least 44 meters/second applied tangentially to the orbit at apoapsis shall be provided.

3.6.1.3.6.2 Operational Flexibility

The operating modes provided shall permit even larger velocity adjustments by using propellants allocated for other functions.

The operating modes provided shall permit velocity adjustments for periapsis altitude control on a more frequent basis than is required to meet the requirements of Section 3.6.1.3.6.1. As a goal, 50 additional velocity adjustment maneuvers shall be possible without exceeding thruster cold start capabilities.

The operating modes provided shall permit periapsis altitudes less than 200 km to be achieved by appropriate ground commands. No propellant capacity shall be required for this function but use of propellant allocated for other functions shall be possible.

3.6.2 Probe and Bus Targeting and Entry Trajectories

3.6.2.1 Large Probe

The multiprobe spacecraft shall have the capability for targeting the large probe to enter the Venusian atmosphere on the daylight side of Venus not closer than 15 degrees to the terminator and within ±10 degrees latitude of the subsolar trace.

3.6.2.2 Small Probes

The multiprobe spacecraft shall have the capability for targeting the three small probes to enter the Venusian atmosphere at locations which satisfy the following requirements:

a) Two of the three small probe entry locations shall be at latitudes which differ in absolute value by at least 30 degrees. There is no requirement for entry into a specific hemisphere.

b) Two of the four probe entry locations shall have a longitudinal separation of at least 90 degrees. There is no requirement for small probe entry into the daylight side of the Venusian atmosphere.

3.6.2.3 Bus

The bus shall be capable of being targeted to enter the atmosphere on the daylight side of Venus at a location where the entry flight-path angle is less steep than -25 degrees.

3.6.2.4 Definition of Entry

Unless otherwise specifically stated, entry as a discrete event is defined at the point when a probe or the bus reaches a distance of 6250 km from the center of Venus (altitude of 200 km above the assumed mean surface). The term entry phase is used to describe events or the trajectory between "entry" as defined above, through the period of peak heating and acceleration, until the vehicle velocity has slowed to approximately Mach 1, at which time the descent phase begins, continuing until the nominal end of the mission when the probe(s) reaches the surface of Venus at a radius of 6050 km.

3.6.2.5 Time of Arrival

The probe bus shall provide for adjustment of velocity prior to and after release of each probe so that probe and bus arrival time may be controlled by ground command. The large and small probe designs shall permit separations in arrival time ranging from 0 to 30 minutes. The probe bus design shall permit its arrival to be delayed up to 1.5 hours after entry of the first probe. The probe bus and probe designs shall be compatible with targeting the probes for simultaneous arrival.

3.6.2.6 Probe Bus Design for Entry

As a goal, the probe bus shall survive and maintain communications with the earth until an altitude of about 130 km is reached. Fields of view for the bus scientific instruments shall meet the requirements of Section 3.5.1 and the angle of attack shall not exceed ±10 degrees between the altitude of 250 to 130 km (achieved by controlling spin axis attitude to the desired orientation prior to entry). The bus shall be spinning at a rate of 60 ± 1 rpm at entry. No other requirements shall be imposed to improve the probability of bus survival past entry.

3.6.2.7 Probe Entry Angle Constraints

3.6.2.7.1 Large Probe

The large probe shall be designed for safe entry and successful completion of its mission when the flight path angle at entry is between -34.5 and -40.5 degrees relative to local horizontal at entry.

3.6.2.7.2 Small Probe

Each small probe shall be designed for safe entry and successful completion of its mission when the flight path angle at entry is between -15 and -60 degrees relative to local horizontal at entry.

3.6.2.8 Probe Entry Velocity Constraints

3.6.2.8.1 Large Probe

The large probe shall be designed for entry at any velocity (relative to Venus) between 11.2 and 11.34 km per second.

3.6.2.8.2 Small Probe

Each small probe shall be designed for entry at any velocity (relative to Venus) between 11.2 and 11.34 km per second.

3.6.2.9 Descent Time and Science Exposure Time

3.6.2.9.1 Large Probe

The hypersonic ballistic coefficient of the large probe shall, for the entry flight path angles and velocities specified in Sections 3.6.2.8 and 3.6.2.9 cause the large probe to decelerate to subsonic velocities by the time it reaches approximately 70 km above the mean surface for the Venus model atmosphere specified in Section 3.4.10.6. Between 71.7 and 69.2 km, the parachute decelerator shall be deployed, and viewing and sampling science instruments shall be exposed. The subsonic ballistic coefficient of the probe parachute shall cause the descent capsule to descend at rates which shall not exceed limits specified in Figure 3.6-1. The descent capsule shall remain on the parachute until it has descended to an altitude in the range from 48 to 40 km, at which time the parachute shall be jettisoned. The subsonic ballistic coefficient of the descent capsule shall provide a terminal descent velocity which shall not exceed the limits specified in Figure 3.6-1. Lower descent rates may be provided but the probe shall

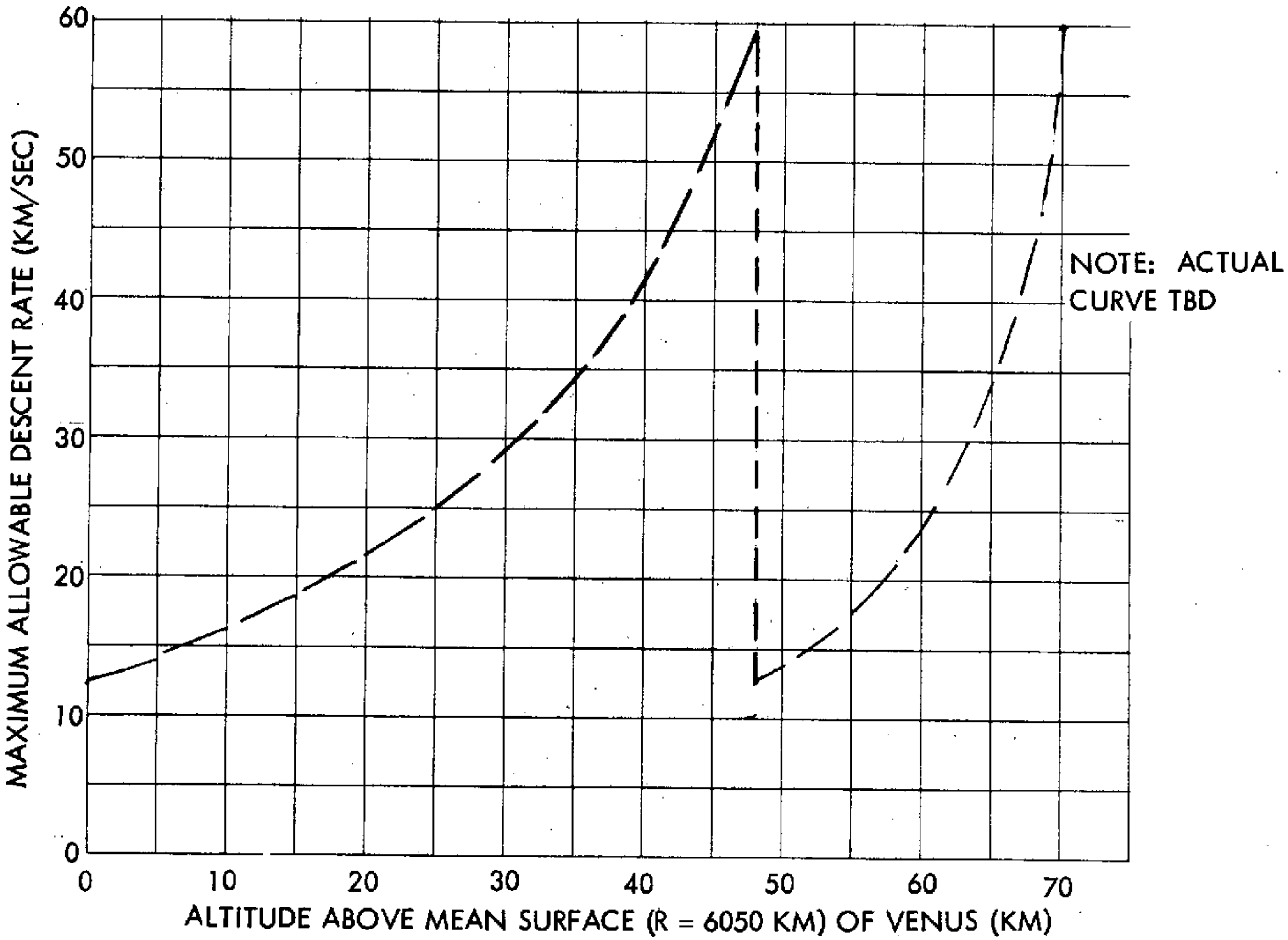

Figure 3.6-1. Large Probe Descent Rate Requirements

be capable of successful operation until it reaches a radius of 6050 km at the minimum probable (3σ) descent rates.

3.6.2.9.2 Small Probes

For the range of entry flight path angles and velocities specified in Sections 3.6.2.8 and 3.6.2.9, the small probes shall decelerate to a Mach number of 1.6 or less by the time they reach an altitude of 67.0 km. The (common) hypersonic ballistic coefficient of the small probes shall be compatible with this requirement. Deployment of small probe science instruments shall be initiated by a timer at a fixed (and common) time delay after the probe has encountered a 50 g level. Because of the different entry flight path angles, the science deployment altitude corresponding to this time delay will be different for the three small probes. The time delay shall be selected so that deployment occurs above an altitude of 67 km and below a Mach number of 1.6. The subsonic ballistic coefficient of the small probes shall provide a terminal descent velocity which shall not exceed the limits specified in Figure 3.6-2. The total descent time from entry to impact may be increased but the small probe shall be capable of successful operation until they reach a radius of 6050 km at the minimum probable (3σ) descent rates.

3.6.2.10 Targeting Constraints on Earth Communication Angle

3.6.2.10.1 Large Probe

The large probe shall be designed to provide adequate antenna gain to meet the communications requirements of Section 3.6.4.11 during descent for any entry and descent trajectory which meets the requirements of Sections 3.6.2.8.1 and 3.6.2.9.1 and which also satisfies the requirement that the angle between the earth line and the local vertical through nominal impact point of the large probe (in the absence of winds) does not exceed 55 degrees. In designing to meet this requirement, the effects of initial angle of attack at entry, and attitude disturbances due to staging shall be separately accounted for in determining the total communications angle between the large probe axis of symmetry and the earth line.

3.6.2.10.2 Small Probe

Each small probe shall be designed to provide adequate antenna gain to meet the dommunications requirements of Section 3.6.5.11 during descent

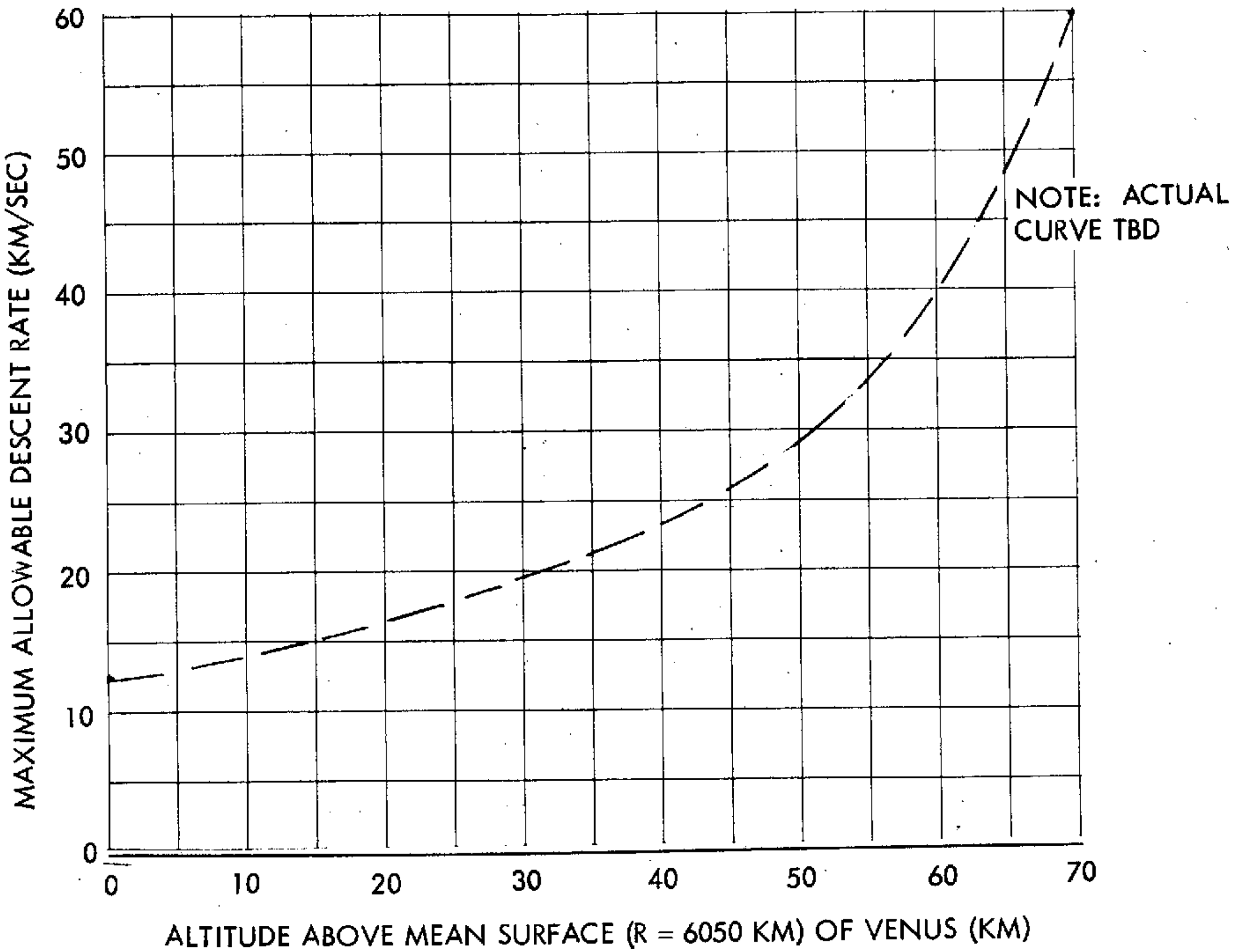

Figure 3.6-2. Small Probe Descent Rate Requirements

for any entry and descent trajectory which meets the requirements of Sections 3.6.2.8.2 and 3.6.2.9.2 and which also satisfies the requirement that the angle between the earth line and the local vertical through the nominal impact point of that small probe (in the absence of winds) does not exceed 55 degrees. In desinging to meet this requirement, the effects of initial angle of attack at entry and attitude disturbances due to the effects of exposing the scientific instruments shall be separately accounted for in determining the total communications angle between the small probe axis of symmetry and the earth line.

3.6.2.10.3 Probe Bus

The probe bus shall be designed to meet the entry communications requirements of Section 3.6.3.5 for entry trajectories which meet the requirements of Section 3.6.2.4 and for which the bus/earth line at entry is within 25 degrees of the velocity vector at entry. In designing to meet this requirement, the effects of uncertainties in spin axis orientation due to attitude control subsystem errors shall be separately accounted for in determining antenna requirements.

3.6.2.11 Probe Free Flight Between Separation and Entry

3.6.2.11.1 Separation Tables

The large probe and the probe bus shall be designed to be compatible with separation 25 ± 3 days prior to large probe entry. Each small probe and the probe bus shall be designed to be compatible with separation of that small probe 17 ± 5 days before entry of that probe.

3.6.2.11.2 Range of Solar Aspect During Probe Free Flight

For the 1978 probe mission, targeting the large probe so that entry angles and velocities are within the ranges specified in Sections 3.6.2.5.1 and 3.6.2.6.1, so that the nominal earth communication angles of 55 degrees (see Section 3.6.2.8) are not exceeded, and so that entry will occur with the probe at zero nominal angle of attack, will result in release of the large probe in a nominal attitude such that the solar aspect angle is TBD degree at release and changes to 72 ± 1 degree at entry. The large probe, and in particular, its thermal control subsystem shall be compatible with the above ranges of solar aspect angles during free flight.

Similarly, the range of nominal solar aspect angles of any particular small probe at release is between TBD and TBD degrees and the range of nominal solar aspect angles for any particular small probe at entry is between TBD and TBD degrees. Each small probe and in particular its thermal control subsystem shall be compatible with the above ranges of solar aspect angles during free flight.

3.6.2.12 Probe Angle of Attack at Entry

Uncertainties (primarily due to tracking errors) in bus position and velocity relative to Venus when the probes are released will result in variations in flight path angle from the planned value and this then will result in uncertainties in probe angle of attack at entry. The probes shall be designed to be compatible with angle of attack variations at entry due to flight path uncertainty within the following limits:

	Maximum contribution to angle of attack error due to entry angle uncertainties (deg, 3σ)
Large Probe	4
Small Probe	4

Each probe will be released in an attitude intended to provide zero angle of attack at entry. In addition to the above variations which may occur in angle of attack at entry, each probe shall be compatible with angle of attack variations resulting from separation dynamics as specified in Section 3.3.6 and with angle of attack variations resulting from effects of the environment (such as solar pressure) on the probe attitude during free flight after release and prior to entry.

3.6.3 Probe Bus Performance Requirements

3.6.3.1 Structural Requirements

3.6.3.1.1 General Requirements

The probe bus structure subsystem shall provide for the support and alignment of the other probe bus subsystem equipment, the probe bus scientific instruments, one large probe, and three small probes.

3.6.3.1.2 Compatibility

The probe bus structure shall be compatible with the Atlas Centaur launch vehicle as specified in Section 3.2.1, with the probes as specified in Section 3.3, and with the scientific instruments as specified in Section 3.5.

3.6.3.1.3 Alignment

Provision shall be provided for initial alignment of attitude sensors, communication antennas, attitude control and ΔV thrusters, and scientific instruments to a common spacecraft datum. The initial alignment and subsequent alignment stability throughout the mission (except during launch) shall be adequate for the spacecraft to meet the requirements of Sections 3.6.3.3., 3.6.3.4, and the scientific instrument pointing requirements of PV 1006.02.

3.6.3.1.4 Rigidity

The structural rigidity of the spacecraft shall be in accordance with the requirements of Section 3.4.9.5.

3.6.3.1.5 Vibration Amplification

The vibration amplification in the vicinity of the scientific instruments on the probe bus shall not exceed the ratios given in Table 3.6-1. Vibration amplification is defined as the ratio of the acceleration experienced at the point of interest to the input acceleration due to primary loads during powered flight at the mechanical interface between the launch vehicle and the spacecraft.

3.6.3.1.6 Weight, Balance, and Inertia

The probe bus structure shall be designed so as to permit mounting of subsystems, scientific instruments, and the probes in position to insure proper weight, balance, and inertial characteristics compatible with all spacecraft subsystems, the scientific instrument scan requirements, probe release dynamics, and launch vehicle payload restraints.

3.6.3.1.7 Primary Loads

The probe bus structure, including all appendages shall withstand the primary loads specified below without incurring any damage.

Table 3.6-1. Maximum Vibration Amplification at Probe Bus Scientific Instrument Mounting Locations

Vibration Axis	Frequency (Hz)	Ratio
Thrust	10 to 30	4
	30 to 100	13
	100 to 130	4
	130 to 2000	1
Lateral	1 to 100	10
	100 to 2000	2
Torsional	(same as lateral)	

Note: Amplification ratios on the equipment compartment side panels in the 30 to 100 Hz range in the thrust axis and the 1 to 100 Hz range in the lateral and torsional axes may exceed the listed values within narrowband frequency ranges. The bandwidths wherein the ratios may exceed the listed values shall be less than 0.2 time the center band frequency.

a) Handling and shipping loads in accordance with Section 3.4.10.1.

b) Aerodynamic loads encountered during spin balance operations as required in Section 4.0.

c) Probe separation loads encountered during the separation tests as required in Section 4.0

d) Loads due to structural and acoustic excitation during launch as specified in PV 1006.04.

3.6.3.1.8 Engineering Telemetry

The condition of all deployable appendages (stowed or deployed), of the launch vehicle/spacecraft interface (attached or separated) of small probe thermal covers (attached or separated), and of each probe (attached or separated) shall be contained in the engineering telemetry data.

3.6.3.1.9 Provisions for Ground Handling

At least three hard points shall be provided on the probe bus structure for attachment of handling, measuring, and shipping equipment. In addition, access shall be provided so that any probe may be removed and replaced at CKAFS using suitable ground handling equipment as required in Section 3.4.12.

3.6.3.2 Electrical Power Subsystem

3.6.3.2.1 Description

The probe bus electrical power subsystem shall consist of a solar array, power controls, a battery, fault protection devices, regulators, converters, and power cabling.

3.6.3.2.2 Capability

The electrical power subsystem shall have the capability of distributing the design power to the probe bus subsystem and scientific instruments as required, converting and regulating the voltage as required, and protecting the subsystem, other subsystems, and the scientific instruments from momentary or prolonged overloads or undervoltage conditions. The electrical power subsystem shall also provide power for probe checkout to meet the requirements of Section 3.3.6.

3.6.3.2.3 Design Power

The electrical power subsystem shall provide power for full normal operation of all spacecraft subsystems required to perform simultaneously and continuously the following functions:

a) Provide downlink communication modulated with telemetry data and with the RF power amplifier at design power (which may be switched as a function of communication range).

b) Provide uplink communication including command processing, storage, and execution, and with the capability of selecting either receiver at any time.

c) Process data for telemetry including write-in or read-out data from the DSU.

d) Supply signals to the scientific instruments.

e) Convert, regulate, and distribute power from the battery or solar array to the spacecraft subsystems as necessary.

f) Reorient the spacecraft using the open-loop attitude control modes and including the necessary power for the precession thruster heaters and the thruster fuel heaters, but excluding transient power required by the fuel valves.

g) Perform velocity corrections including the necessary power for the precession thruster heaters and the thruster-fuel heaters but excluding transient power required by the fuel valves.

h) Provide power to the scientific instruments as required in PV 1006.02.

The solar array shall be sized to provide the design power for all nominal spacecraft cruise attitudes as specified in Section 3.6.3.3.2. The battery may be used to supplement the solar array output when off-nominal maneuver attitudes are required.

The electrical power subsystem shall be capable of providing adequate electrical power so that the spacecraft can be processed to any nominal maneuver attitude, remain there for 5 hours, and precess back to cruise attitude without exceeding 50 percent depth of discharge on the battery. The scientific instruments may be operated at reduced power throughout the maneuver sequence in order to meet this requirement.

3.6.3.2.4 Power Bus Voltage Regulation

The electrical power subsystem shall automatically control the power bus voltage to 28 VDC ± 2 percent except during power switching transients. Regulation shall be achieved by shunting array power in excess of load requirements or by discharging the battery through a regulator when load power exceeds the array capability. Power for using bus subsystems shall be further conditioned by the electrical power subsystem to provide voltages other than 28 VDC.

3.6.3.2.5 Overload Protection

If at any time the load exceeds the power source capability then a portion of the load shall be disconnected, preferably in the following steps with suitable delays between steps:

a) All scientific instruments and/or probe loads

b) All switched loads except the S-band transmitter amplifiers

c) S-band transmitter amplifiers

The overload shall be sensed by a drop in power bus voltage below 28 VDC ±2 percent and shall include delay circuits to prevent inadvertent functioning due to momentary transients. The capability to override the overload protection by ground command shall be provided.

3.6.3.2.6 Fusing of Circuits for Scientific Instrument Power

The probe bus shall provide an individually fused branch circuit for each scientific instrument. The fuses for the scientific instruments shall be located in a module separate from probe bus subsystem fuses and shall be easily accessible for replacement. Fuse sizing shall be in accordance with PV 1006.02 and shall be subject to approval by ARC/PPO.

3.6.3.2.7 Probe Power

The probe bus shall be capable of supplying power to the large probe and small probes through fault protected circuits in order to meet the requirements of Section 3.3.6. Control of power from the probe bus to the probes shall be by ground command.

3.6.3.3 Probe Bus Attitude Control

3.6.3.3.1 Description

The attitude control subsystem consists of electronic control assemblies, sensors, thrusters fuel tanks, fuel lines, heaters (if required), and instruments for measuring the subsystem performance. In addition to performing the functions required for attitude control, this subsystem hardware shall also be used to implement the velocity control capability required in Section 3.6.3.4.

3.6.3.3.2 Control Capability

The attitude control subsystem shall be the capability for

a) Adjusting the initial spin rate after separation from a nominal 2 rpm counterclockwise (viewed from in front of the nose of the launch vehicle) to the nominal cruise spin rate of 4.8 rpm.

b) Adjusting intermittently the spin rate by ground command during the nominal flight profile to correct for the effects of thruster misalignment and other sources of disturbance torque. A capability for adjusting the spin speed during the mission by at least 20 rpm shall be provided.

c) Precessing the spin axis intermittently by ground command during the nominal interplanetary flight profile in order to maintain spin axis pointing in the desired attitudes relative to the earth and sun. A capability to precess the spin axis at least 500 degrees shall be provided to meet this requirement.

d) Changing the attitude of the spacecraft by ground command to that required to perform the number of midcourse velocity corrections necessary to achieve arrival in the vicinity of Venus at the desired time and at the desired position and velocity and to provide velocity corrections for probe targeting and to control bus entry position and time. A precessional capability of at least 500 degrees shall be provided to meet this requirement. Capability shall be provided to precess to any inertial attitude with the sun in the forward hemisphere and up to 95 degrees from the forward spin axis, except that accurate pointing is not required for attitudes with the sun within 10 degrees of the forward spin axis.

e) Adjusting the spacecraft velocity by firing one or more of the attitude control thrusters in order to meet the requirements of Section 3. 6. 3. 4.

3.6.3.3.3 Hardware Design

3.6.3.3.3.1 Thrusters

A total of at least eight attitude and velocity control thrusters shall be provided and mounted on the spacecraft structure with their thrust directions as indicated in Figure 3.6-3. Control modes shall be provided, selectable on ground command, such that any single thruster can be operated on command, or any pair of thrusters whose thrust vectors lie in the same nominal plane can be selected to operate simultaneously. Thrusters shall be located so as to achieve minimum interference or degradation of other subsystems or instruments.

3.6.3.3.3.2 Propellant Capacity

The design shall provide for sufficient propellant capacity to meet the spin rate and precession requirements of Section 3. 6. 3. 3. 2 and the velocity adjustment requirements of Section 3. 6. 3. 4 and the propellant required to meet each separate requirement shall be added rather than statistically combined with the others in order to determine the total propellant requirement. Total propellant capacity shall include allowances for propellant leakage during the nominal flight profile, for in-flight thruster calibration as required, and for contingency.

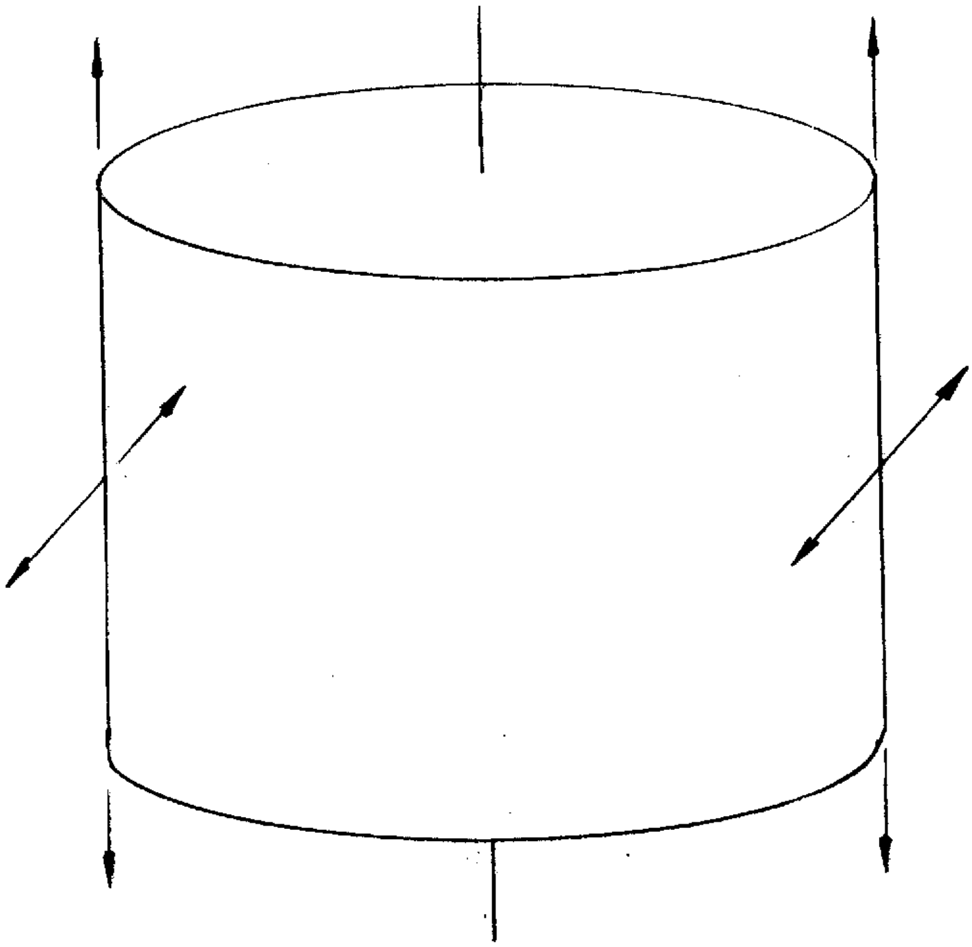

Figure 3.6-3. Attitude and Velocity Control Thruster Arrangement

3.6.3.3.3.3 Sensors

Optical sensors used to initiate thruster action shall be completely redundant and shall be capable of providing operation at sun look angles between 10 and 110 degrees from the forward spin axis. The sensors shall be designed so as to prevent triggering by other than the selected stimulus. If the design requires more than a single sensor or a single source for sensor stimulation during the mission profile, it shall be possible to select the appropriate sensor source by ground command. All photo stimulated sources shall have adequate protection from the environments specified in PC 310.02.

Commands shall be provided as necessary for the activation of any protective devices and the status of each protective device shall be indicated in the telemetry data.

3.6.3.3.3.4 Safety

Tanks, lines, fittings, valves, thrusters, and other components of the propulsion system shall meet applicable safety requirements of:

a) The General Safety Plan of the Kennedy Space Center as specified in Attachment A to KMI 1710.1.

b) The Range Safety Manual of the AFETR as specified in AFETRM 127-1.

c) Any other applicable federal, state, or local regulations.

3.6.3.3.3.5 Fill/Drain

The subsystem shall be designed to permit filling and draining the propellant and pressurant from the exterior of the spacecraft.

3.6.3.3.3.6 Valves

Thruster valves shall have redundancy for open and closure action.

3.6.3.3.3.7 Leak Test

The propulsion assembly shall be designed to provide the capability for performing suitable and adequate leak tests.

3.6.3.3.3.8 Propellant Freezing

If necessary because of propellant characteristics and constraints by other spacecraft subsystems, the design shall incorporate means to prevent the propellant from solidifying or to liquify the propellant after solidification by turning heaters on by ground command. Where the latter option is selected, it shall be possible to liquify the propellant within 120 hours after commanding the heaters on following an unlimited time in which the heaters were off.

3.6.3.3.4 Thrust Control

Operation. The attitude control subsystem shall have at least two modes of operation for each thruster or thruster pair.

Short Pulse. A single pulse of short duration for each ground command with a different command for each direction of thrust. This capability may be achieved in combination with the extended thrust capability by loading the appropriate extended-thrust register with only a short time for thrusting.

Extended Thrust. A thrust of variable duration. In this mode of operation, it shall be possible to select by ground command:

a) A time delay between receipt of the thrust enable signal and the thrust execute from 0 to at least 120 minutes in two-minute increments.

b) The duration of thrust from 0 to about 2000 seconds in increments of about 1 second.

The information necessary to effect capabilities (a) and (b) shall be contained in an on-board storage register. It shall be possible to select by ground command the register for subsequent entry of information. The register shall remain enabled only for the next series of commands that are identified as "store" commands and sufficient to fill the register, or until another register is selected. Thereafter, the register shall accept no additional data until reselected. The register shall be loaded with quantitative commands from the ground having octal numbers corresponding to the selected time delay and thrust duration. The octal numbers for the quantitative commands shall not exceed 177. The extended thrust shall be enabled by a signal from the attitude-control open-loop series mode at the completion of the first precession sequence. A ground command shall be available which, upon receipt by the spacecraft, will inhibit continuation of the extended thrust. A ground command shall be available to subsequently override the inhibit command and continue with the thrust.

Resolution. The thrust level and time unit duration shall be sized to be compatible with other spacecraft subsystems, injection errors, and the required capability of the system.

Cross Coupling. The system shall effect a minimum coupling with spin rate such that the second sequence of the open-loop series precession can be made to the accuracy specified in Section 3.6.3.3.8, and so that any change in spin rate will not adversely affect the operation of other spacecraft systems.

3.6.3.3.5 Spin Rate

The spacecraft contractor shall select the nominal spin rate (subject to approval by the ARC/PPO if outside the range of 4 to 10 rpm), so as to optimize overall performance from the standpoint of spin axis drift, frequency of attitude adjustment to maintain attitude, propellant requirements, and wobble damping requirements.

3.6.3.3.6 Spin Control

The level and duration of thrust used to spin or despin the spacecraft by ground command during the mission shall be such as to permit controlling the spin rate to at least ±0.05 rpm.

3.6.3.3.7 Spacecraft Dynamics

3.6.3.3.7.1 Wobble

The precession control system shall be designed to minimize wobble as a result of asymmetric inertia or adverse effects of synchronous conditions of wobble and thrust.

3.6.3.3.7.2 Wobble Damping

The spacecraft shall be designed to damp wobble induced from asymmetrical deployment of appendages, from probe deployment, from separation from the launch vehicle as specified in PV1006.04, from on-board changes in angular momentum caused by spacecraft subsystems or the instruments as specified in PC324.00, and from spin, despin, precession and ΔV thrust effects. The damping rate and the selected precession step size shall be adequate to meet the requirements for accuracy of the ΔV thrust direction, post-injection orientation, in-flight precession calibration, and attitude determination.

3.6.3.3.8 Attitude Control and Stability

3.6.3.3.8.1 <u>Capability</u>

The spacecraft shall have the capability which provides for control and which allows the determination of spacecraft attitude throughout the mission, with accuracies which satisfy the spacecraft system requirements for the mission and the pointing requirements of the scientific instruments as specified in Section 3.2.2.2.8.

3.6.3.3.8.2 <u>Stability</u>

Attitude stability shall be achieved through the gyroscopic effect from spinning the spacecraft about the axis with the largest moment of inertia.

3.6.3.3.8.3 <u>Attitude Control</u>

The spacecraft shall be capable of maintaining attitude control while performing the in-flight maneuvers required, including velocity change, spin rate control and precession.

3.6.3.3.8.4 <u>Damping</u>

The time constant for damping the spacecraft wobble shall be conservatively consistent with the mission operation timeline, but in no case greater than 1 hour. The threshold for damping shall be less than 0.1 degree.

3.6.3.3.8.5 <u>Drift</u>

The allowable drift of the spacecraft spin axis resulting from solar radiation pressure shall be such that attitude correction maneuvers, as required to maintain normal spacecraft operations, are required no more frequently than once per week.

3.6.3.3.8.6 <u>Spin Rate</u>

The spacecraft spin rate shall be controllable by ground command. The spin rate during the bus entry phase shall be approximately 60 rpm. Spin rate control accuracy shall be within 1 percent of the desired value. Telemetry shall contain information to permit determining the spin rate to an accuracy of at least 0.1 percent.

3.6.3.3.8.7 Roll Position Reference

The spacecraft shall provide a roll position reference signal, with each spacecraft revolution, for use by the scientific instruments. This signal shall be such as to allow each instrument to control its internal operations and reference its data to an inertial pointing position to an accuracy of 1 degree or better in the spacecraft spin plane. Telemetry shall contain information to permit relating the time of the roll position reference signal to any bit with the bit stream.

3.6.3.3.8.8 Attitude Data Acquisition Time

The time for the spacecraft to obtain and telemeter the measurements necessary to determine the spacecraft attitude to the accuracy limits specified herein for the scientific instruments and as required for spacecraft operations shall not exceed 1 hour.

3.6.3.3.8.9 Positionable Platforms and Assemblies

The orientation of any positionable spacecraft platforms and assemblies, with respect to the spacecraft reference axes, shall be continuously determinable whenever such platforms and assemblies are in active operation.

3.6.3.3.9 Propulsion

The spacecraft shall have the capability for thrust and torque impulses to perform precession of spin axis, velocity changes, and spin control as required for mission accomplishment. Sufficient propellant shall be provided to accommodate the maximum system leakage and to perform the required mission maneuvers with reserves for 99.5 percent probability, considering launch vehicle performance and orbit determination errors specified in PV 1006.04 plus spacecraft execution errors. Propellant leakage shall not exceed 0.1 pound (0.0045 km) per year.

3.6.3.4 Velocity Control

3.6.3.4.1 Description

The hardware used to implement the spacecraft velocity control functions is included as part of the attitude control subsystem as specified in Section 3.6.3.3.

3.6.3.4.2 Capability

The attitude control subsystem shall have the capability for adjusting spacecraft velocity at any time from five days after launch until termination of the nominal mission (bus entry into the Venus atmosphere). Capability shall be provided to make velocity adjustments at least once every 2 days in order to correct for the injection errors induced by the launch vehicle as specified in PV 1006.04 to correct for velocity errors introduced by the spacecraft systems themselves at the time of velocity adjustments, and to achieve probe deployment with velocity, position, and timing accuracies consistent with probe entry conditions as required in Sections 3.6.4 and 3.6.5.

No spacecraft capability shall be required to correct for errors in tracking data at the time of velocity correction beyond that inherently provided by the attitude control and thruster firing capabilities required in Section 3.6.3.3 and the total velocity change capabilities required in Section 3.6.3.4.3.

3.6.3.4.3 Propellant Capacity

The design shall provide for sufficient propellant capacity to permit the spacecraft velocity to be adjusted by at least 10 meters/sec prior to release of any probes, to be adjusted by an equivalent of 30 meters/sec after large probe release, and to adjust the bus velocity by 30 meters/sec after all probes have been released.

3.6.3.5 Probe Bus Communication Subsystem

3.6.3.5.1 Description

The communication subsystem consists of receivers, transmitter drivers, power amplifiers, switches, filters, diplexers, antennas, associated coaxial lines, hybrid couplers, and instrumentation to measure performance.

3.6.3.5.2 Compatibility

The communication subsystem shall be compatible with other spacecraft subsystems, and the DSN as specified in PV 1006.03.

3.6.3.5.3 Capability

The communications subsystem shall have the capability to:

a) Radiate a noncoherent RF signal to enable radio acquisition of the spacecraft by the DSIF. After acquisition of an uplink signal, establish and maintain phase coherent transmission with the signal received from the DSIF for purposes of measuring two-way doppler shift. The transfer from noncoherent to coherent operation, and vice versa, shall occur automatically and shall be controlled by signal present detection circuitry within the spacecraft receivers. The subsystem shall also have the capability of transmitting in the noncoherent mode, while receiving an uplink signal, when so directed by ground command.

b) Receive commands transmitted from the DSIF

c) Telemeter to the DSIF scientific and engineering data as generated by the data subsystem

d) Transmit and receive continuously from before liftoff through the nominal mission profile.

3.6.3.5.4 Subsystem Design

3.6.3.5.4.1 Redundancy

The communication subsystem shall be designed with maximum redundancy within the overall constraints of the spacecraft and mission profile. As a minimum, the subsystem shall have two transmitter drivers, two power amplifiers, and two receivers.

3.6.3.5.4.2 Switching

The communication subsystem shall be designed to permit selecting by ground command either of the transmitter systems and each transmitter-antenna configuration that is required to meet the downlink performance requirements. The receiver shall be selectable by frequency address. The subsystem shall be designed to provide the capability of receiving with the spacecraft in any attitude even in the event of a failure of either receiver. If this requirement necessitates automatic switching of the omni system alternative from one receiver to the other, the switching shall occur no less or more frequently than once every 30 to 50 hours and only if neither receiver has had a signal present indication during the period. The period shall restart whenever the signal-present indicator in both receivers disappears. It shall be possible to inhibit and restore the automatic switching capability by ground command.

3.6.3.5.4.3 Switching Performance

The switching hardware shall contain sufficient filtering so that the power of spurious frequencies, including carrier harmonics, shall be more than 40 db below the level of the unmodulated carrier as measured at the input to any antenna of the system.

3.6.3.5.5 Receiver Design

3.6.3.5.5.1 Type

The receivers shall be of the phase-lock-loop type and shall be designed to minimize uplink acquisition time.

3.6.3.5.5.2 Frequency

The receivers shall be designed to operate at any frequency between 2110 and 2120 MHz by means of crystal selection. The frequency of each of the redundant receivers shall be different. Within the limits of stability and temperature effects, the "rest" or acquisition frequency of each receiver shall be the same at all times. See specification PV 1006.03 for specific frequency allocation.

3.6.3.5.5.3 Warm Up

The receivers shall be designed to minimize the time to reach the final operating frequency after power is applied to the receiver consistent with other requirements. As a goal, the time should not exceed 15 minutes.

3.6.3.5.5.4 Frequency Stability

With no signal at the receiver input terminals and during any 10-hour period, the best lock frequency shall not deviate more than 10 kHz for a temperature change from 30 to $90^{\circ}F$.

3.6.3.5.5.5 Demodulation

The receivers shall be designed to demodulate a FSK/PM signal and supply the FSK tones to the command decoder input. The frequency of the FSK tones shall be 204.8 and 128.0 Hz. The modulation index of the received RF signal is specified in PV 1006.03.

3.6.3.5.5.6 Bandwidth

The receiver bandwidth and phase-lock-loop characteristics shall be compatible with the command-information transmission requirements, as specified in PV1006.03, the doppler shift, and the rate of doppler shift resulting from the mission trajectories after launch, and until the probe bus reaches an altitude of 110 km above the mean surface of Venus (surface assumed at R = 6050 km).

3.6.3.5.5.7 Automatic Acquisition

Automatic acquisition techniques that change the receiver VCO rest frequency when the receiver is not in lock shall not be incorporated in the design.

3.6.3.5.5.8 Control of Transmitter Frequency

Each command receiver shall be capable of controlling the output frequency of either transmitter driver at any time a receiver is locked to an uplink signal (coherent mode). This control frequency shall be such that the transmitter driver S-band output frequency is precisely 240/221 times the received uplink frequency. The change to receiver control of the transmitted frequency shall be automatic and shall be controlled by the presence, or lack of, a received signal. It shall be possible to inhibit and restore the coherent mode of operation by ground command.

3.6.3.5.6 Transmitter Design (Transmitter Driver plus Power Amplifier)

3.6.3.5.6.1 Frequency

Operation. The transmitter shall be designed to operate at any frequency between 2290 and 2300 MHz. See specification PC 322.00 for specific frequency allocation.

Control. The transmitter frequency shall be controlled by a crystal-controlled oscillator during periods when neither receiver is being addressed or when the "frequency control by receiver" operating mode has been inhibited by ground command. When the "frequency control by receiver" mode is operating (coherent operation), the output of the crystal-controlled oscillator shall be disabled.

Short-Term Frequency Stability. When the temperature and supply voltage for the spacecraft transmitter are steady and during noncoherent operation, the peak-to-peak frequency deviation within the 10 successive one-minute time periods shall be less than 0.5 Hz for a 1/4-second integration time.

Long-Term Frequency Stability. During noncoherent operation and any 10-hour period, the transmitter frequency shall not vary more than 10 kHz for a temperature change from 30 to 90°F.

Phase Stability. When operating with a phase coherent test receiver and a test transmitter having the characteristics specified below, the spacecraft transponder and power amplifier shall meet the following requirements:

a) In the coherent mode under strong uplink signal conditions, the phase stability of the unmodulated carrier measured at the output of the transmitter driver shall cause no more than 2.8 degrees rms or 8.4 degrees peak phase error.

b) In the noncoherent mode, the phase stability of the unmodulated carrier measured at the output of the transmitter driver shall cause no more than 3.6 degrees rms or 10.8 degrees peak phase error.

c) The power amplifier shall add no more than 3 degrees rms phase error to the above values.

The test receiver shall have a strong signal double side bandwidth ($2B_L$) of 12 Hz or less and a high frequency cutoff of 1 kHz or more. The test transmitter and receiver shall have a self-test mode to establish their contribution to the measured phase error; the contribution shall be less than 10 degrees rms. The phase error attributed to the spacecraft shall be calculated by the root-sum-square method.

3.6.3.5.6.2 Power Amplifier

As a minimum, the power output from each amplifier shall be the same, single, nominal value and sufficient to meet the performance requirements specified herein.

3.6.3.5.6.3 Modulation Index

The modulation index shall be adjustable between 0.5 and 1.2 radians by minor changes of components in the transmitter driver. The flight value selected for the modulation index shall be within this range, shall be adjusted prior to acceptance testing of the transmitter driver, and shall remain within ±0.10 radians of the selected value. The value of the modulation index selected is subject to approval by the ARC/PPO.

3.6.3.5.7 Probe Bus Antenna

3.6.3.5.7.1 Performance

The antenna subsystem shall be designed to meet the communication performance requirements specified below (gain, directionality, etc.) and operate at the frequencies specified in Sections 3.6.3.5.5.2 and 3.6.3.5.6.1.

3.6.3.5.7.2 Polarization

The polarization of the omni antennas and the bus medium gain antenna for both transmit and receive signals shall be circular (RHCP).

3.6.3.5.7.3 Beamwidth

The beamwidth of the medium gain antenna shall be 24 degrees as measured at the - 3 db points.

3.6.3.5.7.4 Omni Coverage

The omni antenna coverage with interconnection to the two available receivers shall have a compatible pattern with effective gains as follows:

Minimum Effective Gain (dbi)	Earth Aspect Angles (deg)
-2	0→180
-1	0→160

3.6.3.5.8 Performance – Uplink

The uplink system shall be capable of receiving commands for the conditions specified below and for the modulation index as specified in PV 1006.03 with a bit error probability no greater than 10^{-5} as measured at the output of the spacecraft command detection equipment.

3.6.3.5.8.1 Powered Flight

There is no requirement for uplink communication capability during powered flight.

3.6.3.5.8.2 Post-Launch

The spacecraft shall be capable of receiving a signal having a total power density at the spacecraft of -45 dbm per square meter when in any attitude relative to the spacecraft/earth line during the post-launch portion of the mission after separation from the launch vehicle.

3.6.3.5.8.3 Flight

During the nominal flight mission, the probe bus shall be capable of receiving a signal having the total power density given below for the directions indicated:

Power Density (dbm per square meter)	Earth Aspect (deg)
-108	0 to 180 (omni)
TBD	180 ± 13 (medium gain)

3.6.3.5.9 Performance – Downlink

The frame (word) deletion rate shall be $\leq 10^{-2}$.

3.6.3.5.9.1 Powered Flight

TBD

3.6.3.5.9.1 Flight

During the nominal flight mission after separation from the launch vehicle, the spacecraft shall be capable of transmitting not less than the following total EIRP in the directions indicated:

Total EIRP (dbm)	Earth Aspect Angle (deg)
31	0 to 180 (first 80 days)
47	180 ± 13 (first 80 days)
34	0 to 180 (> 80 days)
50	180 ± 13 (>80 days)

3.6.3.5.10 Telemetry Data

Telemetry data from the spacecraft shall contain information about:

a) The status of the communication subsystem such as switch position and signal present in either receiver

b) Performance of the communication subsystem such as temperatures of critical components, receiver signal strength and static phase error, and power amplifier voltages, currents, and output power.

3.6.3.6 Probe Bus Data Subsystem

3.6.3.6.1 Description

The data subsystem consists of a digital telemetry unit (DTU).

3.6.3.6.2 Capability

The data subsystem shall provide the capability for:

a) Supplying, as required, timing and operational signals to the scientific and engineering instruments on the spacecraft.

b) Sampling these instruments

c) Conditioning and quantizing the data and arranging them into formats suitable for further processing

d) Coding the data and processing them into a form suitable for modulating the down link RF carrier

3.6.3.6.3 DTU Design

3.6.3.6.3.1 Redundancy

The data subsystem shall have redundancy of critical circuits and functions.

3.6.3.6.3.2 Operation

The DTU shall be designed to operate in the modes and bit rates and to format the data in the manner specified hereinafter. At turn on, the DTU shall operate in the real time mode at 512 bits/sec, in Format C (see Section 3.6.3.6.6), and in the coded mode.

3.6.3.6.3.3 Bit Rate

The DTU shall be designed to operate at data bit rates of 2048, 1024, 512, 256, 128, 64, 32, and 16 bits/sec. The bit rate shall be selectable by ground command and shall be independent of format and mode.

3.6.3.6.3.4 Signals and Timing Frequencies

The DTU shall supply the signals to the scientific instruments as specified in spacecraft/scientific instruments interface document, PC324.00. The DTU shall generate timing signals for use by the spacecraft subsystems and scientific instruments. The clock source for these timing signals shall have a frequency of 32.768 kHz or greater.

3.6.3.6.3.5 Word Gates

The DTU shall be designed to have a capability of supplying up to 20 discrete word gates, each of some multiple of 3 bits long, to the scientific instruments for use when sampling measurements for the science formats, Formats A and B (see Section 3.6.3.6.6). The shape and time of occurrence of the leading and trailing edge of each word gate signal shall be such that the signal amplitude during the period of an "extended word gate" formed by combining one or more adjoining discrete word gates shall be within the tolerances for a discrete word gate. The DTU shall also be designed to have a capability of supplying 20 discrete word gates, each 6 bits long, to the scientific instruments for use when sampling measurements for the subcommutated science formats. It shall be possible to supply the signal for each word gate specified above to two scientific instruments simultaneously; the second instrument will use the gate as a timing signal only and there will be no readout of telemetry data from the second instrument during the word gate.

3.6.3.6.3.6 Input Signals

The DTU shall be capable of processing analog, digital, and steady-state signals from scientific instruments and spacecraft subsystems.

3.6.3.6.3.7 Analog Data

The DTU shall provide for the analog-to-digital conversion of analog data (0 to 3 volts range) to six bits resolution.

3.6.3.6.3.8 Data Conditioning

Data conditioning shall be provided for the spacecraft and scientific instrument data. Data conditioning shall include encoding, conversion, commutation, multiplexing, and synchronizing. As a goal, fixed word and data word bits that are used as status indicators shall be of a positive logic so that a "logical one" signifies a condition or function that is on or operating normally and a "logical zero" signifies a condition that is off or not normal. Also as a goal, engineering measurements shall be conditioned for telemetering such that the digital value of the telemetry word will increase for:

a) An increase in the absolute value of the measurement for those not passing through zero

b) An increase in the algebraic value of the measurement for those passing through zero

3.6.3.6.3.9 Output

The output of the DTU shall be two signals with a capability for input to either transmitter driver. The format of the signal shall be non-return-to-zero level (NRZ-L). The data output of the DTU shall be a biphase modulated square wave with a frequency of 32.768 kHz ±0.02 percent. The variation in the average time period of these square waves shall be less than 0.5 percent (short term phase stability).

3.6.3.6.3.10 Signal Buffering

Output signal and data lines from the DTU shall be adequately buffered within the DTU such that an open circuit or short circuit failure of any given output line external to the DTU shall not result in the loss of other signal or data lines.

3.6.3.6.3.11 Format Rearrangement

The DTU shall be designed to provide the capability of rearranging the word allocation in the scientific format (Section 3.6.3.6.6.3) by patchboard or other suitable means so as not to require redesigning of, changing permanent wiring in, adding or removing modules from, or removing any components from the structure of the DTU. This provision will be used in the event of a change in a scientific instrument.

3.6.3.6.4 Convolutional Encoder Design

3.6.3.6.4.1 Operation

The data subsystem shall provide for the convolutional coding of the data bit stream transmitted by the spacecraft. The spacecraft shall be capable of operating in either the coded mode or the uncoded mode. Mode selection shall be controllable by ground command. The data subsystem shall automatically operate in the coded mode upon application of electrical power to the spacecraft.

3.6.3.6.4.2 Encoder Parity Bits

When the data subsystem is operating in the coded mode, the encoder shall replace each data bit generated by the data system with two parity bits, P and Q. The value of each parity bit shall be based on the values of 32 selected data bits previously generated. The selected data bits for P are 1, 3, 5, 6, 8, 9, 10, 12, 14, 15, 17, 18, 19, 20, 22, 23, 24, 25, 26, 28, 29, 30, and 32, and the selected data bits for Q are 1, 2, 3, 5, 6, 8, 9, 10, 12, 14, 15, 17, 18, 19, 20, 22, 23, 24, 25, 26, 28, 29, 30, and 32, in which bit 1 refers to the most recently generated data bit. Each parity bit shall be a logical "one" if there are an odd number of "ones" in the selected data bits and a logical "zero" otherwise. • The encoding cycle begins at the end of the last bit of each frame synchronization word at which time each stage of the shift register containing the value of the previously transmitted 32 data bits and the 33rd flip-flop used to generate the code are reset to a logical "zero." For the main frames containing formats D1 through D8, the coder shall reset every 384 bits, for all other formats the reset shall occur every 192 bits.

3.6.3.6.5 Modes of Operation

The data subsystem shall be capable of operation in the modes specified below, each of which can be selected by ground command. A change in the mode of operation, by ground command or by automatic means on board the spacecraft, shall be synchronous with the start of the next main frame. Mode identification shall be included in every telemetered main frame except those containing data in Format D. The bit arrangement for the mode identification word shall be as follows:

	First Bit		Last Bit	
Real time	0	0	X	
Telemetry store	1	0	X	Not applicable or required on probe bus
Memory readout	0	1	X	
Redundant DTU-A on	X	X	0	
Redundant DTU-B on	X	X	1	

where X denotes a binary one or zero.

3.6.3.6.5.1 Real Time Mode

In this mode scientific and engineering data shall be transmitted without intermediate storage, at the bit rate and format selected by ground command.

3.6.3.6.5.2 Telemetry Storage Mode

(This mode is not used on the probe bus mission.) In this mode, data shall be stored at any of the bit rates selected by ground command and may be in excess of the real time ground station telemetry receiving capability. Consecutive frames of data shall be simultaneously transmitted and read into storage on a 100 percent duty cycle basis. When the telemetry store mode is terminated by ground command, or when the memory is filled, the data system shall automatically switch to the real time mode in the format and bit rate used during the telemetry store mode.

3.6.3.6.5.3 Memory Readout

(This mode is not used on the probe bus mission) In this mode, scientific and engineering data shall be read out from the DSU at a bit rate selected by ground command. When memory readout is completed, the data system shall automatically switch to the real time mode and the format used before memory readout, and remain in the bit rate used during memory readout.

3.6.3.6.6 Data Formats

3.6.3.6.6.1 General

The data subsystem shall have the capability of telemetering various arrangements or "formats" of scientific and engineering data. The format being used shall be selectable by ground command and shall be independent of bit rate and mode of operation. Format changes shall take place immediately upon receipt by the spacecraft of the appropriate command. Frame and word synchronization shall be maintained during format changes. Allocation and arrangement of scientific and engineering data in each format shall be subject to approval of the ARC/PPO. It shall be possible to transmit the formats at one or more of the following rates:

a) "Main frame" rate in which each bit of the complete format is transmitted sequentially and at the bit rate being used.

b) "Subcommutated" rate in which only one word at a time is transmitted in a main frame.

3.6.3.6.6.2 Common Characteristics

The length of each data format shall be 192 bits. In addition, certain characteristics are common to all data formats described herein when transmitted as a main frame (standard format) except the special format, identified as Format D. These characteristics are as follows:

a) Bits 1 through 6 of each format shall contain a 6-bit word identifying the operation mode and bit rate being used.

b) Bits 7 through 24 of each format shall contain an 18-bit word of fixed sequence for frame identification. The sequence shall be

   111100110101000000

   in which the "1" is the first bit transmitted.

c) Bits 97 through 101 shall contain a 5-bit word identifying the format being used.

d) Bits 102 through 108 shall contain a 7-bit word identifying the frame count and the engineering and science subcommutated words.

e) Bits 109 through 114 shall contain a 6-bit word subcommutated from one of the engineering formats, identified as Format C1 through C4. The subcommutated word will cycle sequentially through all four Format C's at all times.

f) Bits 115 through 120 shall contain a 6-bit word subcommutated from the science format identified as Format E.

3.6.3.6.6.3 <u>Science Formats (Formats A and B)</u>

Two science formats shall be available. The selected format shall be telemetered as a main frame only. The word length is 3 bits. Forty-eight such words shall be available for scientific measurements. The input signal for each of these words shall be digital only.

3.6.3.6.6.4 <u>Special Format (Format D)</u>

In this format, all 192 bits shall be allocated to a single scientific instrument or other data source and the selected instrument will supply data on a single input channel. The particular source supplying data to this format shall be selected by ground command and provision shall be made for selecting up to eight different sources. One of the eight formats provided shall be arranged to accept data from two sources with the first 24 bits allocated to one source and the remaining 168 bits to the second. The format shall be telemetered only at the main frame rate, only in combination with the selected science format (A or B), and shall alternate with the science format. The input shall be digital data only. Identification of the particular source supplying data to this format shall appear in the 5-bit format identification word in the science format.

3.6.3.6.6.5 <u>Engineering Format (Format C)</u>

Four engineering formats shall be available. The length of the words in these formats shall be either 1 bit in groups of six bits for accepting as input signals bilevel status data; or 6 bits for accepting as input signals analog or digital data from the spacecraft instrumentation. The most significant bit shall be transmitted first. Where possible, the allocation of words in each format shall be such that they contain engineering measurements associated with the following subsystems:

a) Format C1: data subsystem

b) Format C2: spacecraft electrical power subsystem

c) Format C3: spacecraft communication subsystem

d) Format C4: spacecraft orientation and propulsion subsystems

The four formats shall be telemetered sequentially at all times in the subcommutated engineering word of the main frame. In addition, it shall be possible to telemeter as a main frame any of the four formats alone or all four in combination. When so transmitted, data bits 1 through 24 and 97 through 120 in each format shall be inhibited so as not to interfere with the fixed words of main frame data. Because of this requirement, these inhibited bits should be allocated to those types of data that change infrequently.

3.6.3.6.6.6 Subcommutated Science Formats (Format E)

Two subcommutated science formats shall be available. The length of the words in these formats shall be either 1 bit in groups of six bits for accepting as input signals bilevel status bits; or 6 bits for accepting as input signals analog or digital data from the scientific instruments. The two formats shall be telemetered sequentially at all times and only in the subcommutated science word of the main frame.

3.6.3.6.7 Frame Identification

It shall be possible to identify unambiguously the frame count up to 8192. The seven least significant bits of the frame counter appear in bits 102 through 108 of the main frame. The six most significant bits of the frame counter (extended frame counter) shall appear in one word of the engineering format. The frame counter shall increase by unity for each transmission of a main frame containing the common (fixed) words.

3.6.3.6.8 Telemetry Data

In addition to containing data as specified, the telemetry data shall contain information about:

a) The status of the data subsystem such as which unit of a redundant subassembly is being used.

b) The performance of the data subsystem such as temperature of the units and A-D converter calibration.

3.6.3.7 Command Subsystem

3.6.3.7.1 Description

The command subsystem consists of two decoders (DDU) and a command distribution unit (CDU).

3.6.3.7.2 Capability

The command subsystem shall provide the capability for controlling the operating modes of the spacecraft equipment, scientific instruments, and probes from information received by means of RF transmission to the spacecraft and from signals generated on board at discrete events. The capability shall exist during ground tests, on-stand activities before launch, during the nominal flight profile, and as long as possible during the post-encounter mission profile consistent with the performance of other spacecraft subsystems.

3.6.3.7.3 Subsystem Design

3.6.3.7.3.1 Spurious Commands

The command subsystem shall be designed so that voltage fluctuations, noise from the receivers, voltage transients, or normal signals generated by spacecraft equipment and scientific instruments will not result in the execution of any uncommanded change in operating mode of the equipment and instruments.

3.6.3.7.3.2 Commands Using RF Information

The command subsystem shall be designed to accomplish the following functions upon receipt of information in the RF transmission to the spacecraft:

a) Turning power on and off individually to each spacecraft assembly as required, to four probes, and to at least five scientific instruments. Turning power on to the probes and instruments shall be by command only and shall not occur as a result of any other changes in spacecraft operating modes or electrical power transients.

b) Turning power off to all scientific instruments simultaneously.

c) Switching between redundant equipment with the exception of the spacecraft receivers and decoders.

d) Switching between the various operational modes of the communications and data subsystems.

e) Switching between the operational modes of the scientific instruments. At least 50 discrete commands shall be provided to accomplish this function.

C - 2

f) Controlling the operation of the attitude control subsystem.

g) Loading qualitative and quantitative commands into on-board storage units for later execution.

3.6.3.7.3.3 Commands Using On-Board Events

The command subsystem shall be designed to provide signals to the spacecraft equipment to perform the functions at the event time as follows:

a) Initiation of despin between spacecraft/launch vehicle separation and shortly after emergence of spacecraft from earth shadow.

b) Deployment of appendages and the like upon reduction of spin rate to an acceptable level.

c) Turnoff of appropriate subsystems when load reduction is required by the electrical power subsystem.

d) Operational changes as required and at times as directed by information in command storage.

Back-up ground commands shall be available for items (a) and (b). Items (a) and (b) are not required for the probe bus but may be implemented as capabilities to maintain commonality with the orbiter.

3.6.3.7.3.4 Stored Commands

The command subsystem shall have the capability of being programmed by ground command to store at least six spacecraft commands for execution at a later time and to store the time delays between sequence enable and sequence execution and between each command of the sequence. It shall be possible to provide for each of the delay periods, a time of at least 129,600 seconds to a resolution of at least 2 seconds.

It shall be possible to select by ground command in any order any of the register triplets used to store the commands (discrete or serial) and the corresponding lower order and higher order delay times. Each register tripplet shall remain enabled only for the next three commands received by the spacecraft that are identified as store commands, or until another register is selected. Thereafter, the register shall accept no further data until reselected.

A single ground command shall be available to set all registers to zero. Also all registers shall be set to zero when power is applied, including after momentary power outages, to stored command electronics.

3.6.3.7.3.5 Command Number

Where possible, the octal number of 200 or greater shall be used in identifying all critical commands such as:

a) Turnoff of power amplifier

b) Initiation of stored command sequences in the orientation, propulsion, or command subsystems

c) Initiation of on-board ordnance devices

d) Changes to operating modes from which it is impossible to return

The octal number 000 shall not be used for any discrete command.

3.6.3.7.3.6 Command Message

The command message and command equipment shall be designed such that the probability execution of an erroneous command message (with a bit error rate of 10^{-5}) shall be no greater than 1.1×10^{-9}. The message shall also contain the capability for selecting either of the two decoders and for identifying whether the command is to be implemented immediately upon receipt by the spacecraft (real time command) or is to be stored (store command).

3.6.3.7.3.7 Command Bit Rate

The command bit rate will be 1 bit/second as generated by the DSN.

3.6.3.7.4 DDU Design

3.6.3.7.4.1 Redundancy

Two DDU's shall be provided with the capability of selective operation by ground command.

3.6.3.7.4.2 Operation

The DDU's shall be either automatically switched to the receiver that is in lock, or each DDU shall be connected to both receivers at all times. The DDU's shall accept the digital tone outputs from either command receiver and shall accept only those command words satisfying the command word format.

3.6.3.7.4.3 Failures

The DDU's shall be connected to the receivers in a fail-safe mode. Failure of the squelch circuit of one receiver, so that noise from that receiver is present at the input to both DDU's, shall not prevent either DDU from processing command signals received from the other receiver.

3.6.3.7.5 CDU Design

3.6.3.7.5.1 Operation

The CDU shall be designed to accept signals from the DDU, process them, and supply pulse, level, or serial signals to spacecraft equipment, probes, and scientific instruments.

3.6.3.7.5.2 Capacity

The CDU shall be designed to accommodate the number of commands required by the spacecraft equipment, four probes and five scientific instruments.

3.6.3.7.6 Telemetry Data

3.6.3.7.6.1 Status and Performance

Telemetry data from the spacecraft shall contain information about:

a) The subsystem status such as an indication of a signal in either decoder and ordnance status

b) The temperature of selected assemblies.

3.6.3.7.6.2 Stored Command Data

The contents of the command memory shall be contained in one of the four engineering formats. As a goal, the complete contents of the command memory shall be contained in the format. As a minimum, the format shall contain the contents of a single command memory register tripplet (delay time and stored command) with the contents of each register being telemetered in sequence. In this latter case, the identification of the register shall also be in the format. The value of the command in the register shall immediately follow that of the corresponding time delay in the format.

3.6.3.7.6.3 Command Verification

Verification of changes in spacecraft operating modes or status by ground command shall be required where such verification is not implicit in the telemetry data.

3.6.3.8 Thermal Control Subsystem

3.6.3.8.1 Description

The probe bus thermal control subsystem consists of insulation to provide passive thermal control for spacecraft components and louvers to provide active thermal control of the equipment compartment.

3.6.3.8.2 Capability

The thermal control subsystem shall have the capability of maintaining spacecraft components at the temperatures required for normal operation, the scientific instruments at temperatures as specified herein for the thermal environments as specified, and meeting the probe thermal interface requirements.

3.6.3.8.3 Design Environments

The thermal control subsystem shall be designed to maintain the spacecraft and instrument temperatures at the required levels with negligible degradation of materials and components when exposed to the following environments:

a) Prelaunch: Within the environment inside the fairing achieved by the on-stand air conditioning as specified in PV 1006.04.

b) Powered Flight:

   1) Within the environment inside the fairing resulting from heating of the fairing as specified in PV 1006.04

   2) Within the environment encountered after fairing jettison for the powered flight profile specified in PV 1006.04, including the effects of aerodynamic heating, eclipse, or radiation from the sun and earth.

c) Flight: Within the thermal environment imposed during the nominal mission profile, including direct radiation from the sun on the sides of the spinning spacecraft for an unlimited time and on the aft side of the spacecraft for at least <u>four</u> hours.

3.6.3.8.4 <u>Design</u>

3.6.3.8.4.1 <u>Equipment Compartment</u>

The probe bus shall be designed so that throughout the mission until the end of the nominal flight mission:

a) Temperatures within the vicinity of spacecraft equipment will be maintained as necessary

b) Temperatures within the equipment compartment in the vicinity of each scientific instrument specified in PV 1006.02 will be maintained between 0 and 90 F when the instrument is powered and its net heat loss or gain is within the limits specified.

3.6.3.8.4.2 <u>Instrument Appendages</u>

Appendages or external structure that support scientific instruments shall be designed to minimize the heat flow into or away from the instrument.

3.6.3.8.4.3 <u>Instrument Apertures</u>

Insulation shall be provided to fill any gaps between scientific instrument apertures and the spacecraft exterior insulation.

3.6.3.8.4.4 <u>External Instruments</u>

Scientific instruments mounted external to the equipment compartment will provide their own thermal control.

3.6.3.8.4.5 <u>Probes</u>

The probe bus shall be designed so that the requirements of Section 3.3.5 are met.

3.6.3.8.5 <u>Powered Flight Constraints</u>

The spacecraft shall impose no requirements on the orientation of the launch vehicle/spacecraft combination during powered flight up to and including separation to satisfy thermal limitations of spacecraft materials and components.

3.6.3.8.6 <u>Telemetry Data</u>

Telemetry data from the spacecraft shall include, in addition to the temperatures of other subsystem units as specified elsewhere, the temperature at several locations in the equipment compartment and if possible, the angle or full open/full close status of at least one of the louvers.

3.6.3.9 Allocated Weight and Reliability Requirements

The allocated weight and reliability requirements as specified in Sections 3.4.9.2.1 and 3.4.1.2 shall be observed.

3.6.4 Large Probe Performance Requirements

3.6.4.1 Stability

3.6.4.1.1 Hypersonic Stability Margin

The static dynamic and stability margins and the aerodynamic damping during entry shall be adequate to ensure that oscillations resulting from the large probe entering with attitudes and rates within the limits specified in Section 3.6.2.10 shall be damped to less than 1 degree half amplitude by the time the probe deceleration reaches 10 g's. In addition, any subsequent excursions in angle of attack shall be damped to less than 10 degrees at time of parachute deployment.

3.6.4.1.2 Parachute Descent Phase Stability

The static and dynamic stability of the combined parachute, descent and aeroshell shall be adequate to damp any oscillations resulting from parachute deployment to less than 1 degree half amplitude within 4 seconds after parachute deployment in the absence of winds.

Prior to and after release of the aeroshell, the combined parachute and probe configuration shall have adequate static and dynamic stability margins so that the maximum departure of the descent capsule attitude from vertical shall not exceed 20 degrees in response to the design wind profile specified in Section 3.4.10.7. Angle of attack oscillations resulting from this transient shall not exceed 5 degrees peak half amplitude and shall be damped to 1 degree or less (half amplitude) within 5 seconds after the wind returns to steady state velocity.

3.6.4.1.3 Terminal Descent Phase Stability

The static and dynamic stability margins of the descent capsule after parachute release shall be adequate to limit the maximum departure of the descent capsule attitude from vertical to less than 15 degrees in response to the design wind profile specified in Section 3.4.10.7. Angle of attack oscillations from this transient shall not exceed 5 degrees peak half amplitude and shall be damped to 1 degree or less (half amplitude) within 5 seconds after the wind returns to steady state velocity.

The maximum steady state trim angle of attack shall be less than 1 degree in the absence of winds. As a goal, in the absence of winds the descent capsule shall not exhibit any tendency to limit cycle or oscillate about the trim angle of attack. If such motions do occur they shall not exceed 2 degrees half amplitude.

3.6.4.1.4 Spin Rate During Entry and Descent

The large probe shall be designed so that for attitudes and rates at entry within the limits specified in Section 3.6.2.10 the probe spin rate shall not exceed rpm at any time prior to parachute deployment. The descent capsule shall be provided with positive aerodynamic means to control spin rate to meet the requirements of Section 3.5.2.7.

3.6.4.2 Mission Termination

The large probe shall not be required to survive or operate beyond the time it reaches the surface of Venus or a radius of 6050 km, whichever occurs first. If the probe does survive after reaching the surface, scientific measurements will be made until end of life.

3.6.4.3 Aeroshell

3.6.4.3.1 Description

The large probe aeroshell subsystem shall consist of structural elements which form the aerodynamic shape during entry, resist the aerodynamic loads during entry, and enclose and support all other large probe subsystems (except the heat shield) and scientific instruments throughout the mission until parachute deployment.

3.6.4.3.2 Capability

The aeroshell subsystem shall have the capability of maintaining the designed aerodynamic shape within mechanical limits which assure that hypersonic stability requirements will be met. It shall have the capability of providing mechanical support to the heat shield subsystem on the exterior of the aeroshell and the other probe equipment and instruments on the interior. The aeroshell subsystem shall include provisions for venting its internal volume during test, launch, and entry to avoid excessive pressure differentials across the aeroshell and shall provide a separation joint and aperture in the afterbody to permit separation and exit of the descent capsule from the aeroshell. The aeroshell subsystem shall also provide a smaller aperture which is compatible with the mortar used for parachute deployment, and it shall provide an aperture for the communications antenna mounted on the descent capsule.

3.6.4.3.3 Hypersonic Ballistic Coefficient

The aeroshell shall provide an aerodynamic shape with a hypersonic ballistic coefficient of TBD + TBD slugs/ft^2 with the large probe mass at the maximum value allocated in Section 3.4.9.2.1.

3.6.4.3.4 Aeroshell Afterbody Separation

The maximum force required to remove the aeroshell afterbody shall not exceed 80 lbs exclusive of the force required for separation of the heat shield joint as specified in Section 3.6.4.4.5 after release mechanisms have been actuated.

3.6.4.3.5 Rigidity

The aeroshell shall have adequate structural rigidity to maintain the aerodynamic shape within limits which will assure mission success and to maintain overall probe structural frequencies above the frequency range of probe angle of attack oscillation in order to minimize possibilities of dynamic coupling.

3.6.4.3.6 Thermal Environment

The aeroshell shall be compatible with temperatures between -40 and 150°F prior to entry. The aeroshell shall be capable of meeting performance requirements during entry if the bond line temperature between the heat shield and the aeroshell is maintained below 300°F until after entry stagnation pressures have decreased below 0.1 bar, and the aeroshell shall continue to maintain structural integrity and alignment sufficient for separation of the afterbody and the descent capsule when the heat shield bondline temperature is at 600°F and stagnation pressures remain below 0.1 bar.

3.6.4.3.7 Provisions for Ground Handling

The aeroshell structure shall be compatible with being supported by a specially designed sling with the forebody facing either upward or downward and with being supported in a suitably padded support fixture with the forebody down. The sling loads may be applied near the maximum diameter in order to avoid interference between the sling and the probe bus when mating or demating the large probe to the bus.

3.6.4.4 Heat Shield

3.6.4.4.1 Description

The large probe heat shield subsystem shall include a quartz window at the forward stagnation point for the shock layer radiometer, other ablative materials bonded to the aeroshell forebody and afterbody, and a protective radome which covers the communications antenna during entry.

3.6.4.4.2 Capabilities

The heat shield subsystem shall have the capability to provide thermal protection for all other large probe equipment and scientific instruments during entry, to control surface erosion and mass loss during entry so that no significant changes in probe aerodynamic shape occur, to accept appropriate thermal control coatings for protection during other mission phases, to maintain mechanically sound attachment to the aeroshell throughout test and all mission phases, and to permit deployment of the parachute and separation of the aeroshell afterbody.

3.6.4.4.3 Thermal Performance During Entry

The heat shield subsystem shall limit the temperature of the bondline between itself and the aeroshell during entry to less than 300°F during the time period that entry stagnation pressures exceed 0.1 bar and to less than $600^{o}F$ after stagnation pressures have dropped below 0.1 bar up until the time the descent capsule is separated from the aeroshell forebody (nominally 1.5 minutes after entry).

3.6.4.4.4 Radome RF Performance

The RF attenuation due to the radome shall not exceed 0.2 db at frequencies between 2100 and 2200 MHz anywhere within a cone with 65 deg half angle centered along the aft axis of symmetry of the aeroshell. This requirement shall be met both before and after exposure to entry heating.

3.6.4.4.5 Mechanical Requirements

The heat shield subsystem materials shall be selected to resist the aerodynamic (shear) loads and erosion during entry, to withstand without permanent deformation the loads which are imposed by ground handling slings and fixtures, and to withstand the concentrated loads at the probe bus attach point without deformation which is detrimental to mission success.

The radome shall be capable of withstanding a pressure differential of 1 psi in either direction without permanent deformation.

The joint in the heat shield at the aeroshell afterbody separation line shall maintain thermal integrity during entry and shall not require a force of greater than 20 lbs for separation after entry. The joint in the heat shield provided for parachute deployment shall maintain thermal integrity during entry and shall not require a force greater than 20 lbs for separation after entry.

3.6.4.4.6 Storage and Mission Life

The heat shield subsystem shall be designed for storage for at least 24 months prior to launch and to withstand the interplanetary cruise prior to entry as specified in Section 3.6.1.

3.6.4.5 Parachute

3.6.4.5.1 Description

The parachute subsystem shall consist of a parachute, swivel, 3 cable bridle, and mortar for deployment.

3.6.4.5.2 Capabilities

The parachute subsystem shall have the capability to operate successfully after 4 months of storage within the large probe prior to launch (including exposure to probe environmental test environments) and after the interplanetary cruise specified in Section 3.6.1. The parachute subsystem shall be capable of subsonic deployment in the Venus atmosphere, of extracting the descent capsule from the aeroshell forebody and, in combination with the descent capsule drag providing reduced descent rates in the altitude range of approximately 70 to 50 km.

3.6.4.5.3 Ballistic Coefficient

The parachute shall be sized to maintain a ballistic coefficient of TBD + TBD slugs/ft^2 in conjunction with the descent capsule while the two are attached.

3.6.4.5.4 Deployment

The parachute subsystem shall be capable of deployment at any combination of altitude and Mach numbers within the following ranges:

a) Altitudes between 71.7 and 69.2 km

b) Mach numbers between 0.7 and 0.8.

3.6.4.5.5 Minimum Altitude

The parachute shall be capable of operation down to at least 48 km altitude.

3.6.4.6 Descent Capsule

3.6.4.6.1 Description

The descent capsule subsystem shall include a pressure vessel with windows and ports for scientific instruments and with internal structure to support the probe electronic subsystems and certain scientific instruments, external insulation, a vented aerodynamic cover for the external insulation, an aerodynamic flare structure, and spin fins.

3.6.4.6.2 Capabilities

The descent capsule subsystem shall be capable of providing thermal protection and a controlled pressure environment throughout the mission to the probe equipment and scientific instruments mounted inside it. The descent capsule subsystem shall also provide structural support for internally mounted equipment and instruments throughout the mission and provide viewing and sampling windows or ports with the fields of view or exposure for the scientific instruments as specified in Section 3.5.2.5.

3.6.4.6.3 Ballistic Coefficient

After separation from the parachute and aeroshell the descent capsule shall provide a ballistic coefficient of TBD + TBD slugs/ft^2.

3.6.4.6.4 Stability

The requirements of Section 3.6.4.1.3 shall be met by the descent capsule subsystem including the probe equipment and scientific instruments which it supports and encloses.

3.6.4.6.5 Pressure Vessel

The pressure vessel including windows and penetrations shall be capable of withstanding at least one exposure to combined pressure and ambient temperature profiles equivalent to descent to the surface of Venus (6050 km) without catastrophic failure. Permanent deformation of the pressure vessel after one exposure to the Venus environment shall not be large enough to cause failure of pressure seals, internally mounted equipment, or scientific instruments except that the descent capsule shall not be required to meet scientific instrument alignment requirements after such exposure.

The pressure vessel, its windows, and the seals provided at assembly joints shall be designed so that the requirements of Section 3.5.2.3.3 (Operating Atmosphere Within the Pressure Vessel) are met after exposure of the descent capsule to acceptance test environments as specified in Section 4.0.

3.6.4.6.6 Structure

In addition to the requirements of Section 3.6.4.6.5 the entire descent capsule shall be designed to support the probe equipment and scientific instruments and to withstand the entry loads and acceleration environment without permitting damaging mechanical interference. The descent capsule shall have sufficient structural rigidity so that scientific instrument alignment requirements as specified in PV 1006.02 are met after exposure to the entry environment and throughout descent to the surface (6050 km).

The descent capsule internal structure shall be designed to support the scientific instrument sensors adjacent to their viewing or sensing ports as required in PV 1006.02 and to permit the probe equipment to be arranged so the probe mass property and center of gravity constraints derived from stability requirements are met.

3.6.4.7 Mechanisms

3.6.4.7.1 Description

The large probe mechanisms subsystem shall include the clamping rings, explosive bolts, pin pullers and similar hardware to attach and release the probe elements which separate or deploy during entry and descent.

3.6.4.7.2 Capabilities

The mechanisms subsystem shall be capable of providing the structural attachments between the descent capsule and the forebody aeroshell, and between the descent capsule and the afterbody of the aeroshell. The mechanisms subsystem shall accept firing currents from the Electric Power subsystem in order to initiate pyrotechnically operated release devices to separate the descent capsule and aeroshell afterbody from the aeroshell forebody and then to separate the aeroshell afterbody from the descent capsule.

3.6.4.7.3 Performance Requirements

The mechanisms subsystem shall provide for release of the separable elements in a manner which assures that no damaging recontact of the two bodies involved can occur during the transient following release. The use of guide rails or similar devices to meet this requirement is permissible but is not encouraged. All end products of separation such as gases and cut bolts shall be contained and constrained.

3.6.4.8 Thermal Control

3.6.4.8.1 Description

The large probe thermal control subsystem shall consist of thermal insulation, heaters, and coatings which in conjunction with the heat shield and descent capsule provide a controlled thermal environment throughout the mission for all probe equipment and scientific instruments.

3.6.4.8.2 Capabilities

The thermal control subsystem shall be capable of maintaining all probe equipment and scientific instruments within specified temperature limits throughout the mission except that instruments mounted outside the descent capsule will include their own thermal control provisions for the period after separation of the descent capsule from the aeroshell forebody.

3.6.4.8.3 Windows

The thermal subsystem shall maintain the outer elements of the windows provided for scientific instruments at least $\underline{10}^{\circ}C$ above the ambient gases for the descent trajectory beginning at an altitude of at least 75 km and extending to the end of the nominal mission profile.

3.6.4.9 Electrical Power Subsystem

3.6.4.9.1 Description

The large probe electrical power subsystem shall consist of a rechargeable battery, wiring harnesses, and other equipment including switching devices required to provide the control DC power to ordnance devices, other large probe subsystems, and to the large probe scientific instruments. Ordnance firing circuitry shall comply with requirements of AFETRM 127-1 where inadvertent actuation of ordnance could injure personnel or adversely affect the basic mission.

3.6.4.9.2 Capability

The electrical power subsystem shall be capable of accepting electrical power via the test connectors and distributing this power to other probe subsystems and probe scientific instruments so that tests can be conducted without discharging the probe battery. The electrical power subsystem shall be capable of safing, arming, and firing probe pyrotechnic devices on command from the data handling and command subsystem and of providing regulated 28 VDC power for test and operation of scientific instruments and other probe subsystems. The electric power subsystem shall provide fault isolation to protect against shorting of conductors in the umbilical which connects to the probe bus, and to prevent failures in any scientific instrument from degrading the power supplied to other instruments or to the probe subsystems.

3.6.4.9.3 Voltage Regulation

The electric power subsystem shall provide power at 28 ± 2.8 VDC to the science instruments and to other probe subsystems. Transients during power switching and pyro firing may exceed the above voltage limits for periods not to exceed 15 milliseconds but in no case shall the voltage exceed the range of 22 to 33 volts. The electrical power subsystem shall not be required to regulate power provided via the test connectors.

3.6.4.9.4 Energy Storage Capacity

The electrical power subsystem shall have adequate energy storage capacity so that during the nominal mission it shall be capable of supplying power after the last charge cycle for all of the following:

a) Operation of the coast timer for at least 30 days.

b) Operation of the large probe and its scientific instruments for at least 2 checkout sequences of at least 5 minutes duration each, including operation of the transmitter at full power for at least 1 minute during each checkout.

c) Operation of any heaters required during the period between launch and entry including thermal control for the coast timer and the battery in particular.

d) Operation of the probe subsystems commencing 15 ± 5 minutes prior to entry in order to provide checkout and calibration data.

e) Operation of all scientific instruments except the wind/altitude radar beginning 15 ± 5 minutes prior to entry and continuing until the end of the nominal mission. At least TBD W steady state power shall be available for this purpose with at least TBD W available during the period between entry and deceleration to subsonic speeds.

f) Operation of the wind/altitude radar from the time an altitude of TBD ± TBD km is reached until the end of the nominal mission. At least 48 W of power shall be available for this purpose.

g) Operation of the window heaters in order to meet the requirements of Section 3.4.5.3.1.

h) Operation of all probe subsystems as required to meet their functional requirements (including warmup as required) throughout entry and descent until the end of the nominal mission.

3.6.4.9.5 Temperature Limits

The electric power subsystem is required to provide power throughout the mission, beginning several days before launch when the coast timer will be initialized. The electric power subsystem shall be compatible with operation in the thermal environment existing within the large probe during prelaunch, launch, interplanetary cruise, free flight before entry, entry, and descent mission phases.

3.6.4.9.6 Life

The electric power subsystem and the battery in particular shall be capable of meeting all requirements after being stored within the large probe for 4 months, being exposed to the effects of large probe and spacecraft acceptance test environments prior to launch, and being subjected to the effects of the launch, cruise, entry and descent environment.

3.6.4.9.7 Fault Protection

The electric power subsystem shall provide an individually fused branch circuit for each scientific instrument. Diode isolation and fusing shall be provided to protect against effects of shorting any power conductor which penetrates the descent capsule pressure vessel and any conductor which is disconnected in flight. The battery shall be protected from the effects of shorting in any electroexplosive device initiator.

3.6.4.10 Data Handling and Command Subsystem

3.6.4.10.1 Description

The data handling and command (DHC) subsystem shall include redundant "g" switches to sense entry, a coast timer to control events prior to entry, a descent timer to control events during and after entry, engineering transducers, and an electronics assembly to accept, process, store (as necessary), and encode engineering and scientific data for transmission to earth.

3.6.4.10.2 Capability

The DHC subsystem shall have the capability to accept digital and bilevel scientific data and analog and bilevel engineering data, condition the data, store the data during blackout and other periods such as calibration, and multiplex both the stored and real time data into a single data bit stream for transmission to earth. The DHC subsystem shall have the capability for accepting commands and transmitting telemetry data through the probe test connectors and shall be capable of producing a sequence of stored commands based on time elapsed from an initializing command prior to launch and a second sequence of commands based on time elapsed from the point at which the DHC acceleration switches sense 50 g increasing at entry. The DHC shall have the capability to measure descent capsule internal pressure and to measure temperatures of the pressure vessel and selected probe electronic units.

3.6.4.10.3 Stored Command Sequences

a. Coast Timer

The DHC subsystem shall provide the capabilities required in Section 3.5.2.10 a, Coast Timer. The same equipment shall be used to implement any additional stored commands required for successful operation of the large probe up until the time of entry.

All event switching shall be programmed relative to the planned entry time. Provision shall be included in the DHC for initializing the coast timer and checking its operation via the large probe test connectors prior to separation.

The DHC subsystem shall be designed so that the degradation to overall mission success due to possible failures in the coast timer are minimized. In particular DHC logic shall be designed to place the probe subsystems and scientific instruments in a minimum power mode when it reaches the end of its normal command sequence and it shall be designed so that it will not recycle the command sequence if, for example, it fails in a manner which causes the timer to run abnormally fast.

b. Entry Timer

The DHC subsystem shall provide the capabilities required in Section 3.5.2.10 b, Entry Timer. The same equipment shall be used to implement any additional stored commands required for successful operation of the large probe after entry and during descent until the end of the nominal mission.

The entry timer shall operate independently from the coast timer and shall automatically disable the coast timer output(s) and place the probe subsystems and scientific instruments in the proper mode for entry (if not already in that mode) immediately upon reaching the entry acceleration threshold of 490 ± 10 m/sec^2 (~50 g's). Redundant acceleration sensitive switches shall be used to sense the above threshold. Closure of either switch shall initiate the entry timer sequence without requiring and independent of any outputs from the coast timer. Mechanical and electrical characteristics of the switches and associated electronics shall be selected to provide reliable operation during entry and to provide immunity from generating spurious outputs due to shock, vibration or acceleration loads at any time prior to entry.

The entry timer logic shall be designed so that if the large probe should survive beyond the end of the nominal mission, it will remain in the normal mode used for terminal descent and continue acquiring and transmitting data until the battery is exhausted or until the natural environment induces failures which terminate probe operation.

Provision shall be included to permit control and monitoring entry timer status via the test connectors in order to facilitate tests of the probe and its scientific instruments.

c. Command Characteristics

Commands generated by both the entry and the coast timer for the scientific instruments shall meet the electrical requirements of PV 1006.02.

Output circuits for commands generated by the entry timer and routed to the electrical power subsystem for arming and firing pyro circuits shall not be required to be redundant but shall be compatible with the circuits in the electrical power subsystem which provide the redundancy required in Section 3.4.1.3.

Commands generated by the coast and entry timers for use by other probe subsystems shall be compatible with the requirements of those subsystems.

3.6.4.10.4 Data Handling

a. Outputs to Scientific Instruments

In addition to the stored commands required in Section 3.6.4.10.3, the DHC subsystem shall meet the requirements of Section 3.5.2.

b. Inputs from Scientific Instruments

The DHC subsystem shall meet the requirements of PV 1006.02, arrange the data in format (or formats as appropriate) along with sufficient timing and identification data so that data from each input from the scientific instruments can be identified uniquely and can be reconstructed as a time related data stream. The status of an on board clock signal shall be included in the data format at least once every TBD seconds so that the relative timing of data samples occurring at any time during the entry and descent sequence can be established to within ±0.2 second even if downlink communications are temporarily interrupted during data transmission.

3.6.4.10.5 Data Output Signals

The DHC shall provide two serial digital data signal outputs as follows:

a) The pre-encoded output routed to the probe test connector shall be NRZ-L at information bit rates of 128, 256, or 512 bps.

b) The post-encoded output to the probe communications system shall be a bi-phase modulated squarewave subcarrier at an amplitude determined by the modulation index requirement of the transmitter and an information bit rate of 128 bps. The subcarrier frequency shall be between 16 and 40 kHz. The variation in the average output squarewave time period shall be less than 0.5 percent (short term phase stability). Rate = 1/2, k = 32 convolutional coding shall be used for this bit stream.

The data (information) contained in each of the above serial digital data signal outputs shall be identical. Probe engineering data may be included in this bit stream on a "space available" basis after all of the data from the scientific instruments has been accommodated along with the required timing and identification data.

3.6.4.10.6 Format Rearrangement

The DHC subsystem shall provide the capability of rearranging the word allocation in the scientific format by minimum hardware changes such that no requalification and a minimum of retest are required.

3.6.4.10.7 Signal Buffering

Signal and data lines shall be buffered such that an open-circuit or short circuit failure of any given line external to the DHC subsystem shall not result in the loss or failure of other signals or data lines.

3.6.4.10.8 Operating Modes

The DHC subsystem shall provide the capability to operate the large probe and its scientific instruments in at least the following modes:

a) Test modes using external commands routed to the DHC subsystem via the large probe test connectors. It shall be possible to turn on and operate the scientific instruments individually or in any combination up to the maximum capability required in flight. It shall be possible to operate the probe subsystems and scientific instruments on either internally or externally supplied power. It shall be possible to control individual probe units including the transmitter in particular in order to facilitate testing and provide diagnostic capability. Battery checkout and charging shall be included as part of the test modes. In at least one test mode it shall be possible to cycle through the entire (120 day) launch and coast mode sequence in an accelerated fashion using an externally supplied clock signal.

b) Launch and coast mode. This mode shall be entered when the coast mode timer is initialized. It shall be possible to verify operation and proper initialization of the coast mode timer via the test connectors while in this mode. It shall also be possible to command the probe out of this mode via the test connector and cause it to enter other modes including the test modes in particular. No hardware change shall be required to initialize the coast timer and enter the launch and coast mode.

c) Post release checkout mode. This mode shall be entered on command from the coast timer and shall cause the probe subsystems and scientific instruments to be energized and operated in a manner so that their health and status can be assessed prior to entry.

d) Pre-entry mode(s). This mode or set of modes shall be entered on command from the coast timer. Sequential power switching to heaters, other probe equipment, and the scientific instruments shall be provided as required to assure, prior to entry, that temperatures of the equipment and instruments have reached the values required for successful operation.

e) Entry modes. These modes shall be entered only as a result of sensing entry acceleration as required in Section 3.6.14.3.2 or in response to specific test commands received via the test connectors. These modes shall provide for control and sequencing of probe subsystem equipment including the firing of pyros to control separation and deployment, and for control and sequencing of the scientific instruments via the stored commands required in Section 3.4.4.6. The entry modes shall control all probe and instrument operations between entry and the end of mission.

3.6.4.10.9 Spurious Commands

The DHC subsystem shall operate in a manner which shall prevent unintentional and potentially damaging commands from being issued to other probe subsystems or to the scientific instruments at any time and particularly when power is turned on to the DHC subsystem or during subsequent power transients within the limits prescribed in Section TBD

3.6.4.11 Communications Subsystem

3.6.4.11.1 Description

The large probe communications subsystem shall consist of a coherent S-band receiver, a transmitter driver, a power amplifier, filters, diplexer, an antenna, RF cabling, and instrumentation to measure performance.

3.6.4.11.2 Compatibility

The communications subsystem shall be compatible with other probe subsystems and with the DSIF as specified in PV 1006.03.

3.6.4.10.3 Capability

The communication subsystem shall have the capability to:

a) Radiate a PSK/PM radio signal to enable radio acquisition of the spacecraft by the DSIF. After acquisition of an uplink signal, establish and maintain phase coherent transmission with the signal received from the DSIF for purposes of measuring two-way doppler shift. The transfer from one-way to two-way operation, and vice versa, shall occur automatically and shall be controlled by signal present detection circuitry within the probe receiver.

b) Telemeter to the DSIF scientific and engineering data as generated by the data handling and command (DHC) subsystem.

3.6.4.10.4 Suppression of Spurious Frequencies

The subsystem shall contain sufficient filtering so that the power of spurious frequencies in the transmitter output, including carrier harmonics, shall be more than 40 dB below the level of the unmodulated carrier as measured at the input to the antenna.

3.6.4.10.5 Receiver Performance

a. Type

The receiver shall be of the phase-lock-loop type with an automatic gain control circuit. Sweep circuitry shall be employed to minimize two-way acquisition time.

b. Receiver Frequency

The receiver shall operate at any frequency selected between 2110.24 and 2119.8 MHz with the exact frequency controlled by means of crystal selection. Within the limits of stability and temperature effects, the acquisition frequency of each receiver shall be the same at all times.

c. Warmup

The receivers shall be designed to minimize the time to reach the final operating frequency after power is applied to the receiver consistent with other requirements. The time required to stabilize so that the requirements of Section 3.6.4.11.5 d (below) are met shall not exceed 10 minutes.

d. Frequency Stability

With no signal at the receiver input terminals and during any 10-hour period, the best lock frequency shall not deviate more than 3 KHz for a temperature change from -17.8°C to 68.3°C.

e. Bandwidth

The receiver bandwidth and phase-lock-loop characteristics shall be compatible with the information transmission requirements as specified in the Doppler shift, and the rate of Doppler shift resulting from mission trajectories. For strong signal conditions, the receiver shall be capable of tracking over a frequency range of ±135 KHz with a phase error less than 6 degrees and at a frequency rate up to 175 Hz/sec. For weak signal conditions, the receiver shall be capable of tracking over a frequency range of ±125 KHz with a phase error less than 25 degrees at a frequency rate up to 5 Hz/sec.

f. Control of Transmitter Frequency

The receiver shall control the output frequency of the transmitter driver at any time the receiver is locked to an uplink signal (coherent mode). This control frequency shall be such that the transmitter driver S-band output frequency is precisely 240/221 times the received uplink frequency. The change to receiver control of the transmitted frequency shall be automatic and shall be controlled by the presence, or lack of, a received signal exceeding the switchover threshold.

3.6.4.10.6 Transmitter Performance (Transmitter Driver Plus Power Amplifier)

a. Frequency

The transmitter shall operate at a frequency between 2290 and 2300 MHz.

b. Control

The transmitter frequency shall be controlled by an auxiliary crystal controlled oscillator during periods when the receiver is not in lock with the ground transmitter.

c. Short Term Frequency Stability

When the temperature and supply voltage for the probe transmitter are steady and during noncoherent operation, the peak to peak frequency deviation within ten successive one-minute time periods shall be less than 0.5 Hz for a 1/4 second integration time, that is, within a 4 Hz bandwidth.

d. Long Term Frequency Stability

During one-way operation and any 10-hour period, the transmitter frequency shall not vary more than 1.0 KHz for a temperature change from $-17.8^{\circ}C$ to $68.3^{\circ}C$.

e. Phase Stability

When operating with a phase coherent test receiver and a test transmitter having the characteristics specified below, the large probe transponder and power amplifier shall meet the following requirements:

1) In the two-way mode under strong uplink signal conditions, the phase stability of the unmodulated carrier measured

at the output of the transmitter driver shall cause no more than <u>2.8</u> degrees rms or <u>8.4</u> degrees peak phase error.

2) In the one-way mode, the phase stability of the unmodulated carrier measured at the output of the transmitter driver shall cause no more than <u>3.6</u> degrees rms or <u>10.8</u> degrees peak phase error.

3) The power amplifier shall add no more than 3.0 degrees rms phase error to the above values.

The test receiver shall have a strong signal double side bandwidth ($2B_L$) of 12 Hz or less and a high frequency cutoff of 1 KHz or more. The test transmitter and receiver shall have a self-test mode to establish their contribution to the measured phase error; the contribution shall be less than 10 degrees rms. The phase error attributed to the spacecraft shall be calculated by the root sum square method.

f. <u>Power Amplifier</u>

The minimum power output from the amplifier including effects supply voltage variation within the limits of Section 3.6.13.3 shall be adequate to ensure that the EIRP requirements of Section 3.6.15.9 are met.

g. <u>Modulation Index</u>

The modulation index shall be adjustable between 0.5 and 1.2 radians by minor changes of components in the transmitter driver. The flight value selected for the modulation index shall be within this range, shall be adjusted prior to acceptance testing of the transmitter driver, and shall remain with ±0.05 radian of the selected value.

h. <u>Warmup</u>

The transmitter shall be designed to minimize the time to reach the final operating frequency after power is applied. The time shall not exceed 10 minutes.

3.6.4.10.7 <u>Antenna</u>

a. <u>Antenna Polarization</u>

The polarization of the antenna for both transmit and receive signals shall be right hand circular and in the same direction.

b. Antenna Beamwidth

The transmission beamwidth of the antenna shall not be less than 130 degrees as measured at the -3 dB points over the frequency range of 2.1 to 2.3 GHz. Antenna gain at any angle within the 65 degrees (half angle) cone surrounding the aft axis of symmetry of the large probe shall be adequate to assure that the communications subsystem performance requirements (EIRP) of Section 3.6.4.11.9 are met with transmitter power amplifier operating at minimum power output.

3.6.4.10.8 Performance-Uplink

During the nominal probe mission the probe shall be capable of receiving and phase locking to a signal having the total power density given below for the directions indicated:

Power Density, dBm per square meter	Earth Aspect Angle, (Degrees)
(a) TBD	180 to 115 where zero is forward (downward) axis of symmetry of probe during entry

3.6.4.10.9 Performance – Downlink

During the nominal mission the probe shall have a minimum effective isotropic radiated power (EIRP) for the following earth aspect angles:

EIRP, dBm	Earth Aspect Angle, (Degrees)
(a) ≥ TBD	180 to 115

3.6.4.10.10 Telemetry Data

The probe telemetry data shall contain information about:

a) Signal presence

b) Component temperatures

c) Output power

d) Receiver static phase error

e) Received signal strength

3.6.5 Small Probe Performance Requirements

3.6.5.1 Stability

3.6.5.1.1 Hypersonic Stability Margin

The static and dynamic stability margins and the aerodynamic damping during entry shall be adequate to ensure that oscillations resulting from the small probe entering with attitudes and rates within the limits specified in Section 3.6.2.10 shall be damped to less than 1 degree half amplitude by the time the probe deceleration reaches 10 g's. In addition, any subsequent excursions in angle of attack shall be damped to less than 10 degrees at time of temperature probe deployment.

3.6.5.1.2 Descent Phase Stability

The transonic and subsonic static and dynamic stability margins of the small probe shall be adequate to limit the maximum departure of the small probe attitude relative to its nominal flight path to less than 15 degrees in response to the design wind profile specified in Section 3.4.10.7. Angle of attack oscillations from this transient shall not exceed 5 degrees peak half amplitude and shall be damped to 1 degree or less (half amplitude) within 5 seconds after the wind returns to steady state velocity.

The maximum steady state trim angle of attack shall be less than 1 degree in the absence of winds. As a goal, in the absence of winds the small probe shall not exhibit any tendency to limit cycle or oscillate about the trim angle of attack. If such motions do occur they shall not exceed 2 degrees half amplitude.

3.6.5.1.3 Spin Rate During Entry and Descent

The small probes shall be designed so that for attitudes and rates at entry within the limits specified in Section 3.6.2.10 the probe spin rate shall not exceed 30 rpm at any time.

3.6.5.2 Mission Termination

The small probes shall not be required to survive or operate beyond the time it reaches the surface of Venus or a radius of 6050 km, whichever occurs first. If the probe does survive after reaching the surface, scientific measurements will be made until end of life.

3.6.5.3 Aeroshell

3.6.5.3.1 Description

The small probe aeroshell subsystem shall consist of structural elements which form the aerodynamic shape during entry, resist the aerodynamic loads during entry, and enclose and support all other small probe subsystems (except the heat shield) and scientific instruments throughout the mission.

3.6.5.3.2 Capability

The aeroshell subsystem shall have the capability of maintaining the designed aerodynamic shape within mechanical limits which assure that hypersonic, transonic and subsonic stability requirements will be met. It shall have the capability of providing mechanical support to the heat shield subsystem on the exterior of the aeroshell and the other probe equipment and instruments on the interior. The aeroshell subsystem shall include provisions for venting its internal volume during test, launch, and entry to avoid excessive pressure differentials across the aeroshell and shall provide two deployment joints and apertures in the forebody to permit deployment of the temperature probe and the nephelometer and a third aperture at the forward stagnation point for the pressure gauge sensing port. The aeroshell subsystem shall also provide an aperture for the communications antenna mounted on the pressure vessel.

3.6.5.3.3 Ballistic Coefficient

The aeroshell shall provide an aerodynamic shape with a hypersonic ballistic coefficient of TBD ± TBD slugs/ft^2 with the small probe mass at the maximum value allocated in Section 3.4.9.2.1. The subsonic ballistic coefficient shall be such that the requirements of Section 3.6.2.10.2 are met.

3.6.5.3.4 Rigidity

The aeroshell shall have adequate structural ridigity to maintain the aerodynamic shape within limits which will assure mission success and to maintain overall probe structural frequencies above the frequency range of probe angle of attack oscillation in order to minimize possibilities of dynamic coupling.

3.6.5.3.5 Thermal Environment

The aeroshell shall be compatible with temperatures between -40 and 150°F prior to entry. The aeroshell shall be capable of meeting performance requirements during entry if the bond line temperature between the heat shield and the aeroshell is maintained below 300°F until after entry stagnation pressures have decreased below 0.1 bar, and the aeroshell shall continue to maintain structural integrity and alignment throughout the descent phase until the end of the nominal mission profile.

3.6.5.3.6 Provisions for Ground Handling

The aeroshell structure shall be compatible with being supported by a specially designed sling with the forebody facing either upward or downward and with being supported in a suitably padded support fixture with the forebody down. The sling loads may be applied at less than the maximum diameter in order to avoid interference between the sling and the probe bus when mating or demating the small probe to the bus.

3.6.5.4 Heat Shield

3.6.5.4.1 Description

The small probe heat shield subsystem shall have the capability to provide thermal protection for all other small probe equipment and scientific instruments during entry, to control surface erosion and mass loss during entry so that no significant changes in probe aerodynamic shape occur, to accept appropriate thermal control coatings for protection during other mission phases, to maintain mechanically sound attachment to the aeroshell throughout test and all mission phases, and to permit deployment of the temperature sensor after peak heating.

3.6.5.4.3 Thermal Performance During Entry

The heat shield subsystem shall limit the temperature of the bondline between itself and the aeroshell during entry to less than 300°F during the time period that entry stagnation pressures exceed 0.1 bar and to less than 600°F for TBD minutes after stagnation pressures have dropped below 0.1 bar.

3.6.5.4.4 Radome RF Performance

The RF attenuation due to the radome shall not exceed 0.2 db at frequencies between 2290 and 2300 MHz anywhere within a cone with 65 deg half angle centered along the aft axis of symmetry of the aeroshell. This requirement shall be met both before and after exposure to entry heating.

3.6.5.4.5 Mechanical Requirements

The heat shield subsystem materials shall be selected to resist the aerodynamic (shear) loads and erosion during entry, to withstand without permanent deformation the loads which are imposed by ground handling slings and fixtures, and to withstand the concentrated loads at the probe bus attach points without deformation which is detrimental to mission success.

The joint in the heat shield provided for temperature sensor deployment shall maintain thermal integrity during entry and shall not require a force greater than TBD pounds for separation after entry.

The radome shall be capable of withstanding the pressure differentials imposed across it during test, launch, entry and descent.

3.6.5.4.6 Storage and Mission Life

The heat shield subsystem shall be designed for storage for at least 24 months prior to launch and to withstand the interplanetary cruise prior to entry as specified in Section 3.6.1.

3.6.5.6 Pressure Vessel

3.6.5.6.1 Description

The small probe pressure vessel subsystem shall include a pressure vessel with windows and ports for scientific instruments, internal structure to support the probe electronic subsystems and scientific instruments (except the magnetometer sensor), and external insulation.

3.6.5.6.2 Capabilities

The pressure vessel subsystem shall be capable of providing thermal protection and a controlled pressure environment throughout the mission to the probe equipment and scientific instruments mounted inside it. The pressure vessel subsystem shall also provide structural support for internally mounted equipment and instruments throughout the mission and shall provide viewing and sampling windows or ports with the fields of view or exposure for the scientific instruments as specified in Section 3.5.3.

3.6.5.6.3 Pressure Vessel

The small probe pressure vessel including windows and penetrations shall be capable of withstanding at least one exposure to combined pressure and ambient temperature profiles equivalent to descent to the surface of Venus (6050 km) without catastrophic failure. Permanent deformation of the pressure vessel after one exposure to the Venus environment shall not be large enough to cause failure of pressure seals, internally mounted equipment, or scientific instruments except that the pressure vessel shall not be required to meet scientific instrument alignment requirements after such exposure.

The pressure vessel, its windows, and the seals provided at assembly joints shall be designed so that the requirements of Section 3.5.2.3.3 (Operating Atmosphere Within the Pressure Vessel) are met after exposure of the descent capsule to acceptance test environments as specified in Section 4.0.

3.6.5.6.4 Structure

In addition to the requirements of Section 3.6.5.6.3 the pressure vessel shall be designed to support the probe equipment and scientific instruments and to withstand the entry loads and acceleration environment without permitting damaging mechanical interference. The pressure vessel shall have sufficient structural rigidity so that scientific instrument alignment requirements as specified in PV 1006.02 are met after exposure to the entry environment and throughout descent to the surface (6050 km).

The pressure vessel internal structure shall be designed to support the scientific instrument sensors adjacent to their viewing or sensing ports as required in PV 1006.02 and to permit the probe equipment to be arranged so the probe mass property and center of gravity constraints derived from stability requirements are met.

3.6.5.7 Mechanisms

3.6.5.7.1 Description

The small probe mechanisms subsystem shall include the hardware to attach and release the probe elements which separate or deploy during entry and descent.

3.6.5.7.2 Capabilities

The mechanisms subsystem shall be capable of accepting currents from the Electric Power subsystem in order to initiate release devices to deploy the temperature sensor and nephelometer.

3.6.5.8 Thermal Control

3.6.5.8.1 Description

The small probe thermal control subsystem shall consist of thermal insulation, heaters, phase change material, and coatings which in conjunction with the heat shield pressure vessel provide a controlled thermal environment throughout the mission for all probe equipment and scientific instruments.

3.6.5.8.2 Capabilities

The thermal control subsystem shall be capable of maintaining all probe equipment and scientific instruments within specified temperature limits throughout the mission except that instruments mounted outside the descent capsule will include their own thermal control provisions for the period after peak heating during entry.

3.6.5.8.3 Windows

The thermal subsystem shall maintain the outer elements of the windows provided for scientific instruments at least 10°C above the ambient gases for the descent trajectory beginning at an altitude of at least 75 km and extending to the end of the nominal mission profile.

3.6.5.8.4 Equipment Temperature Limits

The thermal control subsystem shall maintain the probe equipment and scientific instruments within the following limits:

	Limits While Operating (deg F)	Limits, Non-Operating (deg F)
Probe electronics	0 to 150	-50 to 150
Probe battery (prior to entry)	55 to 90	0 to 85
Probe battery (during entry and descent)	5 to 180	—
Scientific instruments within pressure vessel	0 to 150	-4 to 150
Heat shield and aeroshell	Not applicable	-50 to 200
Scientific instruments external to pressure vessel (prior to entry)	0 to 150	-50 to 200
Scientific instruments external to pressure vessel (entry to separation of aeroshell)	-4 to TBD	—

3.6.5.9 Electrical Power

3.6.5.9.1 Description

The small probe electrical power subsystem shall consist of a rechargeable battery, wiring harnesses, and other equipment including switching devices to provide and control DC power to ordnance devices, other probe subsystems, and to the probe scientific instruments. Ordnance firing circuitry shall comply with the requirements of AFETRM 127-1 where inadvertent actuation of ordnance could harm personnel or adversely affect the basic mission.

3.6.5.9.2 Capability

The electric power subsystem shall be capable of accepting electrical power via the small probe connector and distributing this power to other probe subsystems and probe scientific instruments so that tests can be conducted without discharging the battery. The electric power subsystem shall be capable of safing, arming, and firing probe pyrotechnic devices on command from the data handling and command subsystem and of providing regulated 28 VDC power for test and operation of scientific instruments and other probe subsystems. The electric power subsystem shall provide fault isolation to protect against shorting of conductors in the test connector and to prevent failures in any scientific instrument from degrading the power supplied to other instruments or to the probe subsystems.

3.6.5.9.3 Voltage Regulation

The electric power subsystem shall provide the capability required in Section 3.6.4.9.3.

3.6.5.9.4 Energy Storage Capacity

The electrical power subsystem shall have adequate energy storage capacity so that during the nominal mission it shall be capable of supplying power after the last charge cycle for all of the following:

a) Operation of the coast timer for at least 30 days.

b) Operation of the small probe and its scientific instruments for at least 2 checkout sequences of at least 5 minutes duration each, including operation of the transmitter at full power for at least 1 minute during each checkout.

c) Operation of any heaters required during the period between launch and entry including thermal control for the coast timer and the battery in particular.

d) Operation of the probe subsystems commencing 15 ± 5 minutes prior to entry in order to provide checkout and calibration data.

e) Operation of all scientific instruments beginning 15 ± 5 minutes prior to entry and continuing until the end of the nominal mission. At least TBD W steady state power shall be available for this purpose with at least TBD W available during the period between entry and deceleration to subsonic speeds.

f) Operation of the window heaters in order to meet the requirements of Section 3.6.5.8.3.

g) Operation of all probe subsystems as required to meet their functional requirements (including warmup as required) throughout entry and descent until the end of the nominal mission.

3.6.5.9.5 Temperature Limits

The electric power subsystem is required to provide power throughout the mission, beginning several days before launch when the coast timer will be initialized. The electric power subsystem shall be compatible with operation in the thermal environment existing within the large probe during prelaunch, launch, interplanetary cruise, free flight before entry, entry, and descent mission phases.

3.6.5.9.6 Life

The electric power subsystem and the battery in particular shall be capable of meeting all requirements after being stored within the small probe for 4 months, being exposed to the effects of small probe and spacecraft acceptance test environments prior to launch, and being subjected to the effects of the launch, cruise, entry and descent environments.

3.6.5.9.7 Fault Protection

The electric power subsystem shall provide an individually fused branch circuit for each scientific instrument. Diode isolation and fusing shall be provided to protect against effects of shorting any power conductor which penetrates the descent capsule pressure vessel and any conductor which is disconnected in flight. The battery shall be protected from the effects of shorting in any electro-explosive device initiator.

3.6.5.10 Data Handling and Command Subsystem

3.6.5.10.1 Description

The data handling and command (DHC) subsystem shall include redundant "g" switches to sense entry, a coast timer to control events prior to entry, a descent timer, engineering transducers, and an electronics assembly to accept, process, store (as necessary), and encode engineering and scientific data for transmission to earth.

3.6.5.10.2 Capability

The DHC subsystem shall provide the capability required in Section 3.6.4.10.2.

3.6.5.10.3 Stored Command Sequences

The DHC subsystem shall provide the capability required in Section 3.6.4.10.3.

3.6.5.10.4 Data Handling

a. Outputs to Scientific Instruments

In addition to the stored commands required in Section 3.6.5.10.3, the DHC subsystem shall meet the requirements of Section 3.5.3.

b. Inputs from Scientific Instruments

The DHC subsystem shall meet the requirements of PV 1006.02, arrange the data in format (or formats as appropriate) along with sufficient timing and identification data so that data from each input from the scientific instruments can be identified uniquely and can be reconstructed as a time-related data stream. The status of an on-board clock signal shall be included in the data format at least once every TBD seconds so that the relative timing of data samples occurring at any time during the entry and descent sequence can be established to within ±0.2 second even if downlink communications are temporarily interrupted during data transmission.

3.6.5.10.5 Data Output Signals

The DHC shall provide two serial digital data signal outputs as follows:

a) The pre-encoded output routed to the probe test connector shall be NRZ-L at information bit rates of TBD bps.

b) The post-encoded output to the probe communications system shall be a bi-phase modulated squarewave sub-carrier at an amplitude determined by the modulation index requirement of the transmitter and an information bit rate of TBD bps. The subcarrier frequency shall be between 16 and 40 kHz. The variation in the average output squarewave time period shall be less than 0.5 percent (short term phase stability). Rate = 1/2, k = 32 convolutional coding shall be used for this bit stream.

The data (information) contained in each of the above serial digital data signal outputs shall be identical. Probe engineering data may be included in this bit stream on a "space available" basis after all of the data from the scientific instruments has been accommodated along with the required timing and identification data.

3.6.5.10.6 Format Rearrangement

The DHC subsystem shall provide the capability of rearranging the word allocation in the scientific format by minimum hardware changes such that no requalification and a minimum of retest are required.

3.6.5.10.7 Signal Buffering

Signal and data lines shall be buffered such that an open-circuit or short circuit failure of any given line external to the DHC subsystem shall not result in the loss or failure of other signals or data lines.

3.6.5.10.8 Operating Modes

The DHC subsystem shall provide the capability to operate the small probe and its scientific instruments in at least the following modes

a) Test modes using external commands routed to the DHC subsystem via the small probe test connectors. It shall be possible to turn on and operate the scientific instruments individually or in any combination up to the maximum capability required in flight. It shall be possible to operate the probe subsystems and scientific instruments on either internally or externally supplied power. It shall be possible to control individual probe units including the transmitter in particular in order to facilitate testing and to provide diagnostic capability. Battery checkout and charging shall be included as part of the test modes. In at least one test mode it shall be possible to cycle through the entire (120-day) launch and coast mode sequence in an accelerated fashion using an externally supplied clock signal.

b) Launch and coast mode. This mode shall be entered when the coast mode timer is initialized. It shall be possible to verify operation and proper initialization of the coast mode timer via the test connectors while in this mode. It shall also be possible to command the probe out of this mode via the test connector and cause it to enter other modes including the test modes in particular. No hardware change shall be required to initialize the coast timer and enter the launch and coast mode.

c) Post release checkout mode. This mode shall be entered on command from the coast timer and shall cause the probe subsystems and scientific instruments to be energized and operated in a manner so that their health and status can be assessed prior to entry.

d) Pre-entry mode(s). This mode or set of modes shall be entered on command from the coast timer. Sequential power switching to heaters, other probe equipment, and the scientific instruments shall be provided as required to assure, prior to entry, that temperatures of the equipment and instruments have reached the values required for successful operation.

e) Entry modes. These modes shall be entered only as a result of sensing entry acceleration as required in Section 3.7.13.3 or in response to specific test commands received via the test connectors. These modes shall provide for control and sequencing of probe subsystem equipment including the firing of pyros to control separation and deployment, and for control and sequencing of the scientific instruments via the stored commands required in Section 3.5.4.6. The entry modes shall control all probe and instrument operations between entry and the end of mission.

3.6.5.10.9 Spurious Commands

The DHC subsystem shall operate in a manner which shall prevent unintentional and potentially damaging commands from being issued to other probe subsystems or to the scientific instruments at any time and particularly when power is turned on to the DHC subsystem or during subsequent power transients within the limits prescribed in Section 3.7.12.3.

3.6.5.11 Communications

3.6.5.11.1 Description

The small probe communications subsystem shall consist of a transmitter driver, a power amplifier, an antenna, instrumentation to measure performance, and a stable oscillator.

3.6.5.11.2 Compatibility

The communications subsystem shall be compatible with other probe subsystems with the DSN as specified in PV 1006.03.

3.6.5.11.3 Capability

The communications subsystem shall have the capability to:

a) Radiate a PSK/PM radio signal to enable radio acquisition of the small probe by the DSIF stations.

b) Telemeter to the DSIF scientific and engineering data as generated by the data handling and command subsystem.

3.6.5.11.4 Suppression of Spurious Frequencies

The subsystem shall contain sufficient filtering so that the power of spurious frequencies in the transmitter output, including carrier harmonics, shall be more than 40 dB below the level of the unmodulated carrier as measured at the input to the antenna.

3.6.5.11.5 Stable Oscillator Performance Requirements

TBD

3.6.5.11.6 Transmitter Design

a. Frequency Control

The transmitter carrier frequency shall be controlled by the stable oscillator.

b. Phase Stability

When operating with a phase coherent test receiver having the characteristics specified below, the probe transmitter driver and power amplifier shall meet the following requirements:

1) The phase stability of the unmodulated carrier measured at the output of the transmitter driver shall cause no more than TBD degrees rms or TBD degrees peak phase error.

2) The power amplifier shall add no more than TBD degrees rms phase error to the above value.

The test receiver shall have a strong signal double side bandwidth ($2B_L$) of 12 Hz or less and a high frequency cutoff of 1 kHz or more. The test receiver shall have a self-test mode to establish its contribution to the measured phase error; the contribution shall be less than TBD degrees rms. The phase error attributed to the spacecraft shall be calculated by the root sum square method.

c. Power Amplifier

The minimum power output from the power amplifier including the effects of supply voltage variation with the limits required in Section 3.6.5.9.3 shall be adequate to assure that the EIRP requirements of Section 3.6.5.11.8 are met.

d. Modulation Index

The modulation index shall be adjustable between 0.9 and 1.2 radians by minor changes of components in the transmitter driver. The flight value selected for the modulation index shall be within this range, shall be adjusted prior to acceptance testing of the transmitter driver, and shall remain within ±0.05 radian of the selected value.

3.6.5.11.7 Antenna

a. Antenna Polarization

The polarization of the antenna for transmit signals shall be right-hand circular.

b. Antenna Beamwidth

The transmission beamwidth of the antenna shall not be less than 130 degrees as measured at the -3 dB points over the frequency range of 2.1 to 2.3 gHz. Antenna gain at any angle within the 65 degrees (half angle) cone surrounding the aft axis of symmetry of the small probe shall be adequate to assure that the EIRP requirements of Section 3.6.5.11.8 are met with transmitter power amplifier operating at minimum output power.

3.6.5.11.8 Performance

During the nominal mission the probe shall have a minimum effective isotropic radiated power (EIRP) for the following earth aspect angles:

EIRP dBm	Earth Aspect Angle (Degrees)
(a) ≤TBD	180 to 115

3.6.6 Orbiter Spacecraft Performance Requirements

3.6.6.1 Structural Requirements

3.6.6.1.1 General Requirements

The orbiter structures subsystem shall provide for the support and alignment of the other orbiter subsystem equipment and the orbiter scientific instruments.

3.6.6.1.2 Compatibility

The orbiter structure shall be compatible with the Atlas Centaur launch vehicle as specified in Section 3.2.1, and with the scientific instruments as specified in Section 3.5.4.

3.6.6.1.3 Alignment

Provision shall be provided for initial alignment of attitude sensors, communication antennas, attitude control and ΔV thrusters, and scientific instruments to a common spacecraft datum. The initial alignment and subsequent alignment stability throughout the mission (except during launch) shall be adequate for the spacecraft to meet the requirements of Sections 3.6.6.3, 3.6.6.4, and the scientific instrument pointing requirements of PV 1006.02.

3.6.3.1.4 Rigidity

The structural rigidity of the spacecraft shall be in accordance with the requirements of Section 3.4.9.5.

3.6.6.1.5 Vibration Amplification

The vibration amplification in the vicinity of the scientific instruments on the probe bus shall not exceed the ratios given in Table 3.6-2. Vibration amplification is defined as the ratio of the acceleration experienced at the point of interest to the input acceleration due to primary loads during powered flight at the mechanical interface between the launch vehicle and the spacecraft.

Table 3.6-2. Maximum Vibration Amplification at Probe Bus Scientific Instrument Mounting Locations

Vibration Axis	Frequency (Hz)	Ratio
Thrust	10 to 30	4
	30 to 100	13
	100 to 130	4
	130 to 2000	1
Lateral	1 to 100	10
	100 to 2000	2
Torsional	(same as lateral)	

Note: Amplification ratios on the equipment compartment side panels in the 30 to 100 Hz range in the thrust axis and the 1 to 100 Hz range in the lateral and torsional axes may exceed the listed values within narrowband frequency ranges. The bandwidths wherein the ratios may exceed the listed values shall be less than 0.2 time the center band frequency.

3.6.6.1.6 Weight, Balance, and Inertia

The orbiter structure shall be designed so as to permit mounting of subsystems and the scientific instruments in position to insure proper weight, balance, and inertial characteristics compatible with all spacecraft subsystems, the scientific instrument scan requirements and launch vehicle payload restraints.

3.6.6.1.7 Primary Loads

The orbiter structure, including all appendages shall withstand the primary loads specified below without incurring any damage.

a) Handling and shipping loads in accordance with Section 3.4.10.1.

b) Aerodynamic loads encountered during spin balance operations as required in Section 4.0.

c) Loads due to structural and acoustic excitation during launch as specified in PV 1006.04.

d) Loads due to the nominal 60 RPM spin rate and to structural excitation during thrusting of the solid rocket motor for orbit insertion.

3.6.6.1.8 Engineering Telemetry

The condition of all deployable appendages (stowed or deployed) and of the launch vehicle/spacecraft interface (attached or separated) shall be contained in the engineering telemetry data.

3.6.6.1.9 Provisions for Ground Handling

At least three hard points shall be provided on the orbiter structure for attachment of handling, measuring, and shipping equipment.

3.6.6.2 Electrical Power Subsystem

3.6.6.2.1 Description

The orbiter electrical power subsystem shall consist of a solar array, power controls, a battery, fault protection devices, regulators, converters, and power cabling.

3.6.6.2.2 Capability

The electrical power subsystem shall have the capability of distributing the design power to the orbiter subsystems and scientific instruments as required, including during solar eclipses, converting and regulating the voltage as required, and protecting the subsystem, other subsystems, and the scientific instruments from momentary or prolonged overloads or undervoltage conditions.

3.6.6.2.3 Design Power

The electrical power subsystem shall provide the capabilities required in Section 3.6.3.2.3.

The solar array shall be sized to provide the design power for all nominal spacecraft cruise attitudes as specified in Section 3.6.6.3.2. The battery may be used to supplement the solar array output when off-nominal maneuver attitudes are required. The battery shall be capable of providing the design power during eclipses of up to 0.5 hours duration throughout the nominal mission profile.

The electrical power subsystem shall be capable of providing adequate electrical power so that the spacecraft can be precessed to any nominal maneuver attitude, remain there for 5 hours, and precess back to cruise attitude without exceeding 50 percent depth of discharge on the battery. The electric power subsystem shall be capable of providing adequate electrical power so that the spacecraft can be precessed to the orbit insertion attitude with the sun at 60 ± TBD deg from the spin axis and remain in that attitude for at least 40 hours. The scientific instruments may be operated at reduced power throughout the maneuver sequence in order to meet these requirements.

3.6.6.2.4 Power Bus Voltage Regulation

The electrical power subsystem shall automatically control the power bus voltage to 28 VDC ±2 percent except during power switching transients. Regulation shall be achieved by shunting array power in excess of load requirements or by discharging the battery through a regulator when load power exceeds the array capability. Power for using bus subsystems shall be further conditioned by the electrical power subsystem to provide voltages other than 28 VDC.

3.6.6.2.5 Overload Protection

If at any time the load exceeds the power source capability then a portion of the load shall be disconnected, preferably in the following steps with suitable delays between steps:

a) All scientific instruments

b) All switched loads except the S-band transmitter amplifiers

c) S-band transmitter amplifiers.

The overload shall be sensed by a drop in power bus voltage below 26 VDC ±0.05 percent and shall include delay circuits to prevent inadvertent functioning due to momentary transients. The capability to override the overload protection by ground command shall be provided.

3.6.6.2.6 Fusing of Circuits for Scientific Instrument Power

The orbiter shall provide an individually fused branch circuit for each scientific instrument. The fuses for the scientific instruments shall be located in a module separate from orbiter subsystem fuses and shall be easily accessible for replacement. Fuse sizing shall be in accordance with PV 1006.02 and shall be subject to approval by ARC/PPO.

3.6.6.3 Orbiter Attitude Control

3.6.6.3.1 Description

The attitude control subsystem consists of electronic control assemblies, sensors, thrusters, fuel tanks, fuel lines, heaters (if required), and instruments for measuring the subsystem performance. In addition to performing the functions required for attitude control, this subsystem hardware shall also be used to implement the velocity control capability required in Section 3.6.6.4. The orbiter attitude control subsystem includes the drives and associated electronics for the gimballed ram platform and the gimballed radar altimeter antenna.

3.6.6.3.2 Control Capability

The attitude control subsystem shall be the capability for:

a) Adjusting the initial spin rate after separation from a spin speed of 2 rpm counterclockwise (viewed from in front of the nose of the launch vehicle) to the nominal cruise spin rate of 4.8 rpm.

b) Adjusting intermittently the spin rate by ground command during the nominal flight profile to correct for the effects of thruster misalignment and other sources of disturbance torque. A capability for adjusting the spin speed during the mission by at least 20 rpm shall be provided.

c) Precessing the spin axis intermittently by ground command during the nominal interplanetary flight profile in order to maintain spin axis pointing in the desired attitudes relative to the earth and sun. A capability to precess the spin axis at least 500 degrees shall be provided to meet this requirement. Except during maneuvering periods, the nominal spacecraft attitude during orbital operation shall be with the spin axis normal to the earthline, and normal to the Venus orbit plane as measured in a plane perpendicular to the earthline.

d) Changing the attitude of the spacecraft by ground command to that required to perform the number of midcourse velocity corrections necessary to achieve arrival in the vicinity of Venus at the desired time and at the desired position and velocity. A precessional capability of at least 500 degrees shall be provided to meet this requirement. Capability shall be provided to precess to any inertial attitude with the sun in the forward hemisphere and up to 95 degrees from the forward spin axis.

e) Changing the attitude of the spacecraft by ground command to that required for the orbit insertion velocity adjustment, and increasing the spin speed to 60 ± 1 rpm prior to solid rocket motor firing. A precessional capability of at least 70 deg shall be provided to meet this requirement in addition to the above spin rate changes.

f) Adjusting the spin speed by ground command back to the nominal cruise spin rate after the orbit insertion velocity maneuver and precessing the spacecraft by ground command back to the nominal attitude used for science data acquisition in orbit. A precessional capability of at least 80 degrees and a spin rate change capability of at least 90 rpm shall be provided to meet this requirement.

g) Changing the attitude to that required for orbit adjustment and periapsis maintenance maneuvers as required in Sections 3.6.1.3.5 and 3.6.1.3.6. A precessional capability of at least 400 degrees shall be provided to meet this requirement.

h) Adjusting the spacecraft velocity by firing one or more of the attitude control thrusters.

i) Controlling the angular positions of the ram experiment platform and the radar altimeter antenna relative to the spacecraft main body in order to meet experiment pointing requirements.

3.6.6.3.3 Hardware Design

The orbiter spacecraft shall provide the same capabilities as required in Section 3.6.3.3.3.

3.6.6.3.4 Thrust Control

The orbiter spacecraft shall provide the same capabilities as required in Section 3.6.3.3.4.

3.6.6.3.5 Spin Rate

The orbiter spacecraft shall provide the same capabilities as required in Section 3.6.3.3.5.

3.6.6.3.6 Spin Control

The orbiter spacecraft shall provide the same capabilities as required in Section 3.6.3.3.6.

3.6.6.3.7 Spacecraft Dynamics

The orbiter spacecraft shall meet the requirements of Section 3.6.3.3.7.

3.6.6.3.8 Attitude Control and Stability

The orbiter spacecraft shall meet the requirements of Section 3.6.3.3.8.

3.6.6.3.9 Propulsion

The orbiter spacecraft shall provide the capability required in Section 3.6.3.3.9.

3.6.6.3.10 Spin Axis Attitude Determination for Science Data Reconstruction

3.6.6.3.10.1 Spin Axis Orientation

The orbiter spacecraft shall be designed and balanced so that the true spin axis after any wobble motion has been damped shall be within 1 degree of a spacecraft-fixed reference line throughout the nominal mission profile with the spacecraft in its normal (deployed) orbital configuration. The desired alignment of the scanning scientific instruments shall be maintained or at least calibrated to within ±0.5 degree relative to the spacecraft fixed reference line.

3.6.6.3.10.2 Attitude Determination Accuracy

The orbiter spacecraft shall provide the capability to determine the attitude of the spacecraft-fixed reference line at any time during the orbital portion of the nominal mission profile when the spin axis is nominally perpendicular to the Venus orbit plane except during or within 2 hours after a velocity or attitude adjustment maneuver has been completed. A capability of determining this attitude to within at least 3 degrees (3 σ) shall be provided throughout the nominal mission profile and as a goal, accuracies of 1 degree shall be possible.

3.6.6.3.10.3 Data Processing

Use of ground data processing is encouraged rather than increasing complexity of the on-board spacecraft equipment to meet the attitude determination requirement, but the requirements of Section 3.4.5.1 shall be observed.

3.6.6.3.11 Experiment Gimbals

a) Description. The orbiter attitude control subsystem shall include an electromechanical drive to support and position the ram experiment platform, a second electromechanical drive to support and position the radar altimeter antenna, and electronics to control and supply current to the drive motors as well as to stimulate and readout angular position sensors in the drives.

b) Capability. The ram platform drive shall be capable of withstanding test and launch environments; and of orienting the ram platform as required to maintain the ram instruments' alignment within ±10 degrees of the ram velocity vector once per spacecraft revolution at altitudes less than 500 km during each periapsis pass. The radar altimeter drive shall have the capability of positioning the radar altimeter antenna so that its electrical boresight will sweep through the spacecraft nadir point once per spacecraft revolution at periapsis. Each drive shall provide detent holding torque when not energized so that experiment pointing will not be altered during normal orbital operation except when the drive motors are energized and commanded to change position. Each drive shall include provision to read out its angular position, and the position signals shall be included in the engineering telemetry at all times whether or not the drive motors are energized.

3.6.6.4 Velocity Control

3.6.6.4.1 Description

The hardware used to implement the spacecraft velocity control functions is included as part of the attitude control subsystem as specified in Section 3.6.6.3.

3.6.6.4.2 Capability

The attitude control subsystem shall have the capability for adjusting spacecraft velocity at any time from 5 days after launch until termination of the nominal mission (after 243 days in Venus orbit). Capability shall be provided to make velocity adjustments at least once every 2 days in order to correct for the injection errors induced by the launch vehicle as specified in PV 1006.04 to correct for velocity errors introduced by the spacecraft systems themselves at the time of velocity adjustments, and to provide orbit adjustment and periapsis maintenance capability as required in Sections 3.6.1.3.5 and 3.6.1.3.6.

No spacecraft capability shall be required to correct for errors in tracking data at the time of velocity correction beyond that inherently provided by the attitude control and thruster firing capabilities required in Section 3.6.6.3 and the total velocity change capabilities required in Section 3.6.6.4.3.

3.6.6.4.3 Propellant Capacity

The design shall provide for sufficient propellant capacity to permit the spacecraft velocity to be adjusted by at least 10 meters/sec prior to orbit insertion and to provide the specific velocity adjustment capabilities required in Section 3.6.1.3.5 and in Section 3.6.1.3.6.1.

3.6.6.5 Orbiter Communication Subsystem

3.6.6.5.1 Description

The communications subsystem consists of receivers, transmitter drivers, power amplifiers, switches, filters, diplexers, antennas, coaxial lines, hybrid couplers, and instrumentation to measure performance.

3.6.6.5.2 Compatibility

The communication subsystem shall be compatible with other spacecraft subsystems and the DSN as specified in PV 1006.03.

3.6.6.5.3 Capability

The orbiter spacecraft shall meet the requirements of Section 3.6.3.5.3.

3.6.6.5.4 Subsystem Design

3.6.6.5.4.1 Redundancy

The communication subsystem shall meet the general requirements of Section 3.4.1 and in particular shall be designed with maximum redundancy within the overall constraints of spacecraft and mission profile. As a minimum the subsystem shall have two transmitter drivers, two power amplifiers, and two receivers.

3.6.6.5.4.2 Switching

The orbiter spacecraft shall meet the requirements of Sections 3.6.3.5.4.2 and 3.6.3.5.4.3.

3.6.6.5.5 Receiver and Conscan Processor Design

The orbiter spacecraft shall meet the requirements of Section 3.6.3.5.5. The Conscan processor shall include the capability to measure the amplitude and phase of the signal gano modulation produced when the spacecraft spin axis is pointed slightly away from earth in order to measure the direction of the spin axis pointing relative to the earth.

3.6.6.5.6 Transmitter Design (Transmitter Driver Plus Power Amplifier)

The orbiter spacecraft shall meet the requirements of Section 3.6.3.5.6.

3.6.6.5.7 Orbiter Antennas

3.6.6.5.7.1 Description

The orbiter spacecraft antenna subsystem shall consist of one or more omni directional antennas to give essentially spherical coverage, one directional (high gain) antenna mounted on the forward portion of the spacecraft, and an aft facing medium gain antenna.

3.6.6.5.7.2 Beamwidth

The 3db beamwidth of the high gain antenna shall be at least TBD degrees. The 3 db beamwidth of the medium gain antenna shall be at least TBD degrees. The high gain antenna shall have its primary pattern directed along the forward spacecraft spin axis and shall have the capability on command of squinting the pattern TBD degrees off the spin axis for use with the Conscan processor to make spin axis attitude measurements. An auxiliary feed or antenna may be used to produce a second squinted pattern rather than squinting the main pattern.

3.6.6.5.7.4 Omni Coverage

The omni antenna coverage with the interconnection to the two avail able receivers shall have a compatible pattern with effective gains as follows:

Minimum Effective Gain (dBi)	Earth Aspect Angles (deg)
-2	0 → 180
-1	0 → 160

3.6.6.5.8 Communications Subsystem Performance – Uplink

The uplink system shall be capable of receiving commands for the conditions specified below and for the modulation index as specified in PC 322.00, with a bit error probability no greater than 10^{-5} as measured at the output of the spacecraft command detection equipment.

3.6.6.5.8.1 Powered Flight

There is no requirement for uplink communication capability during powered flight.

3.6.6.5.8.2 Post Launch

The orbiter spacecraft shall be capable of receiving a signal having a total power density at the spacecraft of -45 dbm per square meter when in any attitude relative to the spacecraft/earth line after separation from the launch vehicle.

3.6.6.5.8.2 Cruise and Orbital Operation

During the nominal flight mission, the orbiter spacecraft shall be capable of receiving a signal having the total power density given below for the directions indicated:

Power Density (dbm per square meter)	Earth Aspect (deg)
-108	0 to 180 (omni)
TBD	0 ± TBD (high gain)
TBD	180 ± TBD (medium gain

3.6.6.5.9 Communications Subsystem Performance – Downlink

3.6.6.5.9.1 Powered Flight

TBD

3.6.6.5.9.1 Flight

During the nominal flight mission after separation from the launch vehicle, the spacecraft shall be capable of transmitting not less than the following total EIRP in the direction indicated:

Total EIRP (dbm)	Earth Aspect Angle (deg)
TBD	0 to 180 (first 185 days)
TBD	180 ± TBD (first TBD days)
TBD	0 to 180 (>185 days)
TBD	0 ± TBD (>TBD days)
TBD	180 ± TBD (>TBD days)

3.6.6.5.10 Telemetry

Telemetry data from the spacecraft shall contain information about:

a) The status of the communication subsystem such as switch position and signal present in either receiver.

b) Performance of the communication subsystem such as temperatures of critical components; receiver signal strength and static phase error; power amplifier voltages, currents and output power; and pointing control equipment voltages, currents, and error signals.

3.6.6.6 Orbiter Data Subsystem

3.6.6.6.1 Description

The orbiter data subsystem consists of a digital telemetry unit (DTU) and three data storage units (DSU).

3.6.6.6.2 Capability

The data subsystem shall provide the capability for:

a) Supplying, as required, timing and operational signals to the scientific and engineering instruments on the spacecraft.

b) Sampling these instruments.

c) Conditioning and quantizing the data and arranging them into formats suitable for processing immediately, or for storage and later processing.

d) Coding the data and processing them into a form suitable for modulating the downlink rf carrier.

e) Buffering high speed data bursts from scientific instruments and storing this data for subsequent telemetry.

3.6.6.6.3 DTU Design

The orbiter spacecraft shall provide the same capabilities as required in Section 3.6.3.6.3.

3.6.6.6.4 DSU Design

Each DSU shall consist of two separate memories, each with its own input/output. These modules shall be configured such that the DSU can operate as two 122,880 bit memories or as one 245,760 bit memory; the configuration shall be selectable by command. The DSU shall accept input bit rates up to 10 kbps. It shall be possible to store data in the DSU up to its capacity during a single time interval or during multiple time intervals that are neither contiguous nor separated by the same amount each time before reading out any of the stored data. When data is accepted from the DTU, it shall be formatted but uncoded. When data is accepted from an instrument, it may be unformatted. It shall be possible by ground command to terminate the storage mode at any time before filling the memory to its capacity without destroying the stored data. Termination of the storage mode at the end of each of the noncontiguous intervals can be either by ground command or automatically when the storage capacity for the modual is reached. The memory readout shall be nondestructive and it shall be possible to terminate readout from the storage at any time before the entire memory is clear without destroying or preventing later recovery of the remaining data. The DSU shall issue a discrete signal (flag) when the DSU (or module) reaches its full capacity. Flags shall also be issued by each module at 7,680 bits and 115,200 bits.

3.6.6.6.5 Other

All other aspects of the data subsystem shall be identical to those of the probe bus. In particular the orbiter spacecraft shall have the same capabilities as required in Sections 3.6.3.6.4 to 3.6.3.6.8.

3.6.6.7 Command Subsystem

3.6.6.7.1 Description

The command subsystem consists of two decoders (DDU) and a command distribution unit (CDU).

3.6.6.7.2 Capability and Design

With the exception of the requirements of Section 3.6.3.7.3.4, all aspects of the command subsystem and equipments shall be identical to those of the probe bus as required in Sections 3.6.3.7.2 and 3.6.3.7.3.

3.6.6.7.3 Special Command Storage Mode for SRM Firing

In addition to the command storage mode delineated in paragraph 3.6.3.7.3.4, a special mode shall be provided for the orbiter spacecraft. In this mode, the 6 stored commands shall be configured such that no single failure will cause the omission, premature, or late execution of a single event (SRM firing).

3.6.6.8 Thermal Control Subsystem

3.6.6.8.1 Description

The orbiter spacecraft thermal control subsystem consists of insulation to provide passive thermal control for spacecraft components and louvers to provide active thermal control of the equipment compartment.

3.6.6.8.2 Capability

The thermal control subsystem shall have the capability of maintaining spacecraft components at the temperatures required for normal operation, the scientific instruments at temperatures as specified herein for the thermal environments as specified, and meeting the probe thermal interface requirements.

3.6.6.8.3 Design Environments

The thermal control subsystem shall be designed to maintain the spacecraft and instrument temperatures at the required levels with negligible degradation of materials and components when exposed to the following environments:

a) Prelaunch: Within the environment inside the fairing achieved by the on-stand air conditioning as specified in PV 1006.04.

b) Powered Flight:
   1) Within the environment inside the fairing resulting from heating of the fairing as specified in PV 1006.04
   2) Within the environment encountered after fairing jettison for the powered flight profile specified in PV 1006.04, including the effects of aerodynamic heating, eclipse, or radiation from the sun and earth.

c) Flight: Within the thermal environment imposed during the nominal mission profile, including direct radiation from the sun on the sides of the spinning spacecraft for an unlimited time and on the aft side of the spacecraft for at least two hours.

3.6.6.8.4 Design

3.6.6.8.4.1 Equipment Compartment

The probe bus shall be designed so that throughout the mission until the end of the nominal flight mission:

a) Temperatures within the vicinity of spacecraft equipment will be maintained as necessary.

b) Temperatures within the equipment compartment in the vicinity of each scientific instrument specified in PC 326.00 will be maintained between 0 and 90°F when the instrument is powered and its net heat loss or gain is within the limits specified in Section 3.5.4.

3.6.6.8.4.2 Instrument Appendages

Appendages or external structure that support scientific instruments shall be designed to minimize the heat flow into or away from the instrument.

3.6.6.8.4.3 Instrument Apertures

Insulation shall be provided to fill any gaps between scientific instrument apertures and the spacecraft exterior insulation.

3.6.6.8.4.4 External Instruments

Scientific instruments mounted external to the equipment compartment will provide their own thermal control.

3.6.6.8.5 Powered Flight Constraints

The spacecraft shall impose no requirements on the orientation of the launch vehicle/spacecraft combination during powered flight up to and including separation to satisfy thermal limitations of spacecraft materials and components.

3.6.6.8.6 Telemetry Data

Telemetry data from the spacecraft shall include, in addition to the temperatures of other subsystem units as specified elsewhere, the temperature at several locations in the equipment compartment and if possible, the angle or full open/full close status of at least one of the louvers.

3.6.6.9 Allocated Weight and Reliability Requirements

The allocated weight and reliability requirements as specified in Sections 3.4.9.2.1 and 3.4.1.2 shall be observed.

3.6.6.10 Orbit Insertion Solid Rocket Motor

3.6.6.10.1 Description

The orbit insertion solid rocket motor (SRM) includes redundant initiators, an integral combustion chamber and nozzle assembly, solid propellant, protective insulation, and a protective nozzle plug which is expelled on ignition.

3.6.6.10.2 Capability

The SRM shall be capable of surviving launch and the interplanetary cruise environment and providing sufficient velocity adjustment capability during a single burn to place the orbiter spacecraft in a nominal 24-hour orbit.

3.6.6.10.3 Impulse Requirements

The SRM shall produce a total impulse of 50, 550 ± 500 lb sec. The SRM shall be designed so that at least 10% of the propellant required above can be offloaded at any time within TBD months of launch without requiring requalification, and with the resulting total impulse maintained within a ±1% tolerance.

3.6.6.10.4 Thrust

The maximum thrust shall be less than 6400 lb_f.

3.6.6.10.5 Life

The SRM shall be compatible with ground storage of at least TBD months and shall operate properly after launch and exposure to the interplanetary environment during cruise as specified in Section 3.6.1.

3.6.6.10.6 Thermal Environment

The SRM shall be compatible with storage and cruise temperatures between 30 and 100°F. The SRM shall ignite reliably and meet the impulse requirements of Section 3.6.6.10.3 if the SRM temperature is between 30 and 100°F just prior to ignition.

3.6.6.10.6 Igniters

The SRM shall have at least two redundant electrically isolated initiators. Any initiator shall be capable of reliably starting the SRM and shall be compatible with the electric power subsystem pyro firing circuits and the safety requirements of Section 3.6.6.10.7.

3.6.6.10.7 Safety

The SRM shall meet the requirements of TBD and TBD (range safety and transportation).

3.6.6.10.8 Outgassing and Products of Combustion

The SRM shall be designed so as to minimize contaminants which can flow to the exposed surfaces of the orbiter and its scientific instruments. In particular the nozzle and other external surfaces of SRM shall be enclosed or shielded by inert materials so that substances which may outgas during SRM firing due to elevated surface temperatures will be condensed or reflected away before reaching other portions of the spacecraft.

4. PRODUCT ASSURANCE PROVISIONS

TBD

PRELIMINARY

PIONEER VENUS

SCIENTIFIC INSTRUMENT - PROBE AND SPACECRAFT INTERFACE DOCUMENT

PIONEER VENUS SCIENTIFIC INSTRUMENT - PROBE AND SPACECRAFT INTERFACE DOCUMENT

OUTLINE

Page

PART I

OUTLINE (Continued)

OUTLINE (Continued)

OUTLINE (Continued)

OUTLINE (Continued)

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OUTLINE (Continued)

PART I

PRELIMINARY

PIONEER VENUS

SCIENTIFIC INSTRUMENT-SPACECRAFT (PROBE BUS AND ORBITER) INTERFACE DOCUMENT

SCOPE:

This document defines the characteristics of the Pioneer Venus Probe Bus and Orbiter spacecraft pertinent to the scientific instruments and the requirements of the spacecraft on the scientific instruments.

1. MECHANICAL

1.1 SIZE

1.1.1 Platform-Mounted Instruments

The dimensions of the instruments shall be compatible with the overall area and balance requirements of the equipment platform.

1.2 MASS PROPERTIES

1.2.1 Weight

The total weight of the scientific instruments, including the cabling between boxes, connectors, containers, and boom-mounted equipment, shall not exceed the spacecraft capabilities specified in PC-(TBD), section (TBD).

1.3 MOUNTING

1.3.1 Internally-Mounted Instruments

Internally mounted instruments will be secured to mounting platform with 6-32 UNC screws. Mounting hole centerlines will be located a minimum of 16 mm from the edge of the equipment platform. One of three mounting methods shall be used for securing instruments to the equipment platform.

a) 10 x 10 x 3 mm tabs without gussets.

b) 10 x 10 x 3 mm tabs with gussets (if this method is used, tab width shall be increased by 2 times the gusset thickness).

The attachment hole size in the instrument mounting tab, or through-hole within the instrument, shall be 0.45 cm in diameter and spot faced 10 mm diameter for No. 6 flat washer. The spot face edge radius shall be 0.2 ± 0.2 mm. Mounting hole spacing shall be held to within ±0.1 mm (true position tolerance of ±0.3 mm). Each scientific instrument shall be mounted to the spacecraft at a minimum of four locations. The loads at each mounting location shall be limited to 50 kg in tension and 50 kg in shear.

1.3.2 Externally-Mounted Instruments

The radar altimeter on the deployment mechanism shall be designed to mate with the attach fitting shown in Figure 1-1. The magnetometer shall be designed to be mounted to the flange of the deployable boom. (Details TBD).

1.4 EQUIPMENT ARRANGEMENT AND ORIENTATION

The arrangement and orientation of the instruments on the mounting platform is shown in Figures 1-2 and 1-3.

1.5 VIEWING

The equipment compartment will consist of a platform constructed of aluminum honeycomb with aluminum face sheets; its lateral and forward surfaces will consist of aluminized mylar or aluminized kapton thermal blankets. Openings in the compartment surfaces required by instruments within the compartment will be mainly in the lateral surfaces but may be accommodated in the forward or aft surfaces of the compartment. Dimensions of the compartment and the thermal blankets are shown in Figure 1-4. Within these limits, openings required by the instruments may have any shape which does not compromise the spacecraft structural or thermal integrity.

1.6 MISCELLANEOUS

1.6.1 Joint Between Spacecraft Thermal Blanket and Instruments

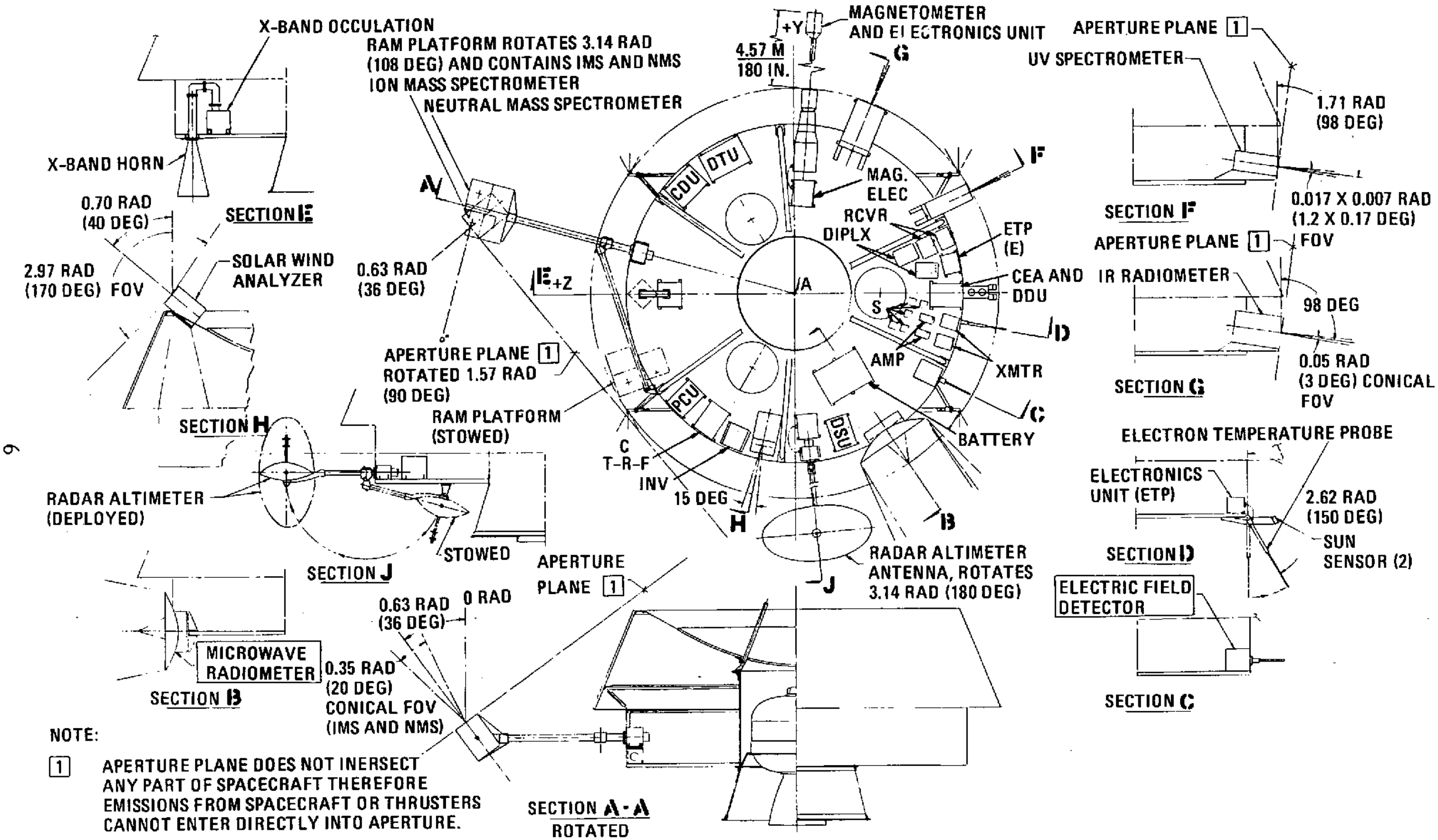
The joint between the thermal blanket and the projection through the spacecraft wall of a viewing instrument will be closed with a thermal shield provided by the spacecraft contractor. The instrument aperture shall extend at least 6 mm beyond the spacecraft thermal blanket. Such projecting surfaces shall have a low emittance finish.

1.6.2 Deployment Mechanism Impact Load

The impact load due to deployment of magnetometer and radar altimeter during release and deployment will not exceed 4-1/2 g's at the instrument.

To Be Supplied

Figure 1-1. Radar Altimeter Attach Fitting

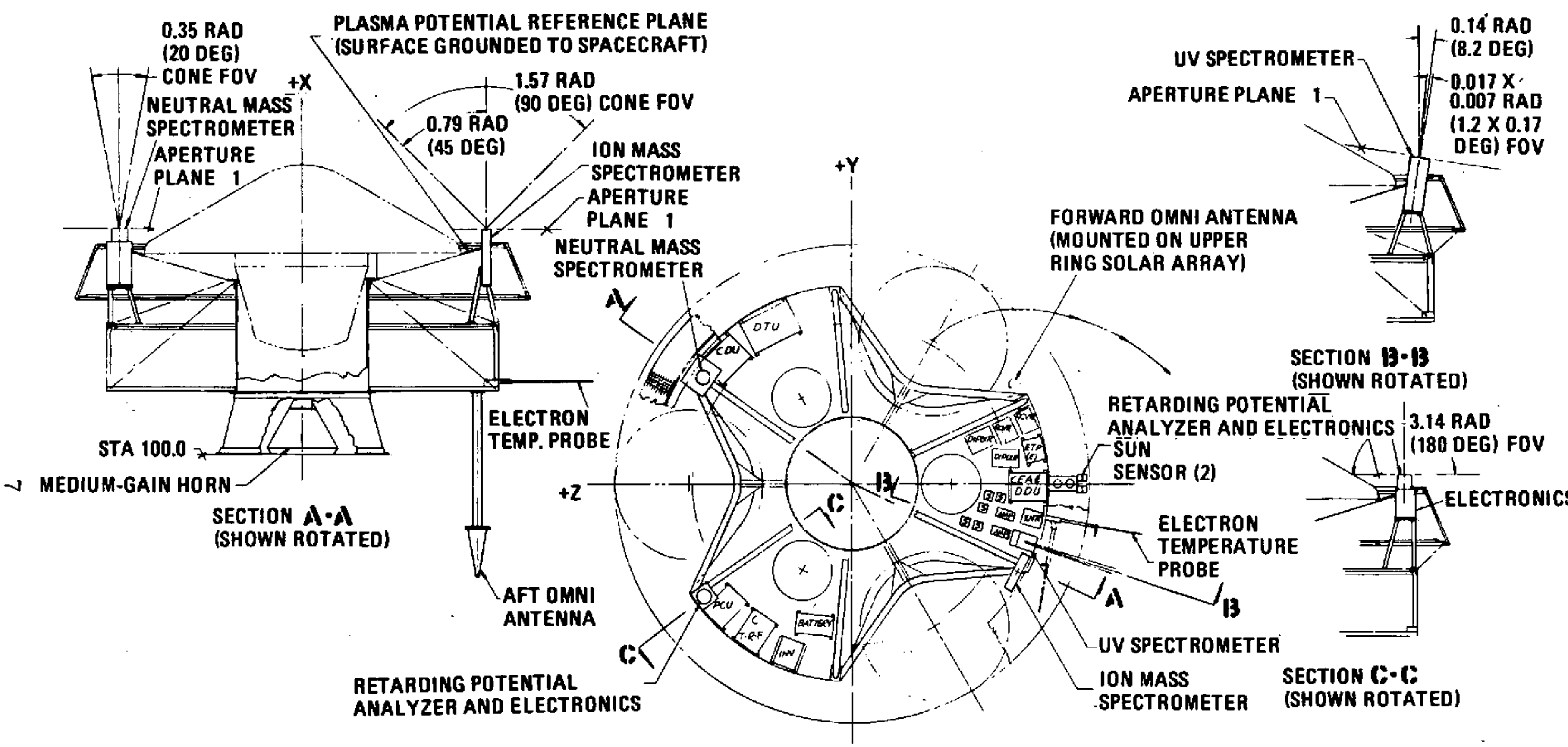


NOTE:

1. APERTURE PLANE DOES NOT INERSECT ANY PART OF SPACECRAFT THEREFORE EMISSIONS FROM SPACECRAFT OR THRUSTERS CANNOT ENTER DIRECTLY INTO APERTURE.

2. ☐ DENOTES OTHER CANDIDATE INSTRUMENTS.

Figure 1-2. Instrument Mounting for Orbiter



NOTES:

1 APERTURE PLANE DOES NOT INTERSECT ANY PART OF SPACECRAFT BUS THEREFORE EMISSIONS FROM SPACECRAFT OR THRUSTERS CANNOT ENTER DIRECTLY INTO APERTURE.

VERSION IV - NOMINAL SCIENCE PAYLOAD AND EQUIPMENT LAYOUT PROBE MISSION SPACECRAFT ATLAS/CENTAUR

Figure 1-3. Instrument Mounting for Probe Bus

To Be Supplied

Figure 1-4. Equipment Compartment and Thermal Blanket Dimensions

2. ELECTRICAL POWER

2.1 HARDWARE

The electrical power subsystem consists of the following spacecraft system assemblies:

a) One inverter assembly (IA)

b) One power control unit (PCU)

c) One shunt radiator (SR)

d) One battery

e) One central transformer-rectifier-filter (CTRF)

f) One solar array

g) Two shunt element assemblies (Orbiter only).

2.2 POWER SOURCES

2.2.1 Ground Power Console

Ground power will be used as required throughout the prelaunch activities.

2.2.2 Solar Array Simulator

A solar array simulator will be used to power the spacecraft for most of the spacecraft system tests. The simulator current and voltage shall be adjustable to simulate temperature intensity and solar aspect effects on the array.

2.2.3 Solar Array

The solar array will have a power output sufficient to supply load and battery charging requirements for all mission conditions as defined in (TBD).

2.3 CONTROL

2.3.1 Voltage

The output voltage is regulated by dumping excess power in the SR, through a dissipative shunt in the PCU. An inverter operating from the 28-volt bus provides a 61.0-volt RMS squarewave to the input of the CTRF. The battery is discharged to the bus through a regulator whenever load power exceeds array capability.

2.3.2 Battery

The spacecraft battery supplies power as required whenever the load exceeds solar array capability and during periods of solar occultation. The battery is recharged at a controlled rate by a charger in the PCU.

2.3.3 Overload Protection

If the total load exceeds the capability of the electrical power subsystem, then loads will be dropped in an automatic sequence. The first group of loads turned off will include all the scientific instruments.

2.4 OUTPUT CHARACTERISTICS

2.4.1 Distribution

Each scientific instrument will receive electrical power through an individual, fused, branch circuit. The branch circuit will normally remain energized. The power allotted to the instrument is measured at the spacecraft/instrument interface connector.

2.4.2 Voltage

Measured in PCU (ahead of fuse).

2.4.2.1 Steady State

Regulated to +28 VDC

a) Short term error: ± 1-percent

b) Long term drift (superimposed): ± 1-percent

NOTE: Voltage at the terminals of each scientific instrument will be lower by the voltage drop due to the fuse and wire resistance of its branch circuit.

2.4.2.2 Noise

Except for the transient voltage excursions specified in Section 2.4.2.3, the peak-to-peak amplitude of any voltage excursion, periodic or aperiodic, will not exceed 560 millivolts at any frequency between 30 Hz and 10.0 kHz decreasing at a 20 dB/decade rate to 280 millivolts at 20.0 kHz and not exceeding 280 millivolts through 100 MHz.

2.4.2.3 Transient Voltage Excursions

a) Performance - No degradation of performance will result when voltage transients having the characteristics shown in Figure 2.4.2.3, sheet 1, are seen on the nominal 28 VDC bus.

b) Damage - No instrument damage, long term degradation, or modes where proper performance is not automatically resumed when the transient is removed, will occur when voltage transient having the characteristics shown in Figure 2.4.2.3, sheet 2, are seen on the nominal 28 VDC bus.

2.4.3 Source Impedance

The source impedance is shown in Figure 2.4.3.

2.5 REQUIRED LOAD CHARACTERISTICS

2.5.1 Switch

Each scientific instrument shall contain a DC to DC converter and shall provide for maintaining its own electrical load ON or OFF operation as controlled by a state signal which is supplied by the spacecraft and controlled by ground command. The state signal is described in Section 3.3.6. The idling power drawn by an instrument during the OFF state shall not exceed 5 milliwatts. As a design goal, the instrument's power control circuitry shall be configured so that no single part failure within the instrument will prevent turn-off whenever the state signal is commanded to the OFF state.

2.5.2 Maximum Load

2.5.2.1 Average

Power: TBD watts total for scientific instruments.

2.5.2.2 Peak

The peak scientific instrument load shall not exceed TBD watts.

2.5.3 Inrush Current

The duration of instantaneous load current of each instrument is a function of fuse size and instrument load characteristics. Upon application of the 28 VDC primary power to the instrument's primary power circuit, and upon application of the power control state signal to the instrument's secondary converter control circuit, the instrument's load current surge shall not exceed the following limit envelope:

TBD

2.5.3.1 Fuse Sizes

The fuse size shall be as follows:

TBD

2.5.4 Load Noise

When operating from a power source impedance of TBD, no single instrument shall feed back into the power input circuit any electrical noise either periodic or aperiodic in excess of 280 millivolts peak-to-peak at any frequency between 30 Hz and 10.0 kHz decreasing at a 20 dB/decade rate to 140 millivolts peak-to-peak at 20.0 kHz and remaining at 140 millivolts peak-to-peak or less from 20.0 kHz through 100 MHz.

2.5.5 Ground

The power supply (front-end) circuit shall be DC-isolated from chassis ground within each instrument prior to electrical connection. (See Section 3.5.3 for general grounding concept.)

2.5.6 Capacitance

TBD

2.6 CONNECTORS AND CABLING

2.6.1 Connector Types

The connector installed on an instrument that connects the instrument to the spacecraft harness shall be a male (straight or coaxial insert) pin connector selected from the Cannon, Golden D, Mark I, nonmagnetic with a NMC-A-106 suffix, series of connectors. The use of two identical connectors on any one box is prohibited. The end pins shall not be used.

2.6.2 Interface Connector Location

The centerline of the interface connector shall be located 3.75 cm (1.5 inches) above the base of the instrument and parallel thereto. The shorter dimension of the connector shall be towards the base.

2.6.3 Interface Connector Pins

The mating surfaces shall be equivalent electrically to bare metal. Pins for the power and signal lines shall be adjacent to their corresponding return lines to facilitate the twisting of the wires in the spacecraft harness. The pin assignments shall be in accordance with Section TBD of Specifications TBD.

2.6.4 Test Connector Location

The location for the test connector shall be on the top surface of the instrument, unless otherwise directed by the ARC/PPO.

2.6.5 Connector Encapsulation

When encapsulated during fabrication, the encapsulation of each backshell shall be done with mated connectors to insure proper pin alignment. ARC/PPO shall be notified of the use of any encapsulated interface connectors.

2.6.6 Connector Identification

Each connector shall be identified by TRW with a number in accordance with Section TBD of Specification TBD.

TBD

Experiments consisting of more than one unit will be identified by assigning a basic reference designator for the main unit and an alphabetic subunit identifier for each additional unit required.

TBD

2.6.7 Wiring

The type of wire between any two units of one instrument shall be unshielded, shielded, twisted pairs or coaxial cables (if required for signal leads). Between the spacecraft and the instrument, or between any two units of one instrument, the maximum wire size will be 20 AWG and the minimum size will be 28 AWG.

3. DATA HANDLING AND SIGNALS

3.1 FUNCTIONAL DESCRIPTION

The spacecraft will accept, from the scientific instruments, information in digital, analog, or state form, convert the analog and state information to digital form, and arrange all information in an appropriate format for the time multiplexed transmission to earth or storage on board the spacecraft. The spacecraft will supply buffer storage for selected digital science data on the orbiter. Storage for periods of occultation is also supplied on the orbiter. There is no storage on the bus. The spacecraft will also supply the instruments with various timing and spacecraft operational status signals as well as functional commands.

3.1.1 Telemetry Word

A telemetry word in the main science formats will consist of 3, 7, 8, or 10 bits, selectable by programming in the digital telemetry unit (DTU). The spacecraft engineering subcommutated words are 6 bits and the science instrument engineering subcommutated words is 10 bits. Spacecraft generated words will be transmitted most significant bit first.

3.1.2 Data Bit Rates

The data subsystem will be capable of processing scientific and engineering data at the following rates:

a) 1024 bits per second

b) 512 bits per second

c) 256 bits per second

d) 128 bits per second

e) 64 bits per second

f) 32 bits per second

g) 16 bits per second

h) 8 bits per second

Bit rate changes will occur within one DTU bit period following the reception of a bit rate command by the data subsystem.

3.1.3 Frame

The data subsystem will assemble the information from the instruments into mainframes composed of a series of 768 bits.

3.1.4 Format and Word Assignments

The words in a frame are assigned in several formats. The formats are organized for specific purposes and are selected by ground command for particular spacecraft operational modes. Format changes will occur within one DTU bit period following the reception of a format command by the data subsystem.

3.1.4.1 Format A-1, A-2, A-3, and A-4

Format A-n is the format used for the orbiter scientific information. Format A-1 is shown in Figure 3-1, Format A-2 in Figure 3-2, Format A-3 in Figure 3-3, and Format A-4 in Figure 3-4.

3.1.4.2 Format B-1 and B-2

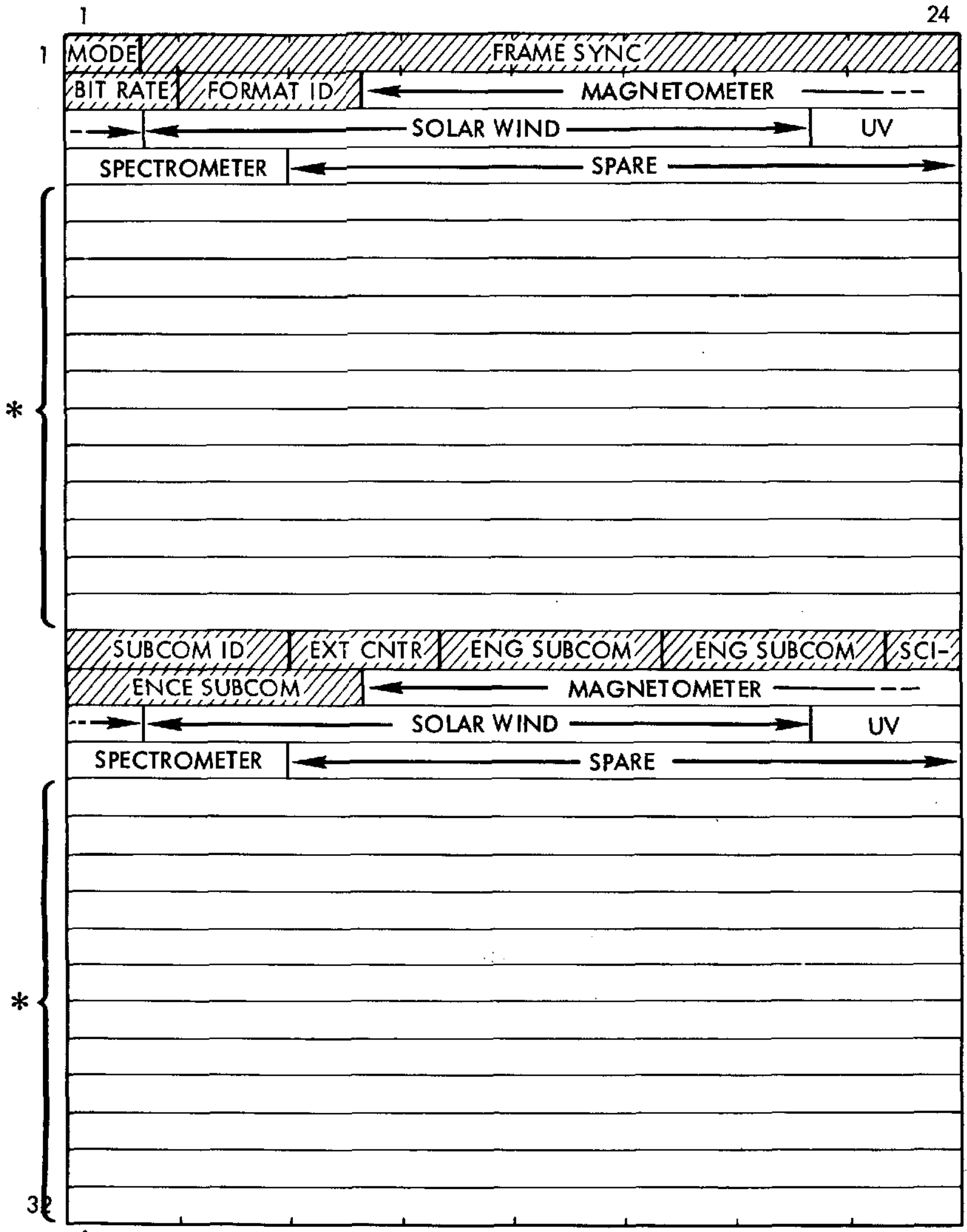
Format B-n is the format used for the probe bus scientific information near Venus. Cruise data for the probe bus will be engineering data only in the C Formats. Word assignments for Format B-1 is shown in Figure 3-5, and for B-2 in Figure 3-6.

3.1.4.3 Formats C-1, C-2, C-3, and C-4

Formats C-1, C-2, C-3, and C-4 contain spacecraft engineering information. Each of the engineering formats may be telemetered as the main frame. However, usually it will be subcommutated in two words in each of the science formats, Format A and B. When one of the engineering formats is commanded to be telemetered as the main frame, engineering words 1 through 4 and 17 through 20 are replaced with the identification, frame sync, and subcom words of the main frame. However, words 1 through 4 and 17 through 20 will be telemetered in the engineering subcom words of the main frame. Instrument power on/off indicators will be telemetered in Format C-1. Word assignments for Formats C-1, C-2, C-3, and C-4 are shown in Table 3-1 for the orbiter and Table 3-2 for the probe bus.

	1				24
1	MODE	FRAME SYNC			
	BIT RATE	FORMAT ID	SPARE		
	MAGNETOMETER				
	SOLAR WIND				
	SPARE				
	SPARE				
	MAGNETOMETER				
	SOLAR WIND				
	SPARE				
	SPARE				
	MAGNETOMETER				
	SOLAR WIND				
	SPARE				
	SPARE				
	MAGNETOMETER				
	SOLAR WIND				
	SUBCOM ID	EXT CNTR	ENG SUBCOM	ENG SUBCOM	SCI-
	ENCE SUBCOM	ANALOG	ANALOG		
	MAGNETOMETER				
	SOLAR WIND				
	SPARE				
	SPARE				
	MAGNETOMETER				
	SOLAR WIND				
	SPARE				
	SPARE				
	MAGNETOMETER				
	SOLAR WIND				
	SPARE				
	SPARE				
	MAGNETOMETER				
32	SOLAR WIND				

Figure 3-1. Word Assignments for Format A-1, Orbiter Cruise (16 bits/s)



*CAN BE USED FOR INTERLEAVING MEMORY DUMP VIA D FORMATS.

Figure 3-2. Word Assignments for Format A-2, Orbiter High Altitude (64 bits/s)

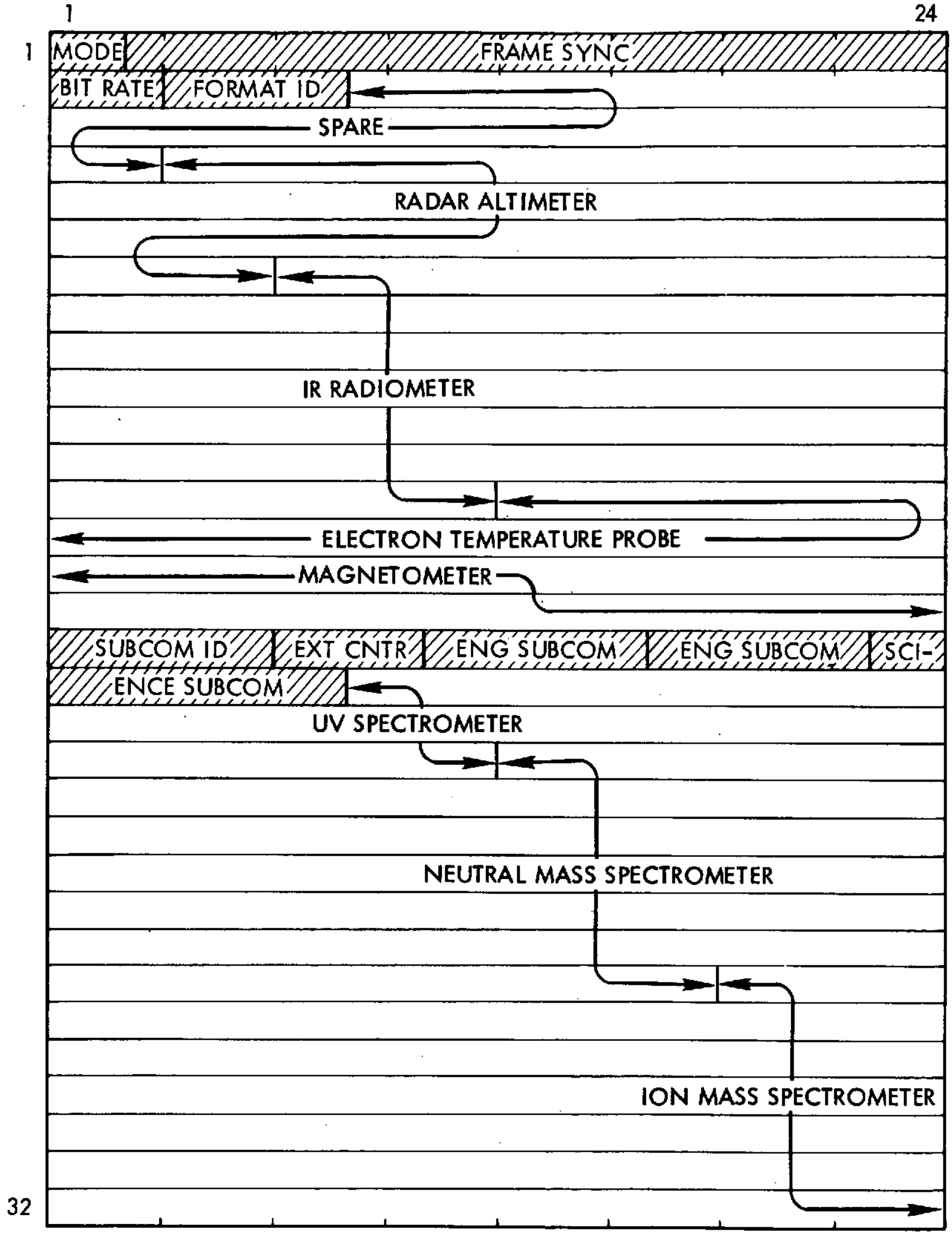


Figure 3-3. Word Assignments for Format A-3, Orbiter Low Altitude (512 bits/s)

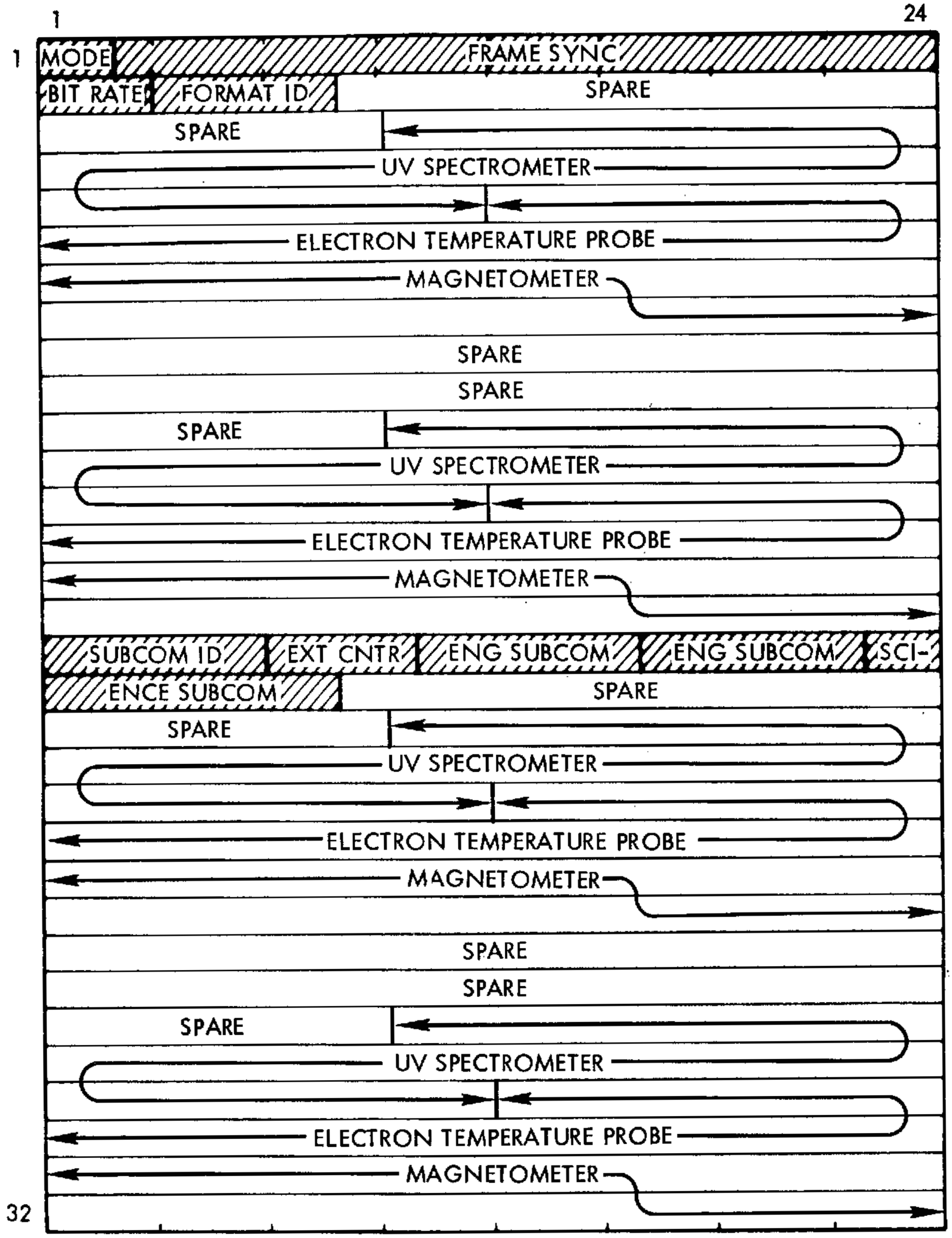


Figure 3-4. Word Assignments for Format A-4, Orbiter Low Altitude Partial (128 bits/s)

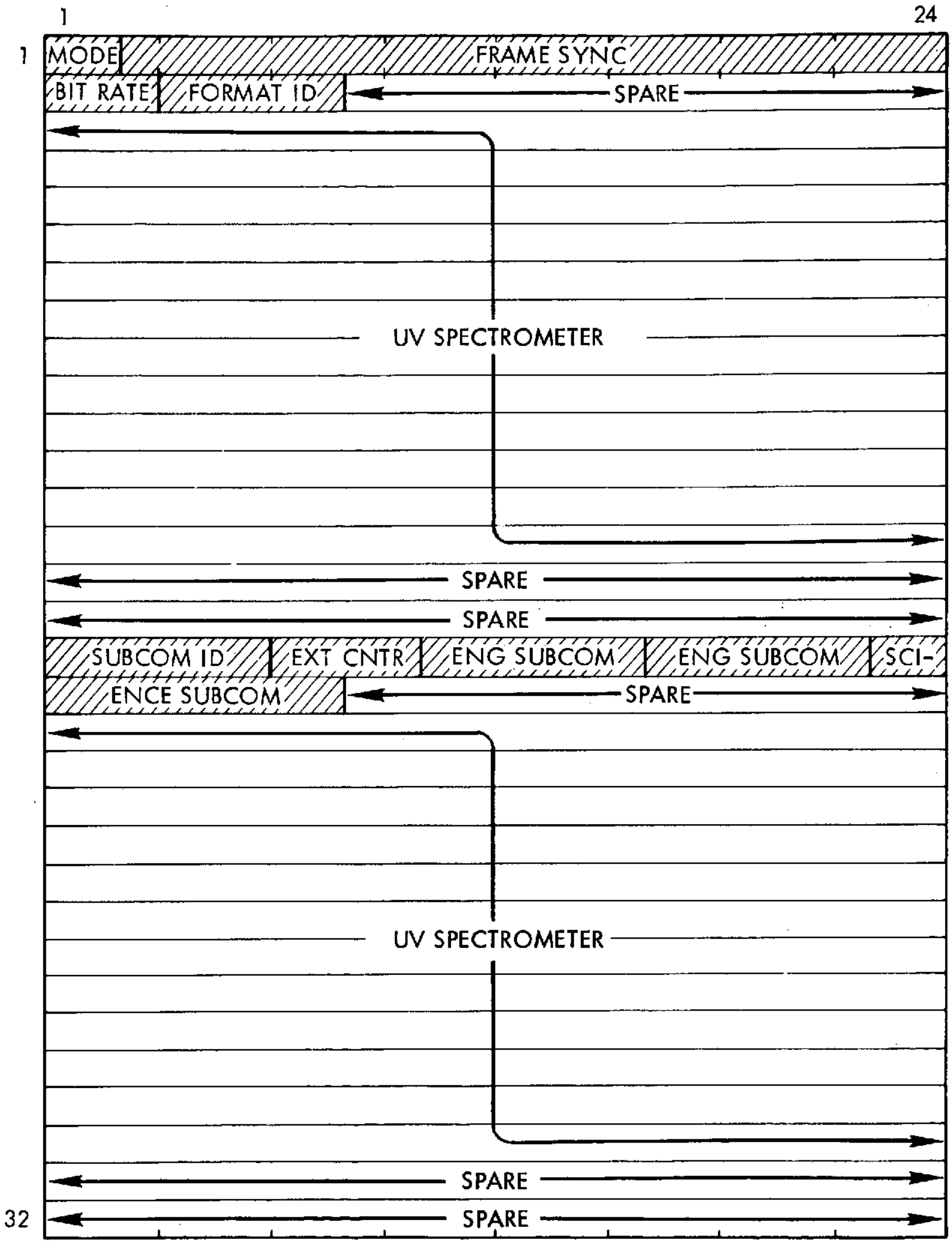


Figure 3-5. Word Assignments for Format B-1, Bus Four Days Before Entry (16 bits/s)

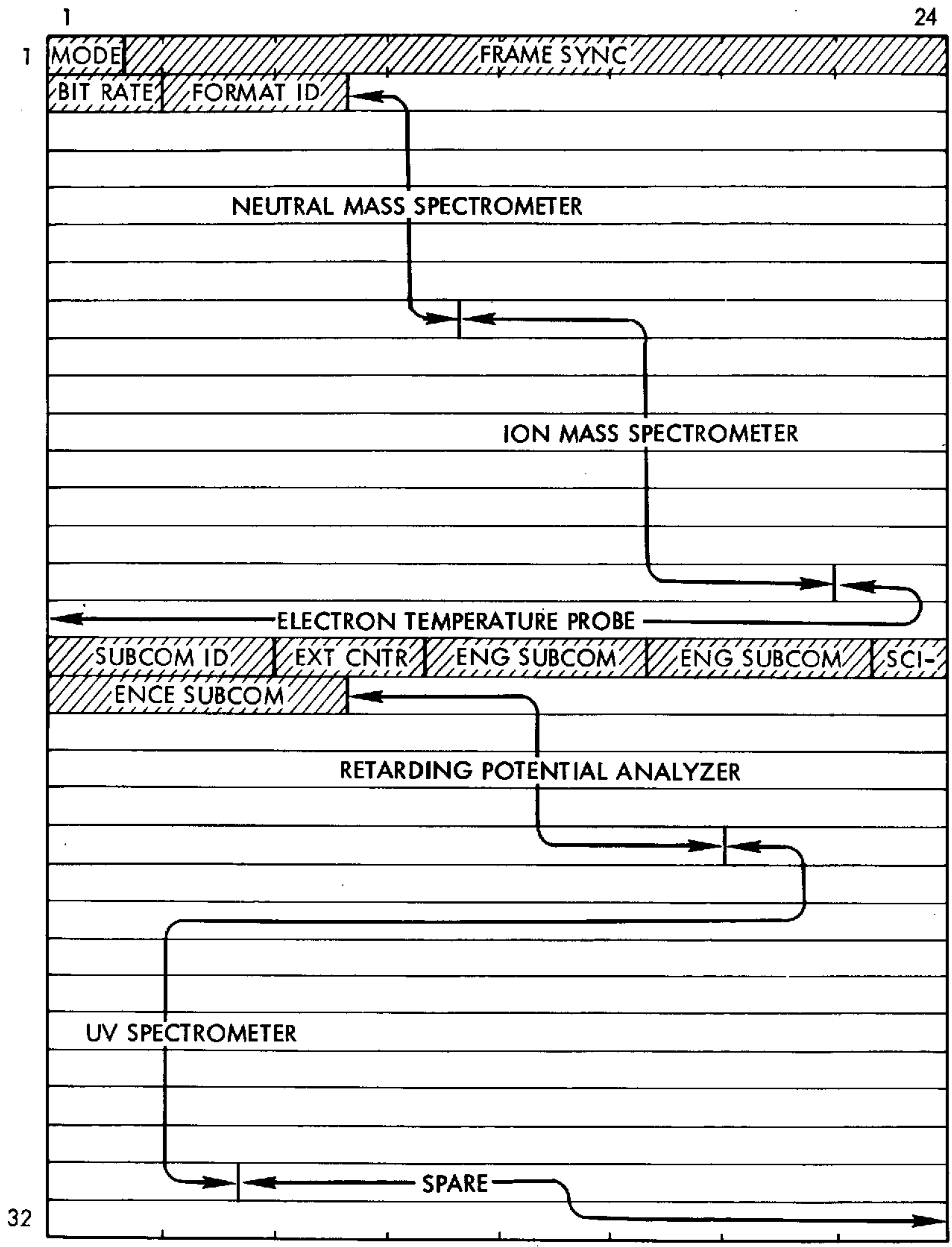


Figure 3-6. Word Assignments for Format B-2,
Bus Entry (1024 bits/s)

Table 3-1. Orbiter Engineering Telemetry List for C Formats

ELECTRICAL DISTRIBUTION

SPACECRAFT SEPARATION STATUS		BILEVEL	
COMMAND EXECUTE STATUS		BILEVEL	
SPACECRAFT ORDNANCE SAFE/ARM RELAY STATUS (PRIME)		BILEVEL	
SPACECRAFT ORDNANCE SAFE/ARM RELAY STATUS (REDNT)		BILEVEL	
SRM ORDNANCE SAFE/ARM MOTOR STATUS		BILEVEL	
OVERVOLTAGE OVERRIDE STATUS		BILEVEL	
RECEIVER REVERSE STATUS		BILEVEL	
EQUIP CONV FAULT ISOLATION RELAY STATUS	(4)	BILEVEL	(THOR/DELTA)
MAGNETOMETER BOOM RETRACTED STATUS		BILEVEL	
MAGNETOMETER BOOM EXTENSION STATUS		BILEVEL	
ELECTRON TEMPERATURE PROBE ANT RELEASE STATUS		BILEVEL	
NEUTRAL MASS SPECTROMETER ION CAP EJECT STATUS		BILEVEL	
UV SPECTROMETER SUN COVER EJECT STATUS		BILEVEL	
RADAR ALTIMETER ANTENNA RELEASE		BILEVEL	
CDU +5 V BUS A SELECT STATUS		BILEVEL	
CDU +5 V BUS B SELECT STATUS		BILEVEL	
RAM PLATFORM RELEASE STATUS		BILEVEL	
CAPACITOR CHARGE STATUS (PRIMARY)	(2)	BILEVEL	
CAPACITOR CHARGE STATUS (REDUNDANT)	(2)	BILEVEL	
COMMAND MEMORY ID TO DTU		BILEVEL	
COMMAND MEMORY ENABLE/DISABLE STATUS		BILEVEL	
COMMAND MEMORY 1 CONTENTS - CMD 8 BITS	(16)	DIGITAL	
COMMAND MEMORY 2 CONTENTS - TIME 8 BITS	(16)	DIGITAL	
COMMAND MEMORY 3 CONTENTS - TIME 8 BITS	(16)	DIGITAL	
COMMAND MEMORY 4 CONTENTS - ROUTING 3 BITS	(16)	DIGITAL	

ELECTRICAL POWER

BATTERY AUTOMATIC CHARGE MODE STATUS		BILEVEL	
BATTERY MAX CHARGE MODE STATUS		BILEVEL	
BATTERY TRICKLE CHARGE STATUS		BILEVEL	
BATTERY DISCHARGE ENABLE/DISABLE STATUS		BILEVEL	
BATTERY HIGH TEMP PROTECTION STATUS		BILEVEL	
BATTERY RECONDITION/DISCONNECT STATUS		BILEVEL	
CTRF INVERTER TRANSFER RELAY STATUS		BILEVEL	
BATTERY CHARGE CURRENT		ANALOG	
BATTERY DISCHARGE CURRENT		ANALOG	
BATTERY VOLTAGE		ANALOG	
BATTERY TEMPERATURE		ANALOG	
DC BUS VOLTAGE		ANALOG	
DC BUS VOLTAGE EXPANDED		ANALOG	
DC BUS CURRENT		ANALOG	
SHUNT BUS CURRENT		ANALOG	
TRF +5 VDC CDU OUTPUT CHANNEL A		ANALOG	
TRF +5 VDC CDU OUTPUT CHANNEL B		ANALOG	
EQUIPMENT CONVERTER TEMP		ANALOG	(THOR/DELTA)
CTRF INVERTER TEMP		ANALOG	

DATA HANDLING

CONVOLUTION CODE GEN STATUS		BILEVEL
ROLL REFERENCE		BILEVEL
SPIN AVERAGING MODE		BILEVEL
HIGH/LOW ALTITUDE STORE FORMAT STATUS		BILEVEL
DSU CONFIGURATION SELECT STATUS	(4)	BILEVEL
SCIENCE DATA STORAGE ENABLE	(2)	BILEVEL
ACS OPERATION MODE		BILEVEL
DECODER A STATUS		BILEVEL
DECODER B STATUS		BILEVEL

Table 3-1. Orbiter Engineering Telemetry List for C Formats (Continued)

AD CALIB VOLTAGE, LOW		ANALOG	
A/D CALIB VOLTAGE, MED		ANALOG	
A/D CALIB VOLTAGE, HIGH		ANALOG	
ROLL ALTITUDE WORD LINES	(2)	DIGITAL	
EXTENDED S.C. ID		DIGITAL	
SPIN PERIOD WORD LINES	(3)	DIGITAL	
COMMUNICATIONS			
RECEIVER A SIGNAL PRESENT STATUS		BILEVEL	
RECEIVER B SIGNAL PRESENT STATUS		BILEVEL	
RECEIVER A COHERENT MODE STATUS		BILEVEL	
RECEIVER B COHERENT MODE STATUS		BILEVEL	
POWER AMP A ON/OFF STATUS		BILEVEL	
POWER AMP B ON/OFF STATUS		BILEVEL	
POWER AMP C ON/OFF STATUS		BILEVEL	(THOR/DELTA)
POWER AMP D ON/OFF STATUS		BILEVEL	(THOR/DELTA)
RECEIVER A OSC ENABLE/DISABLE STATUS		BILEVEL	
RECEIVER B OSC ENABLE/DISABLE STATUS		BILEVEL	
TRANSFER SWITCH 1 POSITION STATUS		BILEVEL	
TRANSFER SWITCH 2 POSITION STATUS		BILEVEL	
TRANSFER SWITCH 3 POSITION STATUS		BILEVEL	
TRANSFER SWITCH 4 POSITION STATUS		BILEVEL	
TRANSFER SWITCH 5 POSITION STATUS		BILEVEL	
TRANSFER SWITCH 6 POSITION STATUS		BILEVEL	
CONSCAN THRESHOLD HIGH/LOW STATUS		BILEVEL	
CONSCAN ON/OFF STATUS		BILEVEL	
RECEIVER A LOOP STRESS		ANALOG	
RECEIVER B LOOP STRESS		ANALOG	
RECEIVER A VCO TEMP		ANALOG	
RECEIVER B VCO TEMP		ANALOG	
RECEIVER A SIGNAL STRENGTH		ANALOG	
RECEIVER B SIGNAL STRENGTH		ANALOG	
CONSCAN DATA WORD LINE		DIGITAL	
ATTITUDE CONTROL			
AXIAL THRUSTER INITIATION STATUS	(4)	BILEVEL	
TRANSVERSE THRUSTER INITIATION STATUS	(4)	BILEVEL	
SUN SENSOR TEMP		ANALOG	
THRUSTER TEMPERATURES	(8)	ANALOG	
PROPELLANT PRESSURE		ANALOG	
DATA WORD LINES	(4)	DIGITAL	
THERMAL			
PROPELLANT HEATERS ENABLE/DISABLE STATUS		BILEVEL	
RAM PLATFORM HEATER ENABLE/DISABLE STATUS		BILEVEL	
SRM HEATER ENABLE/DISABLE STATUS		BILEVEL	
PLATFORM TEMPERATURES	(4)	ANALOG	
PROPELLANT TEMPERATURE		ANALOG	

Table 3-2. Bus Engineering Telemetry for C Formats

ELECTRICAL DISTRIBUTION

SPACECRAFT SEPARATION STATUS		BILEVEL	
COMMAND EXECUTE STATUS		BILEVEL	
SPACECRAFT ORDNANCE SAFE/ARM RELAY STATUS (PRIME)		BILEVEL	
SPACECRAFT ORDNANCE SAFE/ARM RELAY STATUS (REDNT)		BILEVEL	
UNDERVOLTAGE OVERRIDE STATUS		BILEVEL	
RECEIVER REVERSE STATUS		BILEVEL	
CDU +5 VDC BUS A SELECT STATUS		BILEVEL	
EQUIP CONV FAULT ISOLATOR STATUS	(4)	BILEVEL	(THOR/DELTA)
MAGNETOMETER BOOM RETRACT STATUS		BILEVEL	(THOR/DELTA)
MAGNETOMETER BOOM EXTENSION STATUS		BILEVEL	(THOR/DELTA)
ELECTRON TEMPERATURE PROBE ANT RELEASE STATUS		BILEVEL	
NEUTRAL MASS SPECTROMETER ION CAP EJECT STATUS		BILEVEL	
UV FLUORESCENT PROBE ANT RELEASE STATUS		BILEVEL	(THOR/DELTA)
LARGE PROBE RELEASE STATUS		BILEVEL	
LARGE PROBE CONNECTOR RELEASE STATUS		BILEVEL	
SMALL PROBE THERMAL SHIELD RELEASE STATUS	(3)	BILEVEL	
SMALL PROBE CONNECTOR RELEASE STATUS	(3)	BILEVEL	
SMALL PROBE RELEASE STATUS	(3)	BILEVEL	
RECEIVER REVERSE INHIBIT/ENABLE STATUS		BILEVEL	
CAPACITOR CHARGE STATUS (PRIMARY)	(2)	BILEVEL	
CAPACITOR CHARGE STATUS (REDUNDANT)	(2)	BILEVEL	

ELECTRICAL POWER

BATTERY AUTOMATIC CHARGE MODE STATUS		BILEVEL	
BATTERY DISCHARGE ENABLE/DISABLE STATUS		BILEVEL	
CHARGE RATE 1 ON/OFF STATUS		BILEVEL	
CHARGE RATE 2 ON/OFF STATUS		BILEVEL	
CHARGE RATE 3 ON/OFF STATUS		BILEVEL	
LARGE PROBE POWER ON/OFF STATUS		BILEVEL	
SMALL PROBE 1 POWER ON/OFF STATUS		BILEVEL	
SMALL PROBE 2 POWER ON/OFF STATUS		BILEVEL	
SMALL PROBE 3 POWER ON/OFF STATUS		BILEVEL	
CTRF INVERTER TRANSFER RELAY STATUS		BILEVEL	
CTRF INVERTER TEMP		ANALOG	
BATTERY CHARGE CURRENT		ANALOG	
BATTERY DISCHARGE CURRENT		ANALOG	
BATTERY VOLTAGE		ANALOG	
BATTERY TEMPERATURE		ANALOG	
DC BUS VOLTAGE		ANALOG	
DC BUS VOLTAGE, EXPANDED		ANALOG	
DC BUS CURRENT		ANALOG	
SHUNT BUS CURRENT		ANALOG	
TRF +5 VDC CDU OUTPUT CHANNEL A		ANALOG	
TRF +5 VDC CDU OUTPUT CHANNEL B		ANALOG	

DATA HANDLING

CONVOLUTION CODE GEN STATUS		BILEVEL	
ROLL REFERENCE		BILEVEL	
SPIN AVERAGING MODE		BILEVEL	
ACS OPERATION MODE		BILEVEL	
DECODER A STATUS		BILEVEL	
DECODER B STATUS		BILEVEL	
A/D CALIB VOLTAGE, LOW		ANALOG	
A/D CALIB VOLTAGE, MED		ANALOG	
A/D CALIB VOLTAGE, HIGH		ANALOG	
ROLL ATTITUDE WORD LINES	(2)	DIGITAL	

Table 3-2. Bus Engineering Telemetry for C Formats (Continued)

EXTENDED S.C. ID		DIGITAL	
SPIN PERIOD WORD LINES	(3)	DIGITAL	
PROBE DATA WORD LINES	(4)	DIGITAL	
COMMUNICATIONS			
RECEIVER A SIGNAL PRESENT STATUS		BILEVEL	
RECEIVER B SIGNAL PRESENT STATUS		BILEVEL	
RECEIVER A COHERENT MODE STATUS		BILEVEL	
RECEIVER B COHERENT MODE STATUS		BILEVEL	
RECEIVER A OSC ENABLE/DECODER STATUS		BILEVEL	
RECEIVER B OSC ENABLE/DECODER STATUS		BILEVEL	
POWER AMP A/B SELECT		BILEVEL	
POWER AMP A HIGH/LOW STATUS		BILEVEL	(THOR/DELTA)
POWER AMP DRIVER A/B SELECT		BILEVEL	
POWER AMP B HIGH/LOW STATUS		BILEVEL	(THOR/DELTA)
TRANSFER SWITCH 1 POSITION STATUS		BILEVEL	
TRANSFER SWITCH 2 POSITION STATUS		BILEVEL	
TRANSFER SWITCH 3 POSITION STATUS		BILEVEL	
TRANSFER SWITCH 4 POSITION STATUS		BILEVEL	
RECEIVER A LOOP STRESS		ANALOG	
RECEIVER B LOOP STRESS		ANALOG	
RECEIVER A VCO TEMP		ANALOG	
RECEIVER B VCO TEMP		ANALOG	
RECEIVER A SIGNAL STRENGTH		ANALOG	
RECEIVER B SIGNAL STRENGTH		ANALOG	
ATTITUDE CONTROL			
AXIAL THRUSTER INITIATION STATUS	(4)	BILEVEL	
TRANSVERSE THRUSTER INITIATION STATUS	(4)	BILEVEL	
SUN SENSOR TEMP		ANALOG	
THRUSTER TEMPERATURES	(8)	ANALOG	
PROPELLANT PRESSURE		ANALOG	
DATA WORD LINES	(4)	DIGITAL	
THRUSTER PULSE COUNTERS	(8)	DIGITAL	
THERMAL			
PROPELLANT HEATER ENABLE/DISABLE STATUS		BILEVEL	
PROPELLANT TEMP		ANALOG	
PLATFORM TEMPERATURES	(4)	ANALOG	

3.1.4.4 Formats D-1 through D-8

Formats D-1 through D-8 are special purpose scientific formats. In each format, 576 of each 768 bits will be allocated to a single scientific instrument or location in the digital storage unit (DSU) with the selected instrument supplying digital data on a single input channel. The selected Format D will be telemetered only at the main frame rate overlayed on the Format A or Format B. The Format D scientific instrument assignments are as follows:

a) Format D-3, Orbiter: DTU, stored
b) Format D-4, Orbiter: IR radiometer, stored
c) Format D-5, Orbiter: Radar altimeter, stored
d) Format D-6, Orbiter: Neutral mass spectrometer, stored
e) Format D-7, Orbiter: Ion mass spectrometer, stored
f) Formats D-1, D-2, and D-8 are unassigned
g) Format D-3, Bus: Large probe checkout
h) Format D-4, Bus: Small probe one checkout
i) Format D-5, Bus: Small probe two checkout
j) Format D-6, Bus: Small probe three checkout
k) Formats D-1, D-2, D-7, and D-8 are unassigned

3.1.4.5 Formats E-1 and E-2

Formats E-1 and E-2 are for the subcommutation of scientific information in two words of each Format A and B or one word of each Format C-1, C-2, C-3, and C-4 when telemetered as the main frame. The two formats will be telemetered sequentially at all times and only in the subcommutated science word of the main frame. Word assignments for Formats E-1 and E-2 are shown in Table 3-3 for the orbiter, and Table 3-4 for the probe bus.

3.1.5 Operational Modes of Data Subsystem

The data subsystem will be capable of operating in three basic modes as follows:

a) Real Time Mode. Data are transmitted directly without intermediate storage at a bit rate selected by ground command.

Table 3-3. Orbiter Science Telemetry List for E Formats

MAGNETOMETER		
POWER ON/OFF STATUS		BILEVEL
CALIBRATE MODE STATUS		BILEVEL
RANGE HIGH/LOW STATUS		BILEVEL
DATA WORD LINE		DIGITAL
RADAR ALTIMETER		
POWER ON/OFF STATUS		BILEVEL
TRANSMITTER ON/OFF STATUS		BILEVEL
CALIBRATE MODE ON/OFF		BILEVEL
DATA RATE HIGH/LOW STATUS		BILEVEL
MEMORY POWER ON/OFF STATUS		BILEVEL
DATA WORD LINE		DIGITAL
UV SPECTROMETER		
POWER ON/OFF STATUS		BILEVEL
CALIBRATE MODE STATUS		BILEVEL
DATA RATE HIGH/LOW STATUS		BILEVEL
HOUSEKEEPING		ANALOG
DATA WORD LINE		DIGITAL
ION MASS SPECTROMETER		
POWER ON/OFF STATUS		BILEVEL
CALIBRATE MODE STATUS		BILEVEL
DATA WORD LINE		DIGITAL
IR RADIOMETER		
POWER ON/OFF STATUS		BILEVEL
DATA WORD		DIGITAL
NEUTRAL PARTICLE MASS SPECTROMETER		
POWER ON/OFF STATUS		BILEVEL
DATA WORD LINE		DIGITAL
ELECTRON TEMPERATURE PROBE		
POWER ON/OFF STATUS		BILEVEL
DATA WORD LINE		DIGITAL
SOLAR WIND ANALYZER		
POWER ON/OFF		BILEVEL (NEW SCIENCE)
MODE CONTROL STATUS	(3)	BILEVEL (NEW SCIENCE)
DATA WORD LINE		DIGITAL (NEW SCIENCE)
X-BAND OCCULTATION		
POWER ON/OFF STATUS		BILEVEL (NEW SCIENCE)
DATA WORD LINE		DIGITAL (NEW SCIENCE)

C-3

Table 3-4. Bus Science Telemetry List for E Formats

SCIENCE	
MAGNETOMETER	(SCIENCE III)
POWER ON/OFF STATUS	BILEVEL
CALIBRATE MODE STATUS	BILEVEL
DATA RATE HI/LO STATUS	BILEVEL
DATA WORD LINE	DIGITAL
UV FLUORESCENCE	(SCIENCE III)
POWER ON/OFF STATUS	BILEVEL
FURNACE CURRENT MODE 1 ON/OFF STATUS	BILEVEL
FURNACE CURRENT MODE 2 ON/OFF STATUS	BILEVEL
FURNACE CURRENT MODE 3 ON/OFF STATUS	BILEVEL
FURNACE CURRENT MODE 4 ON/OFF STATUS	BILEVEL
CALIBRATE MODE STATUS	BILEVEL
HOUSEKEEPING	ANALOG
DATA WORD	DIGITAL
ION MASS SPECTROMETER	
POWER ON/OFF STATUS	BILEVEL
CALIBRATE MODE STATUS	BILEVEL
DATA WORD LINE	DIGITAL
NEUTRAL PARTICLE MASS SPECTROMETER	
POWER ON/OFF STATUS	BILEVEL
DATA WORD LINE	DIGITAL
ELECTRON TEMPERATURE PROBE	
POWER ON/OFF STATUS	BILEVEL
DATA WORD LINE	DIGITAL
UV SPECTROMETER	
POWER ON/OFF STATUS	BILEVEL
DATA WORD LINE	DIGITAL
RETARDING POTENTIAL ANALYZER	(SCIENCE IV)
POWER ON/OFF STATUS	BILEVEL
CALIBRATE MODE STATUS	BILEVEL
DATA RATE HI/LO STATUS	BILEVEL
DATA WORD LINE	DIGITAL

b) Telemetry Storage Mode. Data are stored and transmitted simultaneously and continuously until either the DSU is full or the data subsystem is commanded to the real time mode. The DSU can be particularly filled and data stored at a later time beginning at the memory location of the last previously stored data. When the telemetry storage mode is terminated by ground command, or when the DSU is filled, the data subsystem will automatically switch to the real time mode in the format and bit rate used during the telemetry storage mode.

c) Memory Readout Mode. Data are read out from the DSU and transmitted at a bit rate selected by ground command. When memory readout is completed, the data subsystem will automatically switch to the real time mode and the format used before memory readout, and remain in the bit rate used during memory readout. Real time science and engineering will be transmitted in the format with the data read out from the DSU via the appropriate D-n format.

3.1.6 On-Board Storage Capacity

A storage capacity of 1, 228, 800 bits will be provided on the Orbiter by five DSU's, each DSU containing two 122, 880-bit modules. (No storage is provided on the Bus.) Storage will be allocated as follows:

1) Neutral mass spectrometer
Up to 122, 880 bits

2) Ion mass spectrometer
Up to 122, 880 bits

3) IR radiometer
Up to 245, 760 bits

4) Radar altimeter
Up to 122, 880 bits

5) DTU, formatted data, includes magnetometer, electron temperature, UV spectrometer, and engineering up to 368, 640 bits

The input rate to the DSU is 10, 000 bits/s maximum. In the buffer mode, the DSU will act as a large raw buffer, reading out only in the D-n format. The radar altimeter is always buffered; the IR radiometer and the UV spectrometer can operate in realtime (see Figure 3.1.4.2-8. Orbiter) or in the buffered mode. Storage is also provided for DTU (preformatted) storage during occultation periods.

3.2 SIGNALS FROM SCIENTIFIC INSTRUMENTS

3.2.1 Characteristics

The characteristics of digital, analog, timing, and operational status signals from the scientific instruments to the data subsystem shall be as defined in Figures 3-7A and Figure 3-7B.

3.2.2 Circuit Diagrams

The DTU/DSU interface circuits for signals from the scientific instruments to the DTU and DSU will be as shown in Figures 3-8 through 3-10.

3.2.3 Analog Signal and Coding Accuracy

An analog signal received by the data subsystem from a scientific instrument will be converted into a 10-bit digital word in the digital telemetry unit. For an impedance of the analog signal source less than 500 ohms, the overall error in converting the signal will be ±0.5 percent accuracy. For a larger source impedance the error will be increased. Up to 16 analog inputs may be handled in the A and B formats.

3.2.4 Signal Fault Voltages

The exposure limit of the spacecraft to fault voltages on the signal lines will be as given in Table 3-5. The fault voltage exposure limit of the spacecraft is defined as the maximum voltage level on the signal lines to which the spacecraft may be exposed with no degradation of operation after removal of the fault.

3.3 SIGNALS TO SCIENTIFIC INSTRUMENTS

3.3.1 Timing Signals

The spacecraft data subsystem will supply the following timing signals to the scientific instruments as required.

a) Main Frame Rate Pulse. A pulse generated at the end of each main frame.

b) Science Subframe Rate Pulse. A pulse generated at the end of each 64th main frame.

c) Word Rate Pulse. A pulse generated each 3-bit periods.

d) Bit Shift Pulse. A pulse generated at the operating bit rate of the DTU that shifts digital bits from the scientific instruments to the data subsystem.

e) 32.768 kHz Clock. A square wave pulse train with a 32.768 kHz repetition rate.

f) 2048 Hz Clock. A pulse train with a 2048 Hz repetition rate.

g) Word Gate. A voltage level provided for the time period that the spacecraft data subsystem will accept digital data from an output line of a scientific instrument. The digital word gate is continuous throughout consecutive word assignments in the main frame. The length of the word gate is also dependent upon the operating bit rate of the data subsystem. The word gate will be capable of driving a second scientific instrument which will use the word gate as a timing signal only. There will be no readout of data from the second instrument.

h) End of Memory. A signal generated whenever the DSU reaches the last memory location. One from each memory module. (Provided on orbiter only.)

3.3.2 DTU Operational Status

The data subsystem will supply state signals to the scientific instruments indicating the following operational conditions of the DTU as required:

a) Bit Rates. The status of the eight bit rates will be presented in a coded form using three lines as follows:

1) 1024 bits/s - 111
2) 512 bits/s - 110
3) 256 bits/s - 101
4) 128 bits/s - 100
5) 64 bits/s - 011
6) 32 bits/s - 010
7) 16 bits/s - 001
8) 8 bits/s - 000

NOMENCLATURE	SIGNAL DESCRIPTION	ORIGIN	LOADED CHARACTERISTICS & TOLERANCES: AMPLITUDE (+) VOLTS On (True)	LOADED CHARACTERISTICS & TOLERANCES: AMPLITUDE (+) VOLTS Off (False)	Duration µS (50% - 50%)	Rise Time µS (10% - 90%)	Fall Time µS (90% - 10%)	IMPEDANCE, OHMS: SPACECRAFT On (True)	IMPEDANCE, OHMS: SPACECRAFT Off (False)	INSTRUMENT	Comments	Noise, Volts (P-P) From Spacecraft	Noise, Volts (P-P) From Instrument
Main-Frame Rate Pulse	A pulse at the end of each frame. 11	DTU	4.5 ±1	0.25 ±0.25	60 ±25	1 to 5	1 to 10	300 max	4 k max	50 k min 50 pf max	Use at least half of 50 k ohm to isolate 50 pf.		
Science Subframe Rate Pulse	A pulse every 64 frames.	"	"	"	"	"	"	"	"	"	"		
Word Rate Pulse	A pulse generated each 3 bit periods.	"	"	"	"	"	"	"	"	"	"		
Bit Shift Pulse	Pulses continuously generated at the operating bit rate.	"	"	"	"	1 to 7.5	1 to 15	"	"	"	"		
Roll Index Pulse	A single pulse per spacecraft revolution.	"	"	"	"	"	"	"	"	"	"		
Sector Generator Pulse	512 pulses per spacecraft revolution.	"	"	"	"	1 to 5	1 to 10	"	"	"	"		
	64 and 8 pulses per spacecraft revolution.	"	"	"	50% ±10% Duty Cycle	"	"	"	"	"	"		
32.768 kHz Clock (High Clock)	A square wave with a 32.768 kHz repetition pulse.	"	"	"	"	"	"	"	"	"	"		
2048 Hz Clock (Low Clock)	A pulse train with a 2048 Hz repetition rate.	"	"	"	15 "	"	"	"	"	"	"		
Bit Rate ID Signal	Continuous states indicating operating bit rate. 3 wire connection (one wire per bit with 000 = lowest bit rate and 111 = highest bit rate).	"	"	"	Duration of Condition	1 to 30	1 to 50	"	"	500 K min 50 pf max	Use at least half of 500 K ohm to isolate 50 pf.		
Mode ID Signal	Continuous states indicating operating mode.	"	"	"	"	"	"	"	"	"	"		
Format ID Signal	States indicating operating format. (Formats A, B, D, and D-3).	"	"	"	"	"	"	"	"	"	"		
Word Gate	Gate to each instrument to indicate time of reading out digital data to DTU. Separate line for main-frame and sub-frame digital words.	"	"	"	"	"	1 to 50*	5 k	123 k max	50 k min 50 pf max	Use at least half of 50 k ohm to isolate 50 pf.		
Digital Data	A pulse shall indicate one and no pulse shall indicate zero.	Sci. Inst.	"	"	One DTU Bit Period	-	-	Current to DTU shall be 5 µA max	Current from DTU will be 100 NA max	5 k max	Duration is bit rate dependent.		
Instrument Operational Status	Bi-level states indicating operational conditions of instruments. Separate line required for each signal.	"	"	"	Duration of Condition	-	-	"	"	"	- - - - - - -		
Analog Data	Normalized analog voltage. Separate line required for each word.	"	Normalized 0 to 3.0	-	-	-	-	Current to DTU shall be 200 NA max	"	500Ω max for ±1 bit A/D conv. accuracy	3.0 V max from instrument		
		"	0 to 5.0	-	-	-	-	"	"	"	5.0 V max from instrument		

*The maximum fall time of the word gate signal will be less than 50 µs when loaded with a parallel combination of word gate output and instrument input resistance of 35 k ohms, and circuit capacitance of 600 pf maximum.

Figure 3-7A. Spacecraft and Instrument Signals

			LOADED CHARACTERISTICS AND TOLERANCES					IMPEDANCE, OHMS				NOISE, VOLTS (P-P)	
NOMENCLATURE	SIGNAL DESCRIPTION	ORIGIN	AMPLITUDE (+) VOLTS On (True)	Off (False)	Duration µs (50% - 50%)	Rise Time µs (10% - 90%)	Fall Time µs (90% - 10%)	SPACECRAFT On (True)	Off (False)	INSTRUMENT	COMMENTS	From Spacecraft	From Instrument
End of Memory	A pulse when DSU reaches last memory location.	DSU	4.5 ±1	0.25 ±0.25	Duration of Condition	1 to 5	1 to 10	300 max	4 k max	50 k min 50 pf max	Use at least half of 50 k ohm to isolate 50 pf.		
IPP Data Store Gate	Gate to DSU to indicate data is to be stored in buffer.	UA/IPP	"	"	Duration of Store Period	1 to 3	1 to 3	Current to DSU shall be 200 µA max	Current to DSU shall be 35 µA max	- 9	Gate will be at a continuous high level during data store.		
IPP Bit Shift Clock	16.384 kHz clock pulses to DSU for shifting digital data into DSU buffer.	UA/IPP	"	"	50% ±10% Duty Cycle	"	"	"	"	- 9	Pulses occur in bursts of 6 or 10 bits		
IPP Digital Data	NRZ data with low state to indicate a zero and high state to indicate a one.	UA/IPP	"	"	One Clock Period	"	"	"	"	- 9	Data bits stored in bursts of 6 or 10 bits		
Function Commands	Pulse upon receipt of a ground command. Separate line for each command.	CDU	0.25 ±0.25	$4^{+1.4}_{-1.5}$	50 ±20 Millisec.	100 max	10 max	Current to CDU shall be 200 µA max	Current from CDU will be 100 µA max	500 pf max	Pulse of decreasing voltage to instrument upon receipt of function command. 5.0 V max from instrument to CDU during non-cmd.		
On-Off Commands	Step signal upon receipt of a ground command. Single line connection.	CDU	$4^{+1.4}_{-1.5}$	0.25 ±0.25	Duration of Condition	"	"	Current from CDU will be 100 µA max	Current to CDU shall be 200 µA max	- 9	Step signal is of increasing voltage to instru. upon receipt of on-command. 5.0 V max from instru. to CDU during off-state.		

Figure 3-7B. Spacecraft and Instrument Signals

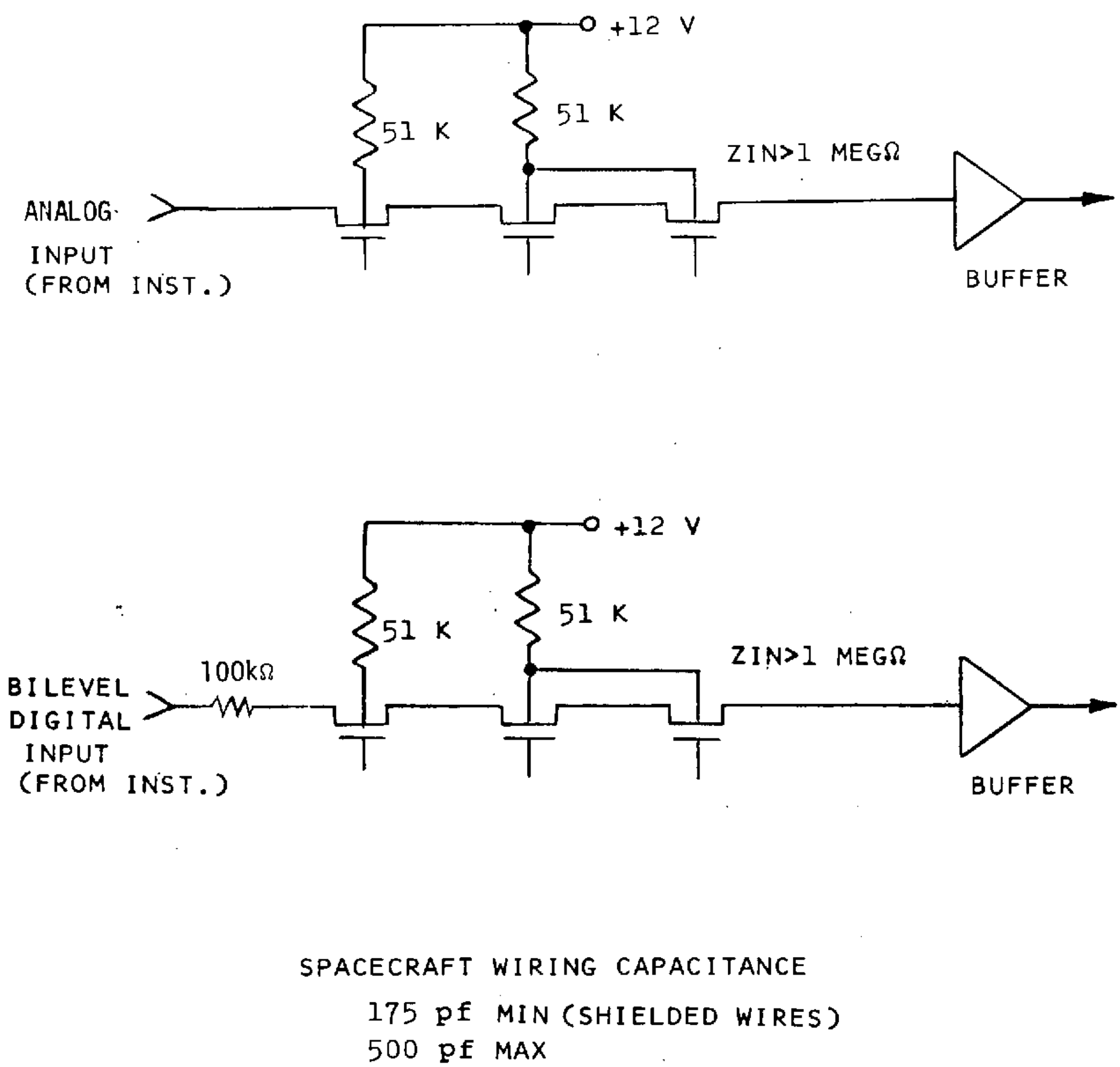


Figure 3-8. DSU Circuit for Science Bit Shift Clock, Data Store Gate, and Digital Data

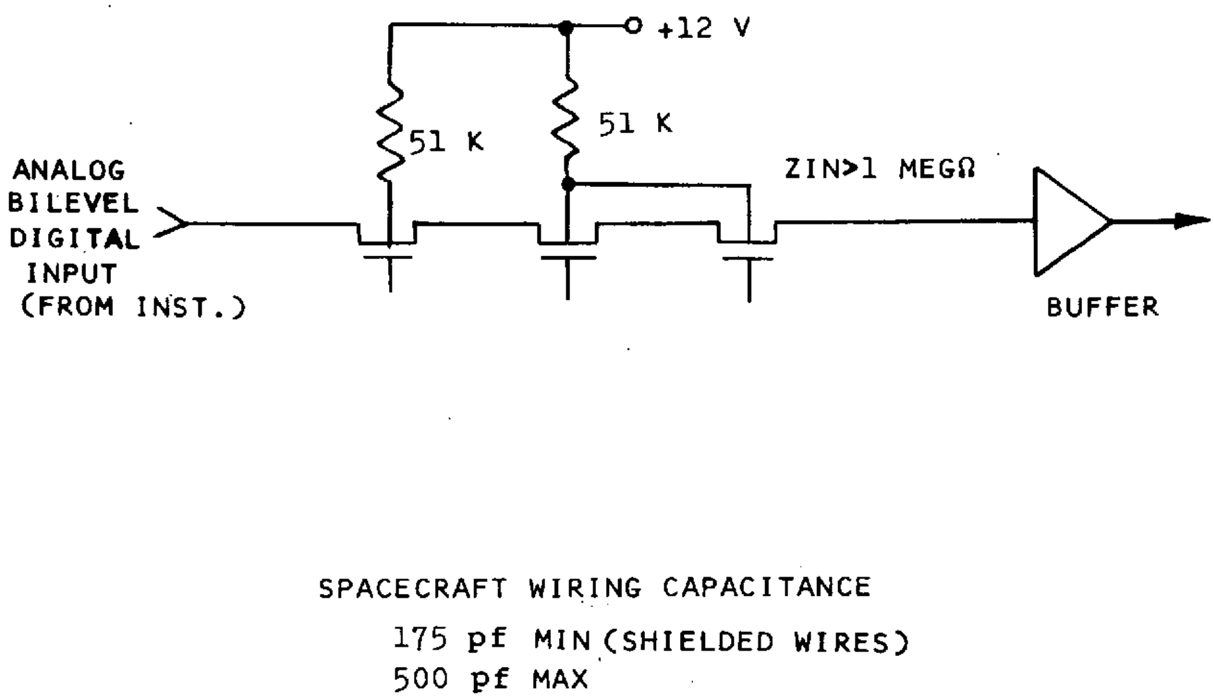
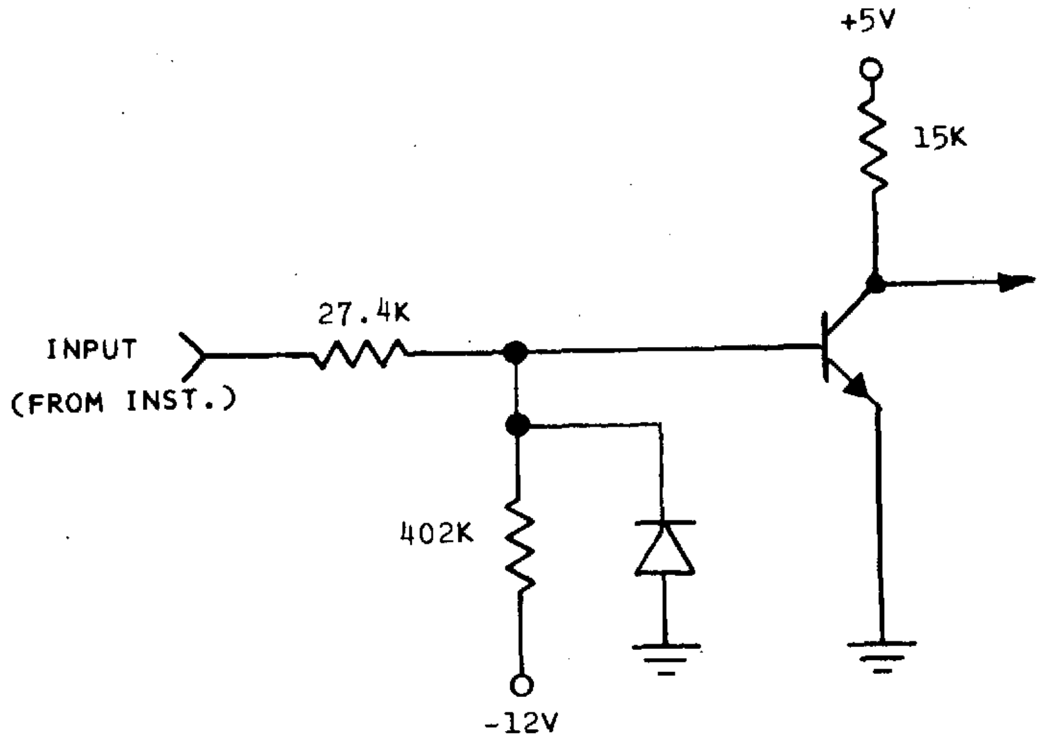


Figure 3-9. Circuit for Subframe Analog Bilevel and Digital Data



SPACECRAFT WIRING CAPACITANCE
175 pf MIN
500 pf MAX

Figure 3-10. Circuit for Mainframe Digital Data

Table 3-5. Exposure Limits for Fault Voltages

Data Inputs	Fault Voltage Limits
Multiplexer	
Analog	±10 VDC
Bilevels	±10 VDC
Digital	±10 VDC
Outputs	
Experiment Word Gate Output	±10 VDC
Experiment Output Buffer	0 to +8 VDC
Transmitter Output Buffer	0 to +8 VDC

b) <u>Formats.</u> State lines will indicate which format the DTU is operating in as follows:

TBD

c) <u>Modes.</u> Telemetry store and memory readout on individual lines. (Not required on probe bus.)

3.3.3 <u>Roll Index Pulse</u>

The spacecraft will provide to the instrument, as required, a roll-index pulse which indicates when the spacecraft-to-sun line passes through a plane containing the spin axis and rotating with the spacecraft body.

The roll index pulse is generated by the spin period sector generator (SPSG) of the DTU. The input to the SPSG is the roll pulse which is generated by the sun sensor assembly (SSA). Should the SPSG fail to receive roll pulses from the SSA, the SPSG will continue to output roll index pulses based on the most recently measured roll pulse time interval.

During periods of time when the SSA is providing the roll pulse to the CEA, the roll index pulse can occur as described above with the reference line (R) directed toward the sun, or away from the sun, selectable by ground command.

3.3.3.1 <u>Deleted</u>

3.3.3.2 <u>Sun for Source Stimulus</u>

When the stimulus source is the sun (and the look angle is between 10 and 110 degrees) the angle between the fixed reference line and the plane containing the spin axis (x) and the sun will:

a) Vary no more than 0.2 degree on a short-term basis (jitter).

b) Vary no more than ±3 degrees on a long-term basis (days).

3.3.3.3 Roll Attitude

The spacecraft telemetry data will provide information which will permit correlating the attitude of the roll index reference line to an accuracy of ±1/2 degree with a given telemetry word by ground calculation.

3.3.4 Spin Period Sector Generator

The spacecraft will have a redundant spin period sector generator which will perform the following

a) Divide each spacecraft revolution into 512 sectors (to within 150 μs) using the roll index pulse (Section 3.3.3) as a reference.

b) Operate over spacecraft spin rate range from 2 to 100 rpm.

c) Provide the following output pulses each revolution of the spacecraft to the scientific instruments and spacecraft subsystems as required based upon receipt of one roll pulse per revolution:

1) One pulse each 1/8 revolution
Accuracy = $\frac{\text{Spin Period}}{8}$ to within 150 μs.

2) One pulse each 1/64 revolution
Accuracy = $\frac{\text{Spin Period}}{64}$ to within 150 μs.

3) One pulse each 1/512 revolution
Accuracy = $\frac{\text{Spin Period}}{512}$ to within 150 μs.

3.3.4.1 Sector Generator Modes — Filtered Roll Pulse

The spin sector generator will operate in three different modes selectable by ground command.

a) Nonaveraging Mode. Nonaveraged output pulses to the scientific instruments.

b) Spin Averaging Mode. The output pulses to the scientific instruments will be averaged over 64 spacecraft revolutions. In this mode the time between any set of adjacent filtered roll pulses will be the same as that between any other set to within 150 μsec.

c) Hold Mode. Upon entry into eclipse, output pulses continue in the spin averaging mode using the spin period obtained just prior to eclipse entry.

3.3.5 Commands

TBD function commands have been allocated for scientific instrument use. The command assignments and nomenclature are shown in Figure 3-11 for bus and orbiter. The interface circuit for low-level function commands will be as shown in Figure 3-12. The on/off command interface circuit will be as shown in Figure 3-13.

3.3.6 Characteristics

The characteristics of signals from the spacecraft to the scientific instruments will be as defined in Figures 3-7A and 3-7B. A definition of terms for the pulse signals and their characteristics is given in Figure 3-14.

3.3.7 Circuit Diagrams

Interface circuits for signals from the DTU and DSU to the scientific instruments will be as shown in Figures 3-15 through Figure 3-19.

3.3.8 Signal Timing Diagram

The time relationship of the timing signals supplied by the data subsystem to the instruments is shown in Figure 3-20.

3.3.9 Signal Fault Voltage to Instruments

The exposure limit of the instruments to fault voltages on the signal lines shall be as given in Table 3-5. The fault voltage exposure limit is defined as the maximum voltage level on the signal lines to which the instruments may be exposed with no degradation of operation after removal of the fault.

3.3.10 Spacecraft/Instrument Thermistors

The interface circuit for spacecraft-supplied instrument thermistors will be as shown in Figure 3-21. These thermistors are powered and conditioned by the spacecraft, and will therefore, read out instrument temperatures at all times, (i. e., instrument power on or off).

To be supplied

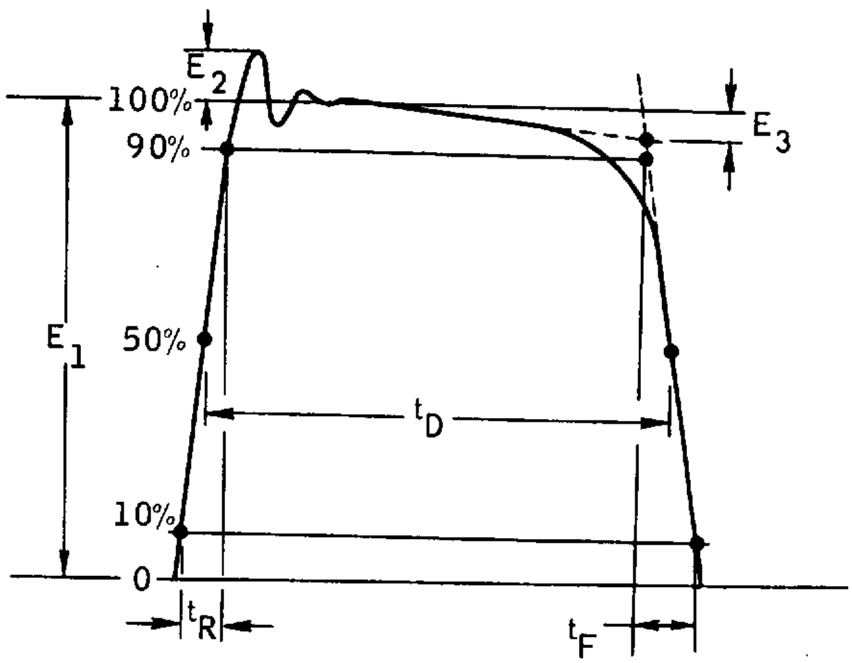
Figure 3-11. Command Assignments and Nomenclature

To be Supplied

Figure 3-12. Interface Circuit for Low-Level Function Commands

To be supplied

Figure 3-13. ON/OFF Command Interface Circuit



t_R = RISE TIME

t_D = PULSE DURATION

t_F = FALL TIME

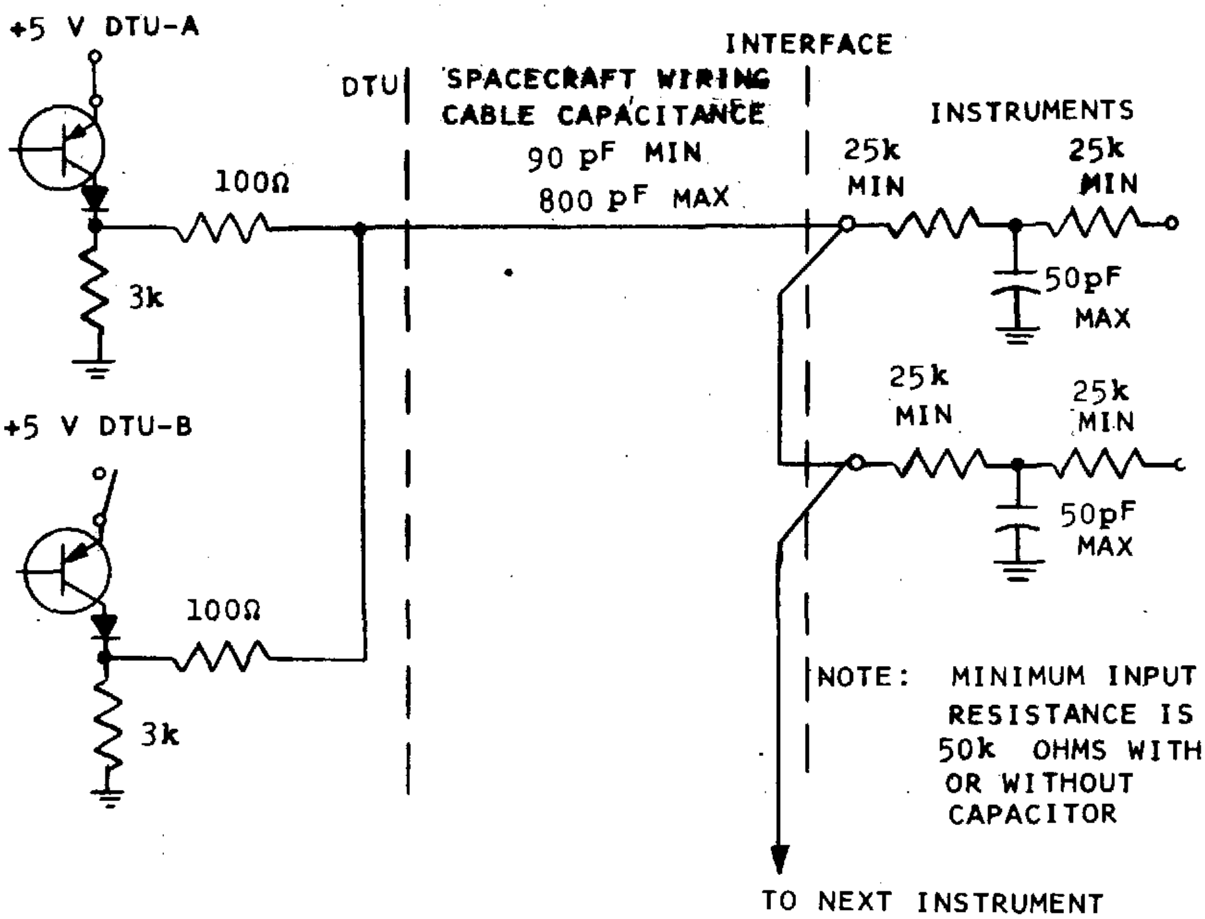
E_1 = PULSE AMPLITUDE

E_2 = MAXIMUM OVERSHOOT = 0.5 V

E_3 = MAXIMUM DROOP = 0.5 V

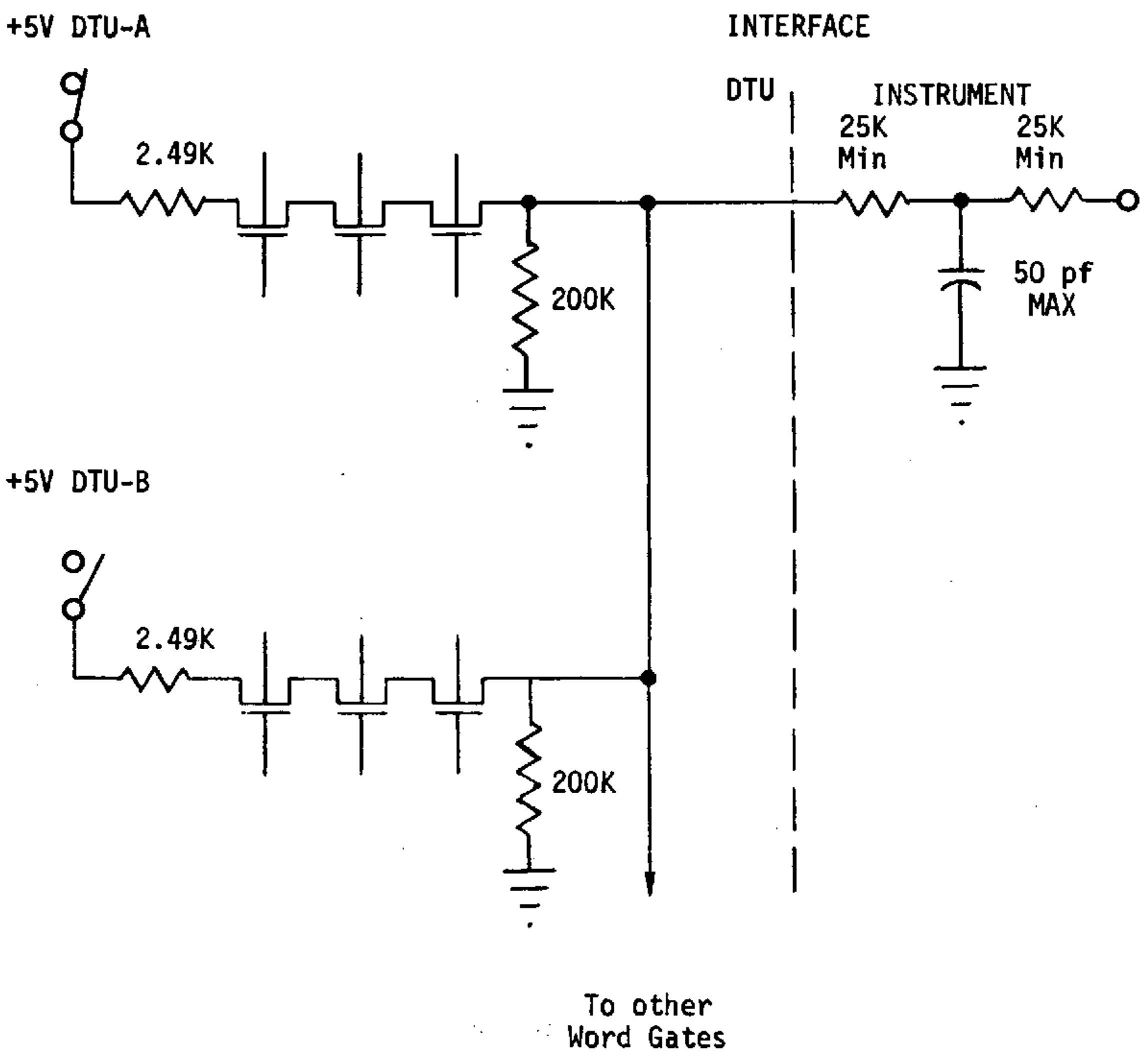
Note: Maximum Undershoot = 0.0 V

Figure 3-14. Definition of Terms and Characteristics of Pulse Signals from the DTU



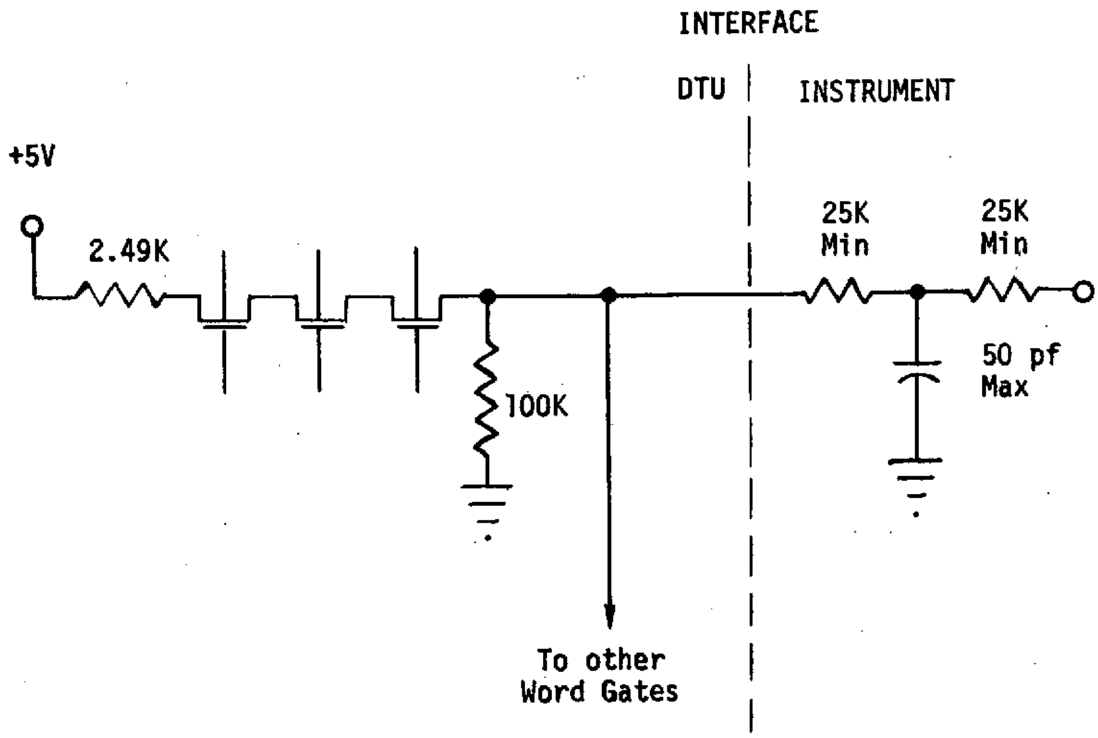
The cable capacitance of the bit shift pulse and the roll index pulse is 1200 pf max. each.

Figure 3-15. Circuit for Typical Timing Pulse Signal, Roll Index Pulse, and Spin Period Sector Pulse



NOTES:
(1) Minimum input resistance is 50K ohms with or without capacitor.
(2) Spacecraft wiring will be 200 pf min and and 600 pf max.
(3) The parallel combination of 200K ohms and instr load shall not be less than 17K

Figure 3-16. Circuit for Mainframe Digital Word Gate



NOTES: (1) Minimum input resistance is 50K ohms with or without capacitor.
(2) Spacecraft wiring will be 200 pf min and 600 pf max.
(3) The parallel combination of 100K ohms and instr load shall not be less than 17K.

Figure 3-17. Circuit for Subframe Digital Word Gate

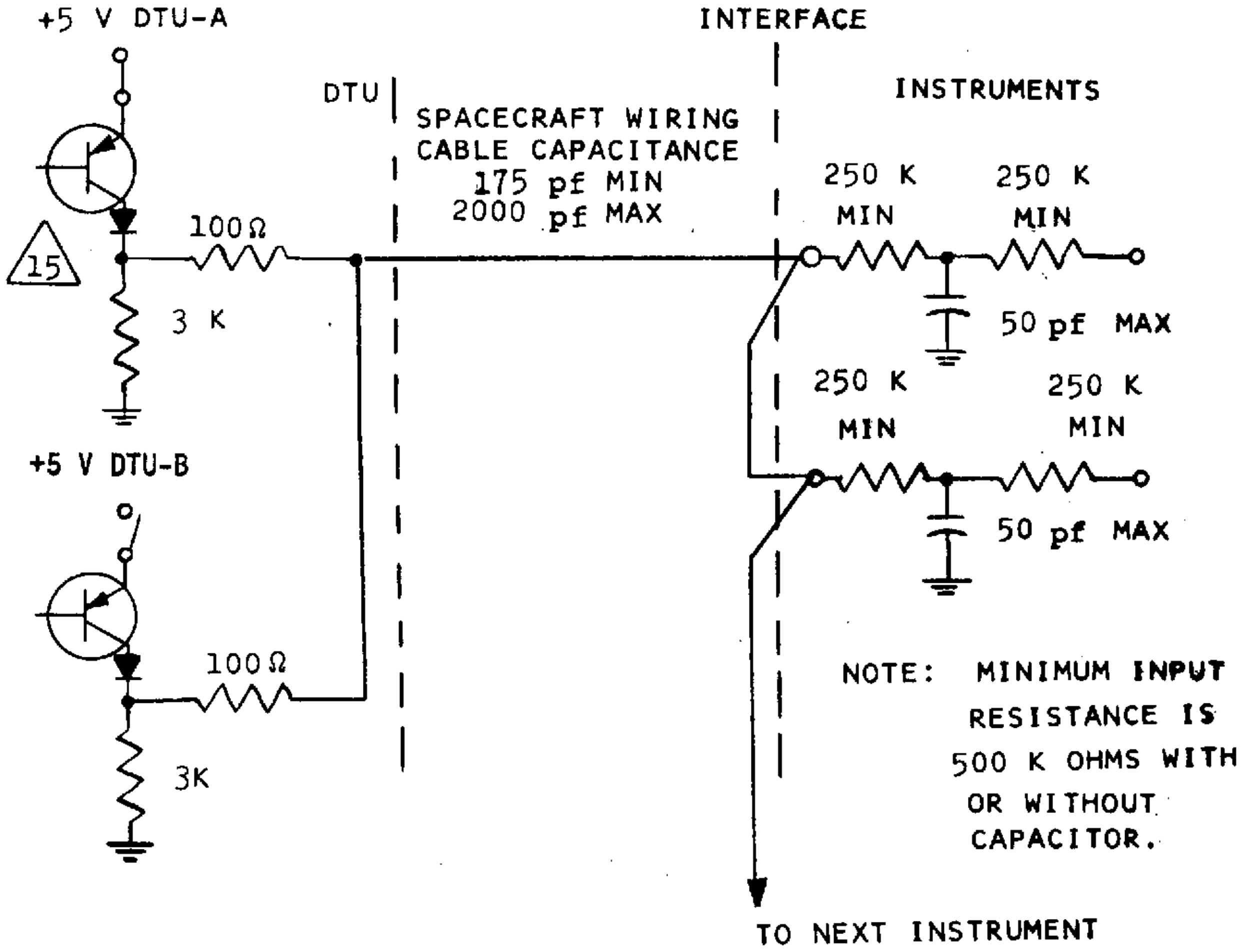


Figure 3-18. Circuit for Typical DTU Operational Status Signal

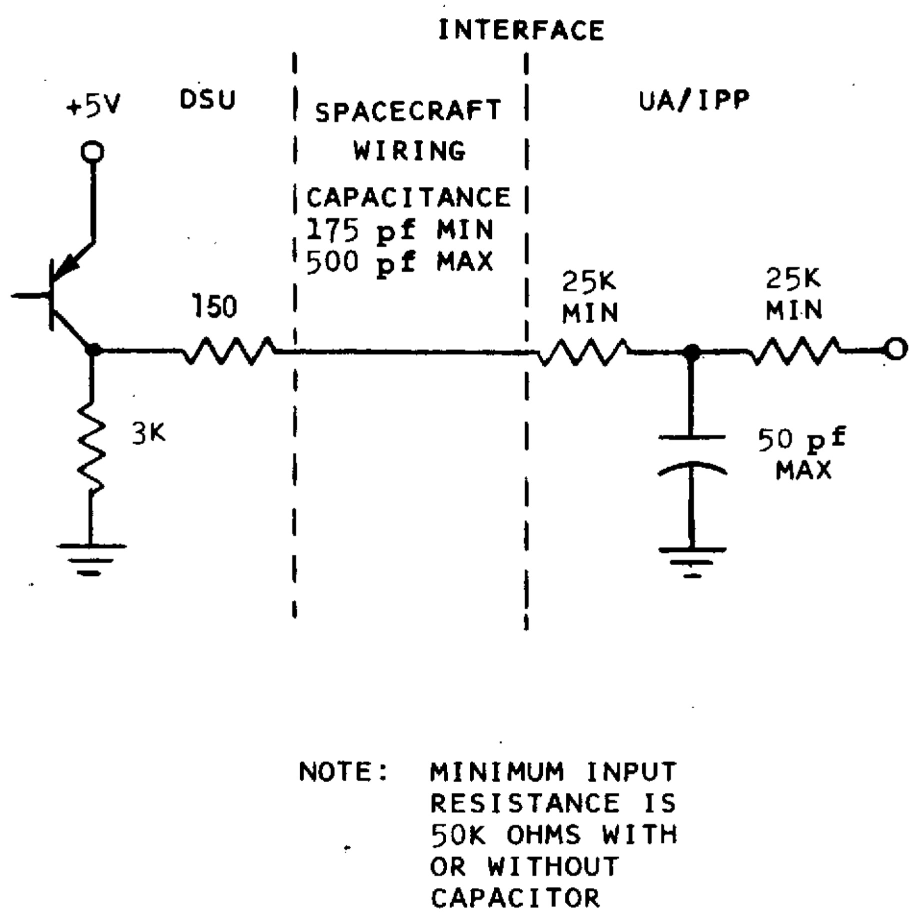


Figure 3-19. DSU Circuit for Science End of Memory Signal

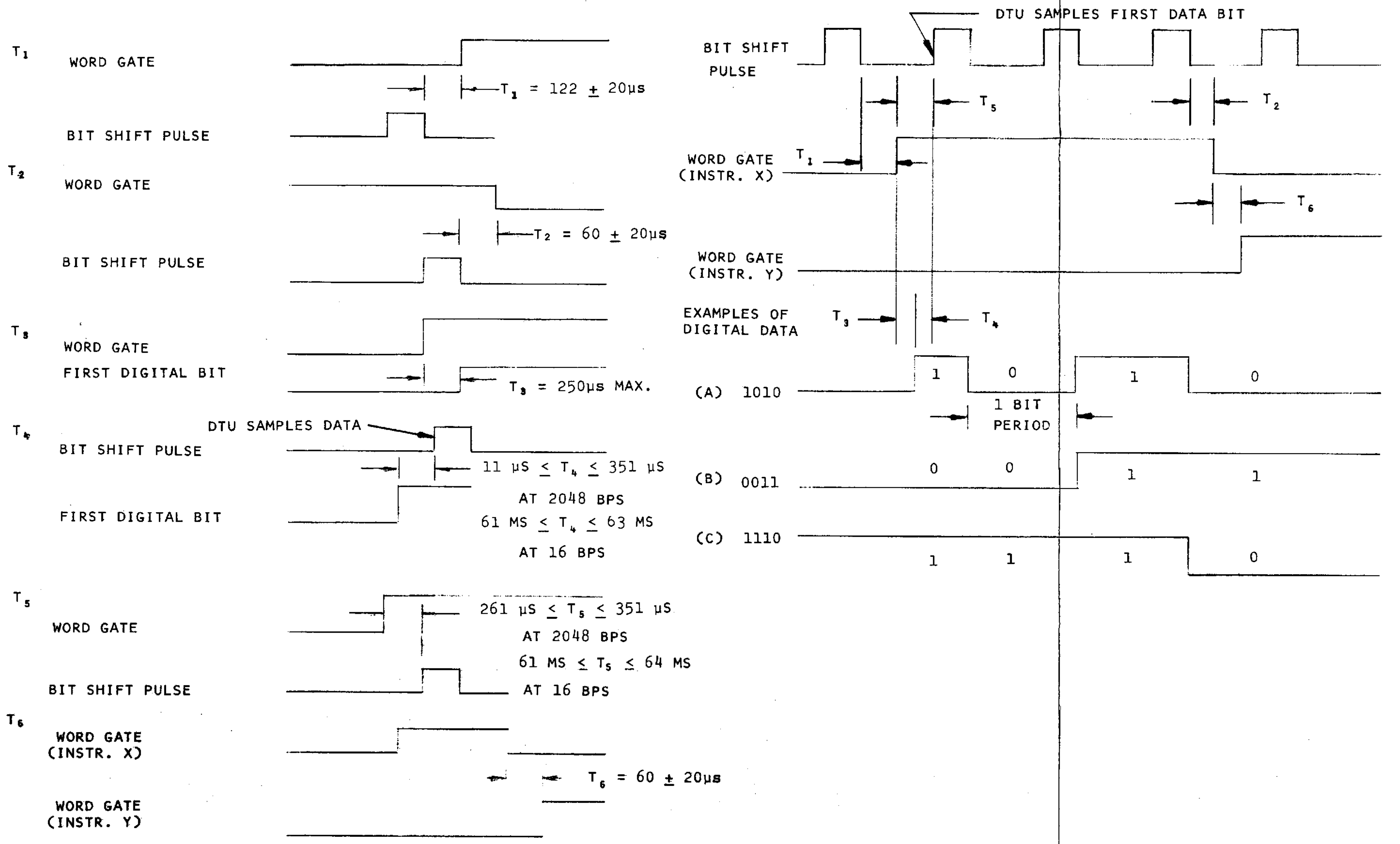


Figure 3-20. Timing Signal Relationships

4. THERMAL

4.1 OPERATIONAL TEMPERATURES

Scientific instruments will be exposed to ambient temperature levels that correspond with the instruments mounting position. The specific levels are listed in Table 4-1.

Table 4-1. Scientific Instruments Ambient Temperature Levels

Spacecraft	Scientific Instrument	Ambient Temperature Levels			
		Minimum °C (°F)		Maximum °C (°F)	
Probe Bus	<u>Equipment Compartment</u>	4	(40)	40	(104)
	Electron temperature probe electronics				
	<u>Solar Array Cavity</u>	-30	(-22)	60	(140)
	UV spectrometer Neutron mass spectrometer Ion mass spectrometer Retarding potential analyzer and electronics				
Orbiter	<u>Equipment Compartment</u>	4	(40)	40	(104)
	Radar altimeter electronics Electron temperature probe electronics UV spectrometer IR radiometer X-band occultation electronics Magnetometer electronics				
	<u>Ram Platform</u>	-30	(-22)	60	(140)
	Neutron mass spectrometer Ion mass spectrometer				

The boom mounted magnetometer can be maintained between -20°C (-40°F) and 60°C (140°F) with its own temperature control system consisting of a second surface mirror radiator, multilayer insulation blankets, and instrument internal power dissipation.

4.2 OPERATIONAL ENVIRONMENT

Instruments that are mounted external to the spacecraft equipment compartment, ram platform, or solar array cavity or have sensors that penetrate the boundaries of the above areas may be exposed to the following environments.

4.2.1 Prelaunch

A temperature range of 4°C (40°F) to 40°C (104°F) during the integration and test, storage, transportation to ETR, and spacecraft erection on-stand phases. Temperatures will be maintained between 16°C (60°F) and 38°C (100°F) once the fairing is in place.

4.2.2 Powered Flight

A fairing wall temperature between 16°C (60°F) and 38°C (100°F) until the shroud is jettisoned. At that time forward facing surfaces will be exposed to an instantaneous aerodynamic heat flux of 220 watts/meter^2 (0.02 BTU/ft^2-sec) which decays linearly to zero in 25 seconds.

4.2.3 Earth/Venus Transit

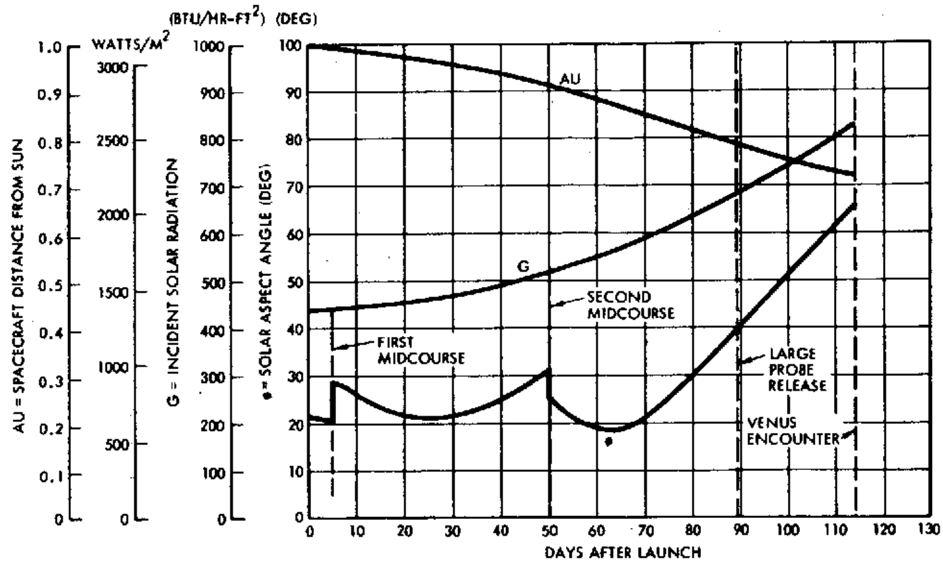
A -273°C (-459°F) vacuum space environment with solar intensity levels and sun incidence angles shown in Figure 4-1.

4.2.4 Venus Orbit Insertion

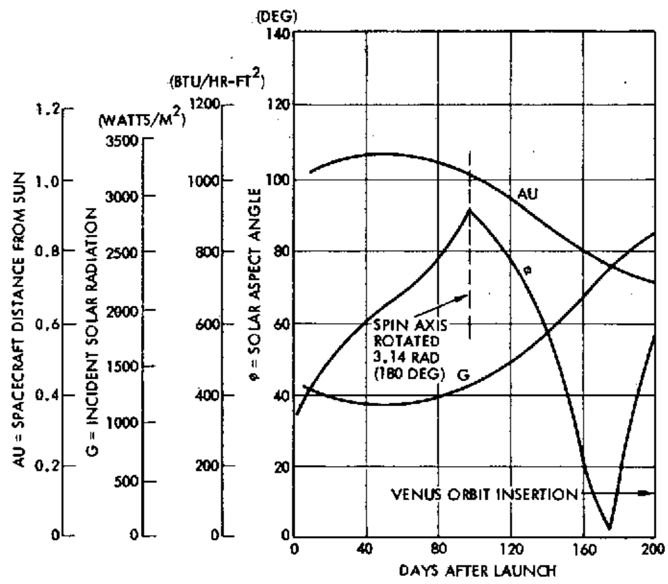
An incident radiant heat flux on aft-facing surfaces produced by the 23 second insertion motor burn. The magnitude of this heating is presented in Figure 4-2.

4.2.5 Venus Orbit

An additional planetary emission and albedo heat input beginning approximately 150 days after orbit insertion. This heat input occurs during the period when the spacecraft undergoes the longest eclipse.



PROBE BUS



ORBITER

Figure 4-1. Spacecraft Transit Thermal Parameters

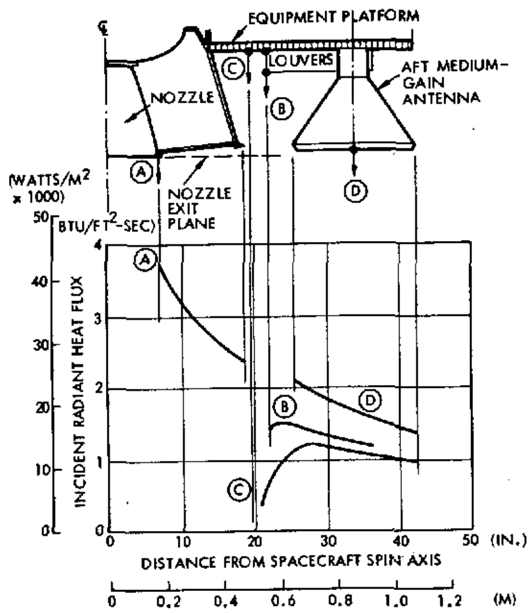


Figure 4-2. Injection Motor Plume Incident Radiant Heat Flux

Figure 4-3 specifies the peak orbital heat inputs in relation to the eclipse time. Orbital sun incidence angles and eclipse periods are presented in Figure 4-4.

4.3 INSTRUMENT REQUIREMENTS

When averaged over any 10-minute period of normal operation, the instrument thermal energy balance shall be such that:

a) The dissipated energy through external viewing ports shall not exceed the internally generated energy plus 1 watt.

b) The sum of the absorbed energy through external viewing ports and the internally generated energy shall not exceed 0.20 watts per square inch of instrument mounting surface.

c) The sum of the dissipated energy through the external viewing ports for all scientific instruments mounted within the equipment compartment shall not exceed 3 watts for the probe bus and 10 watts for the orbiter when the platform temperature near the scientific instruments is 4°C (40°F) or lower.

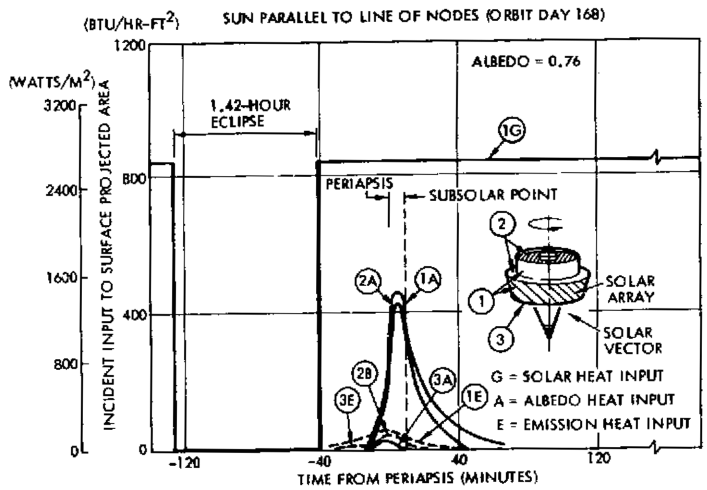


Figure 4-3. Solar and Venus Heat Inputs Near Periapsis

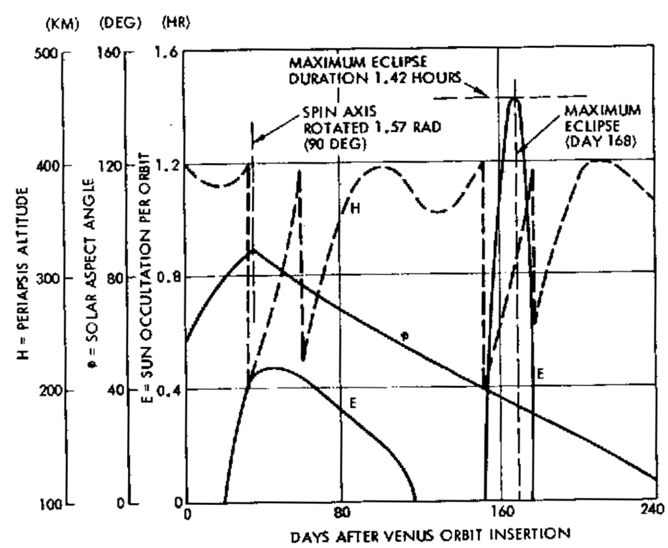


Figure 4-4. Venus Orbit Periapsis Altitude, Solar Aspect Angle, and Eclipse Duration

5. ELECTROMAGNETIC INTERFERENCE, NOISE AND GROUNDS

5.1 ELECTROMAGNETIC INTERFERENCE

Spacecraft equipment and scientific instruments shall comply with the electromagnetic interference requirements specified in SR1-8, EMI requirement Pioneer F/G.

5.2 NOISE

Restrictions on noise introduced into the spacecraft power system are given in Section 2.5.4. Restrictions on noise introduced into the spacecraft and instrument signal lines are given in Figure 5-1.

5.3 GROUNDS

5.3.1 Power Grounds

5.3.1.1 Primary

The primary DC power leads to the internal DC/DC converter scientific instruments shall be unshielded twisted pairs with the return connected to the spacecraft ground at one point only. The single ground point for primary DC power is located inside the power control unit.

5.3.1.2 Secondary

The secondary power developed within the scientific instrument shall be connected to the chassis ground but DC-isolated from the primary power. Secondary DC power distributed from the instrument to an external load (sensor) shall be dc isolated from the load chassis and shall imply a single-point secondary DC power ground within the instrument package.

5.3.2 Signal Returns

Signal returns shall be connected to chassis ground within the instrument. The instrument power on/off command return shall be connected to the instrument primary power return at the spacecraft single point ground.

5.3.2.1 RF, Command, and Digital Pulse Signals

All signals between the spacecraft and the scientific instruments shall be limited to a maximum of TBD microamps each. Shielded

To Be Supplied

Figure 5-1. Electromagnetic Interference Limits cable shall be used for all RF, command, digital, and pulsed interface signals. Coaxial cables shall be used only when signal levels and circuit sensitivities/impedance demand it.

5.3.2.2 Analog Signals

These signals shall be of a magnitude less than TBD microamperes. All analog signal wires shall be unshielded.

TBD

5.3.3 Shield Ground

A shielded wire with shield grounded at one end or both ends of the cable run shall employ a shield ground directly to the spacecraft structure. The shield ground wire shall be as short as possible (generally less than 1.0 inch) using a halo-ring or similar terminating device bonded to the spacecraft structure via the scientific instrument chassis.

5.3.3.1 Shield Termination

a) Shields which are intended to reduce magnetic field coupling shall be grounded at both ends of the cable run.

b) Shields which are intended to reduce electric field coupling may be grounded at one end of the cable run, provided the shield length is shorter than one-fifth of the wavelength of the highest frequency of interest. The shield termination should be at the grounded end of the circuit unless otherwise specified.

c) A shield length longer than one-fifth of the wavelength of the highest frequency of interest shall be grounded at both ends and at all intermediate interfaces.

d) Coaxial cable shields shall be grounded at both ends and at all intermediate interfaces.

5.3.3.2 Shields on Cables

Interconnecting cables for RF (15 kHz and above) or high impedance circuits (1,000 ohms and higher) shall have the shields chassis grounded at both the source and load ends.

5.3.4 Bonding

The mounting base plate of the scientific instrument shall be in total contact with the spacecraft mounting surface. The electrical bond between any interface connector shell via the scientific instrument chassis to the spacecraft platform shall not exceed a DC resistance of 10.0 milliohms per slice joint.

PART II

PRELIMINARY

PIONEER VENUS

SCIENTIFIC INSTRUMENT-LARGE PROBE
INTERFACE DOCUMENT

MARCH 23, 1973

SCOPE:

This document defines the characteristics of the Pioneer Venus Large Probe spacecraft pertinent to the scientific instruments and the requirements of the Large Probe on the scientific instruments.

1.0 MECHANICAL

1.1 Configuration and Dimensions

The approximate size allocation for each experiment instrument shall be as given in Table 1.0. These data are specifically for that instrument complement as presently defined for the Large Probe. Changes to the instrument complement are anticipated and can be accommodated.

The Large Probe descent capsule showing the location for the various experiment instruments is given in Figure 1.0.

1.2 Mass Properties

The weight and volume for each of the instruments shall not exceed those weights listed in Table 1.1. The specified weights do not include any connectors and cabling between the experiment boxes.

1.3 Mounting Technique

To minimize heat leakage into the probe it is preferred that instruments not be mounted physically to the pressure vessel but be mounted in intimate contact with the internal equipment shelf. It is recognized that some instruments will have elements that need to be tied structurally to the pressure vessel surfaces. The mounting of each instrument will be specified within and controlled by the Project Office through the Instrument ICD. The probe design is, however, capable of accommodating various combinations of mounting as follows:

(a) Single or multiple boxes

(b) Internal pressure vessel or shelf attachment

(c) External pressure vessel or non-pressure vessel attachments

1.3.1 Mechanical Attachment

(a) Pressure shell penetrations

Any instrument parts that require a penetration of the pressure shell will be mounted with threaded fitting and compression nut assembly similar to that shown in Figure 1.3. The gasket will be mounted in a groove in the shoulder of the fitting and will seal against a smooth, flat surface machined into the pressure shell.

(b) Equipment Ring Assembly

The Instruments with parts that require a penetration of the pressure shell shall make that penetration through the pressure equipment ring shown in Figure 1.0. The portion of instrument internal to pressure vessel will be mounted on an equipment platform attached to this ring.

1.4 Thermal Attachment

The thermal characteristics of the mechanical attachment shall be designed to promote heat transfer between the instruments and the equipment platform. Assuming such heat transfer properties the equipment shelf temperatures will reach the values shown in Table 1.2 at the indicated times during the large probe descent. The temperatures of the equipment ring assembly are also shown to identify the thermal environment for those parts of the experiments which must be mounted directly on the ring.

1.5 Dynamic and Static Environments

The significant dynamic and static environments to be encountered by the science instruments occur during (a) launch and trans-Venus injection, (b) interplanetary cruise, (c) Venus encounter, preseparation and probe cruise, (d) Venus entry and (e) descent. The Dynamic and Static environments are summarized in Table 1.3.

1.6 Alignment

Instrument mounting surfaces will be held to alignment tolerances of $\pm$ 0.5 degree with respect to the probe coordinate system. Instruments requiring closer alignments will be assigned individual mounting provisions or the instrument shall provide individual capability to satisfy critical alignment requirements. Equipment will be mounted in a manner that will be compatible with thermal control and structural integrity as required for the system. Mounting points on the equipment platform will have mounting surfaces with an out of plane tolerance not to exceed .0127 cm (.005 in).

1.7 Operating Atmosphere

The probe internal pressure may vary during different parts of the mission. The probe will be sealed at 103 kN/m^2 N_2 (1 ATM) and may leak down to a pressure not less than 41.4 kN/m^2 (0.4 ATM).

1.8 Windows and Feed-Throughs

1.8.1 Window Mechanical Interfaces. All windows through probe pressure vessel will consist of an inner and an outer lens element. The outer element will be extended beyond the probe insulation. The inner element will be located just outboard of the walls of the pressure vessel. Each element is capable of withstanding a maximum pressure differential of 9.42 MN/m^2 at the Venus surface.

The basic envelope and geometrical configuration for the various instrument window types is given in Figure 1.2.

All windows are located in the equipment pressure vessel ring frame (structural shell segment) of the Large Probe.

Heating for all windows will be supplied by the probe to minimize deposits of any condensation products. Window heater power will be turned on at aeroshell separation (at which time the window temperature is approximately 305K and the atmospheric temperature is 225K and will remain on until probe impact.

1.8.2 Mechanical Feed-Throughs. A pressure vewwel penetration will be provided for the inlet to the pressure gauges. The pressure gauge feed-through is illustrated in Figure 1.0.

1.8.3 Electrical Feed-Throughs. Electrical feed-throughs for the pressure vessel will be provided for the Temperature Sensor, Hygrometer, Accelerometer Calibration Connector, Window Heaters and Thermistors. In addition, provisions for RF-coax feed-throughs is required for the Wind Altitude Radar instrument. A maximum of 148 pins are available for science instruments.

The electrical feed-throughs for the large probe are shown in Figure 1.0.

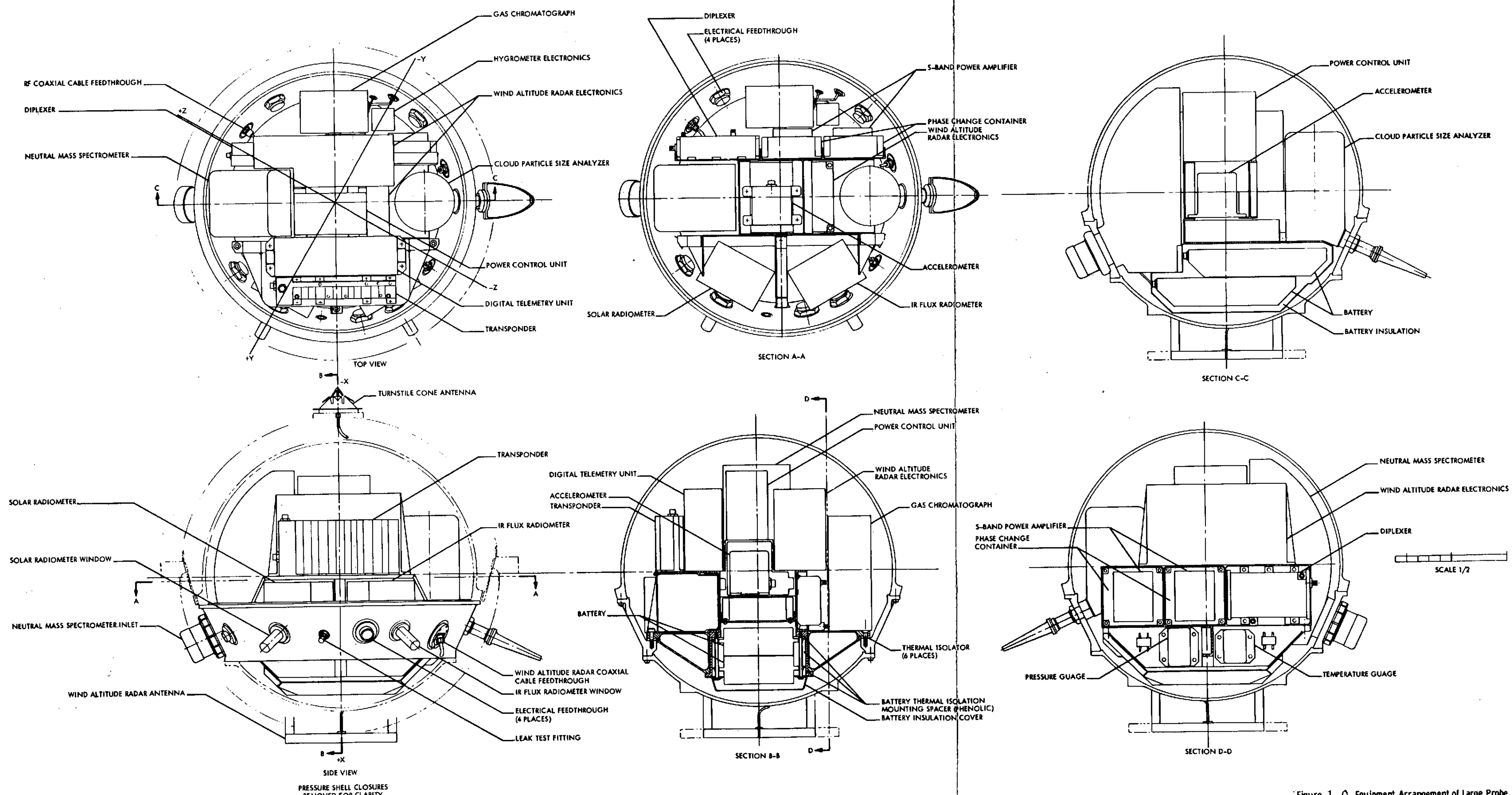


Figure 1.0 Equipment Arrangement of Large Probe

FOLDOUT FRAME 1

FOLDOUT FRAME 2

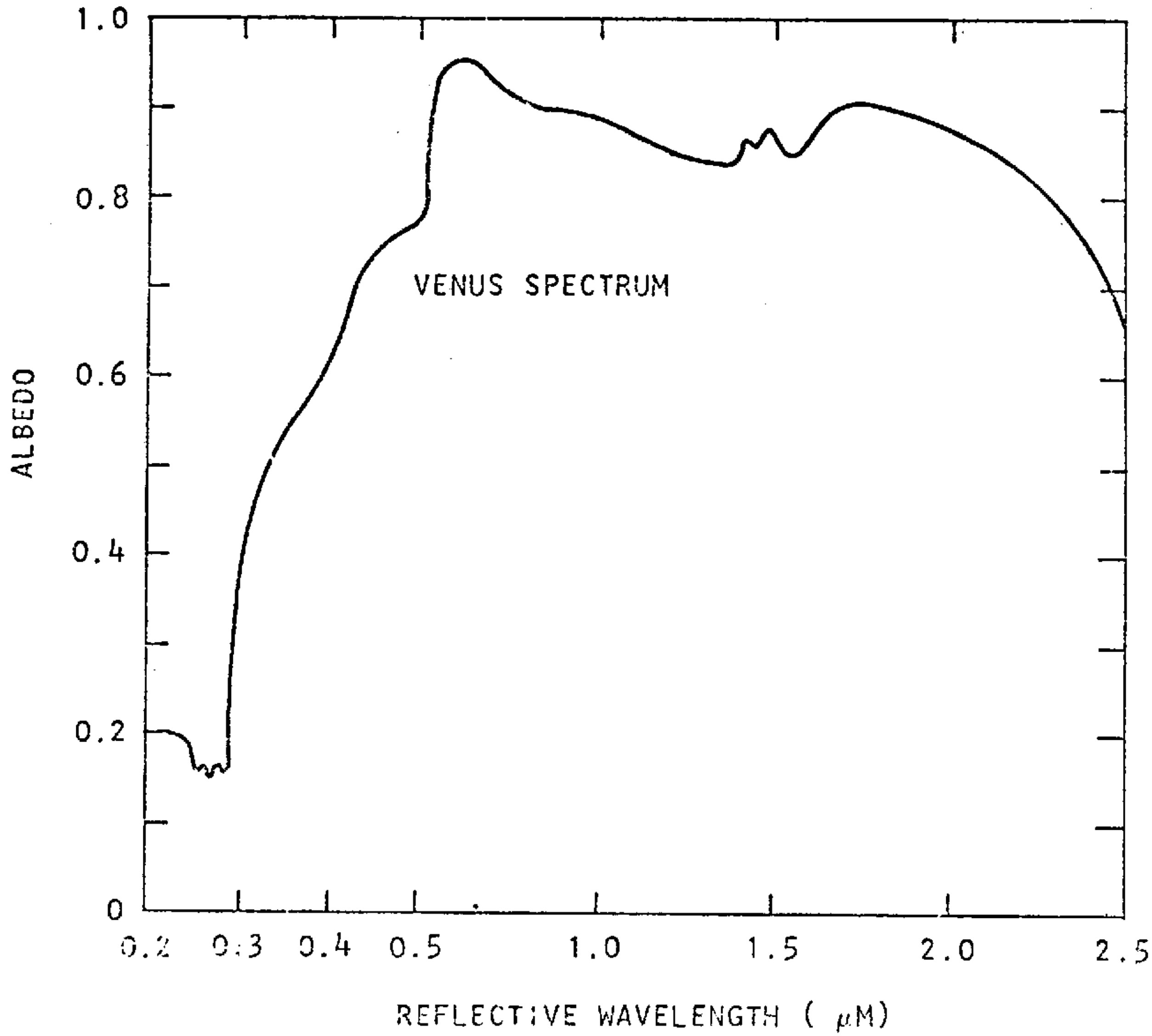


FIGURE 1.1 - REFLECTIVE SPECTRUM OF VENUS

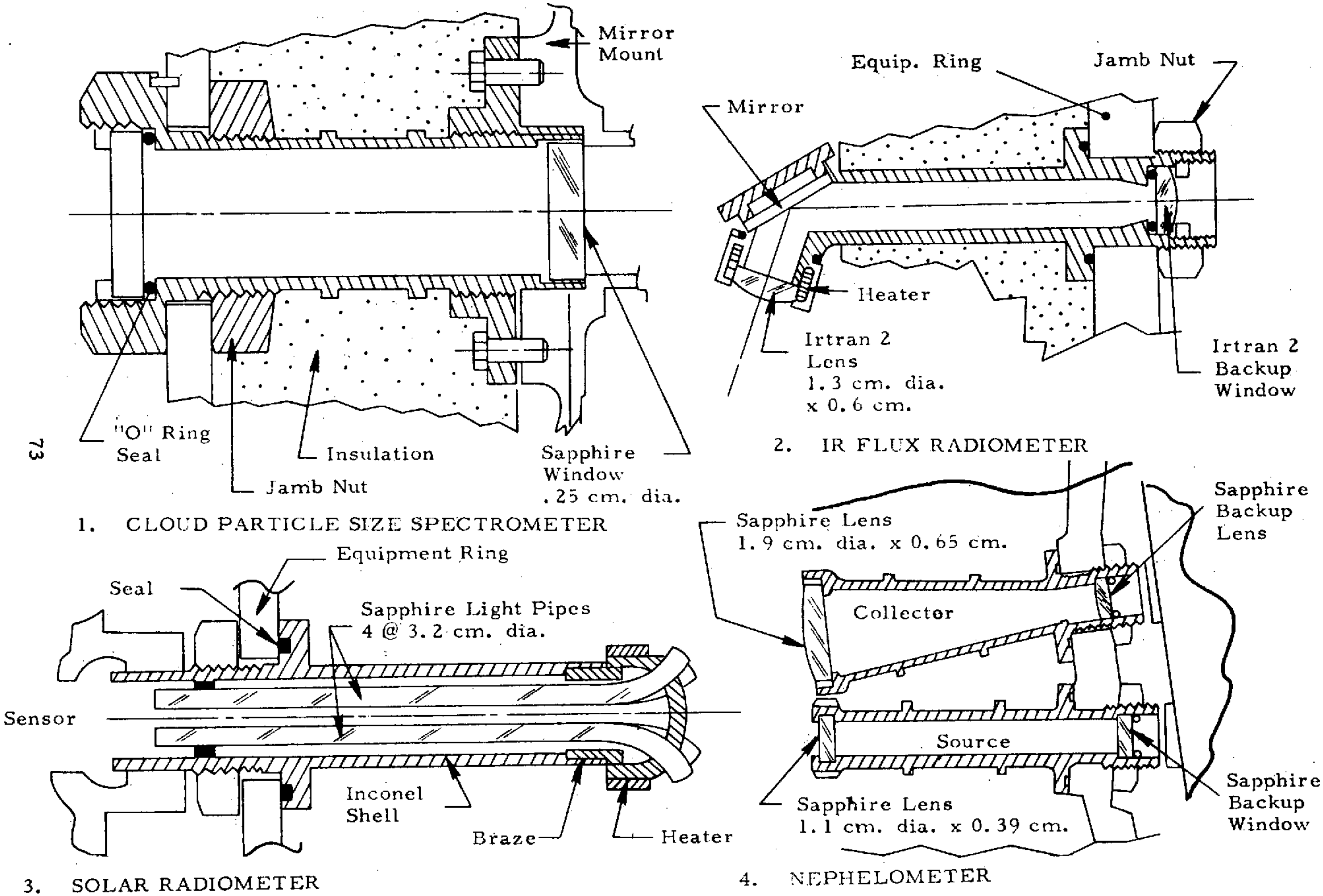


Figure 1.2 WINDOW CONFIGURATION DATA

TABLE 1.0 LARGE PROBE INSTRUMENT ALLOCATION

Instrument	Max. Envelope Dimensions Excluding Connectors cm	Volume (Nominal) $\pm$ 15% cc
o Accelerometer	8.9 x 8.9 x 9.5	665
o Temperature Gauge (2) (Excluding Sensor)	2.5 x 5.5 x 8.0	100
o Pressure Gauge (Excluding Sensor)	2.7 x 5.3 x 9.3	115
o Cloud Particle Size Analyzer	12.7 Dia. x 25.4	3275
o Solar Radiometer	10.5 x 13.2 x 13.0	1600
o IR Flux Radiometer	10.5 x 13.2 x 13.0	1600
o Neutral Mass Spectrometer	Refer to Figure 1.0 for envelope constraints	9830
o Gas Chromatograph	10.6 x 15.9 x 27.9	4100
o Hygrometer (Excluding Sensor)	2.6 x 10.7 x 13.3	330
o Wind Altitude Radar	15.0 x 33.0 x 19.0	8195

TABLE 1.1 SCIENCE MASS PROPERTIES (LARGE PROBE)

Instrument	Weight Lbs + 15% - 10%	Weight kg + 15% - 10%
Accelerometer(s)	2.5	1.15
Temperature Gauge(s)	0.9	0.4
Pressure Gauge(s)	0.65	0.3
Cloud Particle Size Analyzer	8.0	3.65
Solar Radiometer	5.0	2.25
IR Flux Radiometer	5.0	2.25
Neutral Mass Spectrometer	20.0	9.07
Gas Chromatograph	8.5	3.6
Hygrometer	1.1	.50
Wind Altitude Radar	8.8	4.0
Transponder	3.5	1.59
Large Probe Total	63.95 Lbs	28.76 kg

Table 1.2

TEMPERATURES OF EQUIPMENT PLATFORM
AND EQUIPMENT RING ASSEMBLY

Event	Time (hours)	Shelf	Ring
Aeroshell Separation	0	305 K	305 K
Chute Release	.65	312 K	310 K
	.94	315 K	324 K
Surface Impact	1.21	322 K	370 K

TABLE 1.3

LARGE PROBE - OUTSIDE DESCENT CAPSULE - (INSIDE AEROSHELL) - DYNAMIC ENVIRONMENTS

MISSION PHASE / ENVIRONMENT	LAUNCH AND TRANS VENUS INJECTION	INTER-PLANETARY CRUISE	VENUS ENCOUNT, PRESEP, SEP & PROBE CRUISE	VENUS ENTRY	DESCENT
DURATION	≈ 30 Min	≈ 91 Days	≈ 25 Days	≈ 41 Sec.	≈ 73 Min.
PROBE FUNCTIONAL MODE	NONOPERATE	NONOPERATE	*OPERATION **NONOPERATE	OPERATION	OPERATION
ACOUSTICS	142 db OAL	NEGLIGIBLE	*NEGLIGIBLE	137 db OAL	NEGLIGIBLE
VIBRATION	(NO SINE DURING OPERATION AND NO RAMDOM DURING NON-OPERATION)				
SINE	(1)	NEGLIGIBLE	**NEGLIGIBLE	NEGLIGIBLE	NEGLIGIBLE
RANDOM	NEGLIGIBLE	NEGLIGIBLE	*NEGLIGIBLE	6.1 g rms -20-300 HZ	NEGLIGIBLE
SHOCK					
OTHER THAN PYRO	NEGLIGIBLE	N/A	N/A	N/A	N/A
PYRO	1400 g @ 2000 HZ	N/A	N/A	3100-5400 g's 2000 -10,000 HZ	N/A
ACCELERATION					
LINEAR-THRUST AXIS	20 g Max.	N/A	N/A	N/A	N/A
SPIN (CENTRIFUGAL)	2.8 g per radius foot from spin axis	.008 per radius foot from spin axis	.008 g per radius foot from spin axis	.008 g per radius foot from spin axis	.008 g per radius foot from spin axis
DECELERATION					
LINEAR-THRUST AXIS	N/A	N/A	N/A	358 g	NEGLIGIBLE
-LATERAL	N/A	N/A	N/A	2 g	
DESPIN-(ANGULAR DECEL)					

(1) Axial: 1.5 g; 5-15 & 21-100 HZ
4.5 g; 15-21 HZ
Lateral: 1.5 g; 5-14 HZ
1.0 g; 14-100 HZ

TABLE 1.3 (CONTINUED)

LARGE PROBE - INSIDE PRESSURE SHELL - DYNAMIC ENVIRONMENTS

MISSION PHASE / ENVIRONMENT	LAUNCH AND TRANS VENUS INJECTION	INTER-PLANETARY CRUISE	VENUS ENCOUNT, PRESEP, SEP & PROBE CRUISE	VENUS ENTRY	DESCENT
DURATION	≈ 30 Min.	≈ 91 Days	≈ 25 Days	≈ 41 Sec.	≈ 73 Min.
PROBE FUNCTIONAL MODE	NONOPERATE	NONOPERATE	*OPERATION **NONOPERATE	OPERATION	OPERATION
ACCUSTICS	142 db OAL	NEGLIGIBLE	NEGLIGIBLE	142 db OAL	NEGLIGIBLE
VIBRATION					
SINE	1.5 to 4.5 g	NEGLIGIBLE	NEGLIGIBLE	NEGLIGIBLE	NEGLIGIBLE
RANDOM	NEGLIGIBLE	NEGLIGIBLE	NEGLIGIBLE	6.1 g rms	
SHOCK					
OTHER THAN PYRO	NEGLIGIBLE	N/A	N/A	N/A	N/A
PYRO	N/A	N/A	N/A	2000 g @ 3000 HZ	N/A
ACCELERATION					
LINEAR	16 g Max.	N/A	N/A	N/A	N/A
SPIN	2.8 g per radius foot from spin axis	.008 g per radius foot from spin axis	.008 g per radius foot from spin axis	.008 g per radius foot from spin axis	.008 g per radius foot from spin axis
SPIN UP					
DECELERATION					
LINEAR	N/A	N/A	N/A	350 g	N/A
DESPIN					

TABLE 1.3 (CONTINUED)

LARGE PROBE - OUTSIDE DESCENT CAPSULE - (INSIDE AEROSHELL) - STATIC ENVIRONMENTS

MISSION PHASE / ENVIRONMENT	LAUNCH AND TRANS VENUS INJECTION	INTER-PLANETARY CRUISE	VENUS ENCOUNT, PRESEP, SEP & PROBE CRUISE	VENUS ENTRY	DESCENT
DURATION	≈ 30 Min.	≈ 91 Days	≈ 25 Days	≈ 41 Sec.	≈ 73 Min.
PROBE FUNCTIONAL MODE	NONOPERATE	NONOPERATE	*OPERATION **NONOPERATE	OPERATION	OPERATION
TEMPERATURE					
BULK ATMOS. FOREBODY	294 ± 11K	265 to 541 K	*270 to 326K **256 to 340K	262 to 278K	
BULK ATMOS. AFTERBODY	294 ± 11K	177 to 422 K	*168 to 219K **156 to 205K	205 to 227K	
AEROSHELL INSIDE SURFACE — FOREBODY		255 to 370K			N/A
AEROSHELL INSIDE SURFACE — AFTERBODY					N/A
PRESSURE	N/m^2	10.13 kN/m^2	$1.333 \times 10^{-12} N/m^2$	.01 to 9.42 mN/m^2 ext. .10 mN/m^2 int	9.42 mN/m^2
HUMIDITY	N/A	N/A	N/A	N/A	N/A
ATMOSPHERIC COMPOSITION	AIR TO SPACE (RADIATIONS, METEOROIDS, ETC.)	SPACE (RADIATIONS, METEOROIDS, ETC.)	SPACE (RADIATIONS, METEOROIDS, ETC.)	CO_2: 95% by Vol Below turbopause: CO_2-97%, N_2-3% At turbopause: CO_2-95.99%, N_2-3%, He-0.01%, 0-1%	

TABLE 1.3 (CONTINUED)

LARGE PROBE - INSIDE PRESSURE SHELL - STATIC ENVIRONMENTS

ENVIRONMENT \ MISSION PHASE	LAUNCH AND TRANS VENUS INJECTION	INTER-PLANETARY CRUISE	VENUS ENCOUNT, PRESEP, SEP & PROBE CRUISE	VENUS ENTRY	DESCENT
DURATION	≈ 30 Min.	≈ 91 Days	≈ 25 Days	≈ 41 Sec.	≈ 73 Min.
PROBE FUNCTIONAL MODE	NONOPERATE	NONOPERATE	*OPERATION **NONOPERATE	OPERATION	OPERATION
TEMPERATURE: BULK ATMOSPHERE / INSIDE WALL	294 ± 11K	305K to 266K to 310K	**255 to 308K *264 to 308K	255 to 272K	272 to 336K
PRESSURE	103 kN/M^2	103 kN/m^2	103 kN/m^2	103 kN/m^2	103 kN/m^2
HUMIDITY	≈ 0% RH	≈ 0% RH	≈ 0% RH	≈ 0% RH	≈ 0% RH
ATMOSPHERIC COMPOSITION	DRY N_2	DRY N_2	DRY N_2	DRY N_2	DRY N_2

TABLE 1.3 (CONTINUED)

LARGE PROBE - OUTSIDE DESCENT CAPSULE - (INSIDE AEROSHELL) - STATIC ENVIRONMENTS

MISSION PHASE / ENVIRONMENT	LAUNCH AND TRANS VENUS INJECTION	INTER-PLANETARY CRUISE	VENUS ENCOUNT, PRESEP, SEP & PROBE CRUISE	VENUS ENTRY	DESCENT
DURATION	≈ 30 Min.	≈ 91 Days	≈ 25 Days	≈ 41 Sec.	≈ 73 Min.
PROBE FUNCTIONAL MODE	NONOPERATE	NONOPERATE	*OPERATION **NONOPERATE	OPERATION	OPERATION
RADIATION					
SOLAR	N/A	N/A	$14 \mu w/m^2$ @ EARTH TO $28 \mu w/m^2$ @ VENUS		
VENUS	N/A	N/A	≈ BLACK BODY AT A TEMP OF 700K	TBD	TBD
VENUS ALBEDO	N/A	N/A	See Fig. 1.1		
SPACE					

2.0 ELECTRICAL POWER AND CABLING

2.1 Hardware

The Large Probe Electrical Power Subsystem consists of a battery, power switches, fault protection elements, and pyrotechnic firing circuitry.

2.2 Power Sources

2.2.1 Ground Power. Ground power (via the ground checkout connector) and Probe battery power will be used as required for prelaunch activities. Use of Probe battery power will be limited during all testing phases and prohibited after final battery charge.

2.2.2 Cruise Power Source. Power during the cruise portion of the mission (launch insertion through Probe separation from the Probe Bus) will be supplied from the Probe Bus. The voltage shall be 28 VDC ±10%.

2.2.3 Post Separation Power. Power after Probe separation will be derived from the Probe battery.

2.3 Power Control

2.3.1 Voltage Control. The output voltage of the Probe is regulated by monovalent operation of the silver-zinc battery and shall be 28 VDC ± 10%.

2.3.2 Fault Protection. Power to science instruments will be fused individually at the levels shown in Table 2.2. The fuses will be located in the Probe power subsystem.

2.3.3 Switching. Instrument turn-on/off will be accomplished by switching of the branch circuit in the Probe power subsystem.

2.4 Probe Output Characteristics

2.4.1 Distribution. Each scientific instrument will receive electrical power through an individual, fused, branch circuit. The branch circuit

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will be energized/de-energized by Probe sequencer control. Power conversion at each instrument will be synchronized by a standard oscillator drive signal from the Probe. The power allotted to the instrument is measured at the spacecraft/instrument interface connector. See Table 2.1.

2.4.2 Voltage. Steady state power to the instruments will be 28 VDC $\pm$ 10%, as measured at the instrument terminals.

2.4.3 Noise and Ripple. Except for the transient voltage excursions specified below, the peak-to-peak amplitude of any voltage excursion, periodic or aperiodic, will not exceed 1.0 volt at any frequency between 30 Hz and 10.0 KHz decreasing at 6 dB/octave to 0.5 volts at 20.0 KHz and remaining at 0.5 volts through 100 MHz.

2.4.4 Transient Voltage Excursions

(a) Performance - The instruments shall be designed to accommodate without performance degradation, voltage transients up to +42 VDC or down to +18 VDC for durations of 10 microseconds or voltages down to +20 VDC for durations of 500 milliseconds on the nominal +28 VDC bus.

(b) Damage - The instruments shall be designed so that no damage, long term degradation, or modes where proper performance is not automatically resumed when the transient is removed, shall occur when 10 microsecond voltage transients up to +56 VDC or down to 0 VDC are seen on the nominal +28 VDC bus.

2.4.5 Source Impedance. The source impedance will not exceed 0. 5 ohms.

2.4.6 Other Voltages - None defined.

2.5 Instrument Load Characterstics

2.5.1 Load Current

2.5.1.1 Average - See Table 2.2.

2.5.1.2 Peak - See Table 2.2.

2.5.2 Duty Cycle. Not applicable.

2.5.3 Inrush Current. Upon application of the 28 vdc primary power to the instrument's primary power circuit the instrument's load current surge shall not exceed the larger of 400 milliamperes or 100 percent above the steady state requirement. The current shall return to steady-state $\pm$ 10% within 10 milliseconds.

2.5.4 Load Noise. When operating from a power source impedance of 1.0 ohms, no single instrument shall feed back into the power input circuit any electrical noise either periodic or aperiodic in excess of 560 millivolts peak-to-peak at any frequency between 30 Hz and 10.0 KHz decreasing at 6 dB/octave to 280 millivolts peak-to-peak at 20.0 KHz and remaining at 280 millivolts peak-to-peak or less from 20.0 KHz through 100 MHz.

2.5.5 Grounding. The input power supply circuit shall be isolated from chassis ground within each instrument by at least 10 megohms at 500 VDC.

2.5.6 Bonding. All instrument cases shall be bonded to probe structure and shall have a maximum permissible bonding impedance of 10 milliohms. Bonding of science instruments shall be in accordance with MIL-B-5087 (latest revision).

2.6 Connectors and Cabling

2.6.1 Connector Types. The connector installed on an instrument that connects the instrument to the spacecraft harness shall be a male (straight or coaxial insert) pin connector selected from the Cannon non-magnetic series of connectors (NMC-A-106 suffix). The use of two identical connectors on any one box is prohibited. Keying shall be provided. Use of end pins shall be avoided, if possible.

2.6.2 Interface Connector Location. The centerline of the interface connector shall be located as defined in Section 1.0, Figure 1.0.

2.6.3 Test Connector Location. The location of test connector (if required) shall be defined in Section 1.0., Figure 1.0.

2.6.4 Connector Pins. Cannon non-magnetic pins shall be used. Connector savers shall be provided for each instrument connector. The pin assignments shall be in accordance with Table 2.1.

2.6.5 Connector Encapsulation. If encapsulated, the encapsulation shall be done with mated connectors to insure proper pin alignment. ARC shall be notified of the use of any encapsulated interface connectors.

2.6.6 Connector Identification. Each connector shall be identified by a number in accordance with Section 1.0., Figure 1.0.

2.6.7 Wiring. Wires between any two units of one instrument shall be twisted pairs whenever possible. Shielded twisted pairs, single conductor, or coaxial cables may be used if necessary, except that coaxial cables shall be avoided where the wiring between the two units must pass through Probe structural interfaces. The maximum wire size will be No. 20 AWG, and the minimum wire size (except for coaxial cables) will be No. 24 AWG.

ABBREVIATIONS

For

TABLE 2.1

DHC	-	Data Handling & Command Unit
PCU	-	Power Control Unit
TP	-	Twisted Pair wire
STP	-	Shielded Twisted Pair wire
TLM	-	Telemetry
MUX	-	Multiplexer (analog data)
CPSA	-	Cloud Particle Size Analyzer
SFR	-	Solar Flux Radiometer
PFR	-	Planetary Flux Radiometer
NMS	-	Neutral Mass Spectrometer
GC	-	Gas Chromatograph

Table 2.1

ELECTRICAL WIRING LIST
LARGE PROBE/INSTRUMENT INTERFACE

Item	Type of Signal	Source	Connector Pin	Load or Destination	Connector Pin
TEMP GAUGE					
Converter Sync	Signal	Probe PCU		Temp Gauge	
+28 VDC Power	Power	Probe PCU		Temp Gauge	
28 VDC Return	Power	Probe PCU		Temp Gauge	
Analog TLM, Temp	Telemetry	Temp Gauge		Probe DTU	
Analog TLM, Thermistor	Telemetry	Temp Gauge		Probe DTU	
Analog TLM Return	Telemetry	Temp Gauge		Probe DHC	
Temp Sensor No. 1 Signal	Signal	Temp Sensor		Temp Gauge	
Temp Sensor No. 1 Return	Signal	Temp Sensor		Temp Gauge	
Temp Sensor No. 2 Signal	Signal	Temp Sensor		Temp Gauge	
Temp Sensor No. 2 Return	Signal	Temp Sensor		Temp Gauge	
PRESSURE GAUGE					
Sync	Signal	Probe PCU		Press Gauge	
+28 VDC Power	Power	Probe PCU		Press Gauge	
28 VDC Return	Power	Probe PCU		Press Gauge	
Analog TLM, Press	Telemetry	Press Gauge		Probe DTU	
Analog TLM, Thermistor	Telemetry	Press Gauge		Probe DTU	
Analog TLM, Return	Telemetry	Press Gauge		Probe DTU	

Table 2.1

ELECTRICAL WIRING LIST
LARGE PROBE/INSTRUMENT INTERFACE

Item	Type of Signal	Source	Connector Pin	Load or Destination	Connector Pin
ACCELEROMETER					
Converter Sync	Signal	Probe PCU		Accelerometer	
+28 VDC Power	Power	Probe PCU		Accelerometer	
28 VDC Return	Power	Probe PCU		Accelerometer	
Analog TLM, axial	Telemetry	Accelerometer		Probe DTU	
Analog TLM, backup	Telemetry	Accelerometer		Probe DTU	
Analog TLM, Turbulence	Telemetry	Accelerometer		Probe DTU	
Analog TLM, lateral 1	Telemetry	Accelerometer		Probe DTU	
Analog TLM, lateral 2	Telemetry	Accelerometer		Probe DTU	
Analog TLM, return	Telemetry	Accelerometer		Probe DTU	
Checkout No. 1	Ground Checkout	Accelerometer		Checkout Connector	
Checkout No. 2	Ground Checkout	Accelerometer		Checkout Connector	
Checkout No. 3	Ground Checkout	Accelerometer		Checkout Connector	
Checkout No. 4	Ground Checkout	Accelerometer		Checkout Connector	
Checkout return	Ground Checkout	Accelerometer		Checkout Connector	
CLOUD PARTICLE SIZE ANALYZER					
Converter Sync	Signal	Probe PCU		CPSA	
+28 VDC Power	Power	Probe PCU		CPSA	
28 VDC Return	Power	Probe PCU		CPSA	
Serial Digital Data	Telemetry	CPSA		Probe DTU	

Table 2.1

ELECTRICAL WIRING LIST
LARGE PROBE/INSTRUMENT INTERFACE

Item	Type of Signal	Source	Connector Pin	Load or Destination	Connector Pin
Cloud Particle Size Analyzer (continued)					
Bit Clock	Timing	Probe DTU		CPSA	
Word Gate	Timing	Probe DTU			
SOLAR FLUX RADIOMETER					
Converter Sync	Signal	Probe PCU		SFR	
+28 VDC Power	Power	Probe PCU		SFR	
28 VDC Return	Power	Probe PCU		SFR	
Data Readout Sync	Timing	Probe DTU		SFR	
Timing Pulses (1 second)	Timing	Probe DTU		SFR	
Serial Digital Data	Telemetry	SFR		Probe DTU	
Bit Clock	Timing	Probe DTU		SFR	
Word Gate	Timing	Probe DTU		SFR	
IR FLUX RADIOMETER					
Converter Sync	Signal	Probe PCU		IRFR	
+28 VDC Power	Power	Probe PCU		IRFR	
28 VDC Return	Power	Probe PCU		IRFR	
Serial Digital Data	Telemetry	IRFR		Probe DTU	
Bit Clock	Timing	Probe DTU		IRFR	
Word Gate	Timing	Probe DTU		IRFR	

Table 2.1

ELECTRICAL WIRING LIST
LARGE PROBE/INSTRUMENT INTERFACE

Item	Type of Signal	Source	Connector Pin	Destination	Connector Pin
HYGROMETER					
Converter Sync	Signal	Probe PCU		Hygrom.	
+28 VDC Power	Power	Probe PCU		Hygrom.	
28 VDC Return	Power	Probe PCU		Hygrom.	
Analog TLM Humidity	Telemetry	Hygrom.		Probe DTU	
Analog TLM Range	Telemetry	Hygrom.		Probe DTU	
Analog TLM Housekeeping	Telemetry	Hygrom.		Probe DTU	
Analog TLM, return	Telemetry	Hygrom.		Probe DTU	
Humidity Sensor No. 1, High	Signal	Hygrom. Sensor		Hygrom.	
Humidity Sensor No. 1, Return	Signal	Hygrom. Sensor		Hygrom.	
Humidity Sensor No. 2, High	Signal	Hygrom. Sensor		Hygrom.	
Humidity Sensor No. 2, Return	Signal	Hygrom. Sensor		Hygrom.	
Temp Sensor, High	Signal	Hygrom. Sensor		Hygrom.	
Temp Sensor, Return	Signal	Hygrom. Sensor		Hygrom.	
Sensor Power	Power	Hygrom.		Hygrom. Sensor	
Sensor Power Return	Power	Hygrom.		Hygrom. Sensor	
NEUTRAL MASS SPECTROMETER					
Converter Sync	Signal	Probe PCU		NMS	
+28 VDC Power	Power	Probe PCU		NMS	
28 VDC Return	Power	Probe PCU		NMS	
Serial Digital Data	Telemetry	NMS		Probe DTU	
Bit Clock	Timing	Probe DTU		NMS	
Word Gate	Timing	Probe DTU		NMS	

Table 2.1

ELECTRICAL WIRING LIST
LARGE PROBE/INSTRUMENT INTERFACE

Item	Type of Signal	Source	Connector Pin	Destination	Connector Pin
Neutral Mass Spectrometer (continued)					
Pyro Fire No. 1 High	Pyrotechnic Fire	Probe PCU		NMS	
Pyro Fire No. 1 Return	Pyrotechnic Fire	Probe PCU		NMS	
Pyro Fire No. 2 High	Pyrotechnic Fire	Probe PCU		NMS	
Pyro Fire No. 2 Return	Pyrotechnic Fire	Probe PCU		NMS	
Pyro Fire No. 3 High	Pyrotechnic Fire	Probe PCU		NMS	
Pyro Fire No. 3 Return	Pyrotechnic Fire	Probe PCU		NMS	
Pyro Fire No. 4 High	Pyrotechnic Fire	Probe PCU		NMS	
Pyro Fire No. 4 Return	Pyrotechnic Fire	Probe PCU		NMS	
Pyro Fire No. 5 High	Pyrotechnic Fire	Probe PCU		NMS	
Pyro Fire No. 5 Return	Pyrotechnic Fire	Probe PCU		NMS	
Pyro Fire No. 6 High	Pyrotechnic Fire	Probe PCU		NMS	
Pyro Fire No. 6 Return	Pyrotechnic Fire	Probe PCU		NMS	
Pyro Fire No. 7 High	Pyrotechnic Fire	Probe PCU		NMS	
Pyro Fire No. 7 Return	Pyrotechnic Fire	Probe PCU		NMS	
Pyro Fire No. 8 High	Pyrotechnic Fire	Probe PCU		NMS	
Pyro Fire No. 8 Return	Pyrotechnic Fire	Probe PCU		NMS	
Pyro Fire No. 9 High	Pyrotechnic Fire	Probe PCU		NMS	
Pyro Fire No. 9 Return	Pyrotechnic Fire	Probe PCU		NMS	
Pyro Fire No. 10 High	Pyrotechnic Fire	Probe PCU		NMS	
Pyro Fire No. 10 Return	Pyrotechnic Fire	Probe PCU		NMS	
Pyro Fire No. 11 High	Pyrotechnic Fire	Probe PCU		NMS	
Pyro Fire No. 11 Return	Pyrotechnic Fire	Probe PCU		NMS	

Table 2.1

ELECTRICAL WIRING LIST
LARGE PROBE/INSTRUMENT INTERFACE

Item	Type of Signal	Source	Connector Pin	Destination	Connector Pin
WIND ALTITUDE RADAR					
Converter Sync	Signal	Probe PCU		WA Radar	
+28 VDC Power	Power	Probe PCU		WA Radar	
28 VDC Return	Power	Probe PCU		WA Radar	
Serial Digital Data	Telemetry	WA Radar		Probe DTU	
Bit Clock	Timing	Probe DTU		WA Radar	
Word Gate	Timing	Probe DTU		WA Radar	
Analog TLM, Voltage	Telemetry	WA Radar		Probe DTU	
Analog TLM, Temp	Telemetry	WA Radar		Probe DTU	
Analog Return	Telemetry	WA Radar		Probe DTU	
Radar Antenna Signal	Signal	Antenna		WA Radar	
GAS CHROMATOGRAPH					
Converter Sync	Signal	Probe PCU		GC	
+28 Vdc Power	Power	Probe PCU		GC	
28 Vdc Return	Power	Probe PCU		GC	
Serial Digital Data	Telemetry	GC		Probe DTU	
Bit Clock	Timing	Probe DTU		GC	
Word Gate	Timing	Probe DTU		GC	

Table 2.2

INSTRUMENT LOAD CHARACTERISTICS

INSTRUMENT	FUSE RATING (Amps)	VOLTAGE (Volts)	AVERAGE CURRENT (Amps)	PEAK CURRENT (Amps)
TEMP. GAUGE	1/8	28 ± 10%	0.018	
PRESSURE GAUGE	1/8	28 ± 10%	0.018	
ACCELEROMETERS	3/8	28 ± 10%	0.082	0.2 Amp at 400 "g" for 10 seconds
WIND ALTITUDE RADAR	5	28 ± 10%	1.43	
NEUTRAL MASS SPECTROMETER	2	28 ± 10%	0.43 (max)	
CLOUD PARTICLE SIZE ANALYZER	2	28 ± 10%	0.72	
SOLAR FLUX RADIOMETER	3/8	28 ± 10%	0.14	
PLANETARY FLUX DETECTOR	3/8	28 ± 10%	0.11	
GAS CHROMATOGRAPH	1/2	28 ± 10%	0.22	
HYGROMETER	1/16	28 ± 10%	0.009	

Notes: Fuse type is Littlefuse 256 series, Picofuse

3.0 DATA HANDLING AND COMMAND (DHC)

3.1 Functional Description

The Large Probe DHC will accept information in digital, analog, or state form, convert the analog information to digital form, and arrange all information in an appropriate format for time multiplexed transmission to Earth or storage on board the Probe. The Probe will also supply the instruments with various timing and operational status signals and functional commands.

3.1.1 Telemetry Word. A telemetry word in all formats will consist of 7, 8 or 10 bits. Probe generated words will be transmitted with the most significant bit first.

3.1.2 Data Bit Rates. The DHC will be capable of processing scientific and engineering data at bit rates of 512 and 128 bits/second. The 512 bit/s rate is used for the Checkout Mode. The 128 bit/s rate is used for Accelerometers during Blackout, and data are collected and transmitted at 128 bits/second during Terminal Descent Phase.

Bit rate changes will occur within one bit period following the completion of the current data frame after the reception of a bit rate change command by the data system sequencer.

3.1.3 Frame. The data subsystem will assemble the information from the instruments into 768 bit frames. Frame contents are summarized in Figures 3.1 through 3.4.

3.1.4 Format and Word Assignments. The words in a frame are assigned in several formats. The formats are organized for particular probe operational modes and are selected by internal sequencer command. Upon the reception of a format change command by the data subsystem sequencer, format changes will occur within one bit period following the completion of the current frame. Checkout will be accomplished by exercising all of the flight formats except that no pyro events will be initiated.

3.1.5 Format A. Format A is the first format for scientific information and is used during pre-entry and post-landing. Word assignments for Format A are shown in Figure 3.1.

3.1.6 Format B. Format B is used during RF blackout and for a short period immediately thereafter. Word assignments for Format B are shown in Figure 3.2.

3.1.7 Formats D-1 and D-2. Formats D-1 and D-2 are descent formats. Two formats are used to accommodate different science requirements for the high and low altitude portions of the descent. Word assignments for Format D-1 and D-2 are shown in Figure 3.3 and 3.4.

3.1.8 Operational Modes of Data Subsystem. The data subsystem will be capable of operating in three basic modes as follows:

(a) Data are stored.

(b) Data are transmitted real time and stored simultaneously.

(c) Data are transmitted real time and interleaved with the transmittion of stored data.

3.1.9 On-Board Storage Capacity. A storage capacity of 5120 bits will be provided by the data subsystem. Storage is allocated for the accelerometer for entry and blackout.

3.2 Signals from Scientific Instruments

3.2.1 Telemetry List. Table 2.1 lists the data signals from the instruments.

3.2.2 Characteristics. The characteristics of digital, analog, or state signals from the scientific instruments to the data subsystem shall be:

(a) Digital Signals - shall have the following characteristics (reference Figure 3.4).

(1) Amplitude

Logic 0 (false)	0 to 0.4 vdc
Logic 1 (true)	2.4 to 5.5 vdc

(2) Rise and Fall Time (between 10% and 90% points) - less than 1.0 microsecond.

(3) Loading - SN5414 devices shall be used to receive digital signals. Interconnecting cables shall have a maximum capacitance of 200 pico-farads.

(4) Duration (between 50% points) - Minimum duration of digital signals to the Probe DHC shall be 100 microseconds.

(5) Word Length - Digital words may be either 7, 8, or 10 bits.

(6) Signals shall be referenced to chassis.

(7) Overvoltages will result in hardware damage.

(8) A short circuit can be accepted between any two digital signal lines or between signal lines and ground without damage or effect upon other channels.

(b) Analog Signals

(1) Full scale input range shall be 0 to 5 volts.

(2) All input signals shall be referenced to a signal return, which shall be returned to the DHC.

(3) The DHC channel input impedance during sampling periods will be 30K ohm minimum, shunted by an input impedance of 350 picofarads, maximum. Maximum back current during non-sampling periods shall be 100 nanoamperes.

(4) Overvoltage - Voltages up to ± 10 volts can be accepted between any two signal inputs without damage or effect upon other channels. Valid data will not be provided from the affected input channels during application of such over-voltage.

(5) Short Circuit - A short circuit can be accepted between any two signal inputs or between any input and ground without damage or effect on other channels.

(6) A/D Conversion - An analog signal received by the data subsystem will be converted into a 7-bit or 10-bit digital word as required.

(7) Coding Accuracy - For a source impedance of less than 5000 ohms the overall conversion accuracy will be $\pm$ 1 bit. For larger source impedances the error will be increased.

(8) Multiplexing - All channel multiplexing will be done in the central data handling system.

(c) State or Bilevel Signals

(1) Input levels between 0 and 1.2 volts will be encoded as a logical "zero." Input levels between 2 and 10 volts will be encloded as a logical "one."

(2) All input signals shall be referenced to chassis.

(3) Channel input impedance will be 30 K ohms or greater.

(4) Overvoltages up to 10 volts can be accepted between any two signal inputs without damage or effect on other channels. Valid data will not be provided from the affected channels during application of the overvoltage.

(5) Short Circuit - A short circuit can be accepted between any two signal inputs or between any input and ground without damage or effect upon other channels.

(6) Up to eight channel inputs will be grouped into one byte.

3.3 Signals to Scientific Instruments

The DHC shall supply digital timing and command signals to the instruments as identified in Table 2.1. Characteristics of these signals are as follows:

(1) Amplitude

Logic 0 (false)	0 to 0.4 Vdc
Logic 1 (true)	2.4 to 5.5 Vdc

(2) Rise and Fall time (between 10% and 90% points) - less than 1.0 microsecond.

(3) Loading - SN5414 is the recommended load device. Other loads shall have a maximum shunt capacitance of 350 picofarads and a resistance of between 1000 and 10,000 ohms. Interconnecting cables shall have a maximum capacitance of 200 picofarads.

(4) Duration (between 50% points)

Bit Clock	Prevailing bit rate at 50 $\pm$ 5% duty cycle.
Word Gate	10 millisecond pulse
Frame Rate	10 millisecond pulse
Command Signals	As required.

E-2 HRS	E-10 MIN	E	T-14.85 sec	T-7.5 sec
PRE-ENTRY	Format A		ENTRY	

E-2 HRS: Turn on IR Flux reference body heater
E-10: Electronics warmup - axial accel. data begins
E : Entry (alt. - 250 Km)
T-14.8 sec: 4×10^{-4} g's, alt. $\approx$ 140 Km
T-7.5 sec: Blackout begins (g = 0.5)

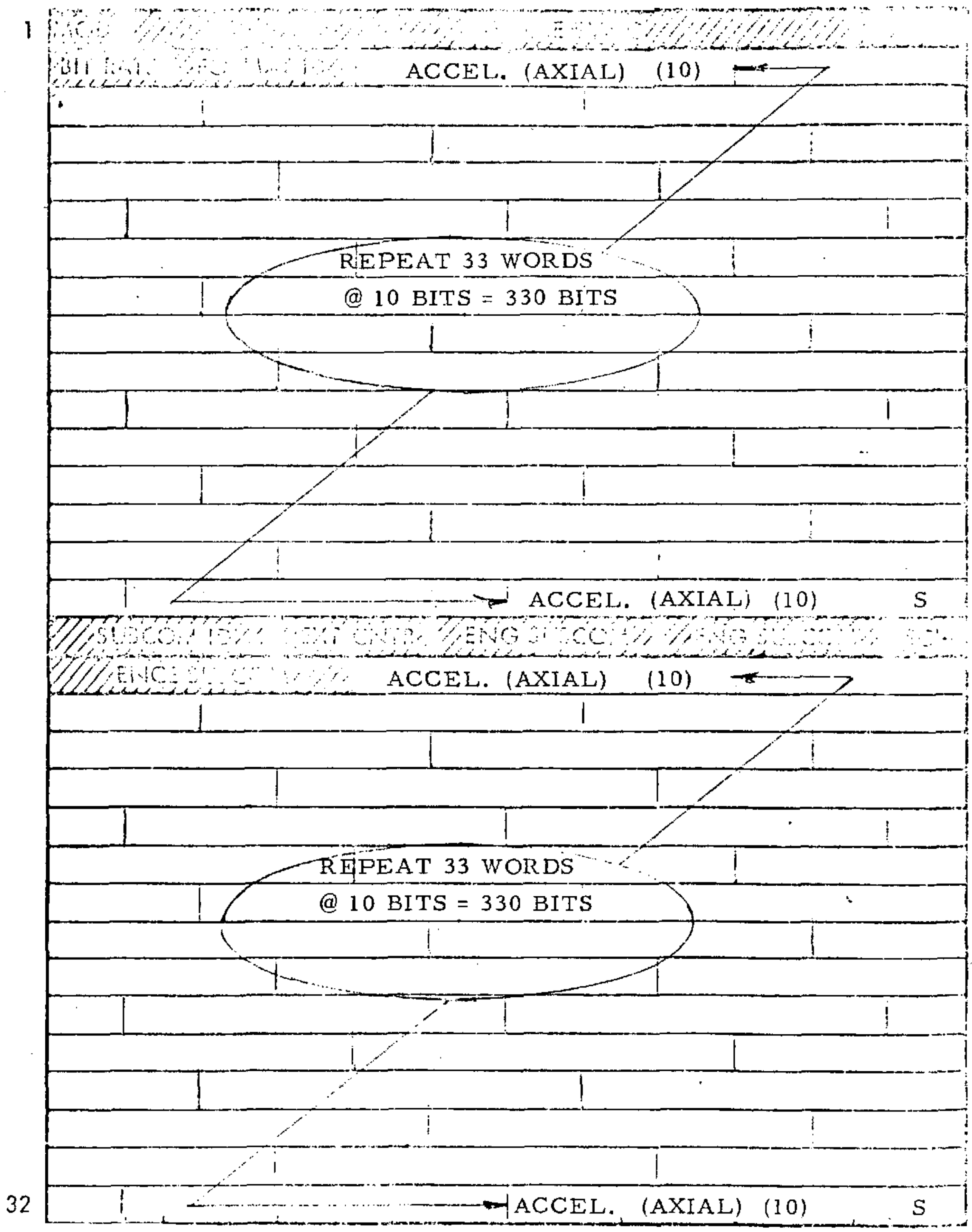


Figure 3.1 LARGE PROBE FORMAT A

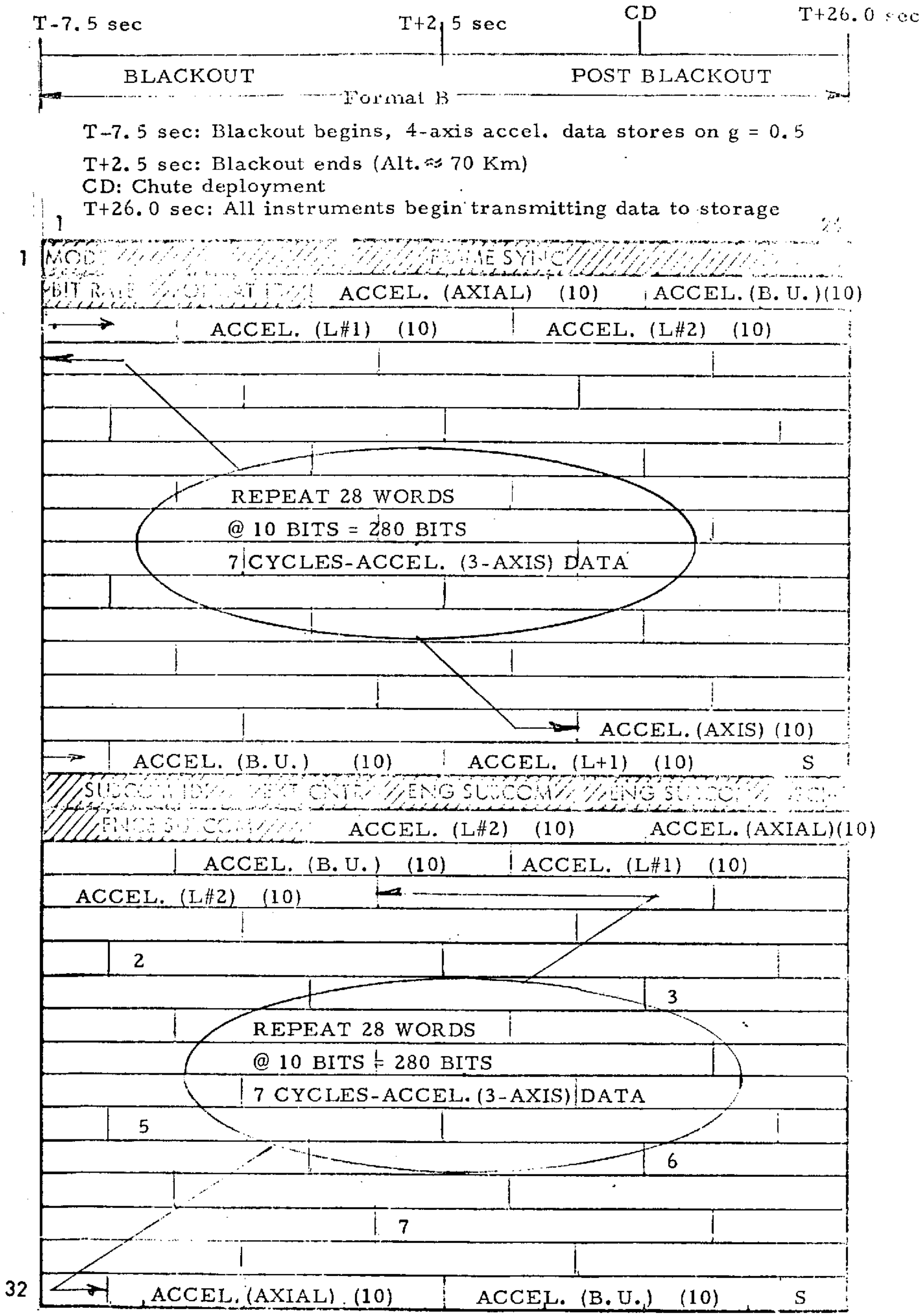
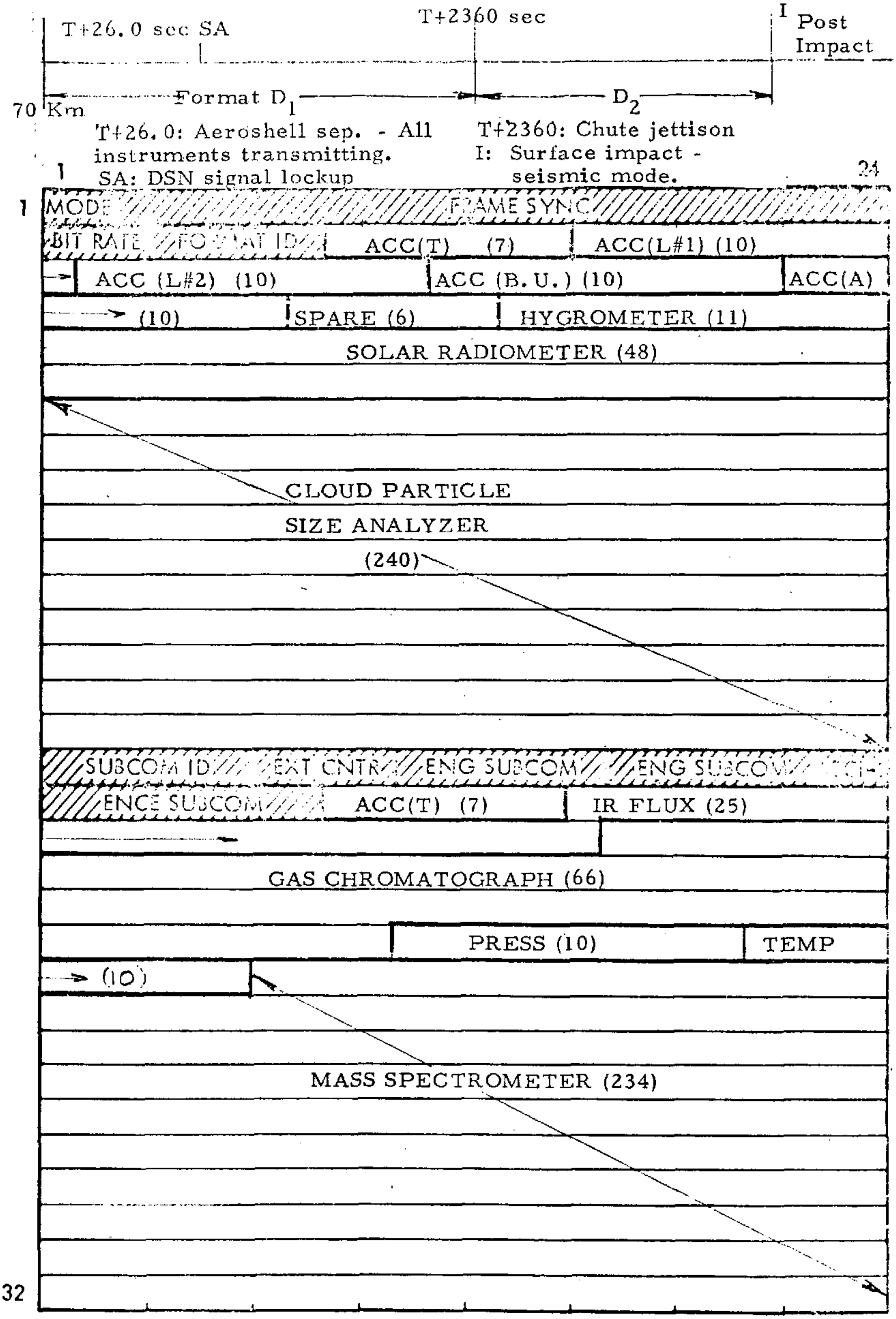
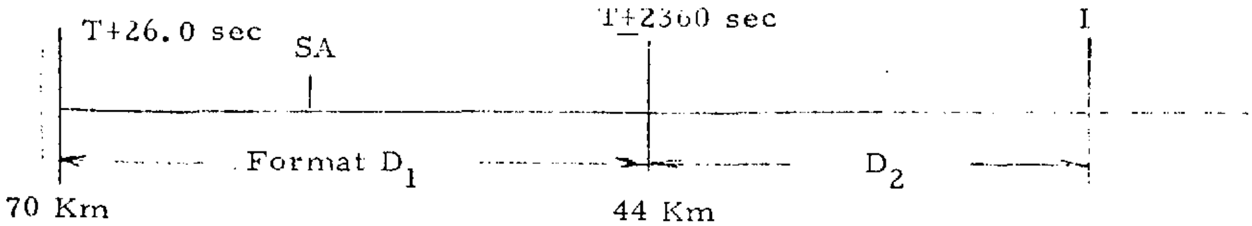


Figure 3.2 LARGE PROBE FORMAT B



Note: Gas Chromatograph will be used to dump memory for 10 min. Mass Spectrometer space will be empty for first 10 frames (1 min.).

Figure 3.3 LARGE PROBE FORMAT D.



T+26.0 sec: Aeroshell separation-all instruments transmitting
SA: DSN signal lockup
T+2360: Chute jettison
I: Surface impact-seismic mode (T-4392 sec)

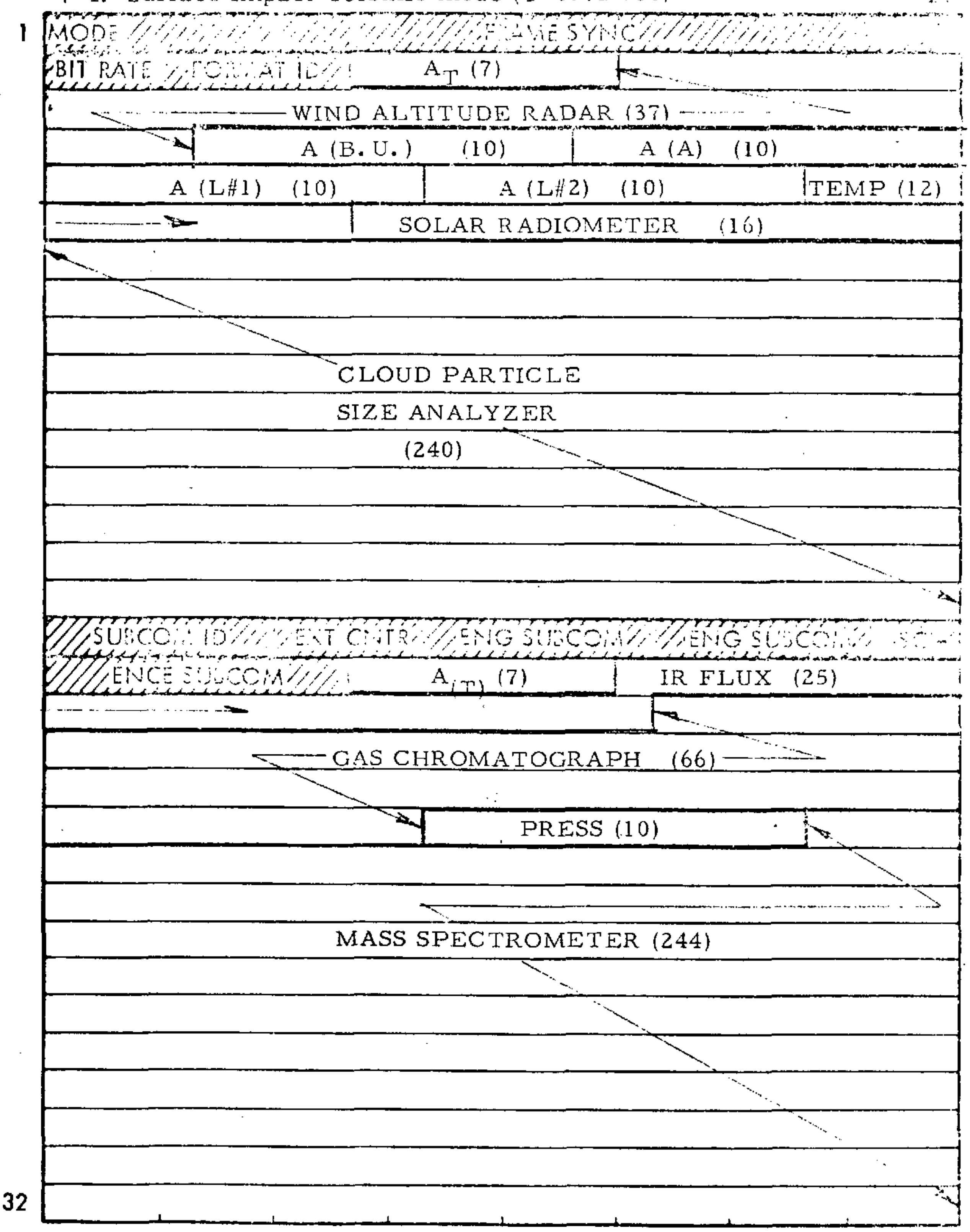


Figure 3.4 LARGE PROBE FORMAT D_2

LEGEND

FC = Frame Count

FS = Frame Synch

P_RID = Probe I. D.

MASS SPEC = Mass Spectrometer

T_1, T_2 = Temperature Gauge 1 and 2

E = Engineering/Housekeeping

P_1, P_2 = Pressure Gauge 1 and 2

S = Spare

HYGR = Hygrometer

SOL RAD - Solar Flux Radiometer

CPSA - Cloud Particle Size Analyzer

ACC = Accelerometer

PLAN FLUX = Planetary Flux Radiometer

SUB = Calibration Subcommutator

WAR = Wind Altitude Radar

GC = Gas Chromatograph

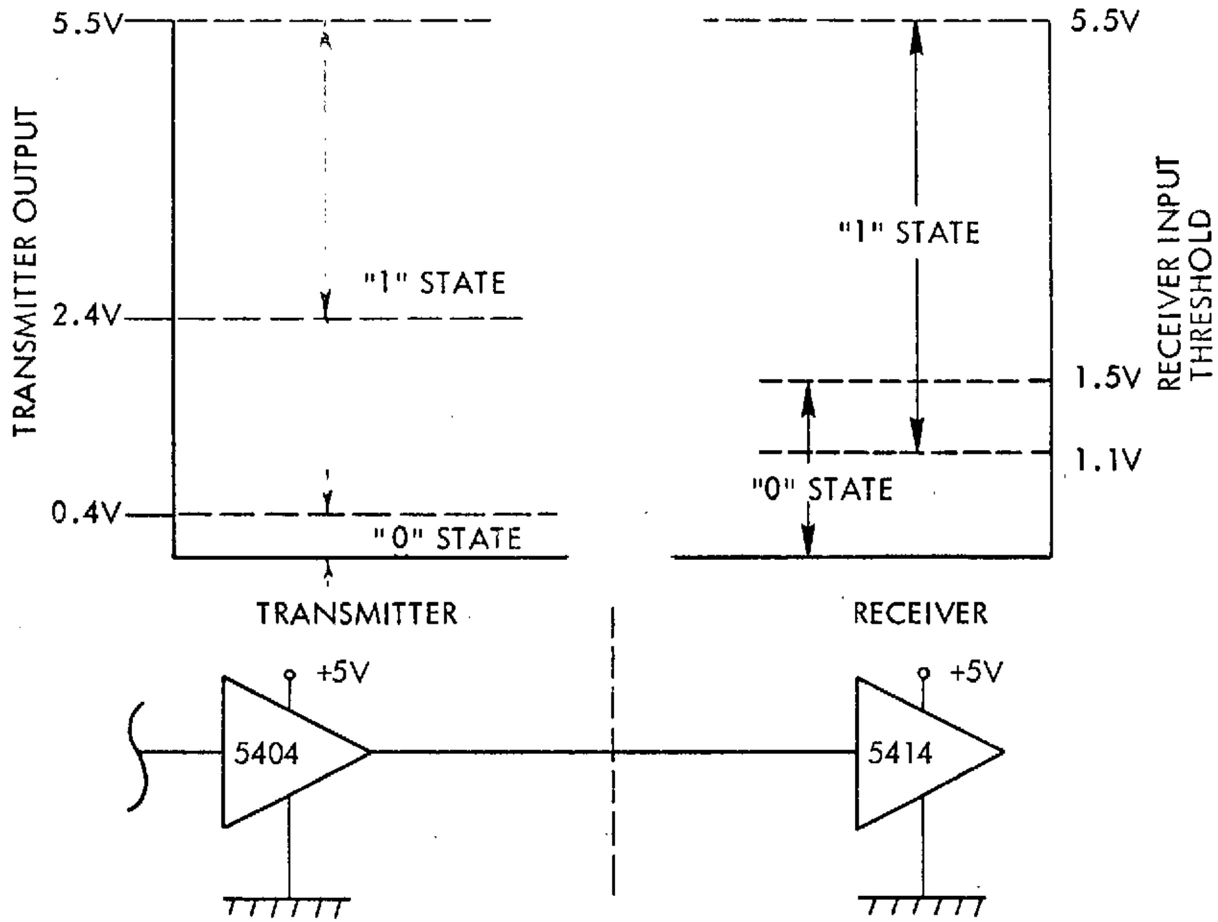

5.5V
2.4V
0.4V
TRANSMITTER OUTPUT
"1" STATE
"0" STATE
5.5V
1.5V
1.1V
RECEIVER INPUT
THRESHOLD
"1" STATE
"0" STATE
TRANSMITTER
RECEIVER
+5V
5404
+5V
5414

FIGURE 3.4

4.0 THERMAL

4.1 Thermal Control

Thermal control is provided by thermal insualtion, coatings, and science window heaters on the descent capsule and the aeroshell heat shield to maintain an environment assuring that all Probe components are within their temperature limits for all mission phases.

The Large Probe temperature limits interior and exterior to the pressure vessel as a function of the mission phase are given in Table 4.1, under both operating and non-operating conditions. The scientific instruments must be capable of proper operation, as applicable for either the operate/non-operate mode, per the limits specified in Table 4.1.

TABLE 4.1

LARGE PROBE TEMPERATURE LIMITS

MISSION PHASE	INTERIOR TO PRESSURE VESSEL	EXTERIOR TO PRESSURE VESSEL
Pre-launch (Operating)	256 to 305K	256 to 325K
Pre-launch (Non-operating)	256 to 302K	227 to 344K
Launch and Cruise (Non-operating)	256 to 302K	227 to 344K
Cruise (Operating)	256 to 305K	256 to 325K
Descent (Operating)	305 to 322K	256K to *

* Each exterior component must be designed with upper temperature limit consistent with maximum atmospheric temperature for which it is intended to operate.

5.0 ELECTROMAGNETIC INTERFERENCE

Spacecraft equipment and scientific instruments shall comply with the electromagnetic interference requirements as given by specification using applicable requirements of MIL-STD-461. Each probe subsystem is electromagnetically compatible with all other probe subsystems, scientific instruments, and the probe bus spacecraft as specified in (TBS) and the associated ground support equipment. Electromagnetic compatibility requires:

(a) The normal operation of each probe subsystem shall not be adversely affected by signals or voltage variations generated by other subsystems or scientific instruments as part of their normal operating mode or intended function.

(b) No probe subsystem shall disturb normal operations of other subsystems or scientific instruments by emission of signals or voltage variations other than those produced to perform its intended function.

6.0 MISCELLANEOUS

6.1 Spin Requirement

6.1.1 Spin Rate. The large probe design will include canted aerodynamic ducts through a stabilization ring assembly for the purposes of imparting a spin of approximately 10 π rad/km to the probe. The maximum spin rate for the probe will be 2.09 rad/s which occurs immediately after the aeroshell jettison at 70 km altitude where the descent velocity is 56 meters per second.

6.1.2 Spin Direction. Either clockwise or counterclockwise spin direction is acceptable. The spin direction of the large probe will be in the same direction as that of the probe bus. The spin of the large probe will be about the geometric axis of symmetry.

6.1.3 Spacecraft Axis Notation. The spacecraft axis notation is as that presented in Figure 1.0.

6.2 Probe Stability

The large probe is aerodynamically stable in the sense that a completely passive body can be stable; i.e., when it is hit by a disturbance it will respond to the disturbance but the amplitude of the response will damp out. The angle of attack response for the descent capsule is given by

$$\alpha_{TF} = K_0 e^{-t/\tau} + K_1 e^{-t \xi \omega_n} \sin\left(\omega_n \sqrt{1 - \xi^2}\ t - \phi\right)$$

The first term represents the attitude offset introduced by a wind shear and has a time constant $\tau \sim$ 5-15sec. The second term represents the damped oscillations about this offset with damping time to half amplitude, $0.693\ (1/\xi\omega_n) \sim 3\text{-}5_{sec}$ after parachute jettison. The oscillation period $2\pi/\omega_n \sim 0.5_{sec}$ before parachute jettison and 4.5-6 sec afterwards; the offset attitude achieved by the probe for a constant wind shear is proportional to the probe terminal velocity. For a wind shear as given in NASA SP-8011 of $dv_\omega/dh = 0.05$ (m/s)/m the offset will range from 3^o-15^o during the descent.

With small wind gradients such as the 0.05 (m/s)/m the first term is dominant i.e. $K_0 \gg K_1$ but if wind gusts approached 1-10 (m/s)/m then $K_1 \gg K_0$ i.e. the oscillatory term is dominant.

PART III

PRELIMINARY

PIONEER VENUS

SCIENTIFIC INSTRUMENT-SMALL PROBE
INTERFACE DOCUMENT

MARCH 23, 1973

SCOPE:

This document defines the characteristics of the Pioneer Venus Small Probe spacecraft pertinent to the scientific instruments and the requirements of the Small Probe on the scientific instruments.

1.0 MECHANICAL

1.1 Configuration and Dimensions

The dimensions of the instruments shall be compatible with the overall size allocation for the small probe.

The approximate size allocation for each experiment instrument shall be as given in Table 1.0. These data are specifically for that instrument complement as presently defined for the small probe. Changes to the instrument complement are anticipated and can be accommodated.

The small probe descent capsule showing the location for the various experiment instruments is given in Figure 1.0.

1.2 Mass Properties

The total weight available for experiments on each small probe is 2.2 Kg (4.9 lbs.). For an instrument complement as shown in Table 1.1, the approximate weight allocations are as listed. The weight for each of the instruments shall not exceed those weights listed in Table 1.1. The specified weights do not include any connectors and cabling between the experiment boxes.

1.3 Mounting Technique

To minimize heat leakage into the probe it is preferred that instruments not be mounted physically to the pressure vessel but be mounted in intimate contact with the equipment platform. It is recognized that some instruments will have elements that need to be tied structurally to the external and internal pressure vessel surfaces. The mounting of each instrument will be specified within and controlled by the Project Office through the instrument ICD.

1.4 Mechanical Attachment

All the instruments are mounted on an equipment platform as shown in Figure 1.0. The windows for the Nephelometer and IR Flux Detector are mounted on the pressure shell. The deployment mechanism for the IR Flux mirror, temperature gauge, and nephelometer are mounted in the aeroshell. The temperature sensor and IR Flux mirror will be mounted in a spring loaded deployable mechanism which will eject a plug from the aeroshell or afterbody cover at the time of sensor deployment. The entrance tube for the pressure gauge is mounted at the apex of the heat shields.

1.5 Thermal Attachment

The thermal characteristics of the mechanical attachment shall be designed to promote heat transfer between the instruments and the equipment platform. The average temperature achieved by the interior system at surface impact will be 322 K and the pressure shell temperature will reach 405 K.

1.6 Windows and Feed-Throughs

1.6.1 Window Mechanical Interfaces. All windows will consist of an inner and an outer lens element. The outer element will be extended beyond the probe insulation. The inner element will be located just outboard of the walls of the pressure vessel. Each element is capable of withstanding the pressure differential that would exist at the Venus surface.

The small probe instrument windows are associated with the nephelometer instrument. A cylindrical window is used for the transmitted laser beam path. A conical window is used for detector observation of scattered light. The windows employed in the small probe are shown in Figure 1.2.

Heating for all windows in the small probe will be supplied by the probe to minimize deposits of any condensation products. Window heater power will be turned on at approximately 70 km altitude and will remain on until probe impact.

1.6.2 Mechanical Feed-Throughs. A pressure vessel penetration will be provided for the inlet to the pressure gauges as shown in Figure 1.0.

1.7 Dynamic and Static Environments

The significant dynamic and static environments to be encountered by the science instruments occur during (a) launch and trans-Venus injection, (b) interplanetary cruise, (c) Venus encounter, pre-separation and probe cruise, (d) Venus entry and (3) descent. The Dynamic and Static environments are summarized in Table 1.2.

1.8 Alignment

Instrument mounting surfaces will be held to alignment tolerances of ± 0.5 degree with respect to the probe coordinate system. Instruments requiring closer alignments will be assigned individual mounting provisions, or the instrument shall provide individual capability to satisfy the requirements. Equipment mounting will be in a manner that will be compatible with thermal control and structural integrity as required for the system. Mounting points on the equipment platform have mounting surfaces with an out-of-plane tolerance not to exceed .0127 cm (.005 in).

1.9 Operating Atmosphere

The probe internal pressure may vary during different parts of the mission. The probe will be sealed at 103 $kM/m^2 N_2$ (1 ATM) and may leak down to a pressure not less than 41.4 kN/m^2 (.4 ATM).

TABLE 1.0 SMALL PROBE INSTRUMENT ALLOCATION

Instrument	Max Envelope Dimensions Excluding Connectors cm	Volume (Nominal) $\pm$ 15% cc
o Temperature Gauge	3.8 x 5.1 x 5.8 (Note 1)	100
o Pressure Gauge	2.9 x 5.1 x 8.9 (Note 1)	115
o Accelerometer	3.2 Dia. x 4.6	33
o IR Flux Detector	3.6 x 6.3 x 16.5	328
o Stable Oscillator	6.6 Dia. Sphere	131
o Nephelometer	7.0 x 7.5 x 11.5	524

Note 1. Instrument electronics are installed into the probe integrated electronic assembly.

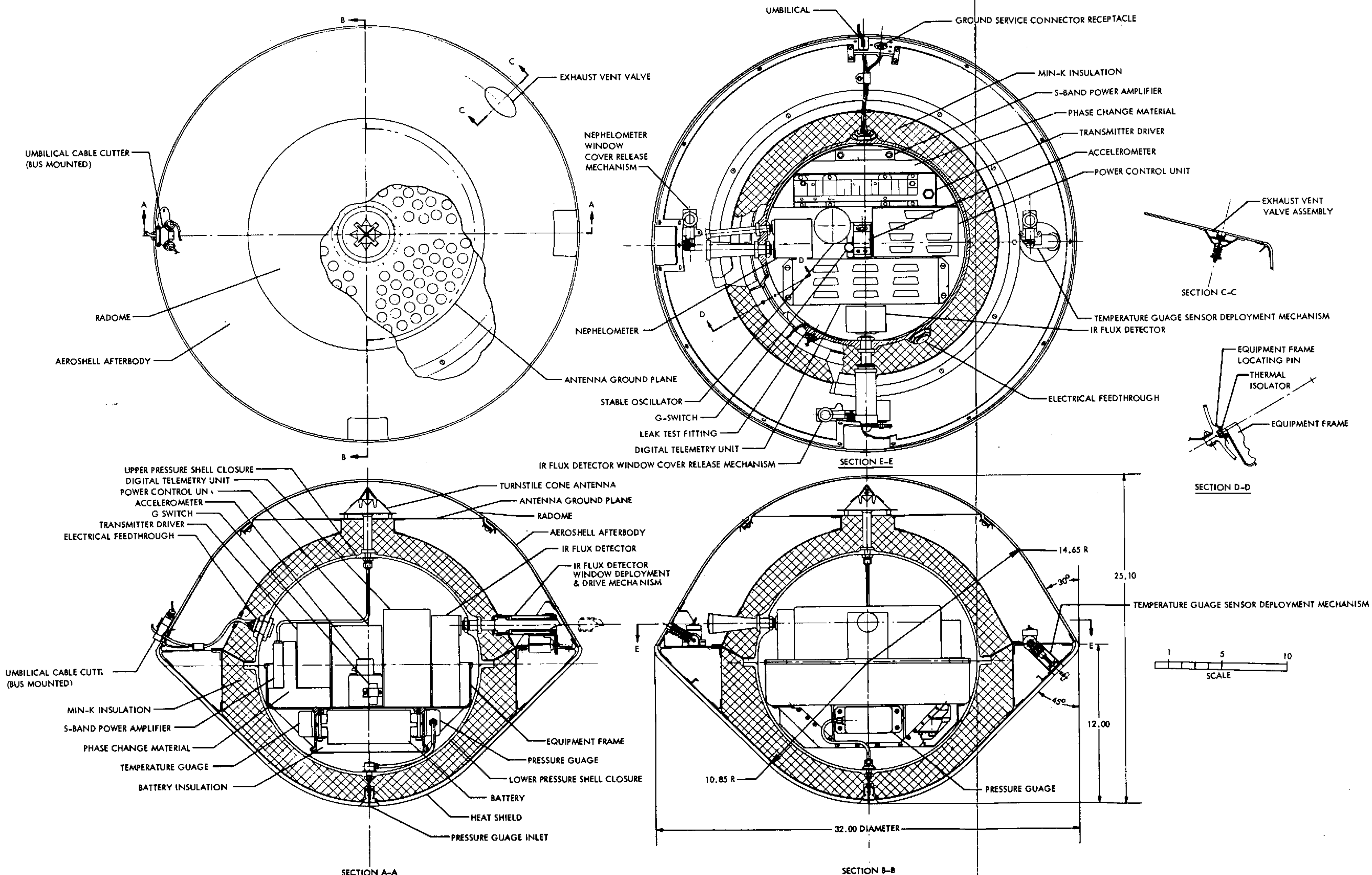


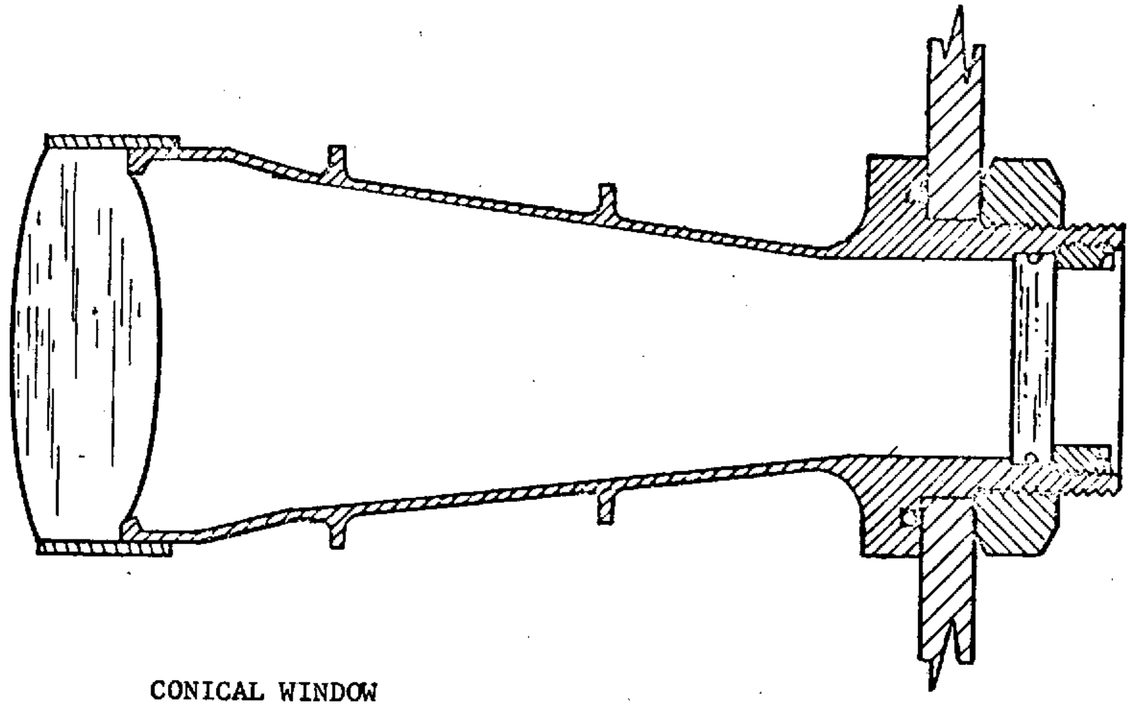
Figure 1.0 General Arrangement of Small Probe

FOLDOUT FRAME /

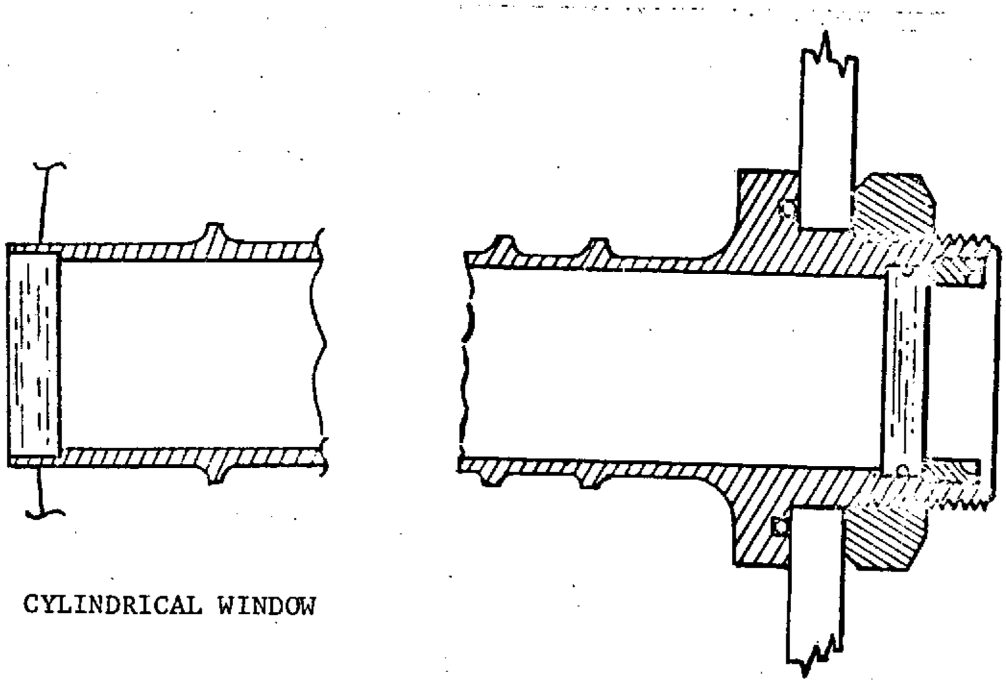
FOLDOUT FRAME 2

TABLE 1.1 SCIENCE MASS PROPERTIES (SMALL PROBE)

Instrument	Weight Lbs + 15% - 10%	Weight kg + 15% - 10%
Temperature Gauge	.65	.3
Pressure Gauge	.9	.4
Accelerometer	.4	.18
IR Flux Detector	1.2	.5
Stable Oscillator	.75	.34
Nephelometer	1.0	.45
Small Probe Total	4.90 Lbs	2.17 kg



CONICAL WINDOW



CYLINDRICAL WINDOW

FIGURE 1.2 - SMALL PROBE NEPHELOMETER WINDOWS

TABLE 1.2

SMALL PROBE - UNIQUE ENVIRONMENTS (Note 1)

MISSION PHASE / ENVIRONMENT	LAUNCH AND TRANS VENUS INJECTION	INTER-PLANETARY CRUISE	VENUS ENCOUNT, PRESEP, SEP & PROBE CRUISE	VENUS ENTRY	DESCENT
DURATION	≈ 30 Min.	≈ 91 Days	≈ 25 Days	≈ 50 Sec.	≈ 65 Min.
PROBE FUNCTIONAL MODE	NONOPERATE	NONOPERATE	*OPERATION **NONOPERATE	OPERATION	OPERATION
TEMPERATURE					
BULK ATMOSPHERE	283 to 305 K	305 to 266 to 310 K	369 to 402 K	255 to 272 K	272 to 336 K
WALL TEMP INSIDE PRESS SHELL	283 to 305 K	255 to 305 K	255 to 305 K	255 to 305 K	416 K
INSIDE SURFACE-SP AEROSHELL	283 to 305K	255 to 305 K	Forebody: 369 to 402 K Afterbody: 255 to 305 K	755 K	767 K
OUTSIDE SURF-SP AEROSHELL	283 to 305K	255 to 305 K	Forebody: 369 to 402 K Afterbody: 228 to 211 K	2533 K	767 K
ACCELERATION					
SPIN			.087 g per radius foot from S P spin axis	.087 g per radius foot from S P spin axis	.087 g per radius foot from S P spin axis
DECELERATION					
LINEAR				400 g	
AEROHEATING				MAX HEATING AT STAGNATION POINT & 40° ENTRY ANGLE IS 3800 WATTS/CM^2 CONVECTIVE & 900 WATTS/CM^2 RADIATIVE	

NOTE 1: Only those unique environments have been listed. If no entry is shown for a given environment phase for the small probe, please refer to large probe tables for respective environments. (See TABLE 1.5 of the Large Probe Interface Document)

2.0 ELECTRICAL POWER AND CABLING

2.1 Hardware - The Small Probe Electrical Power Subsystem consists of a battery, power switches, fault protection elements, and non-explosive device firing circuitry.

2.2 Power Sources

2.2.1 Ground Power - Ground power (via the ground checkout connector) and Probe battery power will be used as required for prelaunch activities. Use of Probe battery power will be limited during all testing phases and prohibited after final battery charge.

2.2.2 Cruise Power Source - Power during the cruise portion of the mission (launch insertion through Probe separation from the Probe Bus) will be supplied from the Probe Bus. The voltage shall be 28VDC ± 10%.

2.2.3 Post Separation Power - Power after Probe separation shall be derived from the Probe battery.

2.3 Power Control

2.3.1 Voltage Control - The output voltage of the Probe is regulated by monovalent operation of the silver-zinc battery and is held 28 VDC ± 10%.

2.3.2 Fault Protection - Power to science instruments will be fused individually at the levels shown in Table 2.1. The fuses will be located in the Probe power subsystem.

2.3.3 Switching - Instrument turn-on/off shall be accomplished by switching of the branch circuit in the Probe power subsystem.

2.4 Probe Output Characteristics

2.4.1 Distribution - Each scientific instrument will receive +28VDC electrical power through an individual, fused, branch circuit. The branch circuit will be energized/de-energized by Probe sequencer control. The power allotted to the instrument is measured at the spacecraft/instrument interface connector. See Table 2.0.

2.4.2 Voltage - Steady state power to the instruments shall be 28 vdc $\pm$ 10%, as measured at the instrument terminals.

2.4.3 Noise and Ripple - Except for the transient voltage excursions specified below, the peak-to-peak amplitude of any voltage excursion, periodic or aperiodic, shall not exceed 1.0 volt at any frequency between 30 Hz and 10.0 KHz decreasing at 6 dB/octave to 0.5 volts at 20.0 KHz and remaining at 0.5 volts through 100 MHz.

2.4.4 Transient Voltage Excursions

(a) Performance - The instruments shall be designed to accommodate without performance degradation voltage transients up to +42 vdc or down to +18 vdc for durations of 10 microseconds or voltages to +20 vdc for durations of 500 milliseconds on the nominal +28 vdc bus.

(b) Damage - The instruments shall be designed so that no damage, long term degradation, or modes where proper performance is not automatically resumed when the transient is removed, shall occur when 10 microsecond voltage transients up to +56 vdc or down to 0 vdc are seen on the nominal +28 vdc bus.

2.4.5 Source Impedance - The source impedance will not exceed 0. 5 ohms.

2.4.6 Other Voltages - None defined.

2.5 Instrument Load Characteristics

2.5.1 Load Current

2.5.1.1 Average - See Table 2.1.

2.5.1.2 Peak - See Table 2.1.

2.5.2 Duty Cycle - Not applicable.

2.5.3 Inrush Current - Upon application of the 28 vdc primary power to the instrument's primary power circuit the instrument's load current surge shall not exceed the larger of 400 milliamperes or 100 percent above the steady state requirement. The current shall return to steady state ± 10% within 10 milliseconds.

2.5.4 Load Noise - When operating from a power source impedance of 1.0 ohms, no single instrument shall feed back into the power input circuit any electrical noise either periodic or aperiodic in excess of 560 millivolts peak-to-peak at any frequency between 30 Hz and 10.0 KHz decreasing a 6 dB/octave to 280 millivolts peak-to-peak at 20.0 KHz and remaining at 280 millivolts peak-to-peak or less from 20.0 KHz through 100 MHz.

2.5.5 Grounding - The power supply circuit shall be isolated from chassis ground within each instrument by at least 10 megohms at 500 vdc.

2.5.6 Bonding - All instrument cases shall be bonded to Probe structure and shall have a maximum permissible bonding impedance of 10 milliohms. Bonding of science instruments shall be in accordance with MIL-B-5087 (latest revision).

2.6 Connectors and Cabling

2.6.1 Connector Types - The connector installed on an instrument that connects the instrument to the spacecraft harness shall be a male (straight or coaxial insert) pin connector selected from the Cannon nonmagnetic series of connectors (NMC-A-106 suffix). The use of two identical connectors on any one box is prohibited. Keying shall be provided. Use of end pins shall be avoided, if possible.

2.6.2 Interface Connector Location - The centerline of the interface connector shall be located as defined in Section 1.0, Figure 1.0.

2.6.3 Test Connector Location - The location of test connector (if required) shall be defined in Section 1.0, Figure 1.0.

2.6.4 Connector Pins - Cannon nonmagnetic pins shall be used. Connector savers shall be provided for each instrument connector. The pin assignments shall be in accordance with Table 2.0.

2.6.5 Connector Encapsulation - If encapsulated, the encapsulation shall be done with mated connectors to insure proper pin alignment. ARC shall be notified of the use of any encapsulated interface connectors.

2.6.6 Connector Identification - Each connector shall be identified by a number in accordance with Section 1.0.

2.6.7 Wiring - Wires between any two units of one instrument shall be twisted pairs whenever possible. Shielded twisted pairs, single conductor, or coaxial cables may be used if necessary, except that coaxial cables shall be avoided where the wiring between the two units must pass through Probe structural interfaces. The maximum wire size will be No. 20 AWG, and the minimum wire size (except for coaxial cables) will be No. 24 AWG.

ABBREVIATIONS

FOR TABLE 2.0

DHC	-	Data Handling & Command Unit
PCU	-	Power Control Unit (integrated into DHC)
TP	-	Twisted Pair Wire
STP	-	Shielded Twisted Pair Wire
TLM	-	Telemetry
MUX	-	Multiplexer (analog data)

Table 2.0

ELECTRICAL WIRING LIST
SMALL PROBE/INSTRUMENT INTERFACE

Item	Type of Signal	Source	Connector Pin	Load or Destination	Connector Pin	Remarks
TEMP GAUGE						
Converter Sync	Signal	Probe PCU		Temp Gauge		
+28 Vdc Power	Power	Probe PCU		Temp Gauge		
28 Vdc Return	Power	Probe PCU		Temp Gauge		
Analog TLM, Temp	Telemetry	Temp Gauge		Probe DHC		
Analog TLM, Thermistor	Telemetry	Temp Gauge		Probe DHC		
Analog TLM, Return	Telemetry	Temp Gauge		Probe DHC		
Temp Sensor Signal	Signal	Temp Sensor		Temp Gauge		
Temp Sensor Return	Signal	Temp Sensor		Temp Gauge		
PRESSURE GAUGE						
Converter Sync	Signal	Probe PCU		Press Gauge		
+28 Vdc Power	Power	Probe PCU		Press Gauge		
28 Vdc Return	Power	Probe PCU		Press Gauge		
Analog TLM, Press	Telemetry	Press Gauge		Probe DHC		
Analog TLM, Thermistor	Telemetry	Press Gauge		Probe DHC		
Analog TLM, Return	Telemetry	Press Gauge		Probe DHC		
ACCELEROMETER						
Converter Sync	Signal	Probe PCU		Accelerometer		
+28 Vdc Power	Power	Probe PCU		Accelerometer		
28 Vdc Return	Power	Probe PCU		Accelerometer		

Table 2.0 (Continued)

ELECTRICAL WIRING LIST
SMALL PROBE/INSTRUMENT INTERFACE

Item	Type of Signal	Source	Connector Pin	Load or Destination	Connector Pin	Remarks
Accelerometer (continued)						
Analog TLM, Axial Accel.	Telemetry	Accelerometer		Probe DHC		
Analog TLM, Turbulence	Telemetry	Accelerometer		Probe DHC		
Analog TLM, Thermistor	Telemetry	Accelerometer		Probe DHC		
Analog TLM, Return	Telemetry	Accelerometer		Probe DHC		
Checkout No. 1	Ground Checkout	Accelerometer		Checkout Connector		
Checkout No. 2	" "	"		" "		
Checkout No. 3	" "	"		" "		
Checkout No. 4	" "	"		" ""		
Checkout Return	" "	"		" "		
NEPHELOMETER						
Converter Sync	Signal	Probe PCU		Nephelometer		TP-A
+28 Vdc Power	Power	Probe PCU		Nephelometer		TP-A
28 Vdc Return	Power	Probe PCU		Nephelometer		TP-A
Serial Digital Data #1	Telemetry	Nephelometer		Probe DTU		
Serial Digital Data #2	Telemetry	Nephelometer		Probe DTU		
Bit Clock	Timing	Probe DTU		Nephelometer		
Word Gate #1	Timing	Probe DTU		Nephelometer		
Word Gate #2	Timing	Probe DTU		Nephelometer		

Table 2.0 (Concluded)

ELECTRICAL WIRING LIST
SMALL PROBE/INSTRUMENT INTERFACE

Item	Type of Signal	Source	Connector Pin	Load or Destination	Connector Pin	Remarks
IR FLUX RADIOMETER						
Converter Sync	Signal	Probe PCU		IRFR		
+28 Vdc Power	Power	Probe PCU		IRFR		
28 Vdc Power Return	Power	Probe PCU		IRFR		
Bit Clock	Timing	Probe DTU		IRFR		
Digital Data #1	Telemetry	IRFR		Probe DTU		
Digital Data #2	Telemetry	IRFR		Probe DTU		
Digital Data #3	Telemetry	IRFR		Probe DTU		
Word Gate #1	Timing	Probe DTU		IRFR		
Word Gate #2	Timing	Probe DTU		IRFR		
Word Gate #3	Timing	Probe DTU		IRFR		

C-4

TABLE 2.1 INSTRUMENT LOAD CHARACTERISTICS

INSTRUMENT	FUSE RATING (Amps)	VOLTAGE (Volts)	AVERAGE CURRENT (Amps)	PEAK CURRENT (Amps)
ACCELEROMETER	1/8	+ 28 VDC $\pm$ 10%	0.036	0.16 at 400 G's peak, duration 10 seconds
PRESSURE	1/8	+ 28 VDC $\pm$ 10%	0.02	
TEMPERATURE	1/8	+ 28 VDC $\pm$ 10%	0.02	
IR FLUX DETECTOR	3/4	+ 28 VDC $\pm$ 10%	.071	.28 Max.
NEPHELOMETER	1/4	+ 28 VDC $\pm$ 10%	0.071	
STABLE OSCILLATOR	1/16	+ 28 VDC $\pm$ 10%	.009	

3.0 DATA HANDLING AND COMMAND (DHC)

3.1 Functional Description

The Small Probe DHC will accept information in digital, analog, or state form, convert the analog information to digital form, and arrange all information in an appropriate format for time multiplexed transmission to Earth or storage on board the Probe. The Probe will also supply the instruments with various timing and operational status signals and functional commands.

3.1.1 Telemetry Word - A telemetry word in all formats will consist of 7, 8 or 10 bits. Probe generated words will be transmitted with the most significant bit first.

3.1.2 Data Bit Rates - The DHC will be capable of processing scientific and engineering data at bit rates of 512 and 64 bits/second. The 512 bit/s rate is used for the Checkout Mode. Bit rate changes will occur within one bit period following the completion of the current data frame after the reception of a bit rate change command by the data system sequencer.

3.1.3 Frame - The data subsystem will assemble the information from the instruments into frames of 768 bits. Frame word assignments are shown in Figures 3.1 and 3.2.

3.1.4 Format and Word Assignments - The words in a frame are assigned in several formats. The formats are organized for particular probe operational modes and are selected by internal sequencer command. Format changes will occur within one bit period following the completion of the current frame upon the reception of a format change command by the data subsystem sequencer. Checkout will be accomplished by exercising all of the flight formats.

3.1.5 Format A - Format A is the first format for scientific information and is used during pre-entry and post-landing. Word assignments for Format A are shown in Figure 3.1.

3.1.6 Format D-1 - Format D-1 is used during the high altitude portion of the descent (between 70 and 38 km). Word assignments for Format D-1 are shown in Figure 3.2.

3.1.7 Format D-2 - Format D-2 is a descent scientific format. It is used to accommodate different science requirements for the low altitude portions of the descent. Word assignments for Format D-2 are shown in Figure 3.2.

3.1.8 Operational Modes of Data Subsystem - The data subsystem will be capable of operating in three basic modes as follows:

(a) Data are transmitted real time.

(b) Data are stored.

(c) Data are transmitted real time and interleaved with the transmission of stored data.

3.1.9 On-Board Storage Capacity - A storage capacity of 5120 bits will be provided by the data subsystem.

3.2 Signals from Scientific Instruments

3.2.1 Telemetry List - Table 2.0 lists the data signals from the instruments.

3.2.2 Characteristics - The characteristics of digital, analog, or state signals from the scientific instruments to the data subsystem shall be:

(a) Digital Signals - shall have the following characteristics.

(1) Amplitude

Logic 0 (false)	0 to 0.4 Vdc
Logic 1 (true)	2.4 to 5.5 Vdc

(2) Rise and Fall time (between 10% and 90% points) - less than 1.0 microsecond

(3) Loading - SN5414 devices shall be used to receive digital signals. Interconnecting cables shall have a maximum capacitance of 200 picofarads.

(4) Duration (between 50% points) - Minimum duration of digital signals to the Probe DHC shall be 100 microseconds.

(5) Word length - Digital words may be either 7 or 10 bits.

(6) Signals shall be referenced to chassis.

(7) Overvoltages will result in hardware damage.

(8) A short circuit can be accepted between any two digital signal lines or between signal lines and ground without damage or effect upon other channels.

(b) Analog Signals

(1) Full scale input range shall be 0 to 5 volts.

(2) All input signals shall be referenced to a signal return, which shall be returned to the DHC.

(3) The DHC channel input impedance during sampling periods will be 30 K ohm minimum, shunted by an input impedance of 350 picofarads, maximum. Maximum back current during nonsampling periods shall be 100 nanoamperes.

(4) Overvoltage - Voltages up to $\pm$ 10 volts can be accepted between any two signal inputs or between any input and ground, without damage or effect upon other channels. Valid data will not be provided from the affected input channels during application of such over-voltage.

(5) Short circuit - A short circuit can be accepted between any two signal inputs or between any input and ground, without damage or effect on other channels.

(6) A/D Conversion - An analog signal received by the data subsystem will be converted into a 7 or 10-bit digital word as required.

(7) Coding Accuracy - For a source impedance of less than 5000 ohms the overall conversion accuracy will be $\pm$ 1 bit. For larger source impedances the error will be increased.

(c) State or Bilevel Signals

(1) Input levels between 0 and 1.2 volts will be encoded as a logical "zero". Input levels between 2 and 10 volts will be encoded as a logical "one".

(2) All input signals shall be referenced to chassis.

(3) Channel input impedance will be 30 K ohms or greater.

(4) Overvoltages up to 10 volts can be accepted between any two signal inputs or between any input and ground, without damage or effect on other channels. Valid data will not be provided from the affected channels during application of the overvoltage.

(5) Short circuit - A short circuit can be accepted between any two signal inputs or between any input and ground, without damage or effect upon other channels.

3.3 Signals to Scientific Instruments -

The DHC will supply digital timing and command signals to the instruments as identified in Table 2.0. Characteristics of these signals are as follows:

(1) Amplitude -

Logic 0 (false)	0 to 0.4 Vdc
Logic 1 (true)	2.4 to 5.5 Vdc

(2) Rise and Fall time (between 10% and 90% points) - less than 1.0 microsecond.

(3) Loading - SN 5414 is the recommended load device. Other loads shall have a maximum shunt capacitance of 350 picofarads and a resistance of between 1000 and 10,000 ohms. Interconnecting cables shall have a maximum capacitance of 200 picofarads.

(4) Duration (between 50% points)

Bit Clock	Prevailing Bit Rate at 50 $\pm$ 5% duty cycle
Word Gate	10 millisecond pulse
Frame Rate	10 millisecond pulse
Command Signals	as required

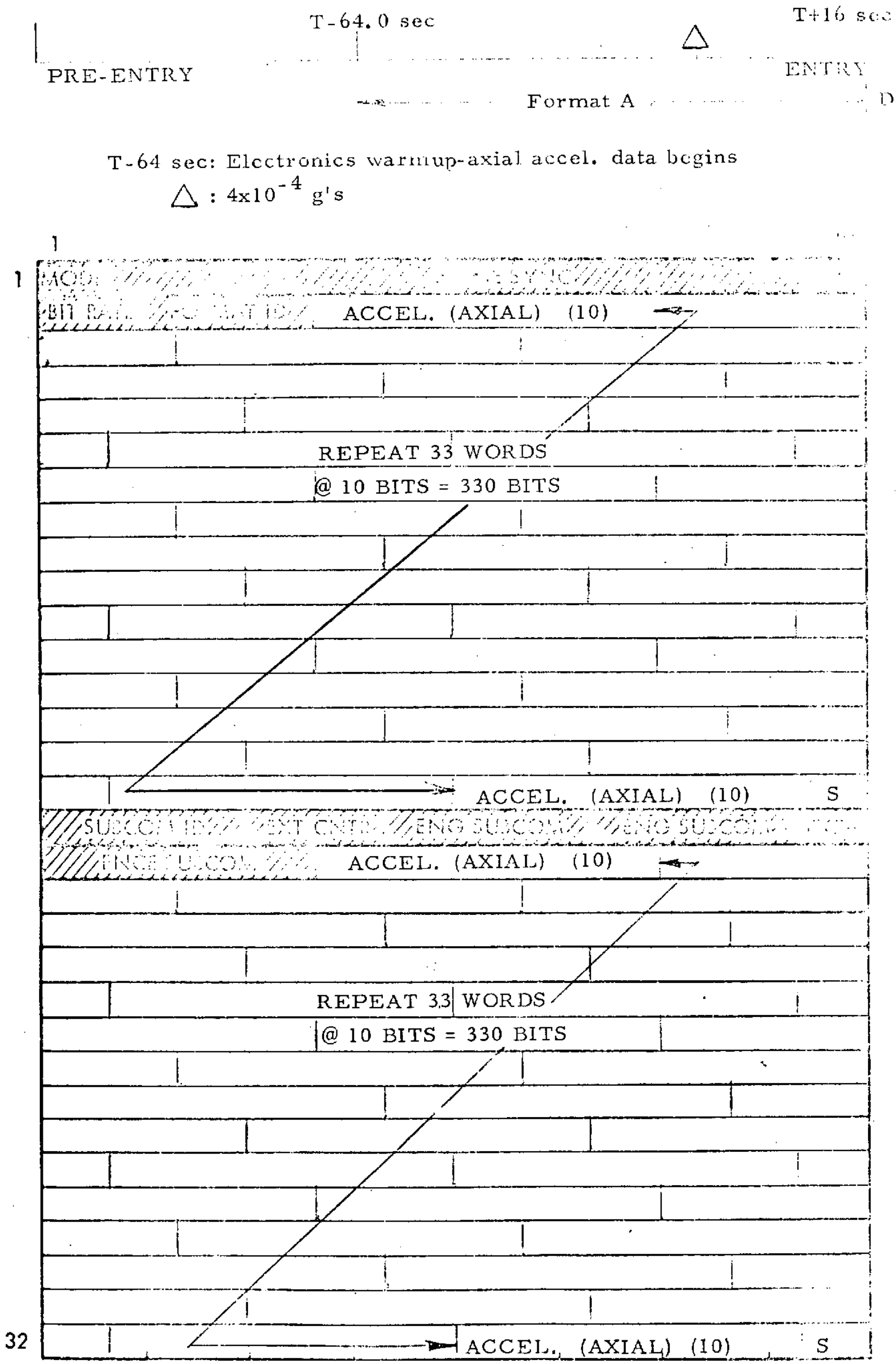
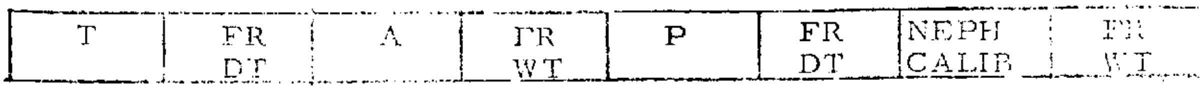


Figure 3.1 SMALL PROBE FORMAT A



T = Temperature Thermistor
P = Pressure Thermistor
A = Accelerometer Thermistor
Blanks avail. for new housekeeping data

FR/DT = Flux Rad. Detect. Temp.
FR/WT = Flux Rad. Window Temp.

NFR: Net Flux Radiometer
AT: Accelerometer, Turb.
A_A: Accelerometer, Axial

P: Pressure
T: Temperature
N: Nephelometer

S: Spare

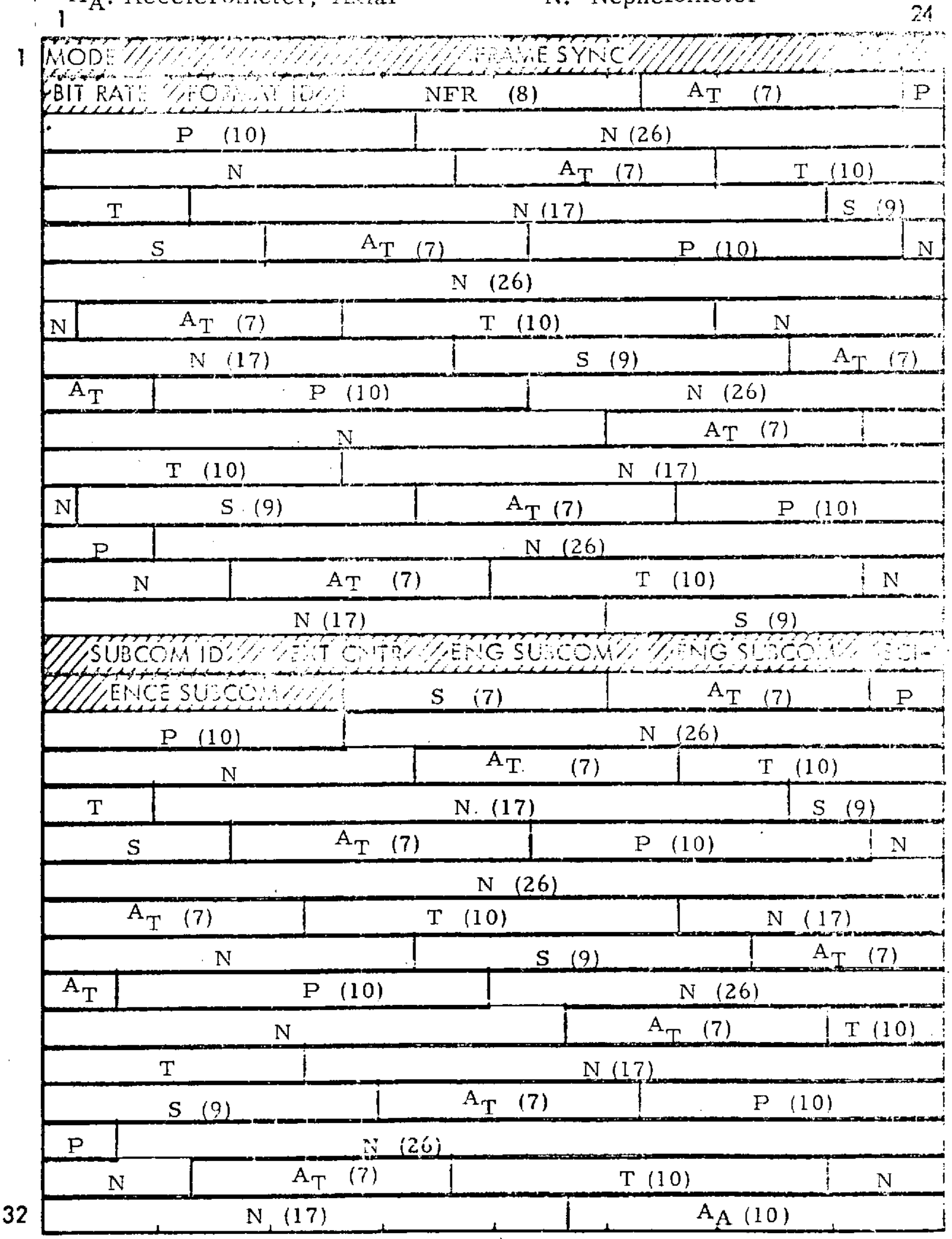


Figure 3.2 SMALL PROBE FORMAT D

4.0 THERMAL

4.1 Thermal Control

Thermal control is provided by thermal insulation, coatings, phase change material and a science window heater on the descent capsule and the aeroshell heat shield to maintain an environment assuring that all Probe components are within their temperature limits for all mission phases.

The Small Probe temperature limits interior and exterior to the pressure vessel as a function of the mission phase are given in Table 4.1 under both operating and non-operating conditions.

The scientific instruments must be capable of proper operation, as applicable for either the operate/non-operate mode, per the limits specified in Table 4.1.

TABLE 4.1 SMALL PROBE TEMPERATURE LIMITS

MISSION PHASE	INTERIOR TO PRESSURE VESSEL, K	EXTERIOR TO PRESSURE VESSEL, K
Pre-Launch (Operating)	256 to 305 K	200 to 366 K
Pre-Launch (Non-operating)	256 to 302 K	200 to 366 K
Launch and Cruise (Non-operating)	256 to 302 K	200 to 366 K
Cruise (Operating)	256 to 305 K	200 to 366 K
Descent (Operating)	266 to 322 K	200 to *

* Each exterior component must be designed with upper temperature limit consistent with maximum atmospheric temperature for which it is intended to operate.

5.0 ELECTROMAGNETIC INTERFERENCE

Spacecraft equipment and scientific instruments shall comply with the electromagnetic interference requirements as given by specification, using the applicable requirements of MIL-STD-461. Each probe subsystem is electromagnetically compatible with all other probe subsystems, scientific instruments, and the probe bus spacecraft as specified in (TBS) and the associated ground support equipment. Electromagnetic compatibility requires that:

(a) The normal operation of each probe subsystem shall not be adversely affected by signals or voltage variations generated by other subsystems or scientific instruments as part of their normal operating mode or intended function.

(b) No probe subsystem shall disturb normal operations of other subsystems or scientific instruments by emission of signals or voltage variations other than those produced to perform its intended function.

(NAS 2-7249)

CR 137509 Volume I Appendices
Section 7 (Part 2 of 3)

CR 137510 Volume I Appendices
Section 8-11 (Part 3 of 3)

CR 137511 Volume II. Preliminary Program Development Plan

CR 137512 Volume III. Specifications

Loretta Vose
for William R. Johnson
Chief, Technical Information Division

Enclosure: 1 cy each subject report

cc: NASA Hqrs, Code KSI (w/o encs.)'

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